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ATR-74(7341)-3

Operations Analysis (Study 2.1)

Payload Designs for Space Servicing

DRA

Prepared by
Advanced Mission Analysis Directorate
Advanced Orbital Systems Division

30 June 1974

Prepared for OFFICE OF MANNED SPACE FLIGHT
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D. C.

Contract No. NASW-2575

Systems Engineering Operations

THE AEROSPACE CORPORATION

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Prepared by

M. G. Wolfe
Advanced Mission Analysis
Directorate

Approved

R. R. Wolfe
NASA Study 2.1 Director
Advanced Mission Analysis
Directorate

L. R. Sitney, Assoc. Group
Director
Advanced Mission Analysis
Directorate
Advanced Orbital Systems
Division

FOREWORD

The objective of Study 2.1, Operations Analysis, is to originate and evaluate various operational concepts for the STS system which may offer a means of reducing future resource expenditures. It is a natural follow-on to Study 2.6, performed under contract NASW-2472, and emphasizes the development of a much more substantiated and detailed satellite design data base than has been available heretofore. Standardization (at either the subsystem module or the component level) and modularization for space servicing are essential factors in the study; in addition, a sophisticated space operations computer simulation program to perform trade-offs of space operations is being developed in parallel. The data that is documented herein will be used to exercise the computer simulation program in order to investigate varying operational policies.

The payload design data presented in this report is developed from the October 1973 NASA Mission Model for automated payload operations. This model, based upon an expendable payload design concept has been revised to reflect space servicing of specific payload programs. The revised payload descriptions and program characteristics are provided in Table 3-1. Those payloads identified as reasonable candidates have been reconfigured for space servicing based upon an in-depth analysis of four NASA and DOD payload designs. These payloads consist of both space replaceable units (SRUs) and nonreplaceable units (NRUs). Their composition, in terms of module definitions is provided in Table 10-2. The detail development of these modules, with traceability to the original NASA requirements is provided by the remaining tabular data.

This part of the study represents approximately 20 percent of the total Study 2.1 effort. At the request of NASA it is being published ahead of the remaining study documentation to provide wide dissemination of the data and to permit the results to be injected into work being carried out under other NASA contracts.

Study 2.1 is one of several study tasks being conducted under NASA Contract NASW-2575 in FY 1974. The NASA Study Director is Mr. V. N. Huff, NASA Headquarters, Code MTE.

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ACRONYMS

AEA	auxiliary electronics assembly
AVCS	Attitude and Velocity Control System
CMG	control moment gyro
DSP	Defense Support Program
DTU	digital telemetry unit
EIRP	effective isotropic radiated power
EOS	Earth Observatory Satellite
ESA	earth sensor assembly
FSSA	fine sun sensor assembly
G&N	guidance and navigation
GCPA	guidance and control processor assembly
GST	gimbal star tracker
IMU	inertial measuring unit
IOC	initial operational capability
IPACS	Integrated Power and Attitude Control System
LEO	low earth orbit
MDAC	McDonnell Douglas Astronautics Co.
NRU	nonreplaceable unit
PTU	processor terminal unit
RCS	reaction control system
RDT&E	research, development, test and engineering
RGA	rate gyro assembly

RTU	remote terminal unit
RW	reaction wheel
SEPS	Solar Electric Propulsion System
SRU	space replaceable unit
SSA	sun sensor assembly
SSDSP	Space Serviceable Defense Support Program
SSEOS	Space Serviceable Earth Observatory Satellite
STDN	Spacecraft Tracking and Data Network
TDRSS	Tracking and Data Relay Satellite System
TDRS	Tracking and Data Relay Satellite
TT&C	telemetry, tracking and command

1. INTRODUCTION

1.1 BACKGROUND

The study documented herein is part of a continuing activity concerned with the investigation of potential modes of operating in space in the Space Shuttle era. The October 1973 NASA Mission Model provides a definition of various NASA and non-DOD automated payload configurations when employed in an expendable mode. The model also specifies a launch schedule for initial deployment of payloads as well as for subsequent replacements at periodic cycles. This model and its associated payload definitions serve as a foundation for the data presented in this report. The reference model has been revised to reflect automated space servicing of payloads as an operational concept instead of the existing expendable approach. In reference 1, the indication is that the bulk of a payload's subsystems and mission equipment require no support over the lifetime of the program. However, failure of a single unit could result in loss of the mission objectives. When space servicing is employed, the approach is to replace only that unit causing the anomaly. This concept affords an opportunity to standardize space replaceable units, as well as to reduce the expense of logistics support, by allowing multiple servicing on any single upper stage/shuttle flight.

Before any meaningful analysis can be performed, it is necessary to develop an entirely new set of payload definitions configured for space servicing. The mission objectives remain unchanged; however, the satellite configuration must reflect a high degree of commonality while minimizing the restrictions to be imposed upon the payload user. Examination of the reference mission model indicates there are 95 automated payload programs scheduled in the time period of 1980 through 1991. Of these, 42 programs are reasonable candidates for space servicing operations. The reference model also specifies that 29 different configurations are required to satisfy these mission objectives, indicating that certain payloads can satisfy more than one program. These 29 payloads therefore have been reconfigured for space servicing. All the fundamental data evolving from a

requirements review, plus definition of servicing mechanisms, and reconfiguration of the candidate payloads is provided in this report. For example, Table 3-1 provides the revised mission model payload characteristics indicating which payloads were selected for space servicing. These payloads are composed of various subsystem and mission equipment modules as defined in Table 10-2. The development of the modules and the associated rationale for segregation of equipments is provided in the text.

This data should serve as a new foundation for further work associated with standardization and space servicing. As mission requirements change, it will be necessary to revise the data of this report. As new subsystem concepts are developed, these also should be included. At this point in time the designs provided in this report represent only the first step. Further optimization is required. To the extent possible, this will be done by use of a simulation program, also developed under this study effort. The simulation program allows the trade-off to be made of SRU reliability versus logistic support costs required to replace less reliable designs. The results of this trade-off, using the payload definitions herein contained, will be documented in the final report of Study 2.1, Operations Analysis, to be issued at the end of the contract period.

1.2 OBJECTIVES

The objectives of this study are as follows:

a. To define the characteristics of the NASA mission model.

b. To identify those missions which are potential candidates for space servicing.

c. To develop all the significant design characteristics of of the mission model automated payloads (including those selected for space servicing.)

d. To document the design data in a form that can be used as input to a space operations computer simulation program.

1.3 APPROACH

Development of the payload and module design definitions presented in this report evolves from several sources of information. An in-depth study was previously performed at The Aerospace Corporation to investigate the feasibility of reconfiguring the Defense Support Program (DSP) (Ref. 5) satellite to a space serviceable configuration. This effort examined many details and conceived specific design approaches which, when examined in the light of other contractor efforts, continue to have merit. This original work was later extended to examine the Defense Support Communications Satellite, DSCS-II, as another test of space servicing reconfiguration. Added to these were two other studies performed on NASA payloads, The Earth Observatory Satellite (EOS) (Ref. 1) and the Large Space Telescope (LST) (undocumented.) These basic design data as well as data from other contractors has provided the foundation for the approach employed in this study.

However, the principal objective of these prior efforts was to perform an in-depth assessment of a single satellite to prove design fesability. Cost trade-offs against logistics support were limited to the one program being analyzed. Therefore, the approach taken for this study was to use these design data where applicable to establish a set of standard reference subsystem modules. These modules, along with mission equipment and nonreplaceable items were used to reconfigure the 29 payload designs of interest for space servicing. In this way the broad application of space servicing to several programs with differing mission requirements can be assessed against the overall payload procurement and logistics support costs.

The October 1973 NASA Mission Model for automated payload programs provides the basic reference for this design effort. A complete review of the mission requirements for each of the 95 programs specified was performed. These requirements were then collected into subsystem

and mission equipment categories. Each group was examined further for commonality of subsystem performance. In this way it was possible to satisfy the performance requirements with a series of components which could be assembled as a space replaceable unit (SRU). These SRUs were then used to configure each of the 29 payloads of interest. Although each SRU is unique, commonality of interfaces is maintained to support SRU replacements and equipment redesign. In each case there are also non-replaceable items and nonstandard mission equipment SRUs. The process of segragating these equipments is somewhat arbitrary and is therefore subject to specific ground rules, as described in the next section (1.4).

Disseminating this large amount of data in a form that can be easily interpreted has presented a formidable problem. For this reason a hierarchy of data tables has been developed which allows the reader to progress from the basic requirements through the design process to arrive at the final payload configurations. This hierarchy is summarized in Figure 1-1. Since many of the mission requirements are vague and obviously subject to change, it was felt important to provide complete traceability from requirements to design. As changes occur, a new entry may be made in the applicable table and the effect of that change traced to the design solution. In addition, it is recognized that in general the weights of components and SRU baseplates are conservative. A redesign effort could possibly reduce these values, thereby enhancing the argument for space servicing. Again, traceability is provided by following the hierarchy of tables shown in Figure 1-1.

A further consideration of the approach employed in this effort is the integration of the service unit design with the payload reconfiguration process. Although other NASA study efforts (References 19, 20, 21) have addressed these two items separately, it has been the experience at Aerospace that the concept must be developed as a total system. A design approach is also desired that is compatible with the Shuttle bay size constraints and at the same time one that allows multiple payload operations. Consequently a section of this report is devoted to servicing concepts and their applicability to the payloads of interest.

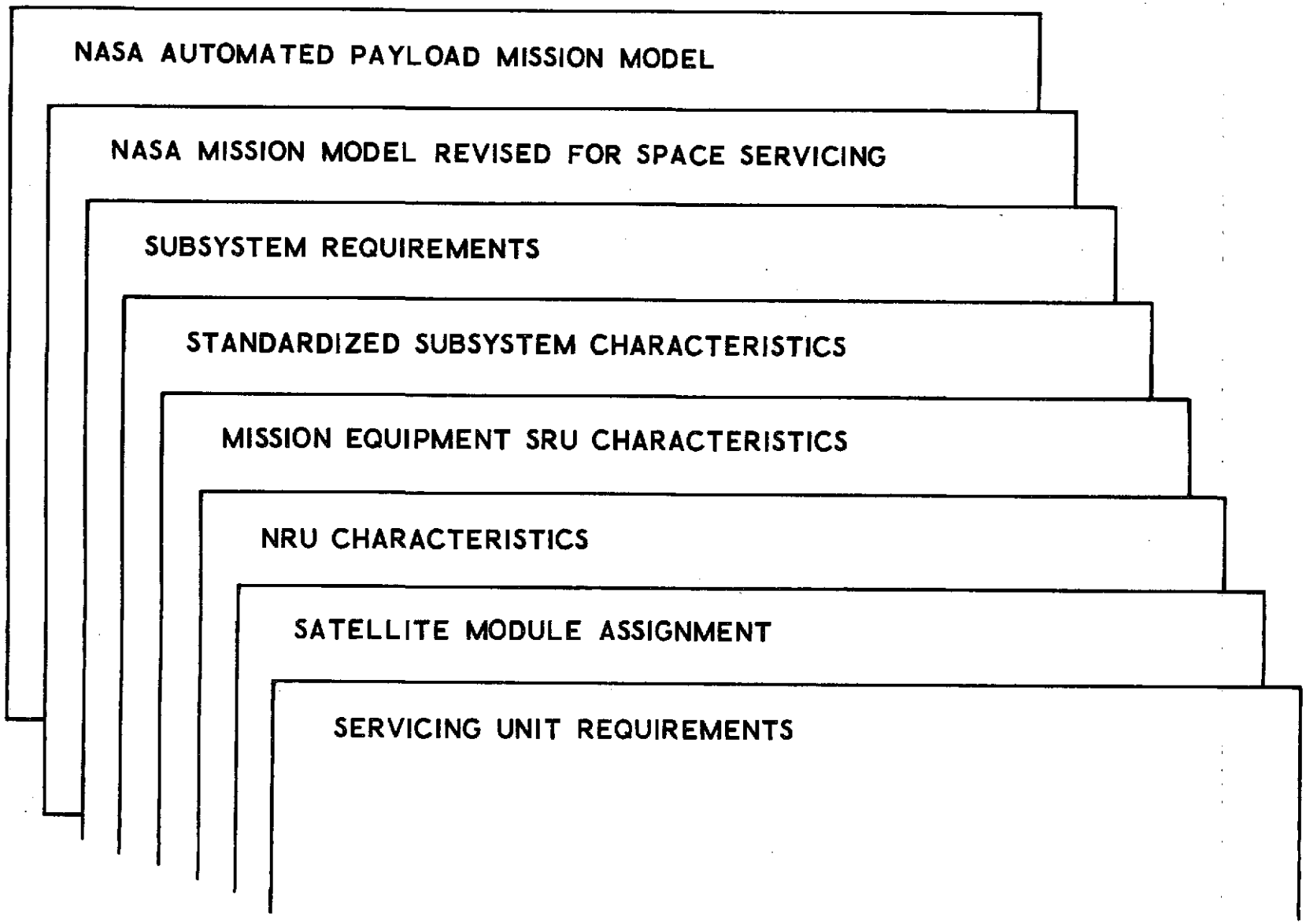


Figure 1-1. Hierarchy of Data Development for Revised NASA Mission Model

Finally, to emphasize the hierarchy of data tables, it is possible to use this data to assess completely new payload definitions beyond those of the reference NASA Mission Model. As new payload programs evolve, their requirements for subsystem support can be examined against the range of data in the requirement tables of each section. If the new payload requirements fall within this range, which at present is quite broad, it is then possible to trace the development to a set of standard subsystem SRUs. Any mission equipment can also be isolated for a new payload design and correlated with the design data provided in this report to estimate the number and weight of SRUs required. Reliability information is usually unavailable on new designs, however rough estimates can be established for the mission equipment based upon its complexity and state of development. In this way, an entirely new payload can be reconfigured and the benefits of space servicing assessed with a minimum of effort. If space servicing appears advantageous a detail design effort can be initiated for the particular payload of interest.

1.4 PAYLOAD SELECTION

References 2, 3 and 4 define to some degree all the NASA and non-NASA/non-DOD payloads anticipated for the time period 1973 to 1991. These payloads were reviewed and those which appeared to be amenable to space servicing were selected as candidates for further study. The selection criteria are as follows:

a. Only unmanned payloads are considered. It was anticipated that the design of a man-rated payload would be different enough from that of an unmanned payload to preclude the application of the same type of servicing technique to both. Furthermore, man-rated equipment must be designed to a higher factor of safety, and thus a weight penalty would be paid in the servicing of automated payloads. In addition, those payloads designated as manned or requiring manned revisits indicate probable requirement for on-orbit adjustment or checkout which could only be carried out via use of a manned system.

b. Only payloads in accessible earth orbits are considered. Lunar and interplanetary missions were eliminated since the satellites could not be reached for servicing.

c. Only active payloads are considered. Any satellite which is strictly passive, i.e., has no moving parts or functioning onboard equipment which would need replacement, has been eliminated.

d. Any payloads with a NASA estimated weight of 230 kg or less are eliminated from consideration. Generally, these are recommended for consideration as part of a multi-mission satellite which perhaps could be space serviced. All the satellites in this category were found to have very light (23 kgor less) mission equipment packages; the rest of the weight consists of housekeeping equipment and structure. Experience has shown that the weight penalties paid for converting a small expendable satellite to one which can be space serviced can be as high as 500 percent. A more profitable approach would be to have various small sensors share the same housekeeping and support equipment on a common satellite base. (This assumes of course that compatible orbits, launch dates, etc. can be established.)

e. Only those payloads which have mission equipment which is considered to be adaptable to space servicing are selected. This is a judgement factor based on experience gained in preparing point designs of space-serviceable satellites; for example, the Defense Support Program (DSP) (Reference 5) and the Earth Observatory Satellite (EOS) (Reference 1). Some payloads, such as the EOP-5 Gravity Gradiometer, contain mission equipment which is too large to be replaced on orbit. These kind of satellites are recommended as candidates for ground refurbishment.

f. For the purposes of this study, if two payloads are identical in all respects except for their title and code number, or are merely located in different orbital positions, they are not considered separately.*

* This is the reason the design activity (as it will be seen later) considered approximately 29 satellites only, whereas there are approximately 42 satellites identified in Reference 2 which appear amenable to automated space servicing.

1.5 SUBSYSTEM DEFINITION

Previous studies (Refs. 1 and 5) have indicated that it is reasonable to consider the space replaceable units to be subsystem peculiar. Early in the study the subsystems selected for modularization and standardization were: (1) guidance and navigation, (2) propulsion, (3) attitude control, (4) command, data processing and instrumentation, and (5) electrical power. (What is considered to be included in the various subsystems is listed in Table 1-1.) After the completion of the review and selection, it could be seen that only four of the candidates required propulsion in addition to attitude control. It was reasoned that, if the satellite was to be visited for servicing, it would have had to have been placed in orbit by the same launch vehicle combination and therefore any propulsion requirements specified in the NASA documentation were assumed not to be required.

As the study progressed, the term "Attitude & Velocity Control System (AVCS) was substituted for "Attitude Control System" and the "Command, Data Processing and Instrumentation System" was divided into the "Telemetry, Tracking and Command System" and the "Data Processing System".

The list of payloads selected for detailed study is listed in Figure 1-2, which also shows typical satellites for each payload category considered.

1.6 STANDARDIZATION

An important part of this study is the effort to define standardized rather than unique system elements. This approach is applied not only to the satellite space replaceable units but also to the servicing unit itself (three modules are used to assemble four different service unit options, as will be discussed later).

A number of different approaches to modularization and standardization of the satellites can be conceived. Standardization can take place at the satellite level (the multipurpose satellite), the subsystem level, the equipment level or the component level. The approach adopted in this study is illustrated in Figure 1-3. The standardized modules are subsystem peculiar and variants of these standardized modules are configured by removing unnecessary components for specific applications.

Table 1-1. Payload Subsystem Definition

Subsystem Element	Typical Hardware
Structures, Mechanisms (All structural and mechanical elements which are not part of the other functional subsystems. Also includes install. of subsystems into spacecraft, attachment of experiments and docking system for retrievable satellites.)	• Spacecraft Structure • Equipment Supports • Sun Baffles • Balance Booms and Extns. Mech. • Antenna Deployment Mech. • Solar Array Deployment Mech. • Retrieval Docking Ring
Environmental Control (All elements which alter and/or control the temperature of the payload and components thereof.)	• Thermal Louvers • Insulation • Thermal Paints and Coatings • Thermostats • Heaters • Radiators, Heat Pipes
Guidance, Navigation, and Stabilization (All elements which provide flight control, orbit positioning, and attitude hold, but excluding thruster system.)	• Position Sensors (Solar, Earth, Star) • Momentum Wheels • Flight Control Electronics • Gyros • Inertial Ref. Units
Propulsion (All elements which are provided for major changes in velocity vectors.)	• Solid-Propellant Motors • Monopropellant or Bi-Propellant Thrusters • Tankage for Propellant, Pressurants • Plumbing and Valves • Propellant, Pressurants
Attitude Control (Elements for control and/or maintenance of attitude which involve mass expulsion.)	• Cold Gas, Monopropellant or Bi-Propellant Thrusters • Tankage for Propellant, Cold Gas, Pressurants • Plumbing and Valves • Propellant, Pressurants

Subsystem Element	Typical Hardware
Telemetry, Tracking and Command (All elements of telemetry, communications and command.)	• Transducers • Transmitters, Beacons, Transponders • Receivers/Decoders • Multiplexers/Encoders • Antennas • RF Power Amplifiers • Command, Data Storage, Timing
Data Processing (All elements of data processing.)	• Data Handling, Processing, Storage Equipment • Signal Conditioners
Electrical (All elements of electrical power generation, control, distribution. Also includes pyrotechnic hardware.)	• Batteries • Solar Arrays (Incl Structural Panels, Solar Cell Diodes, Interconnects; Orientation Assy) • Voltage Regulators, Inverters • Distrib., Primary and Instrumentation Cabling • Pyrotechnic Devices (Squibs, etc.)
Mission Equipment (All elements which are mission-peculiar and not part of the supporting spacecraft. Includes any data processing equipment which is integral with experiments.)	• Telescopes • Cameras • TV Cameras • Physics Experiments • Radiometers, Spectrometers, etc.
Payload Assembly, Integration (All elements which are part of the payload system but do not remain with the payload in orbit.)	• Payload Adapters and Interstages • Fairings (not Std. Exit) • Umbilicals • Safety Devices • Separation Devices

ASTRONOMY PROGRAM

AST-1B	COSMIC BACKGROUND EXPLORER
AST-1C	ADVANCED RADIO ASTRONOMY EXPLORER
AST-3	SOLAR PHYSICS MISSION
AST-9A	FOCUSING X-RAY TELESCOPE - 1.2M
AST-9B	FOCUSING X-RAY TELESCOPE - 3.0M

PHYSICS PROGRAM

PHY-1A	SMALL HIGH ENERGY OBSERVATORY
PHY-1B	UPPER ATMOSPHERE EXPLORER
PHY-1C	MEDIUM ALTITUDE EXPLORER
PHY-2A	GRAVITY PROBE B_2

EARTH OBSERVATIONS PROGRAM

EO-3A	EARTH OBSERVATION SATELLITE
EO-4A	SYNCHRONOUS EARTH OBSERVATORY SATELLITE
EO-6	TIROS
EO-7A	SYNCHRONOUS MET SAT

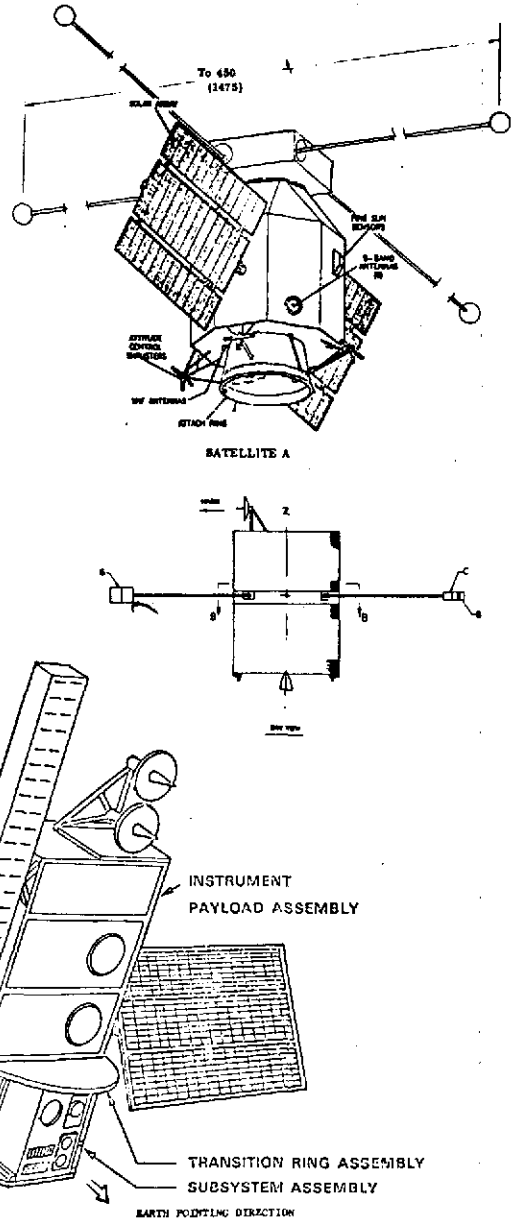
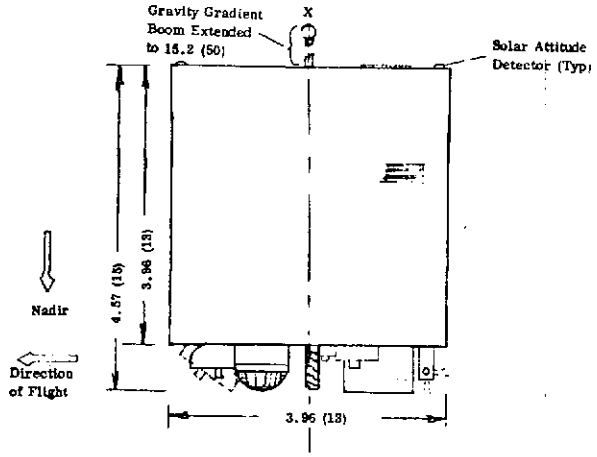


Figure 1-2. Candidate Space-Serviceable Payloads

EARTH AND OCEAN PHYSICS APPLICATIONS PROGRAM

EOP-3	SEASAT A & B
EOP-4	GEOPAUSE
EOP-7	GRAVSAT



NASA ESTIMATED NON-NASA/NON-DOD MISSION PAYLOADS

NND-1A	INTERNATIONAL COMMUNICATION SATELLITE
NND-2A, .B&D	U. S. DOMESTIC SATELLITE
NND-3A	DISASTER WARNING
NND-4A	TRAFFIC MANAGEMENT
NND-5A	FOREIGN COMMUNICATION SATELLITE
NND-6	COMMUNICATION R&D/PROTOTYPE
NND-8	ENVIRONMENTAL MONITORING SATELLITE
NND-11A	LOW EARTH ORBIT EARTH RESOURCES SATELLITE
NND-12A	GEOSYNCHRONOUS EARTH RESOURCES SATELLITE
NND-13A	FOREIGN SYNCHRONOUS EARTH OBSERVATIONS SATELLITE
NND-14	GLOBAL EARTH & OCEAN MONITORING SYSTEM

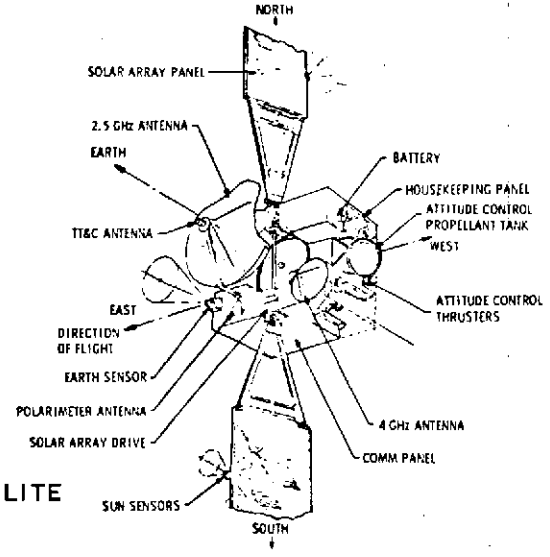


Figure 1-2. Candidate Space-Serviceable Payloads (Continued)

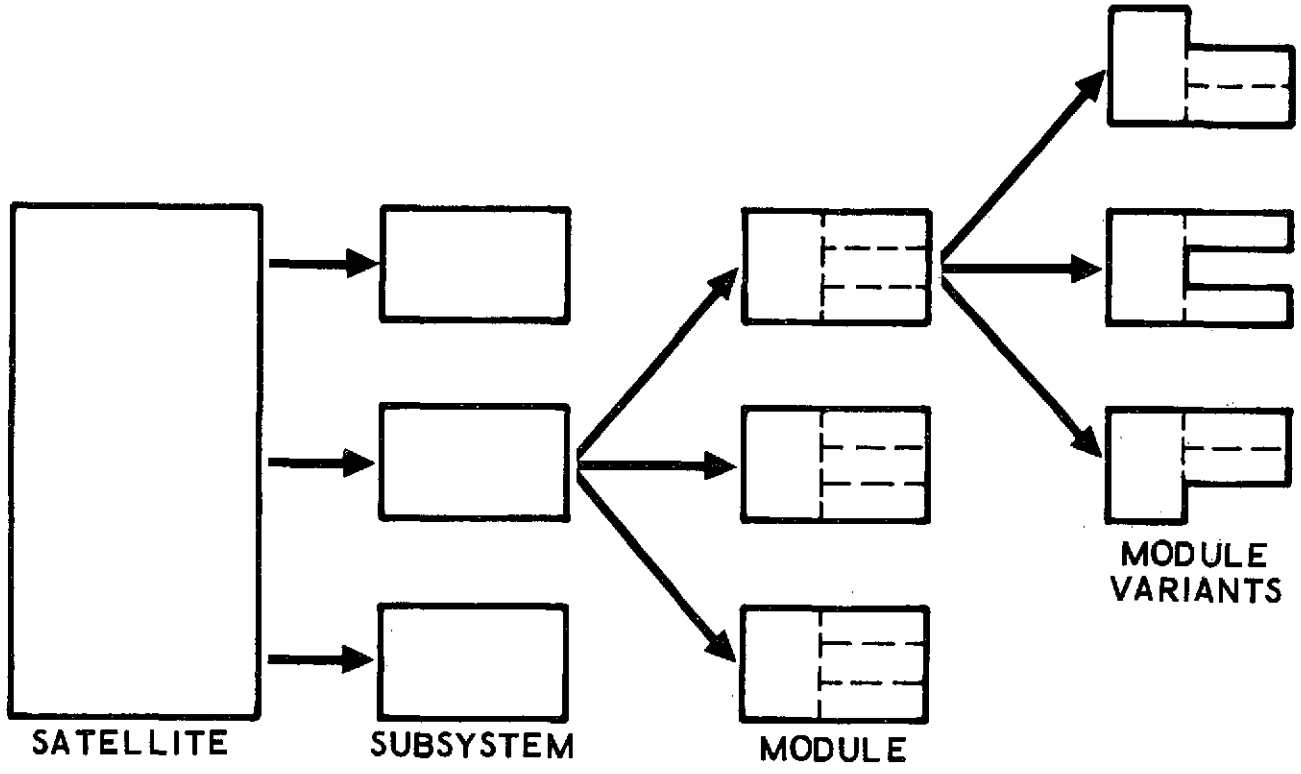


Figure 1-3. Standardization Concepts

2. BASELINE NASA MISSION MODEL

2.1 AUTOMATED MISSION CHARACTERISTICS

2.1.1 Baseline Data

Table 2-1 summarizes the baseline FY73 NASA automated payload mission characteristics and on-orbit payload schedule. The data were derived from References 2, 3, and 4.

2.1.2 Ground Rules

The ground rules that were used in developing the baseline automated payload model are as follows:

a. The payload on-orbit schedule was based on 1 January launch for deployment or service. This approach was used to schedule the payload on-orbit time to equal the payload expected life. The launch can, however, occur any time during the year unless it is noted by a launch date.

b. Only payloads launched from 1979 are shown on the schedule. Payloads that are launched prior to 1979 are not reflected in the payload on-orbit schedule.

c. The payload model in Reference 2 was used to specify launch dates. Payload descriptions in References 3 and 4 were used to specify the orbital parameters, payload characteristics, and on-orbit duration of each payload. The data on the distribution of payloads in a mission were based on the schedule provided in Reference 2.

d. Sortie payloads are not included in Table 2-1.

2.1.3 Symbology

The following symbols are employed in Table 2-1:

↑, ↓↑	New payload with no research, development, test, and engineering (RDT&E)
↑R, ↓↑R	Refurbished payload with no RDT&E
Φ, ΦΦ	New payload with payload RDT&E
ΔR, ΔΔR	Refurbished spacecraft with new mission equipment and mission equipment RDT&E
X/Y	Launch of automated payload in baseline model X Indicates launch Y Indicates payload on-orbit at time of launch

EXP Expendable payloads

GS Ground serviceable payloads

OS On-orbit service payloads

MT Man-tended payloads

2.2 SORTIE PAYLOADS

The Sortie payloads launch schedule is shown in Table 2-2. This schedule was supplied by NASA.

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

ASTRONOMY (AST)

1.
Revised: 25 January 1974

P/L CODE MISS MODEL	P/L CODE SSPDA	PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC ALTITUDE (km)	ORBIT CHARACTERISTIC INCLINATION (deg)	P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE 80	81	82	83	84	85	86	87	88	89	90	91
		EXPLORER																				
AST-1A	AS-02	Extra Coronal Lyman Alpha	EXP	(1)	297 ± 18	28.5 ± 0.5	(1)	1	3	14	2	2	1/1	1	1/1	1	1/1	1	1/1	1	1/1	1
-1B	AS-03	Cosmic Background	EXP	ANY	400 ± 100	ANY	ANY	1	2	14	0/1	0	1	1/0	1	0	1	1/0	1	0	1	1/0
-1C	AS-05	Adv. Radio Astronomy (2)	EXP	ANY	SYNC ± 37	0 ± 5	0 ± 5	2	3	10	0	2	2			0	2	2	2			
-1D	AS-05	Adv. Radio Astronomy (2)	EXP	ANY	SYNC ± 37	0 ± 5	80W ± 20	2	3	10				0	2	2			0	2	2	
AST-3	SO-03	SOLAR PHYSICS MISSION	GS	ANY	500 ± 130 (0)	30 ± 30 (2)	ANY	1	2	13	R		R		R		R		R			R
AST-4	HE-09	HIGH ENERGY ASTR. OBS-MagSpec.	OS/GS	ANY	370 ± 19	28.5 ± 5.5 (6.5)	ANY	1	1	5												
AST-5A	HE-03	HIGH ENERGY ASTR.- Ext. X-Ray	OS/GS	ANY	370 ± 19	28.5 ± 0.5 (1.5)	ANY	1	1	5												
		REVISITS												1	1	1						
-5B	HE-08	HIGH ENERGY ASTR. Gamma Ray	OS/GS	ANY	370 ± 19	28.5 ± 13.5 (0)	ANY	1	1	5												
		REVISITS																1	1	1		
-5C	HE-10	HIGH ENERGY ASTR. Nuclear Cal.	OS/GS	ANY	370 ± 19	28.5 ± 0.5	ANY	1	1	5									R			
		REVISITS																		1	1	1
-5D	HE-05	HIGH ENERGY ASTR. Cosmic Ray	OS/GS	ANY	370 ± 19	28.5 ± 0.5	ANY	1	1	5												R
AST-6	AS-01	LARGE SPACE TELESCOPE	MT/GS	ANY	612 ± 19	28.5 ± 0 (6.5)	ANY	1	1	15				R					R			
		REVISITS										1	1		1	1	1	1		1	1	1
AST-7	SO-02	LARGE SOLAR OBSERVATORY	MT/GS	ANY	350 ± 30 (50)	30 ± 30 (2)	ANY	1	1	15												
		REVISITS															1	1	1	1	1	1
AST-8	AS-16	LARGE RADIO OBSERVATORY	OS	ANY	71600 ± 1000	28.5 ± 0.5	ANY	1	2	6												
		REVISITS																1		1		1
AST-9A	HE-11	FOCUSING X-RAY TEL.- 1.2M	OS/GS	ANY	500 ± 19	15.0 ± 15.0 (13.5)	ANY	1	1	10												R
		REVISITS													1	1						
-9B	HE-01	FOCUSING X-RAY TEL.- 3.0M	OS/GS	ANY	500 ± 19	15.0 ± 15.0 (13.5)	ANY	1	2	10												
		REVISITS																	1		1	

Note:

(1) 21 Sept 1980 @ 1800 LAUNCH DATE INTO PARKING ORBIT FOR FINAL HELIOCENTIC ORBIT

(2) TWO SATELLITES DEPLOYED 20KM to 200 KM APART TO FORM INTERFEROMETER BASELINE

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

SPACE PHYSICS (PHY)

Revised: 25 January 1974

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
		EXPLORER																				
PHY-1A	HE-07	Small High Energy Observatory	EXP	ANY	371 ± 19	$28.5 \pm {}^{D}_{13.5}$	ANY	1	1	8		Δ○			Δ○			①	○	1/○	Δ○	1/○
-1B	AP-01	Upper Atmosphere	EXP	ANY	259/3510 (1)	90 ± 20	ANY	1	1	13		Δ○				Δ○				Δ○		
-1C	AP-02	Medium Altitude	EXP	ANY	1852/37038 (2)	$28.5 \pm {}^{61.5}_{28.5}$	ANY	1	1	13				Δ○				Δ○				Δ○
-1D	AP-03	High Altitude	EXP	ANY	1 A.U.	Ecliptic	ANY	1	1	13	①○		1/○		Δ○		1/○		Δ○		1/○	
		GRAVITY AND RELIABILITY SAT.																				
PHY-2A	AP-04	Earth Orbit	EXP	ANY	938 ± 62	90 ± 0.04	ANY	1	1	5	①○			1/○								
-2B	AP-06	Solar	EXP	ANY	0.3/1.0 A.U.	Helio	ANY	1	1	7							①○					Δ○
		ENVIRONMENTAL PERTURB SAT.																				
PHY-3A	AP-05	Satellite A	EXP	ANY	12,778 ± 926	55 ± 30	ANY	1	3	6		①○	1	1	Δ○	1	1					
-3B	AP-07	Satellite B	EXP	ANY	12,778 ± 926	$55 \pm {}^{35}_{0}$	ANY	1	3	6							①○	1	1	Δ○	1	
PHY-4	AP-08	HELIO & INTERSTELLAR S/C	EXP	(3)	Escape (3)	(3)	ANY	1	7	7								①○	1	1	1	1
PHY-5	HE-12	COSMIC-RAY LABORATORY	MT	ANY	371 ± 19	$28.5 \pm {}^{26.5}_{0.5}$	ANY	1	1	10								①				
		REVISITS																	1	1	1	1

Note:

(1) ± 74 and ± 180 km

(2) ± 370 and ± 9200 km

(3) ESCAPE TRAJECTORY (OUT OF ECLIPTIC TO NEAREST STAR)

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

PLANETARY EXPLORATION (PL)

Revised: 25 January 1974

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		INITIAL LAUNCH DATE (2)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD LAUNCH SCHEDULE (4)											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
PL-7	PL-01	Surface Sample Return	EXP(1)	720	436 ± 164/251	28.5 ± 61.5/27.5	Febr 1984	1	4	4					②							
PL-8	PL-02	Satellite Sample Return (1)	EXP(1)	720	436 ± 164/251	28.5 ± 61.5/14.5	Aug 1990	1	4	4											①	1
PL-10	PL-03	Inner Pl. Follow-On	EXP	2160	436 ± 164/251	28.5 ± 61.5/4.5	Apr 15, 1980	1	1	1	①	2		△			△					
PL-11	PL-07	Venus Radar Mapper	EXP	480	436 ± 164/251	32 ± 58.0/3.5	July 1983	1	1	1				②								
PL-12	PL-08	Venus Buoyant Station	EXP	480	436 ± 164/251	28.5 ± 61.5/24.5	Febr 1985	1	1	1						②						
PL-13	PL-09	Mercury Orbiter (3)	EXP	480	436 ± 164/251	28.5 ± 61.5/10.5	Nov 1987	1	1	1								②				
PL-14	PL-10	Venus Large Lander	EXP	480	436 ± 164/251	32 ± 58.0/3.5	Nov 1989	1	1	1										②		
PL-17	PL-22	Pioneer Saturn Probe	EXP	480	436 ± 164/251	28.5 ± 61.5/10.5	Dec 15, 1980	1	7	7	①											
PL-18	PL-11	Pioneer Sat/Uranus Flyby	EXP	480	436 ± 164/251	28.5 ± 61.5/10.5	Dec 15, 1981	1	>7	>7		①										
PL-19	PL-12	Mariner Jupiter Orbiter	EXP	720	436 ± 164/251	28.5 ± 61.5/27.5	Dec 16, 1981	1	3	3		②										
PL-20	PL-13	Pioneer Jupiter Probe	EXP	480	436 ± 164/251	31 ± 59.0/2.5	Mar 1984	1	3	3					②							
PL-21	PL-14	Mariner Saturn Orbiter	EXP	480	436 ± 164/251	28.5 ± 61.5/26.5	Jan 1985	1	7	7						②						
PL-22	PL-15	Mariner Uranus/Nep. Flyby	EXP	480	436 ± 164/251	28.5 ± 61.5/8.5	Jan 1986	1	>10	>10							②					
PL-23	PL-16	Jupiter Sat. Orbiter/Lander	EXP	480	436 ± 164/251	28.5 ± 61.5/8.5	Oct 1990	1	7	7											①	1
PL-26	PL-18	Encke Rendezvous	EXP	480	436 ± 164/251	45 ± 15/2	Feb 6, 1981	1	3	3		②										
PL-27	PL-19	Halley Flyby	EXP	480	436 ± 164/251	34 ± 56/3	June 1985	1	~3	~3						△						
PL-28	PL-20	Asteroid Rendezvous	EXP	480	436 ± 164/251	28.5 ± 61.5/18.5	June 1986	1	~3	~3							△					

Note:

(1) ORBITER RECOVERS EARTH RETURN VEHICLE
(2) MONTHS CAN BE ± ONE MONTH (DEPENDING ON S/C INSERTION/RETRO CAPABILITY)
(3) DIRECT FLIGHT TO MERCURY (NO VENUS SWINGBY)
(4) PAYLOADS CAN BE LAUNCHED ON SAME FLIGHT

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

4.

Revised: 25 January 1974

LUNAR EXPLORATION (LUN)

LIFE SCIENCES (LS)

SPACE TECHNOLOGY (ST)

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
		LUNAR																				
LUN-2	LU-01	Lunar Orbiter	EXP	2	$436 \pm 251^{1640}_{0}$	$28.5 \pm {}^{61.5}_{0}$	Trans. Lun	1	1	4					(1) 0		1/0					
LUN-3	LU-02	Lunar Rover	EXP	2	$436 \pm 251^{1640}_{0}$	$28.5 \pm {}^{61.5}_{0}$	Trans. Lun	1	1	1							(1) 0	1/0				
LUN-4	LU-03	Lunar Halo	EXP	2	$436 \pm 251^{1640}_{0}$	$28.5 \pm {}^{61.5}_{0}$	Trans. Lun	1	5	5									(1) 0	1	1	
LUN-5	LU-04	Lunar Sample Return (1)	EXP	2	$436 \pm 251^{1640}_{0}$	$28.5 \pm {}^{61.5}_{0}$	Trans. Lun	1	1	1										(1) 0	1/0	
		LIFE SCIENCES																				
LS-1	LS-02	Life Science Research Mod.	GS	ANY	500 ± 100	$28.5 \pm {}^{10}_{0}$	ANY	1	0.5	12	0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0	2/0
		SPACE TECHNOLOGY																				
ST-1	ST-01	Long Duration Exposure Mode.	GS	ANY	500 ± 50	$28.5 \pm {}^{28.5}_{28.5}$	ANY	1	3 (3)	11			R		R		R		R		R	

Note:

(1) ORBITER RECOVER EARTH RETURN SAMPLE

(2) TRANSFER ORBIT

(3) 6 to 9 MONTHS DURATION ON-ORBIT

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

Revised: 25 January 1974

EARTH OBSERVATIONS (EO)

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
EO-3A	EO-8	Earth Observatory Satellite-Res[1]	OS	0.25	914 ± 9	99.15 ± 0.10	9:00 [2]	2[3]	2	12												
		Revisit																				
-3B		Earth Observatory Satellite-Met[1]	OS	0.25	914 ± 9	99.15 ± 0.10	12:00 [2]	2[3]	2	12												
		Revisit																				
-3C		Earth Observatory Satellite-All Wea[1]	OS	0.25	914 ± 9	99.15 ± 0.10	3:00 [2]	2[3]	2	12												
		Revisit																				
-3D		Earth Observatory Satellite-Test[1]	OS	0.25	914 ± 9	99.15 ± 0.10	9:00 [2]	1	2	2	1/0	1										
EO-4A	EO-9	Sync. Earth Obs. Satellite - RD	EXP	ANY	SYNC ± 46	0 ± 0.20	96 W	1	2	11		(1)/0	1	1/0	1	1/0	1					
-4B		Sync. Earth Obs. Satellite - OPER	EXP	ANY	SYNC ± 46	0 ± 0.20	96 W	1	2	11							1/0	2	2/0	2	2/0	
EO-5A	EO-10	Special Purpose Satellite - Sync.	EXP	ANY	SYNC ± 46	0 ± 0.60	80 to 120 W	1	2	10	(1)/0	1				1/0	1					
-5B		Special Purpose Sat. - Polar 3000	EXP	0.50	5500 ± 30	150 ± 0.50	9:00[2]	1	2	5	1/1	1										
-5C		Special Purpose Sat. - Polar 280	EXP	0.25	500 ± 10	97.8 ± 0.10	15:00[2]	1	2	13		2/0	2		1/0	1		1/0	1		1/0	1
-5D		Special Purpose Sat. - Polar 400	EXP	0.25	750 ± 10	98.8 ± 0.10	9:00[2]	1	2	13			1/0	1		1/0	1		1/0	1		1/0
-5E		Special Purpose Sat. - Sync.	EXP	ANY	SYNC ± 46	0 ± 0.60	80 to 120 W	1	2	11				1/0	1					1/0	1	
EO-6	EO-12	TIROS	EXP	0.33	1460 ± 40	102 ± 0.06	9:00[2]	1	2	2			(1)/0	1				1/0	1			
EO-7	EO-7	Sync. Meteorological Satellite	OS	ANY	SYNC ± 46	0 ± 0.10	96 W	1	5	5	0/1	1	1				(1)/0	1	1	1	1	1

(1) Schedule revised from October Space Opportunities and SSPDA documents

(2) Assumed Nodal Crossing Times

(3) Number of payloads operating simultaneously with other Nodal Crossing Time EOS's

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

EARTH AND OCEAN PHYSICS APPLICATIONS (EOP)

6.

Revised: 25 January 1974

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	01
EOP-3	OP-07	SEASAT-B	EXP.	Any	600 ± 100	90 ± 0.10	Any	1	5	10	0/1	1	①/0	1	1	1	1					
EOP-4	OP-01	Geopause	EXP.	0.16	30,000 ± 46	90 ± 0.10	(1)	2	3	6	0/2	2	△/0	2	2							
EOP-5	OP-02	Grav. Gradiometer	EXP.	1.10	200 ± 10	90 ± 0.10	Any	1	1	1	①/0											
EOP-6A	OP-03	Mini-Laser Geodynamic Satellite	EXP.	Any	650 ± 350/250	90 ± (0.10)	(2)	2	5	10	②/0	2	2	2	2	△/0	2	2	2	2		
-6B		Mini-Laser Geodynamic Satellite	EXP.	Any	650 ± 350/250	55 ± (0.10)	(2)	2	5	10	2/0	2	2	2	2	2/0	2	2	2	2		
-6C		Mini-Laser Geodynamic Satellite	EXP.	Any	650 ± 350/250	28.5 ± (0.10)	(2)	2	5	10	2/0	2	2	2	2	2/0	2	2	2	2		
EOP-7	OP-04	GRAVSAT	EXP.	Any	200 ± 5	90 ± 0.10	Any	2	2	2	0/2											
EOP-8	OP-05	Vector Magnetometer Satellite	EXP.	Any	400 ± 10	90 ± 5.0	(3)	3	0.5	10		③/0					△/0				△/0	
EOP-9	OP-06	Magnetic Monitor Satellite	EXP.	Any	1500 ± 500	28 ± 62/28	Any	1	1	10		①/0					△/0				△/0	

(1) Orbit plane normal to ecliptic plane ± 0.5° and both can be launched on same launch.

(2) Each pair of satellites to be deployed from same launch with 1 ft/sec delta velocity imparted to one satellite with respect to the other.

(3) Each satellite is phased 4 hours apart in local time (60 deg.) (to be verified in NASA review cycle).

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

NON-NASA/NON-DOD PAYLOADS (NND)

7.

Revised: 25 January 1974

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
		COMM/NAV																				
NND-1A	CN-51	International Comm. (1) (3)	EXP.	Any	Sync ± 46	0 ± 0.1	40 W	≥ 1	10	12	2/2	4	4	△4	2/5	△7	1/9	9	9	△8	2/7	2/9
-1B		International Comm. (1)	EXP.	Any	Sync + 46	0 + 0.1	180 W	≥ 1	10	12	1/2	3	3	1/3	1/4	5	1/5	6	5	1/4	1/4	5
NND-2A	CN-52	U. S. Domestic - A (3)	EXP.	Any	Sync ± 46	0 ± 0.1	88 to 135W	≥ 1	7	11	1/6	△7	2/9	1/1	11	7	6	5	3	1		
-2B	CN-53	U. S. Domestic - B (ADV)	EXP.	Any	ync ± 46	0 ± 0.1	88 to 135W	≥ 1	10	10					①0	1/1	△2	2/4	3/6	△9	2/11	1/13
-2C	CN-58	U. S. Domestic - C (TDRS) (2) (3)	EXP.	Any	Sync ± 46	2.5 ± 0.1	11 W	1	5	10				△0	1	1	1	1	△0	1	1	1
-2D		U. S. Domestic - C (TDRS)	EXP.	Any	Sync ± 46	2.5 ± 0.1	141 W	1	5	10				1/0	1	1	1	1	1/0	1	1	1
NND-3A	CN-54	Disaster Warning (3)	EXP.	Any	Sync ± 46	0 ± 0.6	94 W	1	5	14		△0	1	1	1	△1	1	1	1	1	△0	1
-3B		Disaster Warning	EXP.	Any	Sync ± 46	0 ± 0.6	124 W	1	5	14			1/0	1	1	1	1					
NND-4A	CN-55	Traffic Management (3)	EXP.	Any	Sync ± 19	2.15 ± 0.31	29 W	1	5	16	2	△2	2	1/2	2	2	1	1	△0	1	1	1
-4B		Traffic Management	EXP.	Any	Sync ± 19	2.15 ± 0.31	52 W	1	5	16	1/2	1/3	4	3	1/2	2	1	1	1		△0	1
-4C		Traffic Management	EXP.	Any	Sync ± 19	2.15 ± 0.31	162 W	1	5	16	1/1	2	1/2	2	2	1	△1	1	1	1	1	
NND-5A	CN-56	Foreign Communication (3)	EXP.	Any	Sync ± 46	0 ± 0.1	60 W	1	7	17	3	△3	4	1/4	3	1/2	3	△3	3	1/3	3	1/3
-5B		Foreign Communication	EXP.	Any	Sync ± 46	0 ± 0.1	96 W	1	7	17	1	1	1/1	2	△1	2	1/2	3	1/3	3	△3	3
NND-6	CN-59	Communication R&D/Proto.	EXP.	Any	Sync ± 46	0 ± 0.2	115 to 140W	1	5	10						①0	1	1	①1	2	①1	2

(1) Launches based on expected traffic between Atlantic and Pacific of 2 to 1 (67% over Atlantic and 33 % over Pacific).

(2) One required in the system but two planned for high availability by providing one on-orbit spare satellite in the nominal model.

(3) NASA developed satellite

Table 2-1

1973 NASA AUTOMATED PAYLOAD MODEL

8.
Revised: 25 January 1974

NON-NASA/NON-DOD PAYLOADS (NND)

P/L CODE		PAYLOAD NAME	TYPE P/L	LCH WIND. (hr)	ORBIT CHARACTERISTIC		P/L LONG. POSITION (deg)	NO. P/L IN SYS.	P/L EXP LIFE (yr)	PROG. LIFE (yr)	PAYLOAD ON-ORBIT SCHEDULE											
MISS MODEL	SSPDA NO.				ALTITUDE (km)	INCLINATION (deg)					80	81	82	83	84	85	86	87	88	89	90	91
		EARTH OBSERVATIONS																				
NND-8	EO-56	Environmental Monitoring Satellite (1)	EXP.	0.33	1685 ± 46	102.97 ± 0.04	1144.5 GMT (2)	1	2	13	△/0	1/1	1/1	1		△/0	1/1	1/1	1/1	1	△/0	1/1
NND-9A	EO-57	Foreign Sync. Met. Satellite (1)	EXP.	Any	Sync ± 46	0 ± 0.6	140 E	1	5	14	0/1	1	1/1	1	1	1	1/1	1	1	1	1/1	1
-9E		Foreign Sync. Met. Satellite	EXP.	Any	Sync ± 46	0 ± 0.6	60 W	1	5	14		△/0	1	1	△/1	2	1	1	△/1	1	1	1
NND-10A	EO-58	Geosync. Oper. Envir. Satellite (1)	EXP.	Any	Sync ± 46	0 ± 0.6	80 W	1	5	16	2	2	1/1	2	1	△/1	2	1	1/1	2	1	△/1
-10E		Geosync. Oper. Envir. Satellite	EXP.	Any	Sync ± 46	0 ± 0.6	120 W	1	5	16	1	△/1	2	1/1	2	2	1	△/1	1	1/1	2	2
NND-11A	EO-61	Earth Resource - LEO (1)	EXP.	(0.33)	907.7 ± 23	99.098 ± 0.10	9:00 (3)	1	2	14	0/1	1/0	1	1/0	1	1/0	1	1/0	1	△/0	1	1/0
-11B		Earth Resource - LEO	EXP.	(0.33)	907.7 ± 23	99.098 ± 0.10	15:00 (3)	1	2	14	△/0	1	1/0	1	1/0	1	△/0	1	1/0	1	1/0	1
NND-12A	EO-59	Earth Resource - Geosync. (1)	EXP.	Any	Sync ± 46	0 ± 0.2	80 W	1	2	14									△/0	1	1/0	1
-12B		Earth Resource - Geosync.	EXP.	Any	Sync ± 46	0 ± 0.2	120 W	1	2	14									1/0	1	1/0	1
NND-13A	EO-62	Earth Resource - Foreign (1)	EXP.	Any	Sync ± 46	0 ± 0.2	60 W	1	2	14									△/0	2/1	2	△/0
		EARTH & OCEAN PHYSICS																				
NND-14	OP-08	Global Earth & Ocean Monit. (1)	EXP.	Any	371 ± 46	98 ± 0.1	Any (4)	3	2	10						△/0	3	3/0	3	△/0	3	

(1) NASA Developed Payload

(2) WTR Launch Time

(3) Assumed Nodal Crossing Times

(4) To Provide Global Coverage Each Satellite Should be Deployed Nominally 60° Longitude Apart

Table 2-2. Sortie Traffic (June 1973 NASA Capture Analysis)

	On-Orbit Time (Days)	80		81		82		83		84		85		86		87		88		89		90		91		TOTAL	
		E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W	E	W
Dedicated Sortie	7	7		12		12	1	14	1	20	1	16	6	17	1	12	5	16	2	16	5	16	5	12	6	170	33
	30									2		2		5		6		8		6		6		7		42	
Sortie + Deploy	7			2		4		5		4	1	5	1	6	1	6	1	4		4		3	1	4	1	47	6
Sortie + Retrieve	7					1								1								2			1	4	1
Sortie + Deploy/Retrieve	7			1			1	1						1							2	1		1		5	3
Sortie + Revisit	7			1		1				1		1		1				2		2		1		2		12	
	30							1												1		1		1		4	
TOTAL	7	7		16		18	2	20	1	25	2	22	7	26	2	18	6	22	2	22	7	23	6	19	8	238	43
	30							1		2		2		5		6		8		7		7		8		46	

3. NASA MISSION MODEL REVISED FOR SPACE SERVICING

The reference NASA Mission Model for automated payload programs has been established by extrapolating today's programs to the time period of interest, 1980 through 1991. Some growth has been provided to account for new mission objectives and mission equipment. However, the model is fundamentally applicable to expendable payload designs and operations similar to those currently employed. The model specifies initial deployment of a payload and estimates when replacements will subsequently be required. These dates can then be used to establish the logistics traffic in yearly increments, thereby providing a basis for estimating total space program costs.

This description is insufficient for assessing space servicing and consequently the first step in this process is to revise (or interpret) the reference NASA Mission Model for space servicing. It is necessary for the subsequent trade-off analyses to specify a definitive launch date and launch window as opposed to only the year of interest. This is not particularly critical except for specific missions, such as planetary operations. Otherwise the launch can be assumed to occur at any time in the year within a 60-day launch window. Further, the replacement schedule, based upon expendable operations, is dismissed altogether. For space servicing, SRU replacements are based upon their individual reliability estimates. The payload is assumed to remain operational with SRU replacements, until its design lifetime has been reached. At this time, if the program is projected to continue, a replacement payload will be scheduled. Replacement payloads may also be required if a nonreplaceable unit fails resulting in a loss of the mission. This is a rare occurrence, but must be accounted for. Therefore, it is seen that this revision of the reference NASA Mission Model is principally an expansion or further definition rather than any real change in substance.

It is also necessary to specify how many payloads of a particular program are required to be operating on orbit for the system to be operational. This information is partially provided in the payload data book (Ref. 2), however, it has been necessary to make assumptions for several programs. These

revisions have all been identified in Tables 2-1 and 3-1 which include estimates of payload lifetime and reliability as well.

For easy reference Table 3-1 provides a summary of all the automated payloads examined in this study. Of the 95 programs in the NASA Mission Model, 42 appear to be reasonable candidates for space servicing. These 42 programs are identified in the column, program type, by the RAS designation. The mission characteristics are also summarized, indicating the orbital altitude, and longitude placement where applicable. The design characteristics shown for the RAS categories represent the reconfigured payloads for space servicing, indicating a substantial weight increase above the reference values. The remaining data is duplicated from the reference NASA Mission Model for the sake of completeness.

The process of arriving at these revisions for the payloads of interest is lengthy, as indicated by the hierarchy of data previously shown. An example is provided below indicating the use of each table. Careful attention to this example should provide a useful aid in understanding the vast amount of data provided by this report and its application to the reference NASA Mission Model.

Example Payload Definition AST-1C

Table 3-1, page 3-3, specifies that AST-1C has been reconfigured for space servicing. Its original weight of 600 kg specified in the reference model has been revised as shown to 1130 kg for space servicing. The mission requirements call for a geosynchronous orbit with a longitude placement at approximately zero deg.

A breakdown of the design features is provided in Table 10-1, page 10-6. This table shows that the AST-1C payload is composed of 12 SRUs and one nonreplaceable unit. For example, six SRUs are required for the attitude and velocity control (AVCS) subsystem. This subsystem requires one AVCS-3 SRU, one AVCS-5 SRU, and four AVCS-7 SRUs, to meet the originally specified AVCS performance requirements.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing
Program Characteristics - Astronomy (AST)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload[4] Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Explorer																	
AST-1A	AS-02	Extra Coronal Lyman Alpha	E	4.06	1.83	1.16	623	1	1	73	(1)	297 ±18	28.5 ±0.5	(1)	7,850	14	5	3	0.25
-1B	AS-03	Cosmic Background	RAS	0.88	4.35	0.38	1264	1	1	76	Any	400 ±100	Any	Any	7,900	14	10	7	0.14
-1C	AS-05	Adv. Radio Astronomy[2]	RAS	0.76	4.35	0.38	1130	2	2	79	Any	Sync ±37	0 ±5	0 ±5	11,700	10	10	7	0.47
-1D	AS-05	Adv. Radio Astronomy[2]	RAS	0.76	4.35	0.38	1130	2	2	85	Any	Sync ±37	0 ±5	80W ±20	11,700	10	10	7	0.47
AST-3	SO-03	Solar Physics Mission	RAS	1.22	4.35	0.40	2148	1	1	78	Any	$500 \pm^{0}_{130}$	$30 \pm^{2}_{30}$	Any	7,950	13	10	7	0.16
AST-4	HE-09	HEAO - Mag Spec	RMS	5.50	4.58	2.75	5429	1	1	77	Any	370 ±19	$28.5 \pm^{6.5}_{5.5}$	Any	7,890	5	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1510	1	1	78	Any	370 ±19	$28.5 \pm^{6.5}_{5.5}$	Any	7,890	N/A	N/A	N/A	N/A
AST-5A	HE-03	HEAO - Ext. X-ray	RMS	5.72	4.58	3.90	7592	1	1	82	Any	370 ±19	$28.5 \pm^{1.5}_{0.5}$	Any	7,890	5	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1500	1	1	83	Any	370 ±19	$28.5 \pm^{1.5}_{0.5}$	Any	7,890	N/A	N/A	N/A	N/A
-5B	HE-08	HEAO - Gamma Ray	RMS	5.22	4.58	2.59	9518	1	1	86	Any	370 ±19	$28.5 \pm^{0}_{13.5}$	Any	7,890	5	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	750	1	1	87	Any	370 ±19	$28.5 \pm^{0}_{13.5}$	Any	7,890	N/A	N/A	N/A	N/A
-5C	HE-10	HEAO - Nuclear Cal.	RMS	5.49	4.58	2.74	5042	1	1	87	Any	370 ±19	28.5 ±0.5	Any	7,890	5	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1500	1	1	88	Any	370 ±19	28.5 ±0.5	Any	7,890	N/A	N/A	N/A	N/A
-5D	HE-05	HEAO-Cosmic Ray	RMS	9.15	4.58	4.58	6430	1	1	91	Any	370 ±19	28.5 ±0.5	Any	7,890	5	2	1	0.27
AST-6	AS-01	Large Space Telescope	RMS	12.70	4.27	4.71	10,401	1	1	80	Any	612 ±19	$28.5 \pm^{6.5}_{0}$	Any	8,020	15	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1500	1	1	81	Any	612 ±19	$28.5 \pm^{6.5}_{0}$	Any	8,020	N/A	N/A	N/A	N/A
AST-7	SO-02	Large Solar Observatory	RMS	17.70	4.58	8.85	10,000	1	1	85	Any	$350 \pm^{50}_{30}$	$30 \pm^{2}_{30}$	Any	8,020	15	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1500	1	1	86	Any	$350 \pm^{50}_{30}$	$30 \pm^{2}_{30}$	Any	7,880	N/A	N/A	N/A	N/A
AST-8	AS-16	Large Radio Observatory	RMS	5.18	3.34	2.59	1300	1	1	85	Any	71,600 ±1000	28.5 ±0.5	Any	11,950	6	3	2	0.25

NOTES:
(1) 21 Sept 1980 @1800 launch date into parking orbit for final heliocentric orbit.
(2) Two satellites deployed 20 km to 200 km apart to form interferometer baseline.
(3) Use HEAO spacecraft.
(4) E = Expendable; NAS = Non-retrievable/Automated in-space servicing; RAS = Retrievable/Automated in-space servicing; RMS = Retrievable/Manned in-space servicing; RSG = Retrievable/Ground refurbished; S = SORTIE.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Astronomy (AST)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Revisits	RSG	1.50	2.00	0.75	750	1	1	83	Any	71,600 $\pm$1000	28.5 $\pm$0.5	Any	11,950	N/A	N/A	N/A	N/A
AST-9A	HE-11	Focusing X-ray Tel. - 1.2M[3]	RAS	10.00	4.35	4.50	6250	1	1	83	Any	500 $\pm$19	15.0 $\pm^{13.5}_{15.0}$	Any	7,950	10	5	3	0.14
-9B	HE-01	Focusing X-ray Tel. - 3.0M	RAS	16.00	4.35	8.00	9393	1	1	86	Any	500 $\pm$19	15.0 $\pm^{13.5}_{15.0}$	Any	7,950	10	5	3	0.04

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Space Physics (PHY)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Explorer																	
PHY-1A	HE-07	Small High Energy Observatory	RAS	0.76	4.35	0.38	1220	1	1	81	Any	371 ±19	$28.5 \pm^{0}_{13.5}$	Any	7,890	8	10	7	0.13
-1B	AP-01	Upper Atmosphere	RAS	0.76	4.35	0.38	1699	1	1	81	Any	259/3510 (1)	90 ±20	Any	8,560	13	5	3	0.29
-1C	AP-02	Medium Altitude	RAS	0.76	4.35	0.38	977	1	1	83	Any	1852/37038 (2)	$28.5 \pm^{61.5}_{28.5}$	Any	11,650	13	5	3	0.29
-1D	AP-03	High Altitude	E	1.83	1.22	0.91	426	1	1	80	Any	1 A.U.	Ecliptic	Any	12,200	13	2	1	0.27
		Gravity and Reliability Sat.																	
PHY-2A	AP-04	Earth Orbit	RAS	0.76	4.35	0.38	1662	1	1	80	Any	938 ±62	90 ±0.04	Any	8,200	5	5	3	0.28
-2B	AP-06	Solar	E	2.10	2.60	1.05	349	1	1	86	Any	0.3/1.0 A.U.	Helio	Any	14,550	7	2	1	0.27
		Environmental Perturb. Sat.																	
PHY-3A	AP-05	Satellite A	E	3.70	2.10	1.83	1488	1	1	81	Any	12,778 ±926	55 ±30	Any	11,800	6	5	3	0.25
-3B	AP-07	Satellite B	E	4.60	3.00	2.30	3946	1	1	87	Any	12,778 ±926	$55 \pm^{35}_{0}$	Any	11,800	6	5	3	0.25
PHY-4	AP-08	Helio. and Interst. Spacecraft	E	3.00	3.00	1.50	280	1	1	88	(3)	Escape (3)	(3)	Any	16,600	7	10	7	0.35
PHY-5	HE-12	Cosmic Ray Laboratory	RMS	8.54	4.27	4.27	18600	1	1	87	Any	371 ±19	$28.5 \pm^{26.5}_{0.5}$	Any	7,890	10	2	1	0.27
		Revisits	RSG	1.50	2.00	0.75	1500	1	1	88	Any	371 ±19	$28.5 \pm^{26.5}_{0.5}$	Any	7,890	N/A	N/A	N/A	N/A

NOTES:
(1) ±74 and ±180 km.
(2) ∓370 and +9200 km.
(3) Escape trajectory (out of ecliptic to nearest star).

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Planetary Exploration (PL)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Gharacteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
PL-7	PL-01	Surface Sample Return	E(1)	6.85	3.80	3.80	3283	1	1	2-84	720	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{27.5}$	N/A	11,500	4	6	4	0.28
PL-8	PL-02	Satellite Sample Return(1)	E(1)	7.62	4.60	3.80	3977	1	1	8-90	720	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{14.5}$	N/A	11,650	4	6	4	0.28
PL-10	PL-03	Inner Pl. Follow-On(2)	E	2.60	2.50	0.60	684	1	1	4-15-80	2160	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{4.5}$	N/A	12,000	1	2	1	0.27
PL-11	PL-07	Venus Radar Mapper	E	6.90	4.26	2.90	3958	1	1	7-83	480	$436 \pm^{164}_{251}$	$32 \pm^{58.0}_{3.5}$	N/A	11,550	1	2	1	0.27
PL-12	PL-08	Venus Buoyant Station	E	3.50	4.60	1.70	5470	1	1	2-85	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{24.5}$	N/A	11,650	1	2	1	0.27
PL-13	PL-09	Mercury Orbiter(3)	E	7.62	4.60	3.80	3496	1	1	11-87	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{10.5}$	N/A	11,850	1	2	1	0.27
PL-14	PL-10	Venus Large Lander	E	5.00	4.60	2.50	1690	1	1	11-89	480	$436 \pm^{164}_{251}$	$32 \pm^{58.0}_{3.5}$	N/A	11,650	1	2	1	0.27
PL-17	PL-22	Pioneer Saturn Probe	E	2.90	2.74	1.50	508	1	1	12-15-80	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{10.5}$	N/A	16,200	7	10	7	0.35
PL-18	PL-11	Pioneer Saturn/Uranus Flyby(2)	E	2.90	2.74	1.50	508	1	1	12-15-80	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{10.5}$	N/A	16,200	7	10	7	0.35
PL-19	PL-12	Mariner Jupiter Orbiter(4)	E	7.60	4.30	2.80	2670	1	1	12-16-81	720	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{27.5}$	N/A	12,450	3	5	3	0.25
PL-20	PL-13	Pioneer Jupiter Probe(2)	E	2.90	2.74	1.50	508	1	1	3-84	480	$436 \pm^{164}_{251}$	$31 \pm^{59.0}_{2.5}$	N/A	15,700	3	5	3	0.25
PL-21	PL-14	Mariner Saturn Orbiter(4)	E	7.62	4.60	3.80	1515	1	1	1-85	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{26.5}$	N/A	15,300	7	10	7	0.35
PL-22	PL-15	Mariner Uranus/Neptune Flyby(4)	E	7.50	4.60	3.70	915	1	1	1-86	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{8.5}$	N/A	17,150	10	10	7	0.35
PL-23	PL-16	Jupiter/Saturn Orbiter/Lander	E	7.62	4.60	3.80	9754	1	1	10-90	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{8.5}$	N/A	11,400	7	10	7	0.35
PL-26	PL-18	Encke Rendezvous	E	3.75	3.63	1.70	2154	1	1	2-6-81	480	$436 \pm^{164}_{251}$	$45 \pm^{15}_{2}$	N/A	12,800	3	5	3	0.25
PL-27	PL-19	Halley Flyby	E	3.50	3.10	1.70	580	1	1	6-85	480	$436 \pm^{164}_{251}$	$34 \pm^{56}_{3}$	N/A	11,750	3	5	3	0.25
PL-28	PL-20	Asteroid Rendezvous	E	5.00	3.10	3.80	1981	1	1	6-86	480	$436 \pm^{164}_{251}$	$28.5 \pm^{61.5}_{18.5}$	N/A	11,900	3	5	3	0.25

NOTES:
(1) Orbiter recovers earth return vehicle.
(2) Pioneer spacecraft.
(3) Direct flight to Mercury (no Venus swingby).
(4) Mariner spacecraft.
(5) Months can be ± one month (depending on spacecraft insertion/retro capability).

FOLDOUT FRAME

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Lunar Exploration (LUN)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
LUN-2	LU-01	Lunar Orbiter	E	4.77	2.03	1.80	757	1	1	84	2	$436^{(2)}_{-251}{}^{+164}$	$28.5 \pm^{61.0}_{0}$	Trans. Lun	11,000	4	2	1	0.27
LUN-3	LU-02	Lunar Rover	E	4.94	3.23	1.24	1380	1	1	87	2	$436^{(2)}_{-251}{}^{+164}$	$28.5 \pm^{61.5}_{0}$	Trans. Lun	11,000	1	2	1	0.27
LUN-4	LU-03	Lunar Halo	E	5.10	2.33	2.50	1120	1	1	89	2	$436^{(2)}_{-251}{}^{+164}$	$28.5 \pm^{61.5}_{0}$	Trans. Lun	11,000	5	7	5	0.30
LUN-5	LU-04	Lunar Sample Return[1]	E	4.94	3.23	1.24	2665	1	1	90	2	$436^{(2)}_{-251}{}^{+164}$	$28.5 \pm^{61.5}_{0}$	Trans. Lun	11,000	1	2	1	0.27

FOLDOUT FRAME

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Life Sciences (LS), Space Technology (ST)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Life Sciences																	
LS-1	LS-02	Life Science Research Module	RSG	2.40	1.50	1.20	682	1	1	80	Any	500 $\pm$100	28.5 $\pm^{10}_{0}$	Any	7,950	12	1	0.5	0.30
		Space Technology																	
ST-1	ST-01	Long Duration Exposure Module	RSG	9.25	4.32	4.57	3860	1	1	80	Any	500 $\pm$50	28.5 $\pm^{26.5}_{28.5}$	Any	7,950	11	5	3(1)	0.25

FOLDOUT FRAME

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)

Program Characteristics - Earth Observations (EO)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
EO-3A	EO-8	Earth Observatory Satellite	RAS	9.00	4.35	2.50	3042	2	2	78	0.25	914 ±9	99.15 ±0.10	9:00[1]	11,600	13	10	7	0.14
-3B		Earth Observatory Satellite	RAS	9.00	4.35	2.50	3042	2	2	79	0.25	914 ±9	99.15 ±0.10	12:00[1]	11,600	13	10	7	0.14
-3C		Earth Observatory Satellite	RAS	9.00	4.35	2.50	3042	2	2	80	0.25	914 ±9	99.15 ±0.10	15:00[1]	11,600	13	10	7	0.14
EO-4A	EO-9	Sync. Earth Observ. Satellite	RAS	4.50	4.35	2.00	1849	1	1	81	Any	Sync ±46	0 ±0.20	80	11,700	11	5	3	0.10
-4B		Sync. Earth Observ. Satellite	RAS	4.50	4.35	2.00	1849	1	1	87	Any	Sync ±46	0 ±0.20	120	11,700	11	5	3	0.10
EO-5A	EO-10	Special Purpose Satellite - Sync.	E	4.14	1.07	2.07	394	1	1	76	Any	Sync ±46	0 ±0.60	80	11,700	10	3	2	0.25
-5B		Special Purpose Satellite - Sync.	E	4.14	1.07	2.07	394	1	1	77	Any	Sync ±46	0 ±0.60	120	9,750	11	3	2	0.25
-5C		Special Purpose Sat. - Polar 3000	E	4.14	1.07	2.07	394	1	1	81	0.50	5500 ±30	150 ±0.50	9:00[1]	7,950	5	3	2	0.25
-5D		Special Purpose Sat. - Polar 280	E	4.14	1.07	2.07	394	1	1	82	0.25	500 ±10	97.8 ±0.10	15:00[1]	8,100	13	3	2	0.25
-5E		Special Purpose Sat. - Polar 400	E	4.14	1.07	2.07	394	1	1	83	0.25	750 ±10	98.8 ±0.10	9:00[1]	11,700	13	3	2	0.25
EO-6	EO-12	TIROS	RAS	0.76	4.35	0.38	2037	1	1	82	0.33	1460 ±40	102 ±0.06	9:00[1]	8,450	2	3	2	0.61
EO-7A	EO-7	Sync. Meteorological Satellite	RAS	0.76	4.35	0.38	1581	1	1	78	Any	Sync ±46	0 ±0.10	80	11,700	5	5	3	0.70
-7B		Sync. Meteorological Satellite	RAS	0.76	4.35	0.38	1581	1	1	87	Any	Sync ±46	0 ±0.10	120	11,700	5	5	3	0.70

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Earth and Ocean Physics Applications (EOP)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
EOP-3	OP-07	SEASAT-B	RAS	2.00	4.35	0.70	1678	1	1	75	Any	600 $\pm$100	90 $\pm$0.10	N/A	8,010	10	10	7	0.45
EOP-4	OP-01	Geopause	RAS	0.76	4.35	0.38	1038	2	2	79	0.16	30,000 $\pm$ 46	90 $\pm$0.10	(1)	12,000	6	10	7	0.31
EOP-5	OP-02	Grav. Gradiometer	E	4.60	4.00	2.04	3244	1	1	80	1.10	200 $\pm$10	90 $\pm$0.10	N/A	7,800	1	2	1	0.27
EOP-6A	OP-03	Mini-Laser Geodynamic Satellite	E	0.50	0.50	0.25	102	2	2	80	Any	650 $\pm^{350}_{250}$	90 $\pm$(0.10)	(2)	8,050	10	7	5	0.30
-6B		Mini-Laser Geodynamic Satellite	E	0.50	0.50	0.25	102	2	2	80	Any	650 $\pm^{350}_{250}$	55 $\pm$(0.10)	(2)	8,050	10	7	5	0.30
-6C		Mini-Laser Geodynamic Satellite	E	0.50	0.50	0.25	102	2	2	80	Any	650 $\pm^{350}_{250}$	28.5 $\pm$(0.10)	(2)	8,050	10	7	5	0.30
EOP-7	OP-04	GRAVSAT	RAS	0.76	4.35	0.38	1625	2	2	79	Any	200 $\pm$5	90 $\pm$0.10	N/A	7,800	2	3	2	0.15
EOP-8	OP-05	Vector Magnetometer Satellite	E	1.37	1.32	0.70	150	3	3	81	Any	400 $\pm$10	90 $\pm$5.0	(3)	7,900	10	1	0.5	0.30
EOP-9	OP-06	Magnetic Monitor Satellite	E	1.37	1.32	0.70	200	1	1	81	Any	1500$\pm$500	28 $\pm^{62}_{28}$	N/A	8,700	10	2	1	0.27

NOTES:
(1) Orbit plane normal to ecliptic plane $\pm$0.5 deg.
(2) Each pair of satellites to be deployed from same launch with one ft/sec delta velocity imparted to one satellite with respect to the other.
(3) Each satellite is phased 4 hours apart in local time (60 deg).

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Non-NASA/Non-DoD (NND)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Comm/Navigation																	
NND-1A	CN-51	International Comm.(1)	RAS	2.50	4.35	0.70	2685	2	2	78	Any	Sync ±46	0 ±0.1	40W	11,700	12	10		0.49
-1B		International Comm.(1)	RAS	2.50	4.35	0.70	2685	2	2	79	Any	Sync ±46	0 ±0.1	180W	11,700	12	10		0.49
NND-2A	CN-52	U.S. Domestic - A	RAS	2.50	4.35	0.40	986	2	2	78	Any	Sync ±46	0 ±0.1	110W	11,700	11	10		0.69
-2B	CN-53	U.S. Domestic - B	RAS	2.50	4.35	0.70	2685	2	2	84	Any	Sync ±46	0 ±0.1	90W	11,700	10	10		0.49
-2C		U.S. Domestic - B	RAS	2.50	4.35	0.70	2685	2	2	83	Any	Sync ±46	0 ±0.1	120W	11,700	10	10		0.49
-2D	CN-58	U.S. Domestic - C (TDRS) (2)	RAS	0.76	4.35	0.38	1325	1	1	83	Any	Sync ±46	2.5 ±0.1	11W	11,700	10	7		0.37
NND-3A	CN-54	Disater Warning	RAS	2.00	4.35	0.50	1349	1	1	81	Any	Sync ±46	2.5 ±0.1	141W	11,700	10	7		0.52
-3B		Disater Warning	RAS	2.00	4.35	0.50	1349	1	1	82	Any	Sync ±46	0 ±0.6	124W	11,700	14	7		0.52
NND-4A	CN-55	Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	77	Any	Sync ±19	2.15 ±0.31	29W	11,700	16	7		0.60
-4B		Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	78	Any	Sync ±19	2.15 ±0.31	52W	11,700	16	7		0.60
-4C		Traffic Management	RAS	3.00	4.35	0.50	1136	1	1	79	Any	Sync ±19	2.15 ±0.31	162W	11,700	16	7		0.60
NND-5A	CN-56	Foreign Communication	RAS	2.80	4.35	0.50	987	1	1	77	Any	Sync ±46	0 ±0.1	0	11,700	17	10		0.49
-5B		Foreign Communication	RAS	2.80	4.35	0.50	987	1	1	78	Any	Sync ±46	0 ±0.1	96W	11,700	17	10		0.49
NND-6	CN-59	Communication R&D/Proto.	RAS	3.40	4.35	0.80	3148	1	1	85	Any	Sync ±46	0 ±0.1	160W	11,700	10	3		0.23

NOTES:
(1) Launches based on expected traffic between Atlantic and Pacific of 2 to 1 (67% over Atlantic and 33% over Pacific).
(2) One is spare since only two are required in the system.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 3-1. NASA Mission Model Revised for Space Servicing (Continued)
Program Characteristics - Non-NASA/Non-DoD (NND)

Payload Code				Design Parameters				Mission Parameters				Orbital Parameters				Lifetime Parameters			
Mission Model	SSPDA	Payload Name	Payload Category	Length (m)	Diameter (m)	CG From Interface (m)	Weight (kg)	Total Number In System	Number Required In System	Initial Launch Date	Launch Window (hours)	Altitude (km)	Inclination (deg)	Longitude (deg)	Characteristic Velocity (m/s)	Program Life	Design Life (yrs)	MMD (yrs)	Reliability at Design Life
		Earth Observations																	
NND-8	EO-56	Environmental Monitoring Sat.	RAS	0.76	4.35	0.38	2602	1	1	80	0.33	1685 ±46	102.97 ±0.04	1144.5 GMT (1)	8,550	13	10	7	0.18
NND-9A	EO-57	Foreign Sync. Met. Satellite	E	3.14	1.91	1.44	257	1	1	80	Any	Sync ±46	0 ±0.6	140W	11,700	14	7	5	0.30
-9B		Foreign Sync. Met. Satellite	E	3.14	1.91	1.44	257	1	1	81	Any	Sync ±46	0 ±0.6	15E	11,700	14	7	5	0.30
NND-10A	EO-58	Geosync. Oper. Environ. Sat.	E	3.14	1.91	1.44	257	1	1	78	Any	Sync ±46	0 ±0.6	80W	11,700	16	7	5	0.30
-10B		Geosync. Oper. Environ. Sat.	E	3.14	1.91	1.44	257	1	1	79	Any	Sync ±46	0 ±0.6	120W	11,700	16	7	5	0.30
NND-11A	EO-61	Earth Resource - LEO	RAS	1.20	4.35	0.50	1460	1	1	79	(0.33)	907.7 ± 23	99.098 ±0.10	9:00 (2)	8,150	14	10	7	0.29
-11B		Earth Resource - LEO	RAS	1.20	4.35	0.50	1460	1	1	80	Any	907.7 ± 23	99.098 ±0.10	15:00 (2)	8,150	14	10	7	0.29
NND-12A	EO-59	Earth Resource - Geosync.	RAS	4.50	4.35	2.00	2396	1	1	88	Any	Sync ±46	0 ±0.2	80W	11,700	14	5	3	0.10
-12B		Earth Resource - Geosync.	RAS	4.50	4.35	2.00	2396	1	1	88	Any	Sync ±46	0 ±0.2	120W	11,700	14	5	3	0.10
NND-13A	EO-62	Earth Resource - Foreign	RAS	4.50	4.35	2.00	2340	1	1		Any	Sync ±46	0 ±0.2	15E	11,700	14	5	3	0.10
		Earth and Ocean Physics																	
NND-14	OP-08	Global Earth & Ocean Monitoring	RAS	2.00	4.35	0.70	2143	3	3	86	Any	371 ±46	98 ±0.1	Any	7,890	10	10	7	0.25

NOTES:
(1) WTR launch time.
(2) Assumed nodal crossing times.

A detailed description of each AVCS space replaceable unit is provided in Table 4-3, (other tables describe the remaining subsystem and mission equipment SRUs.) The AVCS-3 module is described on page 4-26. This SRU incorporates a 10 ft-lb-sec reaction wheel, with wheel electronics and other items common to all SRUs (remote terminal, power conditioning, etc). The weight of this SRU is estimated at 109.2 kg including the baseplate mechanism. It has an expected lifetime of seven years with a reliability at that time of 0.623. This design, coupled with the sensing module AVCS-5 and the hot gas reaction control system AVCS-7, is capable of meeting the performance requirements specified for the AST-1C payload, in Table 4-1, page 4-2. These requirements are traceable to the original data base published with the October 1973 NASA Mission Model.

This then provides the basic design information for the AST-1C payload, reconfigured for space servicing. The detailed information on other payloads may be traced in a similar manner. New payload concepts evolving in the future may also be reconfigured by comparing performance requirements with those specified for each subsystem and selecting the appropriate SRU. Replaceable mission equipment may be approximated by comparison with those of Section 8, or by a simplified buildup using the SRU baseplate definition provided in Section 11. Nonreplaceable components are defined in Section 9 and include such items as basic structure, solar panels and other long-life items. An example of the application of this data is provided in Section 11. The NASA Earth Observatory Satellite has been reconfigured as a point design to indicate SRU selection and placement to meet the specified mission objectives. This particular payload is highly complex and should represent an upper bounds on the demands of flexibility placed on the design concept selected for this study effort.

4. ATTITUDE CONTROL, GUIDANCE AND NAVIGATION

4.1 SUBSYSTEM REQUIREMENTS

The attitude control, guidance and navigation requirements were examined for the 29 different satellite configurations that were identified in the 42 selected for study. Required pointing accuracies, attitude and slew angular rates as well as the probably on-orbit position and velocity accuracies were either extracted from References 3 and 4 or estimated where data were not provided. These requirements were used to estimate the momentum storage and mass expulsion characteristics for the conceptual designs of the space replaceable units.

Summaries of the attitude and velocity control (AVCS) and the guidance and navigation (G&C) requirements are presented in Tables 4-1 and 4-2.

4.2 BASIC SUBSYSTEM ELEMENTS

4.2.1 Computer and the Auxiliary Electronics Assembly

The various control elements needed to perform the attitude and on-orbit control functions (stationkeeping, guidance and navigation) are represented by the general functional control concept shown in Figure 4-1. The central element is seen to be the guidance and control processor assembly (GCPA), or computer, to which an auxiliary electronics assembly may be appended.

The main function of the GCPA is to:

a. Store gains, time constants and saturation limits for all control modes.

b. Perform arithmetic operations using attitude and orbit maintenance control laws.

c. Provide fault detection and diagnostics.

d. Provide time, synchronization, restart and initialization of programs.

FOLDOUT FRAME 1

Table 4-1. Attitude and Velocity Control System Design Parameters

FOLDOUT FRAME 2

Payload Code		Payload Name	Pointing Accuracy ±(deg)	Attitude Rates Less Than (deg/sec)	Slew Rates (deg/sec)	Momentum Storage (8) ft-lb-sec	Attitude and Stationkeeping Thrusters		Orientation Reference
Mission Model	SSPDA						Thrust (lb)	Total Velocity/Impulse Propellant Weight	
AST-1B	AS-03	Cosmic Background	1 min	0.001	0.1 - 1.0	15 (RW)	0.1 and 5	330 ft/sec	Inertial
AST-1C	AS-05	Adv Radio Astronomy	1 sec	10^{-4}	0.1 - 1.0	13 (RW)	0.1 and 5	330 ft/sec	Inertial
AST-3	SO-03	Solar Physics Mission	1.2 sec, 0.25°	1.2 sec/sec	0.1 - 1.0	30 (RW)	0.1 and 5	100 lb GN_2	Sun/Star
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	30 sec, 1 sec	10^{-4}	0.01 - 1.0	500 (CMG) (3 Req.)	0.5 and 10	220 lb GN_2 + Magnet. Torq.	Star (computer)
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	30 sec, 1 sec	10^{-4}	0.01 - 1.0	500 (CMG) (3 Req.)	0.5 and 10	400 lb GN_2 + Magnet. Torq.	Star (computer)
PHY-1A	HE-07	Small High Energy Observatory	1 min	0.001	0.1	14	0.1 and 5	50 lb GN_2	Inertial
PHY-1B	AP-01	Upper Atmosphere Explorer	2	0.001	0.1	None	0.1 and 5	200 lb GN_2	Inertial
PHY-1C	AP-02	Medium Altitude Explorer	2	0.001	0.1	None	0.1 and 5	200 lb-sec	Inertial
PHY-2A	AP-04	Gravity & Rel - Earth Orbit	(1→0.05) sec	10^{-4}	0.1	20 (RW)	Low	300 lb He	Star
EO-3A	EO-08	Earth Observatory Satellite	1→0.01	2×10^{-6}	0.1	18 (RW)	0.1 and 5	200 lb GN_2	Earth
EO-4A	EO-09	Sync Earth Observatory Sat	.0172 (1)	10^{-4}	1.0	6 (RW)	0.1 and 5	151 ft/sec N_2H_4	Earth
EO-6	EO-12	TIROS	.01	2×10^{-6}	0.1	30 (RW)	0.1 and 5	200 lb GN_2	Earth
EO-7	EO-7	Sync Meteorological Sat	0.07 (2)	10^{-4}	0.3	5 (RW) (3 Req.)	0.5	100 lb N_2H_4	Earth
EOP-3	OP-07	SEASAT - B	2 (3)	0.001	None	30 (RW)	0.1 and 5	200 lb GN_2	Earth
EOP-4	OP-01	Geopause	3 (4)	0.001	0.1	30 (RW)	0.1 and 5	250 lb GN_2	Earth
EOP-07	OP-04	GRAVSAT	3 (5)	0.001	0.1	30 (RW)	0.1 and 5	1120 lb GN_2	Earth
NND-1	CN-51	International Comm	0.16	0.001	0.1	20 (RW)	0.0015 Ion, 0.5 N_2H_4	120 lb N_2H_4	Earth
NND-2A	CN-52	U.S. Domestic - A	0.2 (6)	0.001	0.1	5 (RW)	0.1, 0.5 N_2H_4	50 lb N_2H_4	Earth
NND-2B	CN-53	U.S. Domestic - B (Adv)	0.16	0.001	0.1	20 (RW)	0.0015 Ion, 0.5 N_2H_4	120 lb N_2H_4	Earth
NND-2C	CN-58	U.S. Domestic - C (TDRS)	0.58 (7)	0.001	0.1	5 (RW)	0.1, 0.5 N_2H_4	60 lb N_2H_4	Earth
NND-3	CN-54	Disaster Warning	0.11	10^{-4}	0.01	5 (RW)	3×10^{-4} Ion	25 lb Ce	Earth
NND-4	CN-55	Traffic Management	0.3	0.001	0.1	5 (RW)	.1, 5	40 lb GN_2 + N_2H_4	Earth
NND-5	CN-56	Foreign Communication	0.2	0.001	0.1	5 (RW)	.1, 5	27,577 lb-sec	Earth
NND-6	CN-59	Communications R&D/Proto	0.2	0.001	0.1	5 (RW)	0.0015 Ion, 0.5 N_2H_4	170 lb N_2H_4	Earth

(1) Knowledge of pointing ±.00172 deg
(2) Knowledge of pointing ±.00143 deg
(3) Measure attitude ±0.1° relative to an earth centered coordinate system
(4) Measure attitude ±0.5° relative to an earth centered coordinate system
(5) More accurate pointing (to ± 1°) will be required during orbit maintenance
(6) Attitude determination to ± 0.05 (knowledge to ± .0025°)
(7) Attitude determination to ± 0.25°
(8) (RW) = Reaction wheel; (CMG) = Control moment gyro
(9) Dimensions in English units are retained to provide traceability to existing reference material

FOLDOUT FRAME 1

FOLDOUT FRAME 2

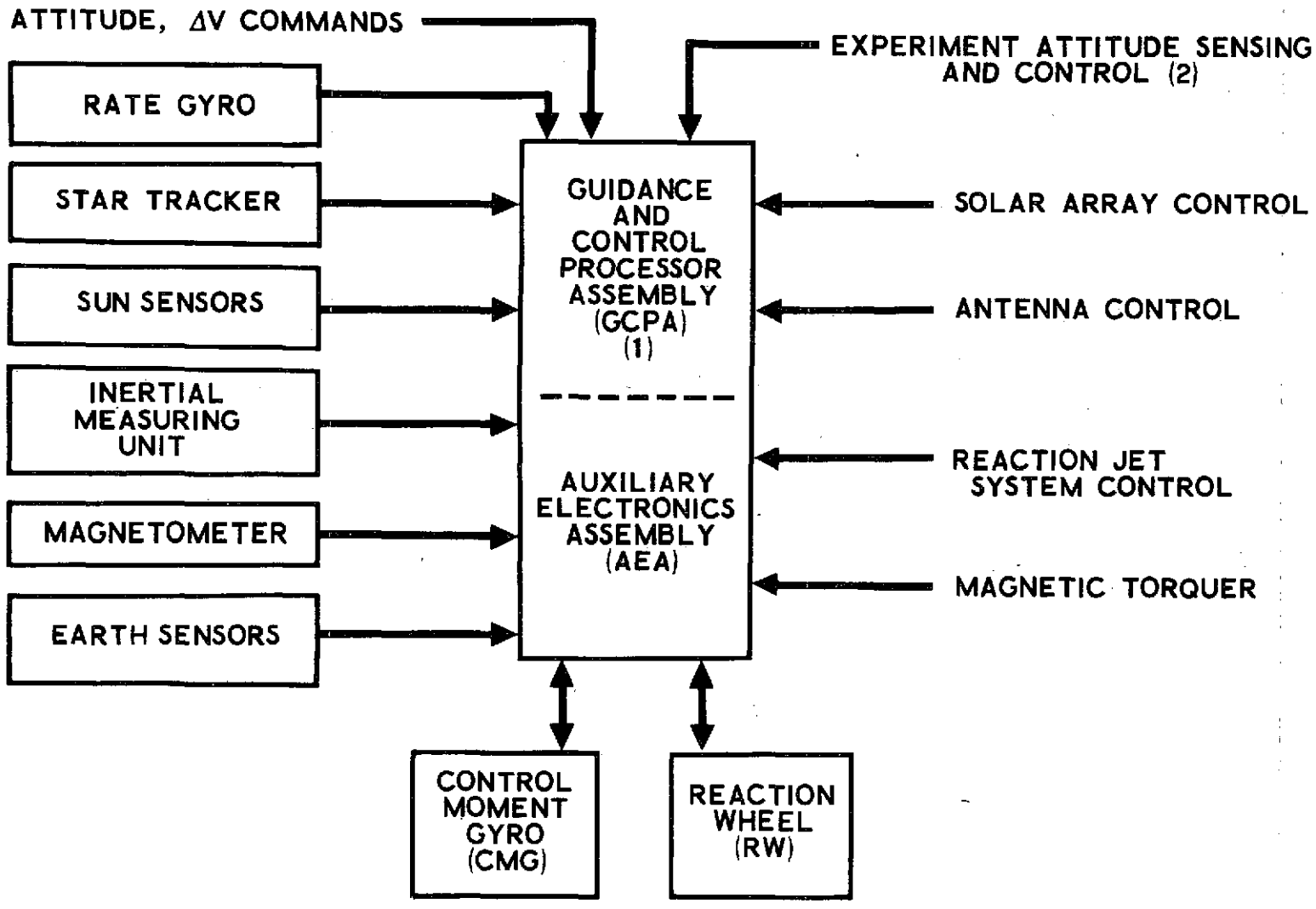
Table 4-1. Attitude and Velocity Control System Design Parameters (Continued)

Payload Code		Payload Name	Pointing Accuracy ±(deg)	Attitude Rates Less Than (deg/sec)	Slew Rates (deg/sec)	Momentum Storage ft-lb-sec	Attitude and Stationkeeping Thrusters		Orientation Reference
Mission Model	SSPDA						Thrust (lb)	Total Velocity/Impulse Propellant Weight	
NND-8	EO-56	Environmental Monitoring Satellite	.01	2×10^{-6}	0.1	30 (RW)	0.1, 5	200 lb GN_2 + N_2H_4	Earth
NND-11	EO-61	Earth Resource - LEO	0.7	.001	0.01	5 (RW)	.1 N_2H_4 + Magnetic	30 lb N_2H_4	Earth
NND-12	EO-59	Earth Resource - Geosync	10 sec	.001	0.1	30 (RW)	.1, 5	180 lb N_2H_4	Earth
NND-13	EO-62	Earth Resource - Foreign	6 min	.001	0.1	30 (RW)	.1, 5	180 lb N_2H_4	Earth
NND-14	OP-08	Global Earth & Ocean Monitoring	0.5	.001	0.1	25 (RW)	.1, 5	180 lb N_2H_4	Earth

Table 4-2. Guidance and Navigation System Design Parameters

Payload Code		Payload Code	Navigation Accuracy (1 σ)[1]		Inertial Measuring Unit[2]	Guidance Computer[3]
Mission Model	SSPDA		Position (ft)	Velocity (ft/sec)		
AST-1B	AS-03	Cosmic Background	100	0.05	S.D.[4]	A.C.[5]
AST-1C	AS-05	Adv Radio Astronomy	200	0.10	S.D.	A.C.
AST-3	SO-03	Solar Physics Mission	100	0.05	S.D.	A.C.
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	100	0.05	S.D.	A.C.
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	100	0.05	S.D.	A.C.
PHY-1A	HE-07	Small High Energy Observatory	100	0.05	S.D.	A.C.
PHY-1B	AP-01	Upper Atmosphere Explorer	100	0.05	N/A[6]	N/A
PHY-1C	AP-02	Medium Altitude Explorer	200	0.10	N/A	N/A
PHY-2A	AP-04	Gravity and Rel - Earth Orbit	100	0.05	S.D.	A.C.
EO-3A	EO-8	Earth Observatory Satellite	100-150	0.05	S.D. (High Perf.)	A.C.
EO-4A	EO-9	Sync Earth Observatory Sat	200	0.10	S.D. + Star Sens.	A.C.
EO-6	EO-12	TIROS	100-150	0.05	S.D. + Star Sens.	A.C.
EO-7	EO-7	Sync Meteorological Sat	200	0.10	4 Gyros + 2 Star Sens.	A.C.
EOP-3	OP-07	SEASAT-B	100	0.05	N/A	N/A
EOP-4	OP-01	Geopause	200	0.10	N/A	N/A
EOP-07	OP-04	GRAVSAT	100	0.05	N/A	N/A
NND-1	CN-51	International Comm	200	0.10	3 Axis Rate Gyro	N/A
NND-2A	CN-52	U.S. Domestic - A	200	0.10	3 Axis Rate Gyro	N/A
NND-2B	CN-53	U.S. Domestic - B (Adv)	200	0.10	3 Axis Rate Gyro	N/A
NND-2C	CN-58	U.S. Domestic - C (TDRS)	200	0.10	3 Axis Rate Gyro	N/A
NND-3	CN-54	Disaster Warning	200	0.10	Single Axis Gyro	N/A
NND-4	CN-55	Traffic Management	200	0.10	N/A	N/A
NND-5	CN-56	Foreign Communication	200	0.10	N/A	N/A
NND-6	CN-59	Communication R&D/Proto	200	0.10	Single Axis Gyro	A.C.
NND-8	EO-56	Environmental Monitoring Satellite	100	0.05	S.D. + Star Sens.	A.C.
NND-11	EO-61	Earth Resource - LEO	100	0.05	N/A	N/A
NND-12	EO-59	Earth Resource - Geosync	200	0.10	S.D. + Star Sens.	A.C.
NND-13	EO-62	Earth Resource - Foreign	200	0.10	S.D. + Star Sens.	A.C.
NND-14	OP-08	Global Earth and Ocean Monitoring	100	0.05	N/A	N/A

(1) Ground Tracking Assumed
(2) 3 Rate and Integrating Gyros + Accelerometers
(3) On Board Data Processing and Control
(4) Strapped Down Type
(5) Attitude Control (15 Bit Data Word Capability Maximum)
(6) Not Applicable
(7) Dimensions in English Units are retained to provide traceability to existing reference material.



(1) ON-BOARD DATA PROCESSING AND CONTROL
(2) PAYLOAD SUPPLIED ERROR SIGNALS

Figure 4-1. General Control Concept

The functional parts of the GCPA are shown in Figure 4-2. The size is on the order of 18 x 20 x 33 cm and the weight is about 9 kg.

The auxiliary electronics assembly (AEA) collects in one package all attitude and velocity control electronics that are not a part of the guidance and control processor assembly. The AEA can thus be used to perform attitude control functions if GCPA is not required or if the power level and processing speed cannot be provided conveniently with the GCPA.

The AEA can be generally divided into the following functional areas:

a. Selection and configuration control of AVCS hardware.

b. Timing and detection (frequency clock, countdown logic, power transient detector).

c. Power amplification and drive to the reaction wheel (RW) or control moment gyro (CMG) spin motor. Also tachometer transmission to GCPA.

d. Gimbal Drive Electronics for RW or CMG (stepper, motor drive, resolver drive and position processing).

The AEA assembly is typically 18 x 20 x 22 cm and weighs about 4.5 kg.

4.2.2 Sensing Elements

The basic sensors which may be needed are indicated in Figure 4-1.

The Earth Sensor Assembly (ESA) is typically a scanning sensor utilizing the 14 to 16 μm spectral range. Four separate heads can be used with separate electronic channels for each head. Any two channels can provide roll and pitch error angles for earth-following vehicles. The ESA telescope field of view is dependent on the orbital altitude. The infra-red discontinuity at the horizon edges is detected by a thermistor bolometer,

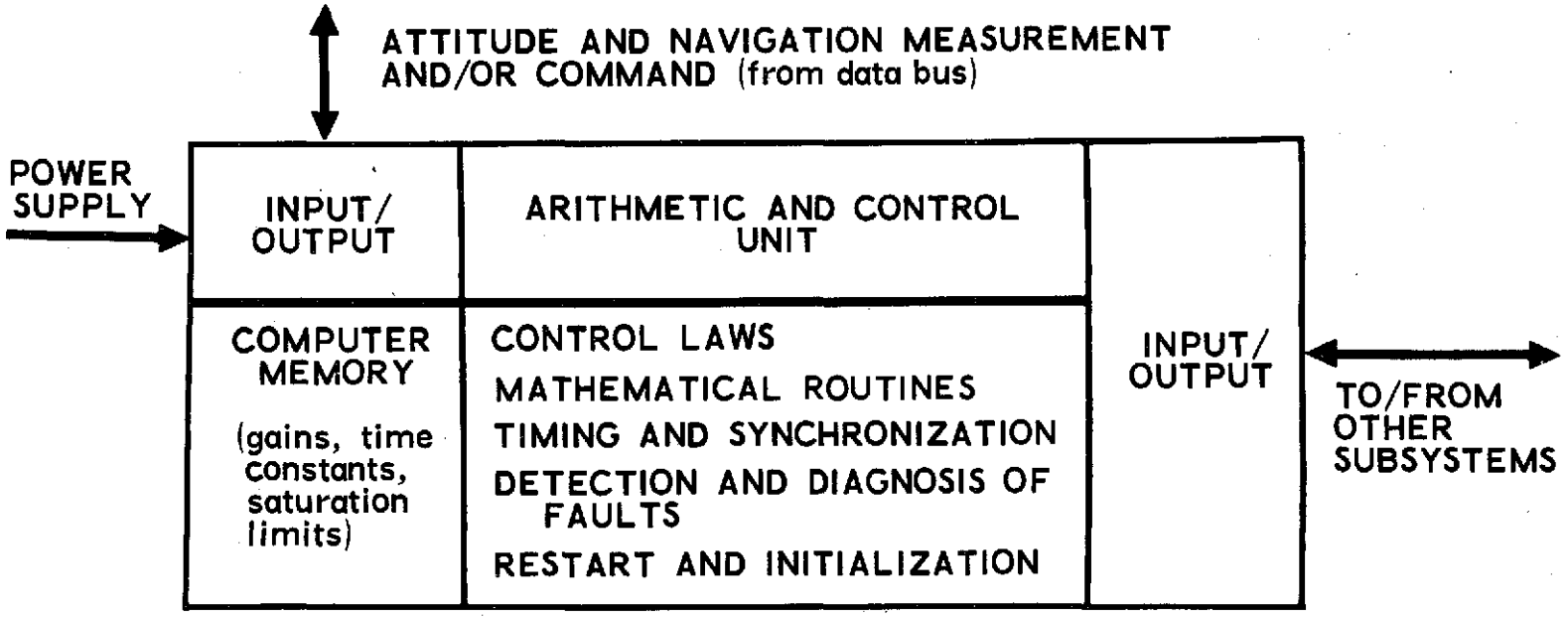


Figure 4-2. Guidance and Control Processor Assembly (GCPA)

the output of which is digital words. Typical accuracy is 0.041° (3σ) for a scanning sensor at synchronous altitude.

Typical power required is 10W, weight is about 10 kg and the size is 25x25x12 cm (including the electronics circuits). A functional block diagram is shown in Figure 4-3.

A coarse Sun Sensor Assembly (SSA) can be used for satellite or solar array control. The solar aspect signals can be generated by the SSA. Each SSA consists typically of six silicon solar cells with the output from opposing pairs subtracted from each other in the AEA. Two SSAs are generally required for a 4π steradian coverage. Typical accuracy (3σ) is 0.32° (null), the weight is 0.4 kg per head and the size is 6.2 x 7.6 x 9.1 cm. Minimum or no power is required.

A Fine Sun Sensor Assembly (FSSA) is required for pointing accuracies on the order of $\pm$ 2 arc min. The field of view is $\pm 15^{\circ}$. The FSSA also uses silicon cells.

An ADCOLE Aspect Digital Sun Sensor can be used where intermediate accuracies ($\pm$ 0.25) are required. Three-axis attitude determination is possible with five sensors each weighing less than 1 lb. A block diagram is shown in Figure 4-4. A high resolution ADCOLE sun sensor can also be used for 14 arc sec resolution and 1 arc minute accuracy.

A two-degree-of-freedom Gimbal Star Tracker (GST) of the Orbiting Astronomical Observatory type can be used for detection, acquisition and tracking of 2.0 magnitude stars in space. The general characteristics are as follows:

Tracking accuracy (two axes)	30 arc sec (1σ)
Field of view	$\pm 0.5^{\circ}$
Resolution of encoder	5 arc sec
Command resolution	10 arc sec
System weight (Gimbal electronics, resolver)	24 kg
High voltage	1000 Vdc
System power	20W

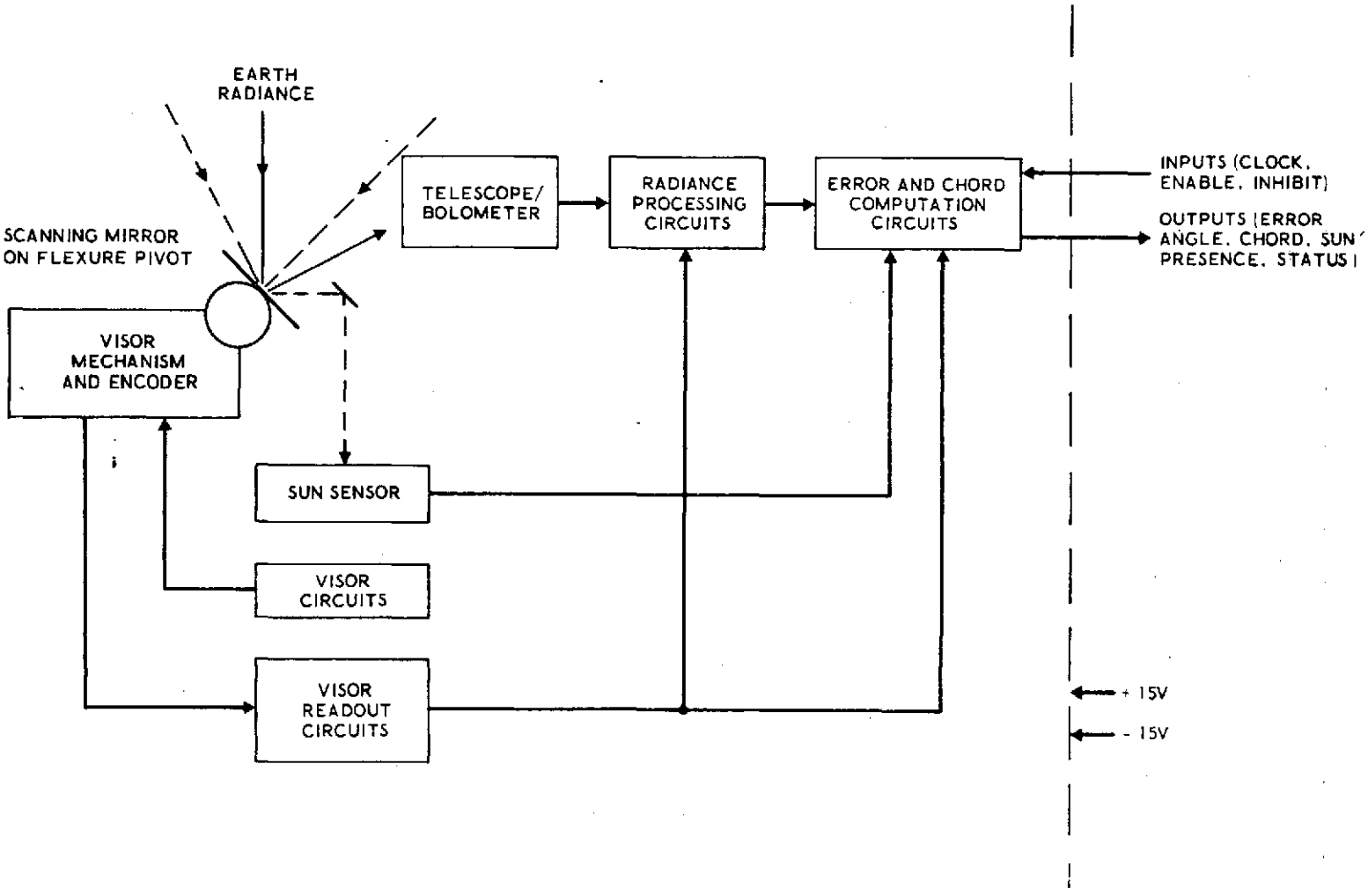


Figure 4-3. Earth Sensor Block Diagram

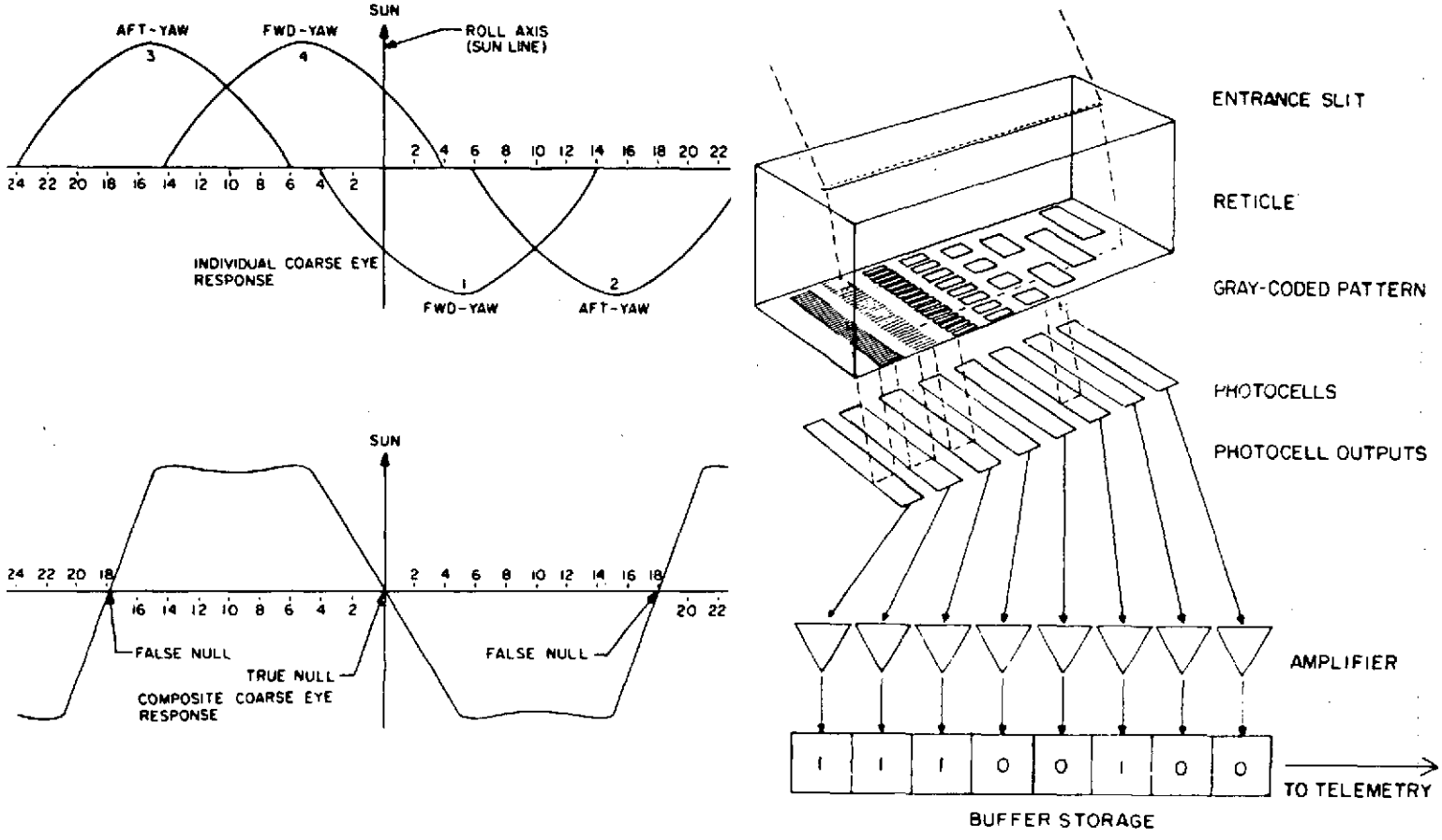


Figure 4-4. Block Diagram of ADCOLE Aspect Sensor Operations

The command and track modes enable the tracker telescope to be pointed by external command signals and track in a closed loop servo-system. A simplified functional block diagram for these modes is shown in Figure 4-5.

The Rate Gyro Assembly (RGA) consists of a single-degree-of-freedom rate integrating gyro with a torque-rebalance servo loop. External power at ± 15 Vdc and a clock signal is required. A block diagram is shown in Figure 4-6. The size is 5 x 15 x 20cm and the weight is 1.3 kg.

A group of three rate gyros with orthogonal sensing axes may be employed for three-axis measurement of angular velocities if an inertial measuring unit is not included. The latter can also provide angular position and translational acceleration information (optional) and would generally obviate the need for a three-axis rate gyro package.

The Inertial Measuring Unit (IMU) is an ultra-precision gyro inertial system. The inertial package is a temperature controlled cube 23 cm on each side. It contains three precision rate integrating gyros, three accelerometers, the gyro control and readout electronics and a frequency source required for the electronics. The gyros are operated with binary pulse restrained torque loop. Torque data from the gyros is processed as rate and rate integrals (i.e., angles) in the electronics and converted to analog signals for use in the spacecraft AVCS. A functional block diagram for the "strapdown" configuration where the gyros are hard mounted to the spacecraft is shown in Figure 4-7. A single integrating rate gyro restrain loop is shown in Figure 4-8.

The IMU characteristics depend on the mode of operation selected. Thus, for SLEW and HOLD modes, the rate measuring capabilities are 2.0 and 0.004 deg/sec, respectively. The power required is 75W (28V unregulated). The weight is on the order of 454 kg (inertial package and electronics). The IMU receives and transmits signals to the guidance and control computer for execution of precise attitude and orbit velocity control commands.

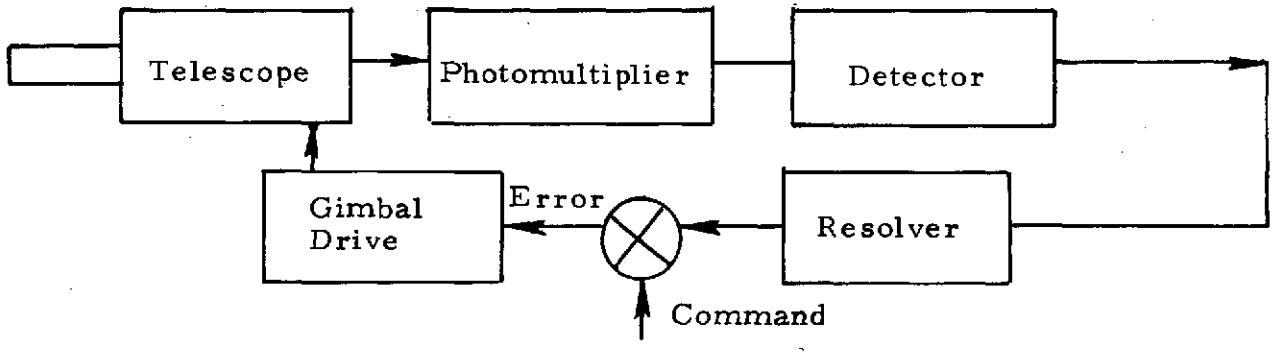


Figure 4-5. Gimbaled Star Tracker Electronics

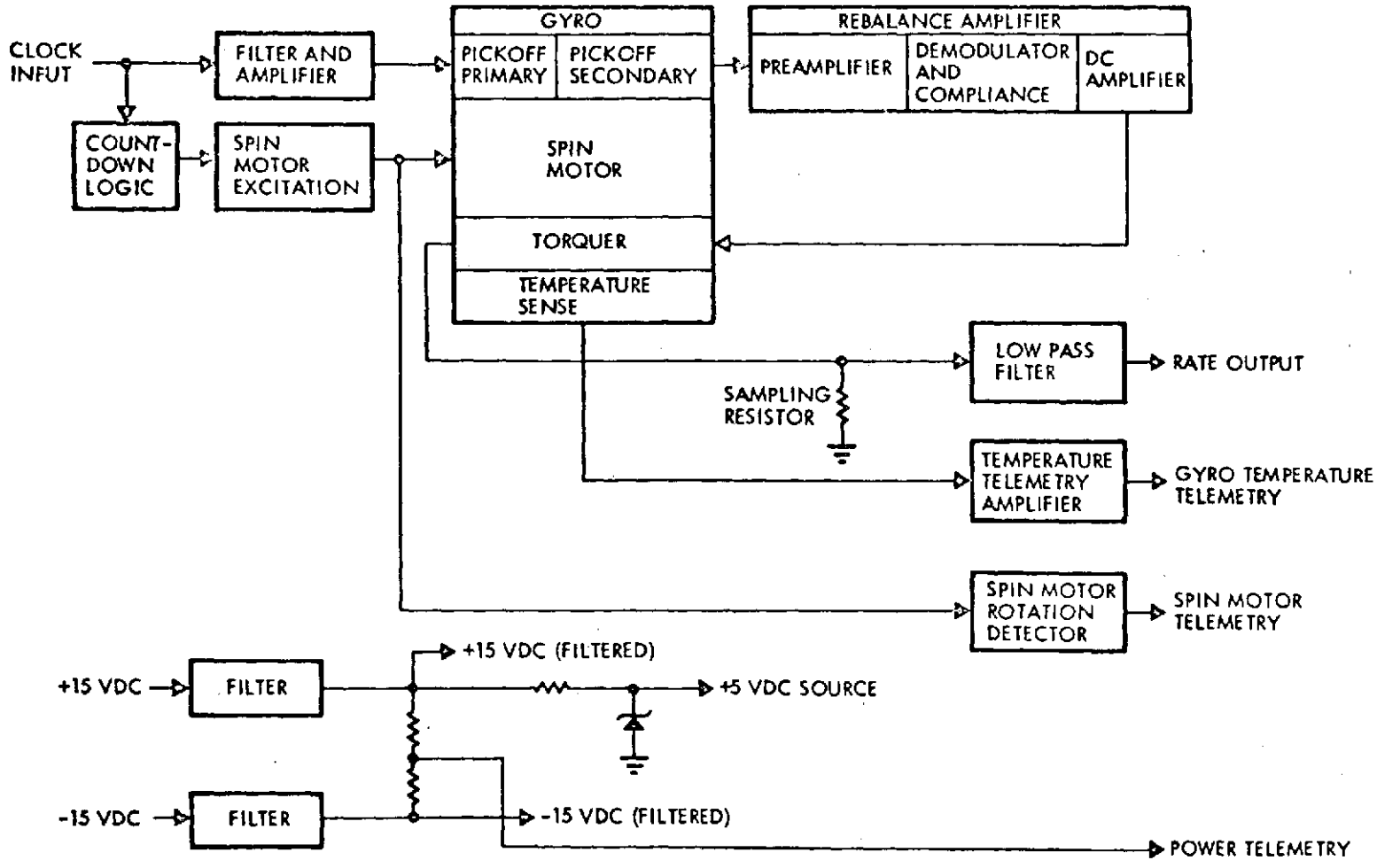


Figure 4-6. RGA Block Diagram

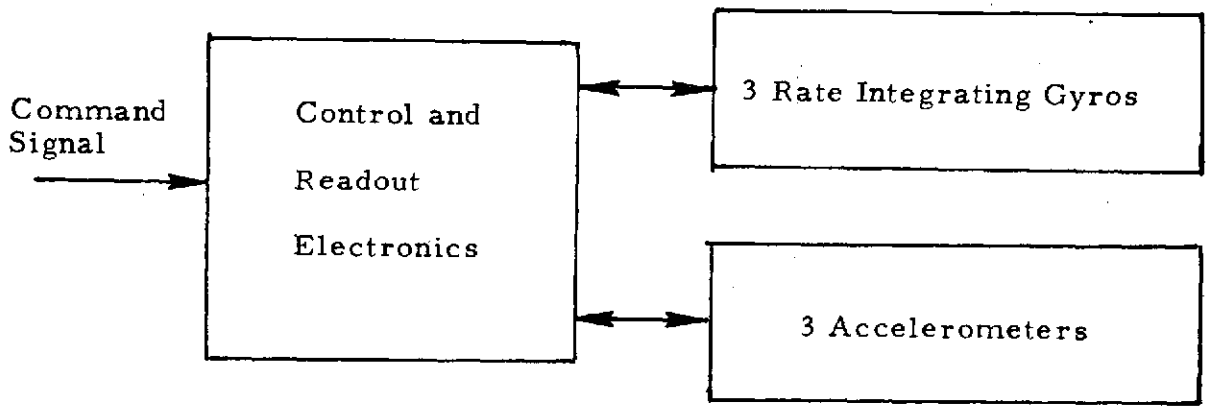


Figure 4-7. IMU Block Diagram

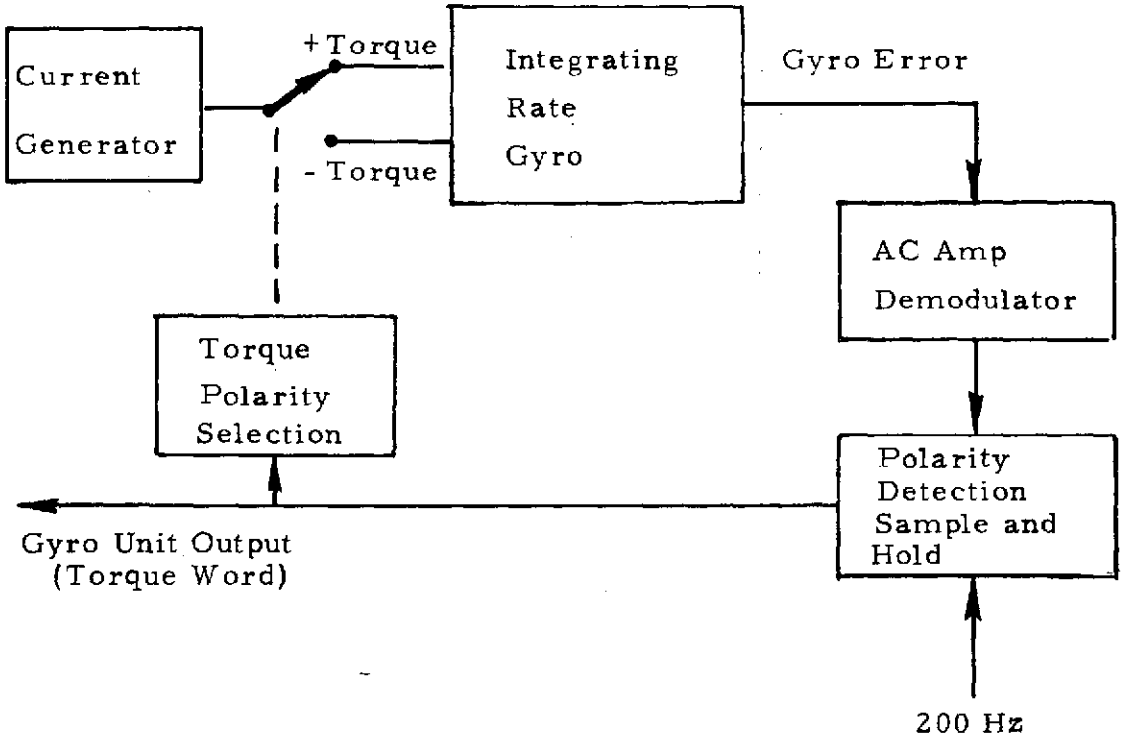


Figure 4-8. IMU Gyro Restraint Loop

4.2.3 Force and Torque Generating Systems

The control authority (force and torque generation) for attitude and on-orbit position maintenance can be obtained from a mass expulsion system. Most common applications involve cold gas (gaseous nitrogen, GN_2), hot gas (hydrazine, N_2H_4) or ion thrusters. Torque generation can be obtained from momentum exchange devices (reaction wheel or control moment gyro) or a magnetic torquer. The basic characteristics of these elements are described below.

Three reaction wheels (RW) may be employed to stabilize a spacecraft about each of three axes or a single wheel (gimbaled, constant speed or modulated) may be utilized to provide gyroscopic stability about two axes and closed loop control about the third axis. A momentum unloading (dumping) mechanism is usually required to maintain the wheel speed below a critical value. This can be a mass expulsion or a magnetic type.

A typical block diagram for closed loop reaction wheel control about an axis is shown in Figure 4-9. In this configuration the lead compensation introduces a phase lead into the control loop to ensure good transient response and sufficient damping. Integral compensation drives the pointing error to zero. The limiter allows the tachometer loop to assume control of the wheel when attitude errors are too large. This prevents wheel speed from exceeding design limits. In normal operation, the tachometer deadzone limits the signal. However, during wheel turn-on and when attitude errors are large, the tachometer loop brings the wheel up to proper speed.

A typical 5 ft-lb-sec wheel requires 1 to 24W (peak) power for operation, weighs 15 lb and its dimensions are 12 x 5 in. Approximate weight of a reaction wheel (weight in lbs) can be found from the equation:

$$W = 7H^{0.4}$$

where H is the angular momentum.

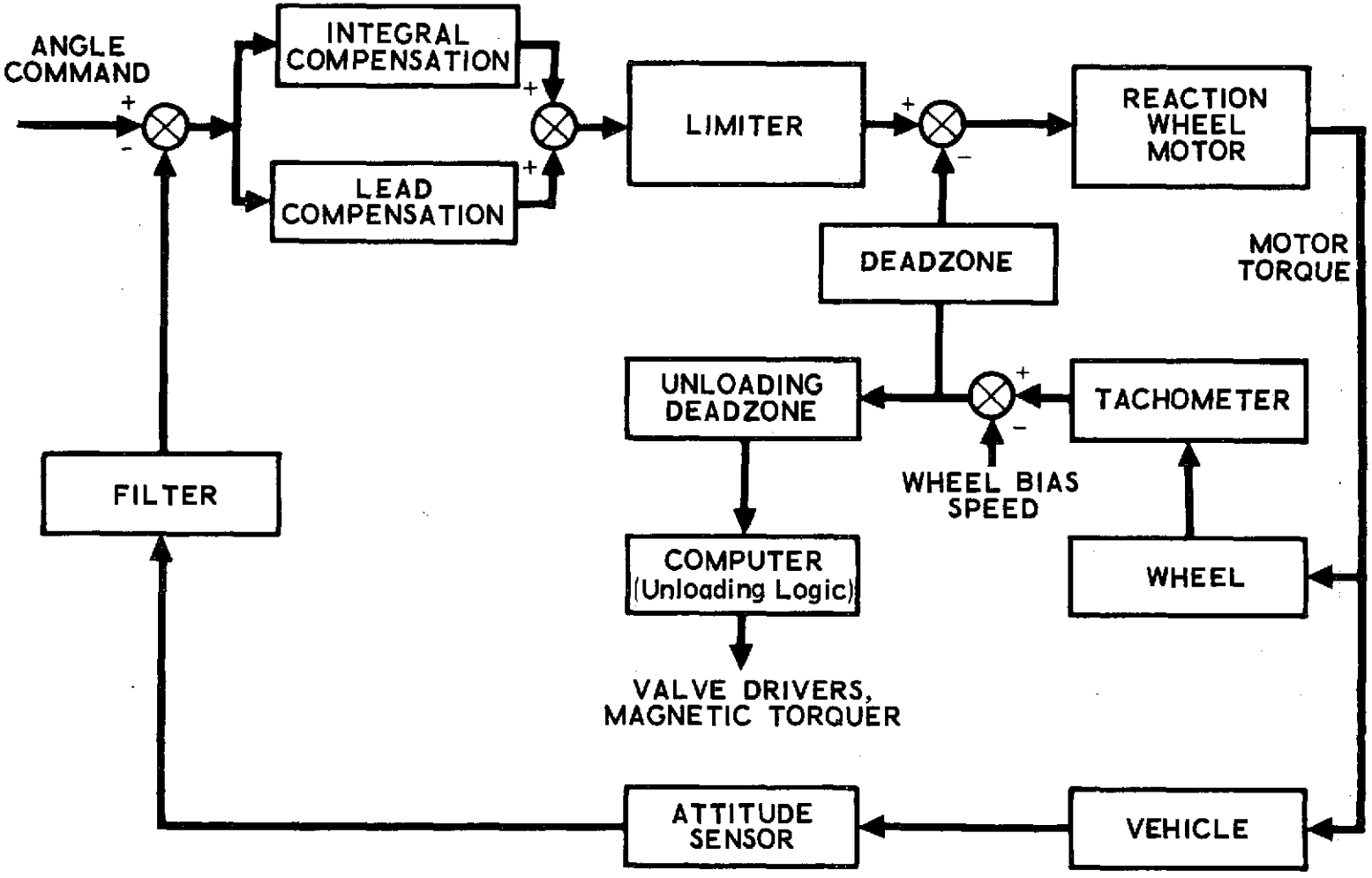


Figure 4-9. Reaction Wheel Attitude Control Subsystem

A two-degree-of-freedom Control Moment Gyro (CMG) is a constant high speed rotor device with two orthogonal gimbals. The CMG is useful primarily when high control torques and pointing accuracy are required. Velocity response of the spacecraft is obtained with low and constant power, high torque gains and minimum reset times. Because of inter-axis dynamic coupling and relatively complex control laws, an onboard computer is required.

The CMGs can be used in clusters of three units for maximum efficiency and reliability. The steering law uses the attitude and attitude rate errors to generate a three-axis torque command vector which is transformed into the CMG command gimbal rates. A matrix relating the six gimbal rates (3 CMGs) to the resultant torque output enables calculation of the desired gimbal rates.

A typical block diagram is shown in Figure 4-10 which is similar to the Apollo Telescope Mount Control System. The three CMG cluster can provide high accuracies and stability (over a specified time interval) such as $\pm$ 0.5 arc-sec in conjunction with a star sensing system and a magnetic momentum unloading torquer. For a 500 ft-lb-sec CMG, the weight is 91 kg, the size is 0.11 m^3 and the average power required is 25 W.

The simplest mass expulsion Reaction Control System (RCS) utilizes gaseous nitrogen (GN_2) or hydrazine (N_2H_4). A cold gas propulsion subsystem which can be used in shown in Figure 4-11 and a hydrazine RCS in Figure 4-12. The nitrogen gas provides a specific impulse of approximately 70 sec while the specific impulse for the hydrazine system is on the order of 200 sec for the higher level thrusters.

A Magnetic Torquer can utilize the earth's ambient magnetic field effectively for satellite angular momentum removal in low earth orbits. This is done by generating a controlled magnetic moment in a vehicle which interacts with the earth's magnetic field producing an

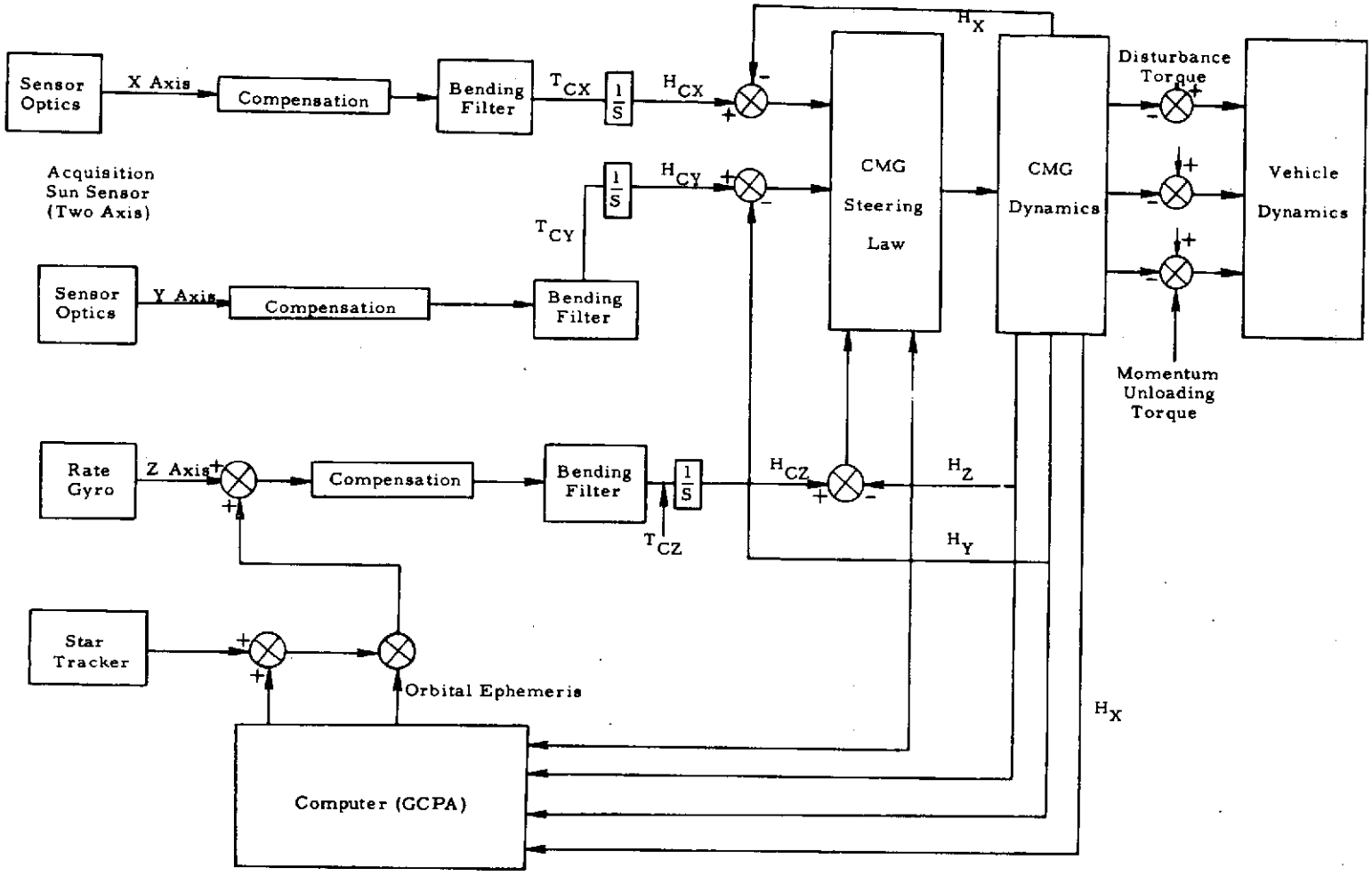


Figure 4-10. Simplified CMG Control Subsystem

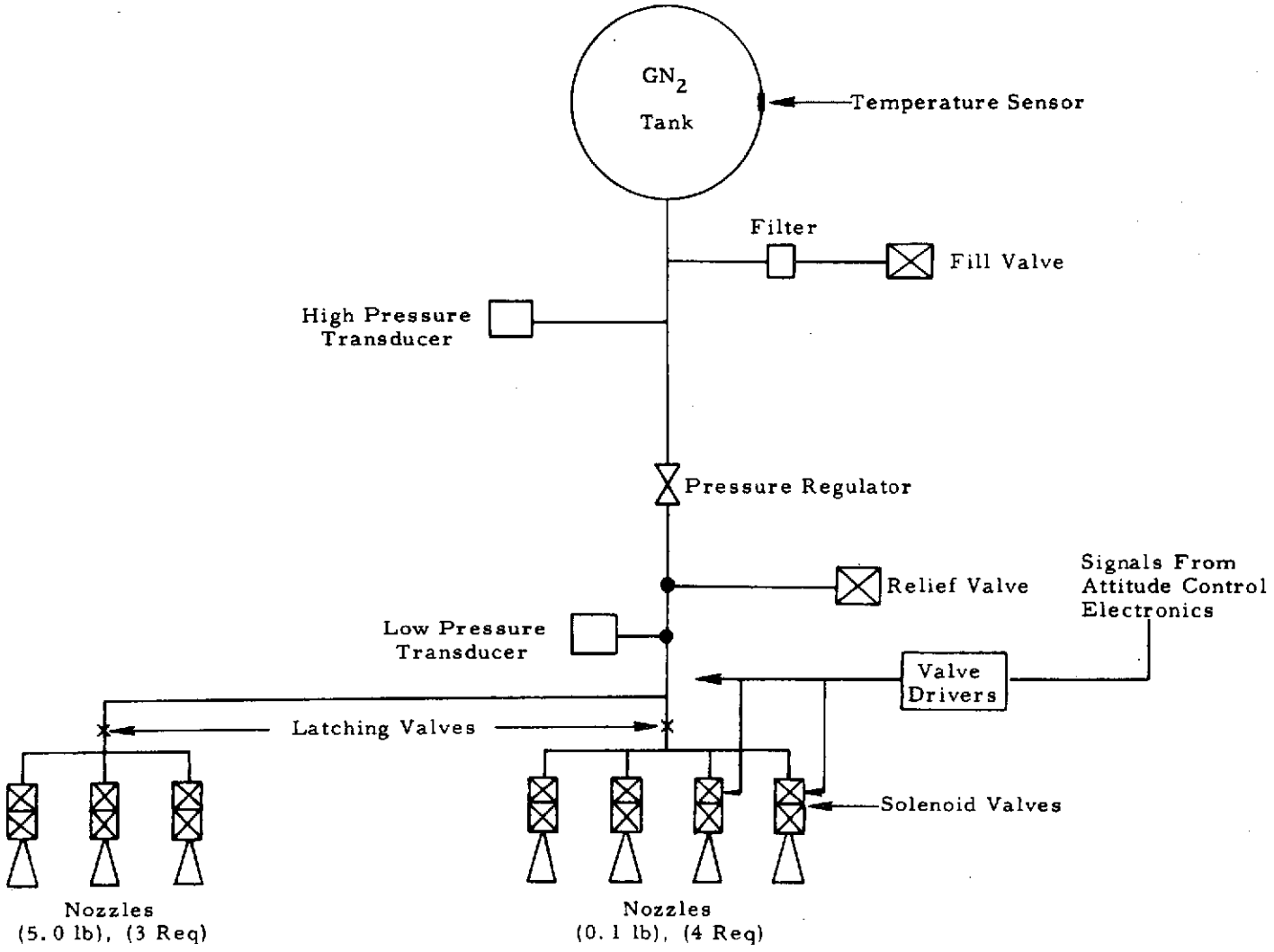


Figure 4-11. Cold Gas (GN_2) Propulsion Subsystem

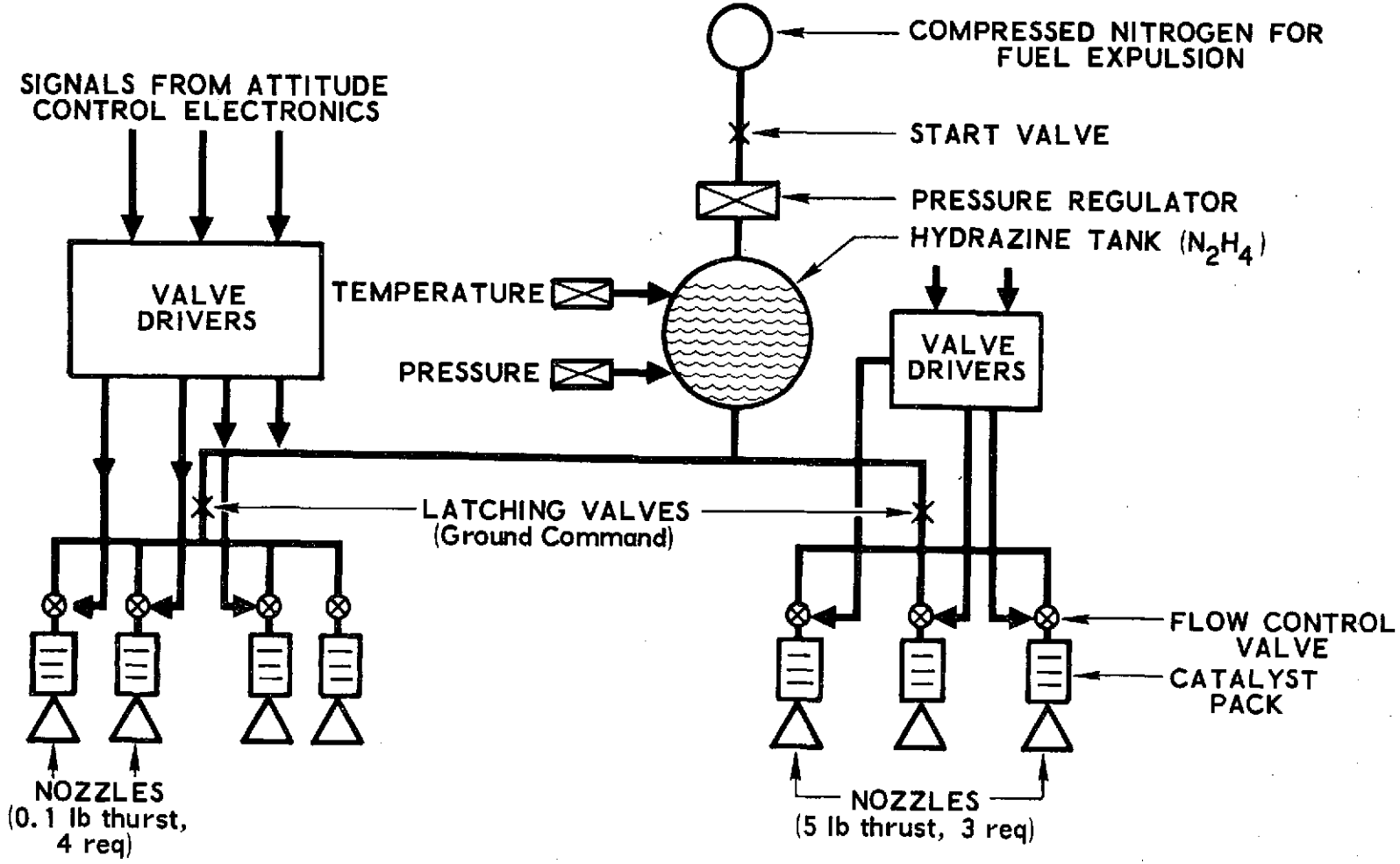


Figure 4-12. Hydrazine Propulsion Subsystem

external torque. The basic advantage of this approach is that no propellant is expended for removing the angular momentum from reaction wheels or control moment gyros. The primary requirements are electrical power and a set of magnetometers (three-axis) as shown in Figure 4-13, which determine the components of the magnetic field in the spacecraft body coordinates. A set of three orthogonal coils for current loops and an appropriate logic (computer) are also required in such a scheme. A block diagram is shown in Figure 4-14.

The size and weight of the magnetic torquer would normally depend on the size and orbit altitude of the vehicle. Such considerations as whether the torquing is continuous, intermittent or optimal also affect the physical characteristics of the torquer and the computer logic required. For example, a representative coil might consist of a thousand turns of aluminum or copper wire, have a 0.3m radius and require a current of 6 milliamperes. Power required would be on the order of 100 mW and the weight of the magnetic torquer about 4.5kg. Three such coils could be required for a typical vehicle application.

4.3 STANDARDIZED SUBSYSTEM MODULES

Based on the AVCS and G&N requirements, the various control elements indicated in Figure 4-1 were grouped into standardized subsystem modules with the aim of minimizing the total number of modules required for the missions of interest. The functional similarities and physical dimensions of the various elements were also considered in the design of the space replaceable units.

A total of 13 SRUs was defined for the attitude control, attitude sensing, propulsion, magnetic torquing, inertial reference measuring and computer processing functions. Several modules are identical in function but differ only in the size of the elements (e.g., reaction wheels) or the type of a sensor unit. For example, the sensing modules AVCS-5

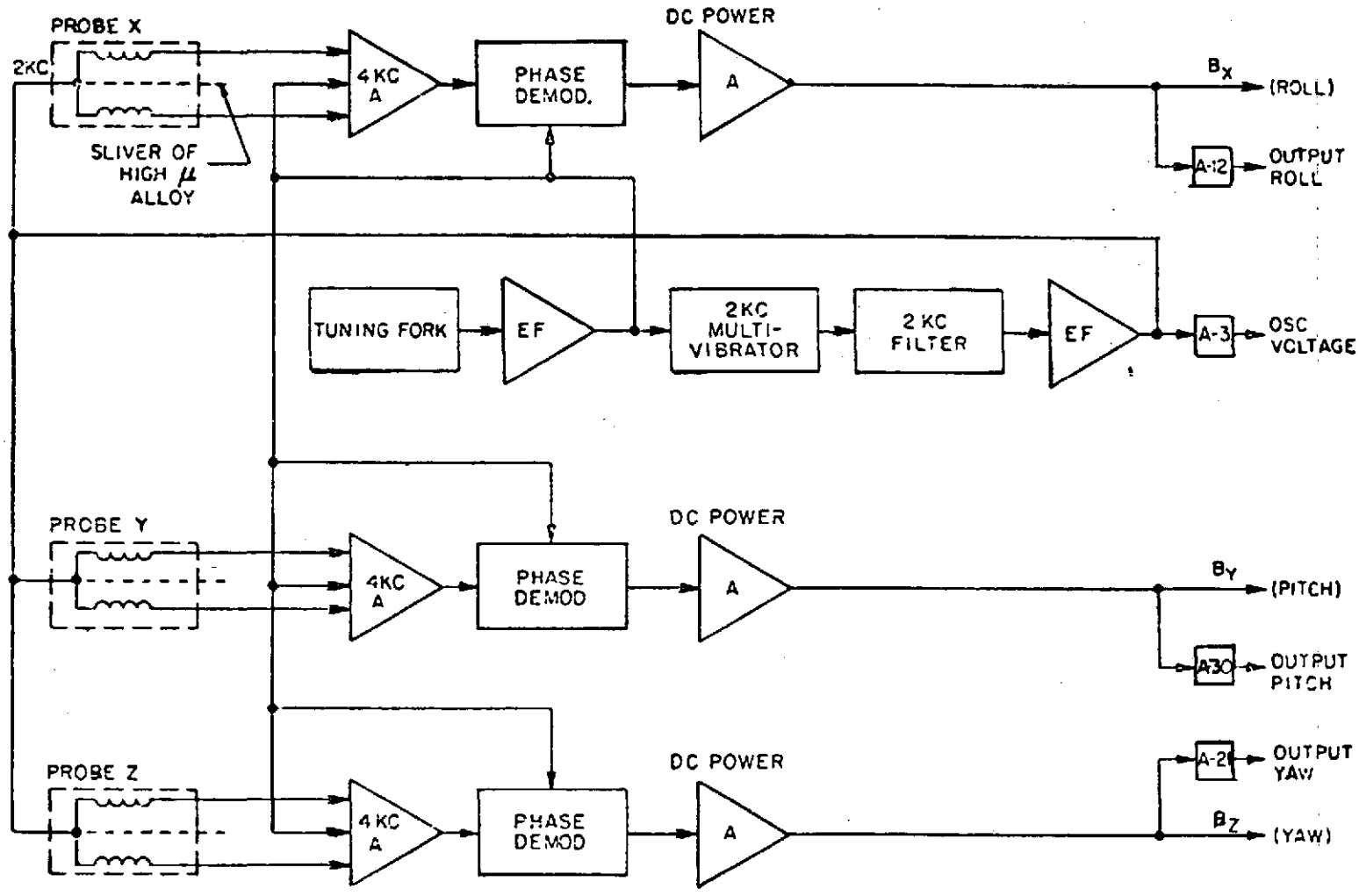


Figure 4-13. Magnetometer Subsystem

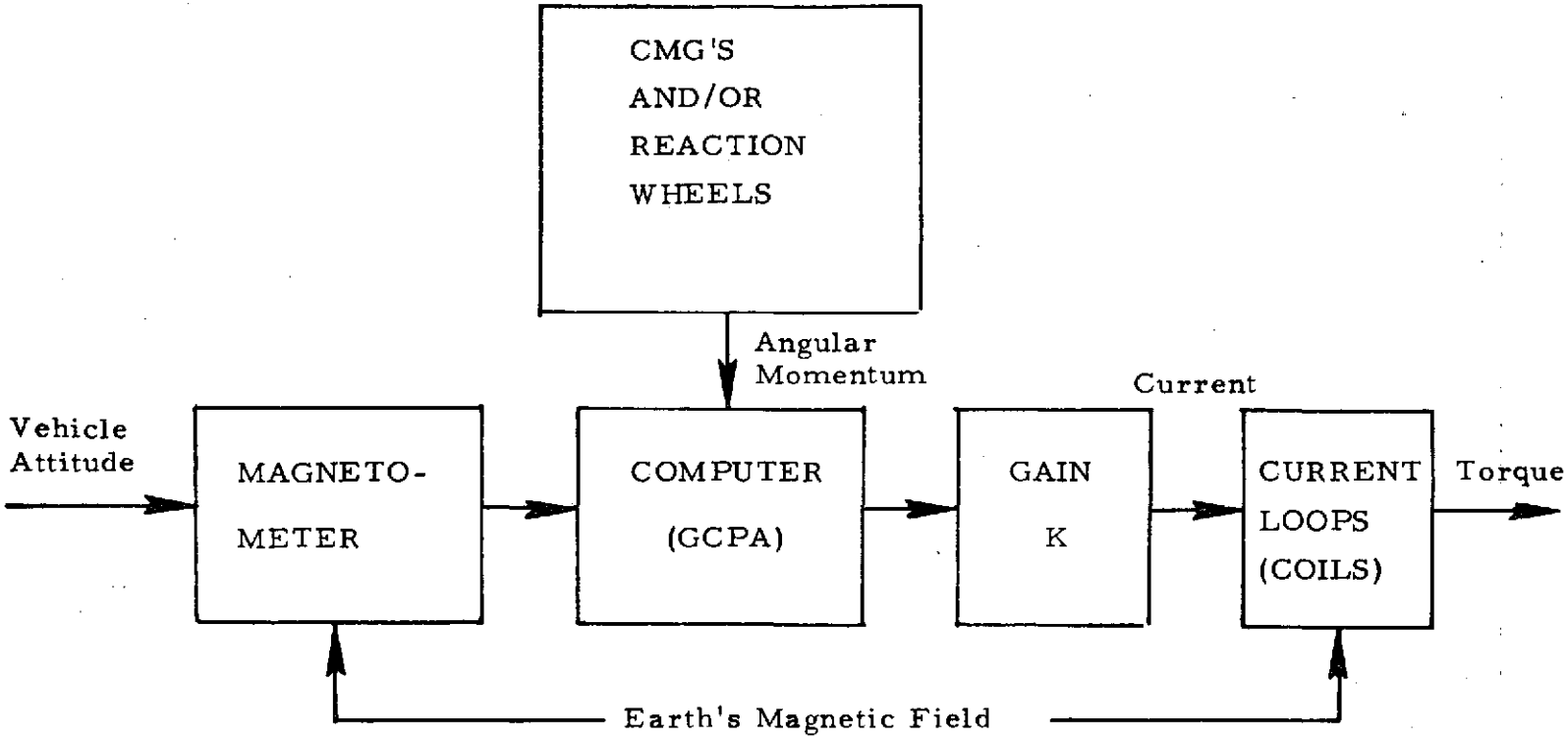


Figure 4-14. Magnetic Torquer Momentum Removal Scheme

and AVCS-5A differ only in that a high or low-altitude horizon sensor is used while several other elements (sun sensor, rate gyro, etc.) are the same. A similar distinction is made in the two sensing modules AVCS-6 and AVCS-6A. AVCS-6A contains a gimbaled star tracker, whereas AVCS-6 does not.

The cold gas (GN_2) and hydrazine (N_2H_4) propulsion modules can be used for attitude control and on-orbit velocity corrections (station-keeping). Either module is usable in principle although the hydrazine system is generally more efficient for large orbital corrections. The magnetic torquer module is included to provide the angular momentum dumping capability for low-altitude orbits. This capability conserves fuel and prevents interference with normal attitude control (limit cycling). The use of the magnetic torquer, however, should be on an intermittent basis to avoid interference with the experiments.

The summary and description of the defined SRU s are given in Tables 4-3 and 4-4. Reliability block diagrams are also shown.

4.4 SATELLITE MODULE ASSIGNMENT

The number and assignment of the standardized space replaceable units for each satellite in the mission matrix is given in Table 10-2. It is believed that this assignment will satisfy the performance requirements as specified in Tables 4-1 and 4-2.

It should be noted that only three reaction wheel sizes are used for all mission spacecraft and that no ion thrusters are included in the standardized modules.. In this way, the performance requirements can be met and the serviceability of automated spacecraft enhanced. It should also be noted that no redundancy is intended except where due to the assumed circumferential placement of the modules. Thus, for example, four propulsion SRU s are employed to provide improved torquing capabilities (longer moment arms). Also, the magnetic torquer is specified for low-altitude missions to provide better performance of the momentum exchange devices.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

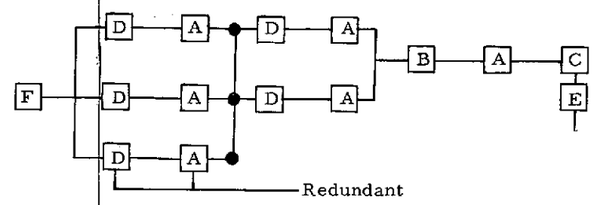
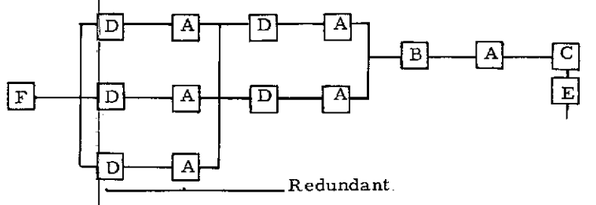
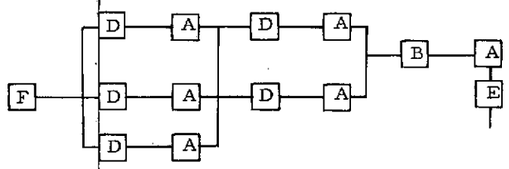
Table 4-3. Standardized Subsystem Modules - Attitude and Velocity Control System

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α(yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
AVCS-1	Reaction Wheel (5 ft-lb-sec)	A	Reaction Wheel	1	4.5	4.5	700	10	.509	14.80	1.0	D A B C E
		B	Wheel Electronics	1	1.4	1.4	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			38.9						
AVCS-2	Reaction Wheel (10 ft-lb-sec)	A	Reaction Wheel	1	8.2	8.2	700	10	.509	14.80	1.0	D A B C
		B	Wheel Electronics	1	1.4	1.4	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			42.6						
AVCS-3	Reaction Wheel (10 ft-lb-sec/ wheel)	A	Reaction Wheel	3	24.5	73.5	700	7	.623	14.80	1.0	D A B C
		B	Wheel Electronics	1	2.7	2.7	6000					
		C	Remote Terminal	1	2.0	2.0	500					
		D	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			109.2						
AVCS-4	Control Moment Gyro (Double Gimbal) (500 ft-lb-sec)	A	CMG Wheel	1	68.0	68.0	700	2	.858	13.10	1.0	E A B C D
		B	Wheel Electronics	1	4.5	4.5	6000					
		C	Torquer, Damper and Resolver	2	4.5	4.0	1000					
		D	Remote Terminal	1	2.0	2.0	500					
		E	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			114.5						

FOLDOUT FRAME 1

FOLDOUT FRAME 2

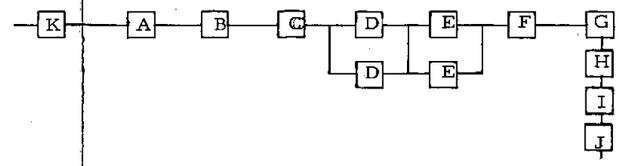
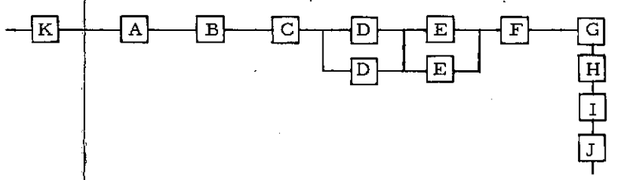
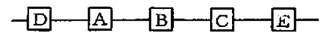
Table 4-3. Standardized Subsystem Modules - Attitude and Velocity Control System (Continued)

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
AVCS-5	Sensing	A	Auxiliary Electronics Assembly (AEA)	1	4.5	4.5	6500	10 (Based on intermittent use)	.174	6.42	1.154	
		B	Rate Gyro Package	1	1.4	1.4	1000					
		C	High Altitude Horizon Sensor	1	5.5	5.5	3000					
		D	Sun Aspect Sensor	5	2.3	11.5	100					
		E	Remote Terminal	1	2.0	2.0	500					
		F	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			55.9						
AVCS-5A	Sensing	A	Auxiliary Electronics Assembly (AEA)	1	4.5	4.5	6500	7 (Based on intermitten use)	.325	6.60	1.134	
		B	Rate Gyro Package	1	1.4	1.4	1000					
		C	Low Altitude Horizon Sensor	1	5.5	5.5	3000					
		D	Sun Aspect Sensor	5	2.3	11.5	100					
		E	Remote Terminal	1	2.0		500					
		F	Power Conditioning	1	2.0		500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			55.9						
AVCS-6	Sensing	A	Auxiliary Electronics Assembly (AEA)	1	4.5	4.5	6500	7	.306	6.28	1.125	
		B	Gimballed Star Tracker	1	18.1	18.1	5000					
		C	High Altitude Horizon Sensor	1	5.4	5.4	3000					
		D	Sun Sensor	5	2.3	11.5	100					
		E	Remote Terminal	1	2.0	2.0	500					
		F	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			72.5						
AVCS-6A	Sensing	A	Auxiliary Electronics Assembly (AEA)	1	4.5	4.5	6500	3	.654	6.84	1.080	Same as AVCS-6
		B	Gimballed Star Tracker	1	18.1	18.1	5000					
		C	Low Altitude Horizon Sensor	1	5.4	5.4	3000					
		D	Sun Sensor	5	2.3	11.5	100					
		E	Remote Terminal	1	2.0	2.0	500					
		F	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			72.5						

FOLDOUT FRAME 1

FOLDOUT FRAME 2

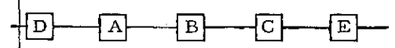
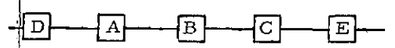
Table 4-3. Standardized Subsystem Modules - Attitude and Velocity Control System (Continued)

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α(yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
AVCS-7	Hot Gas Propulsion (N_2H_4) Small Tank	A	Nitrogen Tank (7.5-in OD)	1	2.3	2.3	1500	7	.496	14.35	1.021	
		B	Start Valve	1	0.5	0.5	100					
		C	Regulator Valve	1	1.8	1.8	100					
		D	Temperature Tranducer	2	0.05	0.1	2000					
		E	Pressure Transducer	2	0.05	0.1	2000					
		F	Hydrazine Tank (15-in OD)	1	4.0	4.0	1500					
		G	Latching Valves	2	0.5	1.0	200					
		H	Thruster (0.1 lb)	4	0.9	3.0	1000					
		I	Thruster (5.0 lb)	3	1.4	4.2	2000					
		J	Remote Terminal	1	2.0	2.0	500					
		K	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			50.6						
AVCS-8	Hot Gas Propulsion (N_2H_4) Small Tank	A	Nitrogen Tank (7.5-in OD)	1	2.3	2.3	1500	7	.496	14.35	1.021	
		B	Start Valve	1	0.5	0.5	100					
		C	Regulator Valve	1	1.8	1.8	100					
		D	Temperature Tranducer	2	0.05	0.1	2000					
		E	Pressure Transducer	2	0.05	0.1	2000					
		F	Hydrazine Tank (24-in OD)	1	11.0	11.0	1500					
		G	Latching Valves	2	0.5	1.0	200					
		H	Thruster (0.1 lb)	4	0.9	3.6	1000					
		I	Thruster (5.0 lb)	3	1.4	4.2	2000					
		J	Remote Terminal	1	2.0	2.0	500					
		K	Power Conditioning	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			57.6						
AVCS-9	Magnetic Torquer	A	Magnetometer (3 Axis)	1	3.2	3.2	200	7	.832	38.05	1.0	
		B	Amplifier	1	1.4	1.4	1600					
		C	Coil	3	4.6	13.7	200					
		D	Power Conditioning	1	2.0	2.0	500					
		E	Remote Terminal	1	2.0	2.0						
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			51.3						

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 4-4. Standardized Subsystem Modules - Guidance and Navigation

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α(yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
GN-1	Intertial Measuring Unit	A	Control and Readout Electronics	1	18.1	18.1	3,000	10	.198	6.17	1.0	
		B	Three Rate and Rate Integration Gyro Assy	1	18.1	18.1	10,000					
		C	Three Accelerometery Assembly	1	9.1	9.1	4,500					
		D	Power Conditioning	1	2.0	2.0	500					
		E	Remote Terminal	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			78.3						
GN-2	Guidance & Control Processor Assembly	A	Input/Output Unit	1	3.2	3.2	5,000	10	.246	7.13	1.0	
		B	Memory Unit	1	3.2	3.2	5,000					
		C	Arithmetic and Control Unit	1	3.2	3.2	5,000					
		D	Power Conditioning	1	2.0	2.0	500					
		E	Remote Terminal	1	2.0	2.0	500					
			Cabling	AR	5.0	5.0						
			Connectors	AR	2.0	2.0						
			Environmental Protection	AR	5.0	5.0						
			Structure	AR	17.0	17.0						
			TOTAL			42.6						

The selected number of SRU s is a first approximation design which can be optimized by further rearrangement of the control elements for more efficient packaging and weight reduction. This can be accomplished after initial design layouts, weight and performance estimates are obtained for the satellites of interest. For example, only three propulsion SRU s may be used if sufficient propellant can be stored onboard. Also, the reaction wheels and sensors may in some cases be repackaged to reduce the total number of SRU s. On the other hand, the geometrical constraints of some satellites may require multiple sensor SRU s should the visibility of the sensed objects be impaired.

5. TELEMETRY, TRACKING AND COMMAND

5.1 INTRODUCTION

Because of the complexity of the Telemetry, Tracking and Command (TT&C) subsystem and limited information on the system requirements, a large number of assumptions were necessary to conduct this section of the study. Underlying many of the assumptions is the more basic assumption that the satellites, except for the communications satellites, are performing scientific missions and not operational missions. Thus, in general, data missed at one opportunity can be obtained at some later opportunity. The assignment of support network priorities can increase the likelihood of obtaining data that may not be repetitive.

The satellites fall into three categories from a TT&C standpoint, and assumptions are made for each category. The categories are:

a. Operational Communications Satellites

b. High-Altitude Scientific Satellites

c. Low-Altitude Satellites

5.2 GROUND RULES

5.2.1 Operational Communication Satellites

The following assumptions are made in the design of low altitude operational satellites:

a. The TT&C subsystem is patterned after INTELSAT IV. (See Ref. 7.)

b. The TT&C subsystem operates at communication frequency.

c. The effective isotropic radiated power (EIRP) divided by the bit rate is equal to that for the INTELSAT IV, which has an EIRP of 0.8 W for 1000 bps.

d. A horn with +11-dB gain at edge-of-earth is used. (This does not apply to NND-3 which operates at VHF and would require a horn of excessive dimensions; an antenna of 0-dB gain is assumed for this satellite.)

e. Onboard losses are 2 dB.

f. The availability requirement for telemetry data does not require additional transmitter power to overcome rain attenuation. (This is pertinent primarily to NND-2D which operates at Ku-band, where the attenuation could be significant during heavy rainfall.)

5.2.2 High-Altitude Scientific Satellites

The following assumptions are made in the design of the high-altitude scientific satellites:

a. All TT&C subsystems operate at S-band.

b. The receiving stations employ 30-ft Unified S-band antennas with a temperature of 170°K. (The 30-ft Unified S-band antenna was selected because it is fairly common in the NASA Spaceflight Tracking and Data Network (STDN) which has 11 of them (Ref. 8). Five of these antennas have a temperature of 170°K, which is the highest listed for this type antenna.)

c. Each TT&C subsystem has a unified link for command and housekeeping telemetry as well as a separate downlink for mission data. (This assumption is based on common current practice. A 3-dB modulation loss was assumed for the unified downlink.)

d. There is no data recording for synchronous satellites. For other high-altitude satellites, only mission data is recorded; it is played back at the record rate and time shares the mission data link.

e. Hemispherical receiving antennas are used to provide broad-beam command reception capability.

f. Off-axis loss for directional transmitting antennas is 3 dB.

5.2.3 Low-Altitude Satellites

The following assumptions are made in the design of the low-altitude satellites:

a. Satellites will be serviced by the Tracking and Data Relay Satellite System (TDRSS) described in Ref. 6. Satellites with apogees below 2000 km (when using TDRSS multiple access service) will be supported by TDRSS because this is the upper altitude limit for 100 percent coverage by the TDRSS.

b. All mission and housekeeping data will be interleaved onto one link. (This is to simplify the satellite TT&C subsystems.)

c. Ranging is time shared with data collection. (The TDRSS guide link calculations make no allowance for modulation losses, as would be the case if simultaneous tracking were done with data collection.)

d. Data recording capability is included for satellites with perigee lower than 1200 km, the upper limit of the zone of exclusion. The recorder capacity is sufficient for 15 min , which is almost 5 min more than the longest time a satellite can be in the ZOE.

e. Recorder playback is at the record rate because of the long viewing time to the TDRSS.

f. Playback of the recorded data time shares the return link with the real-time data.

g. The lowest TDRSS capacity service possible for the application is used. (The TDRSS services in increasing order of capacity are: multiple access, S-band; single access, S-band; and single access, Ku-band.)

h. The rate 1/2 error correcting code is used for the return links only.

i. Hemispherical receiving antennas are used to provide broad-beam command capability.

5.2.4 Other Assumptions

The following additional assumptions are used:

a. Except for communication satellites, all directional antennas are parabolic with a diameter of 1-1/2 ft. (It was determined that a 1-1/2 ft antenna is the largest that could be easily accommodated within the envelope of the space-replaceable units. Use of a larger antenna would have necessitated additional mechanical complexity for storage and deployment.)

b. When unspecified, the command and housekeeping telemetry rates are 1000 bps.

c. Satellite tracking design assumptions are as listed in Table 5-1.

5.3 SUBSYSTEM REQUIREMENTS

The TT&C requirements were interpreted from Refs. 3 and 4 for the 29 missions of interest and are presented in Table 5-2. In the data rate column, a single entry for a low-altitude satellite indicates the sum of the

Table 5-1. Satellite Tracking System Design Assumptions

ITEM (Return Link)	TDRS (MA) S*[1]	TDRS (SA) S*[1]	TDRS (SA) Ku*[1]	GROUND (S-Band)[2]
Antenna Gain (db)	28.0	36.0	52.6	44.0
T_S (^oK)	1079	824	710	170
S/N (Ber = 10^5) (db)	10.0	10.0	10.0	10.0
Margin (db)	3.0	3.0	3.0	3.0
Transponder Loss (db)	2.0	2.0	2.0	2.0
Hardware Degration (db)	3.0	1.5	1.5	3.5
Misc. Losses (db)	1.0	1.0	2.0	1.0
Coding Gain (db)	5.2	5.2	5.2	None

Notes:

(1) *(MA) S: S-Band, Multiple Access
(SA) S: S-Band, Single Access
(SA) Ku: Ku-Band, Single Access
Source: TDRS User's Guide, October 1973

(2) Source: Spacecraft Tracking and Data Network User's Guide; STDN No. 101.1 dated April 1972

Table 5-2. Telemetry, Tracking and Command Requirements

PAYLOAD CODE	ALTITUDE (nmi)	INCLINATION	ORIENTATION	DATA RATE (bps)	STORAGE (bits)	COMPRESSION	COMMAND RATE (bps)
AST-1B	215/CIRC	ANY	OMNI	43,008	3.69×10^7	NO	128
AST-1C	SYNC	0°	EARTH	40,960/ 2,048	NONE	NO	128
AST-3	270/CIRC	30°	SOLAR	8,224	6.48×10^6	NO	25
AST-9A	250/CIRC	15°	EARTH	53,248	4.6×10^7	YES	2048
AST-9B	250/CIRC	15°	RANDOM	53,248	4.6×10^7	YES	2048
PHY-1A	200/CIRC	28.5	RANDOM	24,576	3.69×10^7	NO	250
PHY-1B	1900/140	90°	EARTH	3.69×10^6	9×10^6	NO	1000

Table 5-2. Telemetry, Tracking and Command Requirements (Continued)

PAYLOAD CODE	ALTITUDE (nmi)	INCLINATION	ORIENTATION	DATA RATE (bps)	STORAGE (bits)	COMPRESSION	COMMAND RATE (bps)
PHY-1C	20,000 1,000	28.5	EARTH	10,000/ 1,000	1×10^9	NO	1,000
PHY-2A	500/CIRC	90°	EARTH	3,324	9×10^6	NO	1,000
EO-3A	494/CIRC	99°	EARTH	1×10^7	9×10^9	YES	800
EO-4A	SYNC	0°	EARTH	1×10^7 / 1,000	NONE	NO	800
EO-6	905/915	103°	EARTH	1.65×10^6	NONE	NO	800
EO-7	SYNC	0°	EARTH	1.2×10^6/ 1,000	NONE	NO	1,000
EOP-3	325/CIRC	90°	EARTH	9.5×10^6	8.55×10^9	NO	1,000
EOP-4	16,200/ CIRC	90°	EARTH	2×10^3/ 1,000	9×10^6	NO	1,000

Table 5-2. Telemetry, Tracking and Command Requirements (Continued)

PAYLOAD CODE	ALTITUDE (nmi)	INCLINATION	ORIENTATION	DATA RATE (bps)	STORATE (bits)	COMPRESSION	COMMAND RATE (bps)
EOP-7	108/CIRC	90°	EARTH	1,000	9×10^5	NO	1,000
NND-1	SYNC	0°	EARTH	1,024	NONE	NO	50
NND-2A	SYNC	0°	EARTH	512	NONE	NO	1,000
NND-2B	SYNC	0°	EARTH	1,024	NONE	NO	50
NND-2D	SYNC	0°	EARTH	1,000	NONE	NO	1,000
NND-3	SYNC	0°	EARTH	600	NONE	NO	1,000
NND-4	SYNC	0°	EARTH	512	NONE	NO	1,000
NND-5	SYNC	0°	EARTH	1,000	NONE	NO	50
NND-6	SYNC	0°	EARTH	6,400/ 1,000	NONE	NO	1,000

Table 5-2. Telemetry, Tracking and Command Requirements (Continued)

PAYLOAD CODE	ALTITUDE (nmi)	INCLINATION	ORIENTATION	DATA RATE (bps)	STORAGE (bits)	COMPRESSION	COMMAND RATE (bps)
NN/D-8	905/915	103°	EARTH	5.6×10^5	NONE	NO	800
NND-11	991/CIRC	99°	EARTH	15×10^6	NONE	NO	1,000
NND-12	SYNC	0°	EARTH	1×10^7/1,000	NONE	NO	1,000
NND-13	SYNC	0°	EARTH	1×10^7/1,000	NONE	NO	1,000
NND-14	200/CIRC	98°	EARTH	9.5×10^6	8.55×10^9	NO	1,000

mission data rate and the housekeeping data rate, while a single entry for an operational communications satellite indicates the housekeeping data rate only. In the case of dual entries, the first number indicates the mission data rate, and the second number indicates the housekeeping data rate.

5.4 TRANSMITTER POWER AND ANTENNA REQUIREMENTS

5.4.1 Operational Communications Satellites

The communications satellites are of necessity treated differently from the other satellites being considered. References 3 and 4 define data rates and frequency bands, and it is stated that "data transmission to and reception from the operational communication satellite during nominal on-orbit operations will not utilize the NASA ground network."

Ground station characteristics not being available, it was assumed that the satellites would read out to ground stations in a similar way to INTELSAT IV, and the vehicle TT&C equipment was configured and sized accordingly. Five of the seven satellites of interest operate at C-band (as does INTELSAT IV), one operates at Ku-band, and one at VHF. INTELSAT IV utilizes a +11-dB gain horn antenna at C-band and hence, all of the C-band and the one Ku-band communication satellites were equipped with +11-dB horn antennas. It was decided that a horn at VHF was not practical because of size, and the satellite operating at VHF was equipped with a 0-dB gain onmidirectional antenna. INTELSAT IV has a data rate of 1000 bps, an EIRP of -1 dBW, and a transmitter power of 0.1 W. The transmitters were sized directly from the INTELSAT IV performance characteristics. This was simply the ratios of power as the ratios of data rates for satellites equipped with the +11-dB horn antenna. For the vehicle equipped with the 0-dB omnidirectional antenna, the power was necessarily 11 dB higher than the ratio of the data rates.

The required EIRP's and powers are given in Table 5-3, along with the frequency bands of the satellites.

Table 5-3. Transmitter Power Requirements - Communication Satellites

PAYLOAD	DATA RATE (bps) HSKG	EIRP (dBW)	P_o(W)	FREQUENCY
NND-1	1,024	-1.0	0.1	4 GHz
NND-2A	512	-4.0	0.05	C-BAND
NND-2B	1,024	-1.0	0.1	4 GHz
NND-2D	1,000	-1.0	0.1	Ku-BAND
NND-3	600	-3.2	0.76	VHF
NND-4	512	-4.0	0.05	C-BAND
NND-5	1,000	-1.0	0.1	4.2 GHz

5.4.2 High-Altitude Scientific Satellites

This group of satellites is read out, commanded and tracked directly from the NASA STDN ground stations. Reference 8 gives the characteristics of the 30-ft USB antennas and these are used in calculating the RF power requirements for these satellites.

Two RF downlinks were assumed for each of these satellites. Some of the information indicates separate links for the mission data and housekeeping data, and some indicates only one link for both. The rationale for selecting two links in all cases is that this is the way that all operations to date have proceeded. In the event of a malfunction, the housekeeping becomes vitally important in determining what caused the malfunction and what is needed to correct it. Where a separate housekeeping downlink is assumed, 1000 bps is the assumed rate.

Another rationale for having separate links is that tracking can be done via the uplink and the housekeeping downlink simultaneously with the readout of the mission downlink. By using subcarriers on the housekeeping link, the modulation losses are absorbed on this link with the mission link remaining free of these losses.

Table 5-4 gives the link calculation for a synchronous altitude vehicle reading out to a STDN USB ground station. It is patterned after and includes selected constants from the link analyses in Ref. 6.

The equation:

$$\text{Achievable data rate} = 39.9 + \text{EIRP}$$

is solved for the EIRP of each link and the results are tabulated in Table 5-5 along with the RF power requirements for each vehicle. Again, the calculations are iterative to determine the size and type of antenna to be used in the vehicle TT&C design.

5.4.3 Low-Altitude Satellites

Reference 6 gives the modus operandi for working with TDRSS TDRSS offers the user three different services; multiple-access, S-band

Table 5-4. Link Calculation - Synchronous Satellite to STBN Ground Station

Transmitter power and vehicle antenna gain	EIRP dBW
Space loss	-191 dB
Polarization loss	- 1 dB
STDN antenna gain	+ 44 dB
P_s at output of antenna	-148 dBW + EIRP
T_s	170°K
KT_s	-206.3 dBW
$P_s - KT_s$	58.3 dBW + EIRP
Transponder loss	- 2.0 dB
Demodulation loss	- 1.5 dB
Implementation loss	- 2.0 dB
System margin	- 3.0 dB
Required S/N (10^{-5} BER)	- 9.9 dB
Achievable data rate	39.9 dB + EIRP

Table 5-5. Transmitter Power Requirements - High-Altitude Satellites

PAYLOAD	DATA RATE (bps) MISSION/HSKG	EIRP (dBW)	P_o(W) HEMI, 0 dB	P_o(W) 1-1/2 ft DISH
AST-1C	40,960/ 2,048	6.2 -3.8	4.2 0.42	
PHY-1C	10,000/ 1,000	0.1 -6.9	1.02 0.2	
EO-4A	$1x10^7$/ 1,000	30.1 -6.9	0.2	32.0
EO-7	$1.2x10^6$/ 1,000	20.9 -6.9	0.2	3.9
EOP-4	$2x10^3$/ 1,000	-6.9 -6.9	0.2 0.2	
NND-6	6,400/ 1,000	-1.8 -6.9	0.66 0.2	
NND-12	$1x10^7$/ 1,000	30.1 -6.9	0.2	32.0
NND-13	$1x10^7$/ 1,000	30.1 -6.9	0.2	32.0

[S(MA)], single-access, S-band [S(SA)], and single access, Ku-band [Ku(SA)]. Although not stated explicitly in Ref. 8, it is at least implied that the user should plan on using the lowest capability that can satisfy the requirement from the lowest to the highest capability in S(MA), S(SA), and Ku(SA). Link calculations are given for the forward links (TDRS to user) and the return links (user to TDRS). The link calculations take into account such losses as RF losses, polarization losses, transponder losses, demodulation losses and miscellaneous hardware losses. A system margin of 3 dB is allowed and a bit error rate of 1×10^{-5} is provided in all cases. Link calculations are presented without error correction coding and with error correction coding at rate 1/2 and rate 1/3. Error correction coding at rate 1/2 was selected for the return links; this provides a link improvement of 5.2 dB. Coding was not used on the forward links since the decoding, which is the complicated part of the process, would have to be done onboard the satellite. The link calculations, including the coding improvement for the return links, are:

$$S(MA) = 20.4 + EIRP$$
$$S(SA) = 31.0 + EIRP$$
$$Ku(SA) = 30.3 + EIRP$$

and EIRP = user effective isotropic radiated power in dBW.

The return links for the low-altitude satellites have widely varying data rates from vehicle to vehicle, as can be seen in Table 5-6. It was assumed that the mission and housekeeping data could be interleaved on one link for each vehicle. The data bit rates given in Table 5-6 include both bit rates. No allowance is made for overhead and inefficiencies due to interleaving because the level of detail considered in this study does not justify such a fine adjustment.

Calculations were made to determine the EIRP required for each satellite using the equations for achievable data rates. The calculation process is iterative, since the required EIRP varies with the return link service used. Table 5-6 gives the EIRP's for the bit rates and the service

Table 5-6. Transmitter Power Requirements - Low-Altitude Satellites

PAYLOAD	DATA RATE MISSION + HSKG	EIRP (dBW)	TDRS SERVICE (Watts)			ANTENNA	
			S(MA)	S(SA)	Ku(SA)	1-1/2 ft	HEMI
AST-1B	43,008	25.9	6.2			x	
AST-3	8,224	18.75	1.0			x	
AST-9A	53,248	26.86	7.7			x	
AST-9B	53,248	26.86	7.7			x	
PHY-1A	24,576	23.5	4.25			x	
PHY-1B	3.69×10^6	34.7			2.7	x	
PHY-2A	3,324	14.8	30.2				x
EO-3A	1×10^7	38.7			6.1	x	
EO-6	1.65×10^6	29.2		8.8		x	
EOP-3	9.5×10^6	38.8			6.0	x	
EOP-7	1,000	9.6	9.1				x
NND-8	5.6×10^5	26.4		7.0		x	
NND-11	15×10^6	40.5			11.2	x	
NND-14	9.5×10^6	38.8			6.0	x	

selected along with the required antenna and transmitter power for that service. It should be noted that the directional antennas will require a means for pointing toward TDRSS satellites.

5.5 COMMAND RECEIVING ANTENNAS

Again following INTELSAT IV, the operational communication satellite uplink signals are received with the same antenna used for the downlink.

A broadbeam receiving antenna is included for all other high-altitude satellites. This is consistent with current practice.

The command links for low-altitude satellites were examined more closely because of the introduction of the TDRSS. Coding is not used on the forward link since decoding, which is the complicated part of the process, would have to be done in the vehicle. The forward link capabilities of the TDRSS without error correction coding from Ref. 6 are:

S(MA):	16.3 + Gu
S(SA):	29.3 + Gu (low power)
S(SA):	35.3 + Gu (high power)
Ku(SA):	31.0 + Gu (low power)
Ku(SA):	23.1 + Gu (high power)

where Gu = gain of user antenna in dB.

It is highly desirable (if not mandatory) to have a broadbeam antenna on the user vehicle for acquisition and commanding. The most stringent service required for the forward link is the command bit stream. The highest command rate required for any of the vehicles is 2048 bps. This is about 33 dB in bandwidth. The high powered S(SA) forward link has an achievable data rate of (35.3 + Gu) dB. If there is a 0-dB gain receiving antenna on the user vehicle, the S(SA) forward link will handle this maximum bit rate. Therefore, a 0-dB hemispherical user vehicle antenna is selected to satisfy the forward link needs of all of the low-altitude satellites. The S(MA) forward link will handle some of the lower command bit rates with a 0-dB antenna.

5.6 MISCELLANEOUS HARDWARE REQUIREMENTS

The receiver must operate at the uplink or forward link frequency. The receivers for the communication satellites operate at the communication frequency and all others operate at S-band. Each receiver and signal conditioner must be capable of handling the command rate shown in Table 5-2. The receiver must also provide the ranging code to the transmitter and a drive signal for coherent operation of the transmitters.

The unified links require a baseband assembly unit to accommodate the housekeeping data rate shown in Table 5-2 and a means for coupling the ranging code from the receiver to the transmitter.

The housekeeping and mission data are interleaved for transmission from the down altitude satellites. The interleaved bit stream is recorded on those satellites that go through the Zone of Exclusion. The mission data only is recorded on the nonsynchronous high-altitude satellites. Playback on all satellites is at the record rate. Thus, the record and playback rates both equal the data rate shown in Table 5-2. The storage requirement is also shown in Table 5-2.

5.7 STANDARD COMPONENTS

The antennas are, in essence, standardized as a result of the approach taken in this study.

The transmitter output power requirements and frequency bands are shown in Tables 5-3, 5-5, and 5-6 for all telemetry links. There are 15 different power levels at S-band and four different power levels at Ku-band.

The TT&C equipment requirements are summarized in Table 5-7 and the distribution of the number of S-band transmitters requiring each power level are shown in Figure 5-1. Based on this distribution, two levels of power were selected for all the S-band requirements; the levels chosen were 8W and 40W. Similarly, power levels of 0.1W and 12W were chosen to satisfy all Ku-band transmitters. This approach to standardization will impose additional constraints on the transmitters. For instance, a transmitter of a given power level would be used for both unified and non-unified links; thus, in a non-unified link it would have to be capable of accepting a coherent drive signal even though it would never operate coherently.

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Table 5-7
Telemetry, Tracking and Command Equipment Requirements

Payload Code			Transmitters										Receivers				Command Signal Conditioners	Tracking Circuitry	Baseband Assembly	Antenna						Recorders		Diplexer	Hybrid	Switch
Mission Model	SSPDA	Payload Name	S-40W	S-20W	S-8W	S-2W	S-0.4W	C-0.1W	VHF 1.0 W	Ku 0.1W	Ku-8W	Ku-12W	S-Band	Ku-Band	C-Band	VHF				S Band 1 1/2 Dish	K Band 1 1/2 Dish	VHF Omni	S Band Omni	K Band Horn	C Band Horn	R_1 (3)	R_2 (4)			
AST-1B	AS-03	Cosmic Background			M&H(1)								x				x	x		x track			x			x			x	x
AST-1C	AS-05	Adv Radio Astronomy			M	H	H						x				x		x				x(2)						x	x
AST-3	SO-03	Solar Physics Mission				M&H							x				x	x		x track			x			x			x	x
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M			M&H								x				x	x		x track			x			x			x	x
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M			M&H								x				x	x		x track			x			x			x	x
PHY-1A	HE-07	Small High Energy Observatory			M&H								x				x	x		x track			x			x			x	x
PHY-1B	AP-01	Upper Atmospheric Explorer									M&H		x				x	x			x track		x				x		x	x
PHY-1C	AP-02	Medium Altitude Explorer				M	H						x				x		x				x(2)			x			x	x
PHY-2A	AP-04	Gravity & Rel - Earth Orbit	M&H										x				x	x		x track						x			x	x
EO-3A	EO-8	Earth Observatory Satellite									M&H		x				x	x			x track		x				x		x	x
EO-4A	EO-9	Sync Earth Observatory Satellite	M				H						x				x		x	x			x						x	x
EO-6	EO-12	TIROS		M&H									x				x	x		x track									x	x
EO-7	EO-7	Sync Meteorological Satellite			M		H						x				x		x	x			x						x	x
EOP-3	OP-07	SEASAT-B									M&H		x				x	x			x track		x				x		x	x
EOP-4	OP-01	Geopause					M&H(5)						x				x		x				x(2)			x		x	x	x
EOP-07	OP-04	GRAVSAT		M&H									x				x	x					x				x	x	x	x
NND-1	CN-51	International Comm						H							x		x								x				x	x
NND-2A	CN-52	U.S. Domestic - A						H							x		x								x				x	x
NND-2B	CN-53	U.S. Domestic - B (ADV)						H							x		x								x				x	x
NND-2C	CN-58	U.S. Domestic - C (TDRS)								H				x			x							x					x	x
NND-3	CN-54	Disaster Warning							H							x	x					x							x	x
NND-4	CN-55	Traffic Management						H							x		x								x				x	x
NND-5	CN-56	Foreign Communication						H							x		x								x				x	x
NND-6	CN-59	Communication R&D/Proto				M	H						x				x		x				x(2)						x	x
NND-8	EO-56	Environmental Monitoring Satellite			M&H								x				x	x		x track			x						x	x
NND-11	EO-61	Earth Resource - LEO										M&H	x				x				x track		x						x	x
NND-12	EO-59	Earth Resource - Geosync	M				H						x				x		x	x			x						x	x
NND-13	EO-62	Earth Resource - Foreign	M				H						x				x		x	x			x						x	x
NND-14	OP-08	Global Earth & Ocean Monitoring									M&H		x				x				x track		x				x	x	x	x

Notes: (1) M = Mission; H = Housekeeping
(2) Two items required
(3) Off-the shelf recorder (Mbit/sec)
(4) High technology recorder (10 Mbit/sec)
(5) Two transmitters

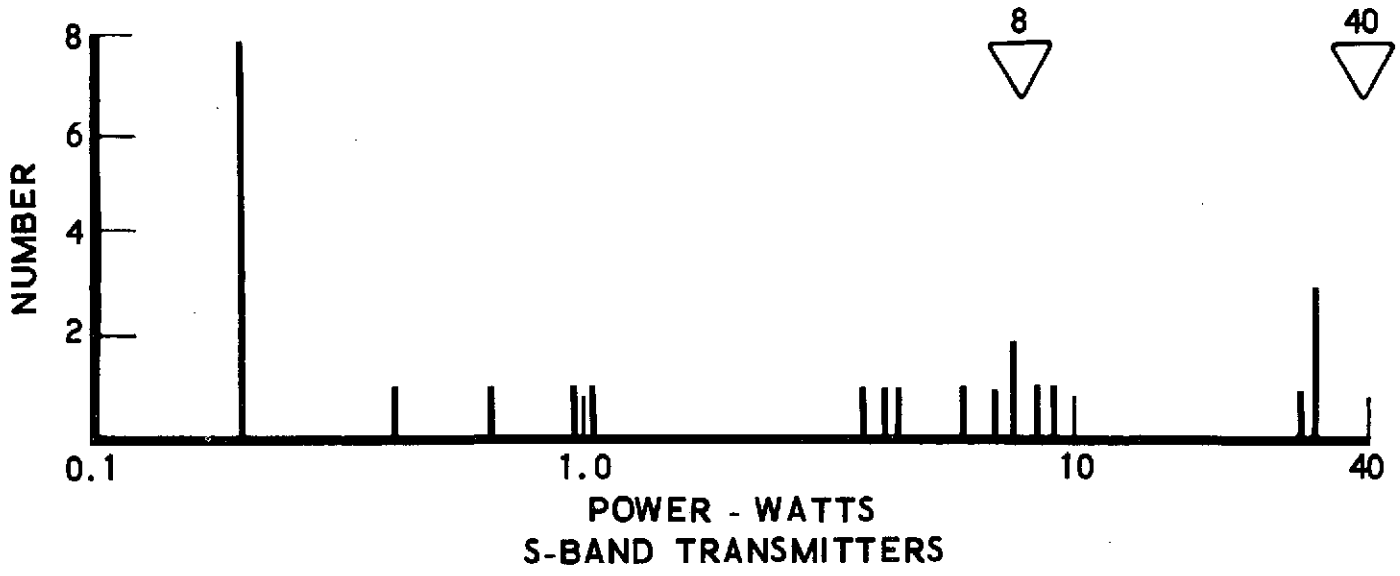

Figure 5-1. Transmitter Power Distribution

It should be noted that even with the additional constraints, transmitters with the same power and operating in the same band will not necessarily be interchangeable. Each will have to operate at a particular frequency assigned to the satellite on which it is used.

All receivers operating in a frequency band are identical in design. All receivers and signal conditioners can handle the highest command rate required. The receivers are also subject to frequency assignments.

The tracking receivers for the steerable parabolic antennas on the low-altitude satellites all operate at S-band and are of identical design. They are also subject to frequency assignment.

The baseband assembly units are all identical and can handle the highest housekeeping data rate.

Most of the low-altitude satellites and two of the high-altitude satellites require data storage. The amount of storage is given in Table 5-2. Present state-of-the-art recorders will record and play back at rates up to 1 Mbps and store about 2×10^4 bits. This type of recorder is used for all requirements within its capability and is designated Recorder R_1. From Table 5-2, it is seen that three satellites require record and play-back rates up to 10 Mbps and storage capacity of 4×19^9 bits. Further development work will be required to meet these requirements. The advanced recorder has been designated Recorder R_2.

5.8 STANDARDIZED SUBSYSTEM MODULES

The standard components were used to configure standard subsystems which were in turn allocated to the satellites, as shown in Tables 5-8, 5-9, and 5-10. The highlights of the TT&C configurations are presented in Table 5-11 which provides an overview of the most significant differences in the configurations. There are a total of 13 configurations; 10 of these are basic configurations and three are variants which require the addition of a recorder to the basic configuration.

Block diagrams for the standard subsystems are presented in Figures 5-2 through 5-5. Hybrids and diplexers have been incorporated as necessary to provide proper distribution of signals between antennas, transmitters, and receivers.

Table 5-8. Operational Communication Satellites

O CONFIGURATIONS

TT&C	TRANSMITTERS	RCVRS.	COMMAND SIG. COND.	ANTENNAS	DIPLEXER	HYBRID	BASEBAND ASSEMBLY UNIT
1	C-O.1W	C	YES	C-HORN	YES	YES	YES
2	Ku-0.1W	Ku	YES	Ku-HORN	YES	YES	YES
3	VHF-1.0W	VHF	YES	VHF-HEMI	YES	YES	YES

O ALLOCATIONS

SATELLITE/TT&C	1	2	3
NN/D-1	x		
NN/D-2A	x		
NN/D-2B	x		
NN/D-2D		x	
NN/D-3			x
NN/D-4	x		
NN/D-5	x		

Table 5-9. High-Altitude Satellites

O CONFIGURATIONS

TT&C	S-BAND TRANSMITTERS 8W	40W	RCVRS.	COMMAND SIG. COND.	BASEBAND ASSEMBLY	S-BAND ANTENNAS 1-1/2ft	Hemi.	RECORDERS	DIPLEXERS	HYBRIDS
4	1	1	S	YES	YES	1	1	NO	YES	YES
5	2	NO	S	YES	YES	NO	2	NO	YES	YES
5A	2	NO	S	YES	YES	NO	2	R_1	YES	YES
6	2	NO	S	YES	YES	1	1	NO	YES	YES

O ALLOCATIONS

SATELLITE/TT&C	4	5	5A	6
AST-1C		x		
PHY-1C			x	
EO-4A	x			
EO-7				x
EOP-4			x	
NN/D-6		x		
NN/D-12	x			
NN/D-13	x			

Table 5-10. Low-Altitude Satellites

- CONFIGURATIONS

TT&C	Transmitters	Rcvrs	Command Sig. Cond.	Track Circuitry	Baseband Assembly	Antennas 1-1/2ft	Hemi.	Recorders	Dixplexers	Hybrids
7	S-8W	S	YES	S	NO	S	S	NO	NO	YES
7A	S-8W	S	YES	S	NO	S	S	R_1	NO	YES
8	S-40W	S	YES	S	NO	S	S	NO	NO	YES
9	Ku-12W	S	YES	S	NO	Ku	S	NO	NO	YES
9A	Ku-12W	S	YES	S	NO	Ku	S	R_2	NO	YES
10	S-40W	S	YES	NO	NO	NO	S	R_1	YES	YES

- ALLOCATIONS

Satellite/TT&C	7	7A	8	9	9A	10
AST-1B		x				
AST-3		x				
AST-9A		x				
AST-9B		x				
PHY-1A		x				
PHY-1B					x	
PHY-2A						x

Satellite/TT&C	7	7A	8	9	9A	10
EO3-A					x	
EO-6			x			
EOP-3					x	
EOP-7						x
NN/D-8	x					
NN/D-11				x		
NN/D-14					x	

Table 5-11. TT&C Configuration Highlights

- COMMUNICATION SATELLITES

	1	2	3
FREQUENCY	C-BAND	Ku-BAND	VHF

- HIGH-ALTITUDE SATELLITES (S-BAND)

	4	5	5A	6
PARABOLA, Watts	40, 8			8, 8
HEMI, Watts		8, 8	8, 8	
RECORDER			R_1	

- LOW-ALTITUDE SATELLITES

	7	7A	8	9	9A	10
FREQUENCY	S	S	S	Ku	Ku	S
PARABOLA, Watts	8	8	40	12	12	
HEMI, Watts						40
RECORDER		R_1			R_2	

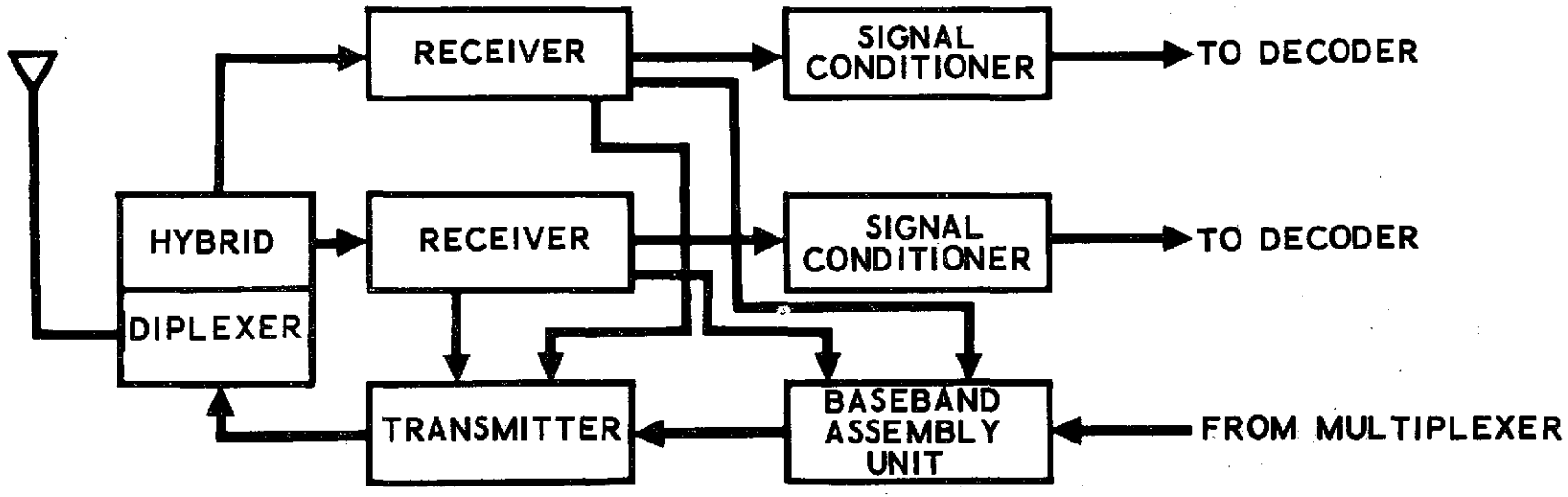

Figure 5-2. Communication Satellites (TT&C 1, 2, 3)

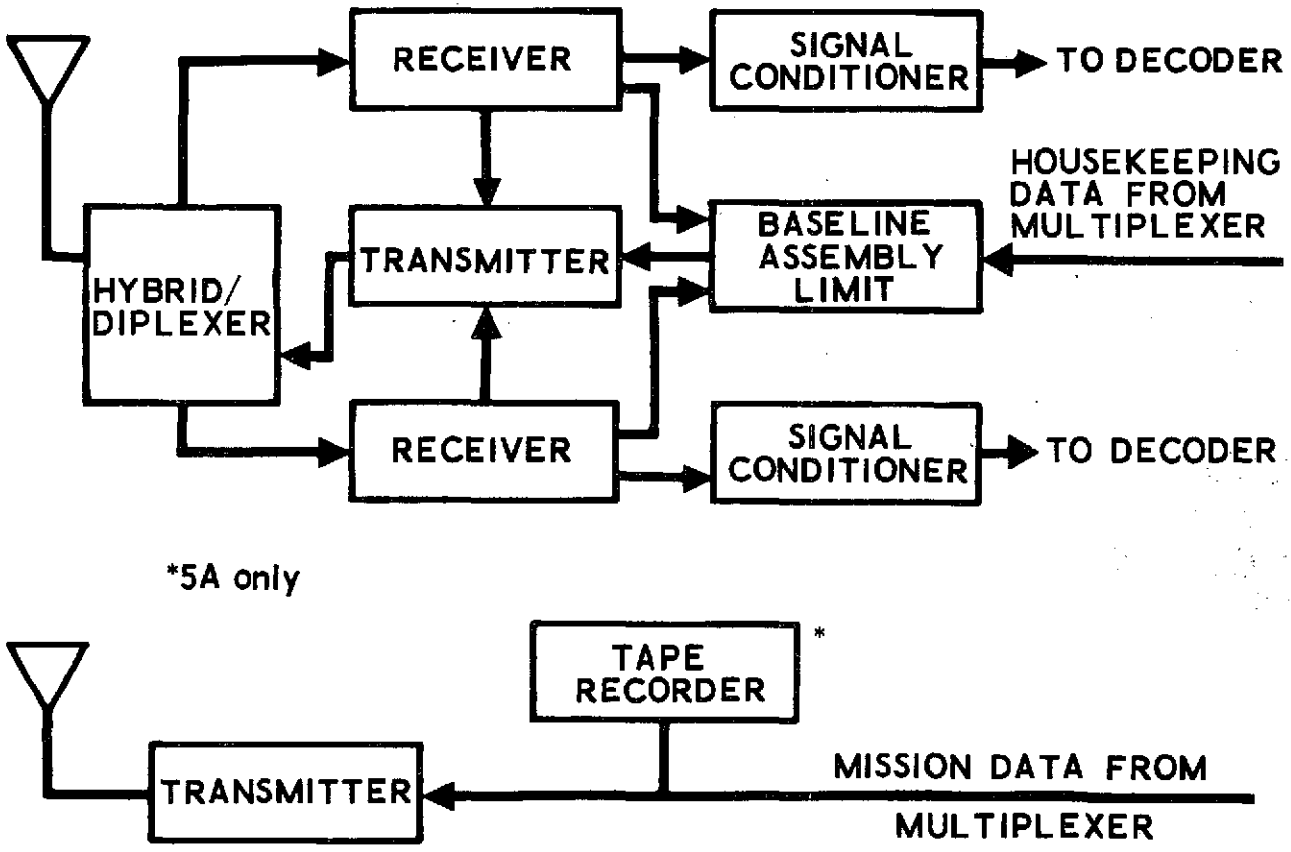

Figure 5-3. High-Altitude Satellites (TT&C 4, 5, 5A, 6, 6A)

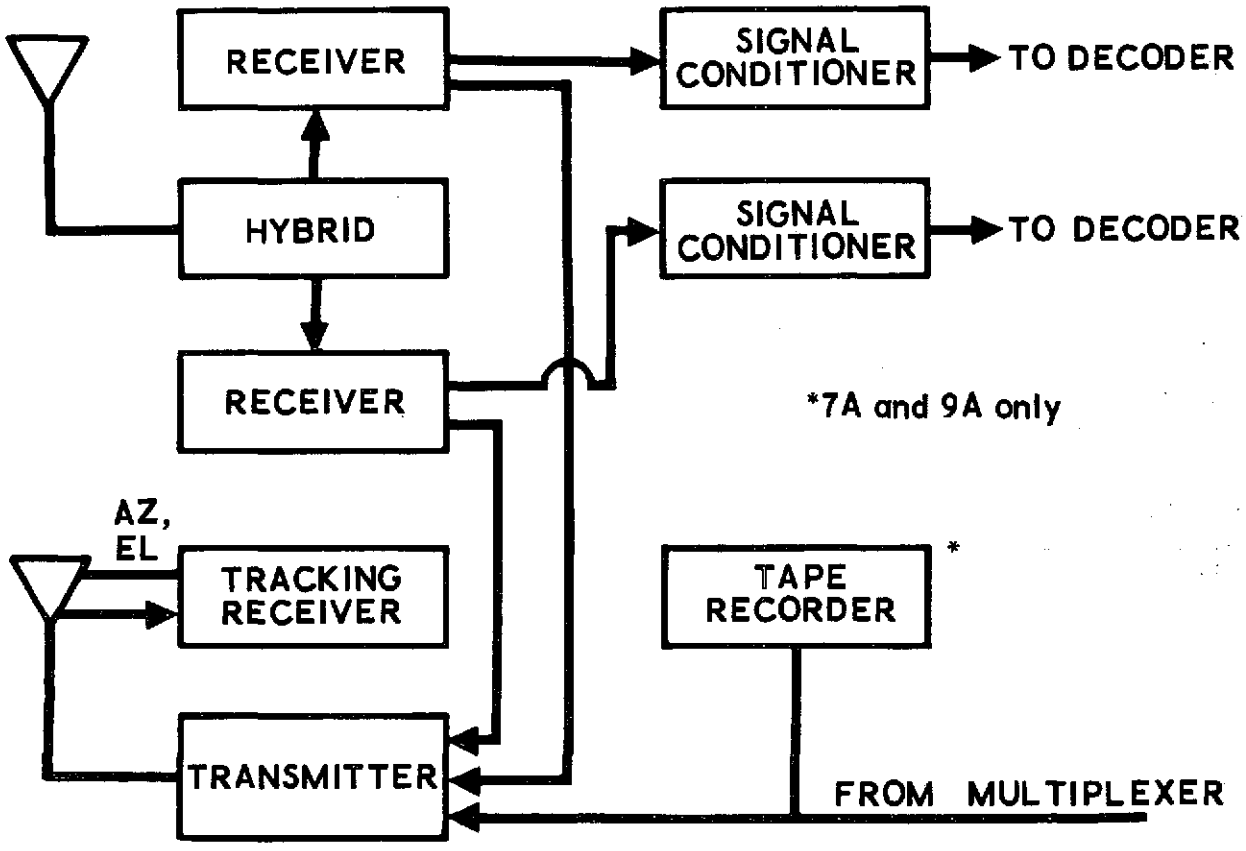

Figure 5-4. Low-Altitude Satellites with Steerable Antenna (TT&C 7, 7A, 8, 9, 9A)

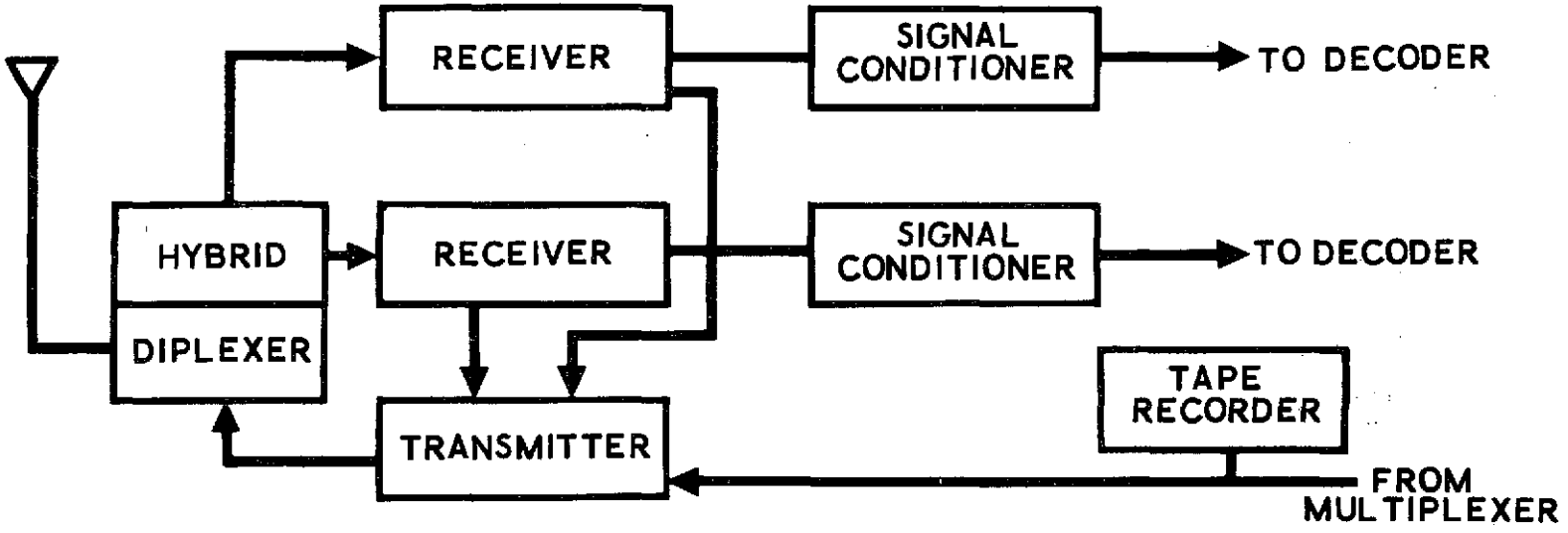

Figure 5-5. Low-Altitude Satellite with Hemispherical Antenna (TT&C 10)

5.9 SUMMARY

TT&C subsystems do not lend themselves gracefully to standardization of the complete subsystems. In an effort to minimize the number of standard subsystems, drastic measures were taken in this study to minimize the number of performance levels of components. For instance, S-band transmitters were limited to two power levels with the result that an 8-W transmitter was used to satisfy a 200-mW requirement (a 40/1 "overkill"). In spite of the drastic measures, essentially 13 standard TT&C subsystems (10 basic and three variants) were required for 29 satellites. Further, the same standardized subsystem is not necessarily interchangeable between satellites because of frequency assignments.

It should be noted, however, that the TT&C system does respond to custom assembly of standardized components into a standardized module. The standardized components can have a more complete set of performance levels; for instance, five standard S-band power levels may be more appropriate than two. Procured as a family, the cost of developing transmitters with five power levels would not be very much greater than the cost of developing transmitters with two power levels. The transmitters and receivers must be changed out of the standard module because of frequency assignments, so a changeout of transmitters for power level would not be materially more difficult. The standardized TT&C subsystem modules are summarized in Table 5-12.

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Table 5-12

Standardized Subsystem Modules - Telemetry, Tracking and Command

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY (3)	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
TTC-1	Telemetry, Tracking & Command	A	Transmitter (C-band, 0.1 W)	1	2	2	3000	10	.504	14.67	1.103	
		B	Receiver (C-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Horn Antenna (C-band)	1	2	2	40					
		E	Diplexer	1	1	1	150					
		F	Hybrid	1	1	1	50					
		G	Power Conditioning	1	2	2	500					
		H	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connectors	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			51						
TTC-2	Telemetry, Tracking & Command	A	Transmitter (Ku-band 0.1 W)	1		2	3000	7	.639	15.24	1.086	
		B	Receiver (Ku-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Horn Antenna (Ku-band)	1	2	2	40					
		E	Diplexer	1	1	1	150					
		F	Hybrid	1	1	1	50					
		G	Power Conditioning	1	2	2	500					
		H	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connectors	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			51						

Notes: (1) Recorder #1 = Off-the-shelf (1 Mbit/sec)
(2) Recorder #2 = High technology (10 Mbit/sec)
(3) AR = As required

FOLDOUT FRAME 1

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Table 5-12

Standardized Subsystem Modules - Telemetry, Tracking and Command

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
TTC-3	Telemetry, Tracking & Commmand	A	Transmitter (VHF, 0.2 W)	1	1	2	3000	7	.637	15.13	1.085	
		B	Receiver (VHF)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Omni Antenna (VHF)	1		1	100					
		E	Diplexer	1		1	150					
		F	Hybrid	1		1	50					
		G	Power Conditioning	1		2	500					
		H	Remote Terminal	1		2	500					
			Cabling	AR		5						
			Connectors	AR		2						
			Environmental Proctection	AR		5						
			Structure	AR		17						
			TOTAL			50						
TTC-4	Telemetry, Tracking & Command	A	Transmitter (S-band 8 W)	1		2	8000	3	.505	4.39	1.012	
		B	Transmitter (S-band 40 W)	1		8	12,000					
		C	Receiver (S-band)	2	4	8	4000					
		D	Signal Condition	2	2	4	2500					
		E	Baseland Assembly	1		1	1500					
		F	Dish Antenna (S-band, 1 1/2')	1		1	25					
		G	Omni Antenna (S-band)	1		1	100					
		H	Diplexer	1		1	150					
		I	Hybrid	1		1	50					
		J	Power Conditioing	1		2	500					
		K	Remote Terminal	1		2	500					
			Cabling	AR		5						
			Connectors	AR		2						
			Environmental Protection	AR		5						
			Structure	AR		17						
			TOTAL			60						

FOLDOUT FRAME 1

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Table 5-12

Standardized Subsystem Modules - Telemetry, Tracking and Command

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
TTC-5	Telemetry Tracking & Command	A	Transmitter (S-band), 8 W)	2	2	4	8000	5	.372	5.10	1.020	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Baseband Assembly	1	1	1	1500					
		E	Omni Antenna (S-band)	2	1	2	100					
		F	Diplexer	1	1	1	150					
		G	Hybrid	1	1	1	50					
		H	Power Conditioning	1	2	2	500					
		I	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connector	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			54						
TTC-5A	Telemetry, Tracking & Command		TTC-5	1	54	54	10,000	5	.240	3.54	1.014	
		J	Recorder #1[1]	1	7	7						
						61						
TTC-6	Telemetry, Tracking & Command	A	Transmitter (S-band 8W)	2	2	4	8000	7	.245	5.08	1.025	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Baseband Assembly	1	1	1	1500					
		E	Dish Antenna (S-band, 1 1/2')	1	1	1	25					
		F	Omni Antenna (S-band)	1	1	1	100					
		G	Diplexer	1	1	1	150					
		H	Hybrid	1	1	1	50					
		I	Power Conditioning	1	2	2	500					
		J	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connectors	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			54						

FOLDOUT FRAME 1

Table 5-12

Standardized Subsystem Modules - Telemetry, Tracking and Command

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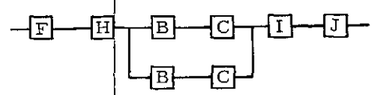
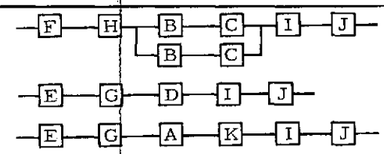
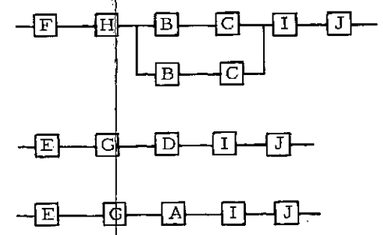
4

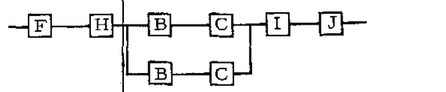
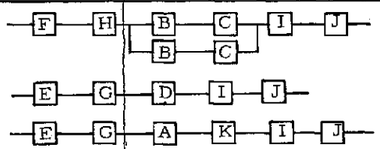
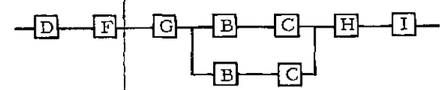
MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
TTC-7	Telemetry, Tracking & Command	A	Transmitter (S-band, 7 W)	1	3	3	8000	3	.589	5.66	1.015	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	25,000					
		D	Track Circuitry	1	2	2	5000					
		E	Tracking Antenna (S-band, 1 1/2')	1	3	3	25					
		F	Omni Antenna (S-band)	1	1	1	100					
		G	Antenna Drive	1	2	2	1500					
		H	Hybrid	1	1	1	50					
		I	Power Conditioning	1	2	2	500					
		J	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connectors	AR	2	2						
			Environmental Control	AR	5	5						
			Structure	AR	17	17						
			TOTAL			57						
TTC-7A	Telemetry, Tracking & Command		TTC-7	1	57	57	10,000	3	.452	3.80	1.010	
			Recorder #1[1]	1	7	7						
			TOTAL			64						
TTC-8	Telemetry, Tracking & Command	A	Transmitter (S-band, 40 W)	1	8	8	12,000	3	.530	4.73	1.013	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Tracking Circuitry	1	2	2	5000					
		E	Tracking Antenna (S-band, 1 1/2')	1	3	3	25					
		F	Omni Antenna (S-band)	1	1	1	100					
		G	Antenna Drive	1	2	2	1500					
		H	Hybrid	1	1	1	50					
		I	Power Conditioning	1	2	2	500					
		J	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connector	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			62						

FOLDOUT FRAME

FOLDOUT FRAME 2

Table 5-12

Standardized Subsystem Modules - Telemetry, Tracking and Command

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
TTC-9	Telemetry, Tracking & Command	A	Transmitter (Ku-band, 12 W)	1	2	2	8000	3	.530	4.73	1.013	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Track Circuitry	1	2	2	5000					
		E	Tracking Antenna (Ku/S band 1 1/2')	1	3	3	25					
		F	Omni Antenna (S-band)	1	1	1	100					
		G	Hybrid	1	1	1	1500					
		H	Power Conditioning	1	2	2	50					
		I	Remote Terminal	1	2	2	500					
		J	Cabling	AR	5	5	500					
			Connectors	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			54						
TTC-9A	Telemetry, Tracking & Command		TTC-9	1	56	54	14,000	7	.116	3.32	1.016	
		K	Recorder #2(2)	1	11	11						
			TOTAL			65						
TTC-10	Telemetry, Tracking & Command	A	Transmitter (S-band, 40 W)	1	8	8	8000	3	.568	5.30	1.014	
		B	Receiver (S-band)	2	4	8	4000					
		C	Signal Condition	2	2	4	2500					
		D	Omni Antenna (S-band)	1	1	1	100					
		E	Recorder #1	1	7	7	10,000					
		F	Diplexer	1	1	1	150					
		G	Hybrid	1	1	1	50					
		H	Power Conditioning	1	2	2	500					
		I	Remote Terminal	1	2	2	500					
			Cabling	AR	5	5						
			Connectors	AR	2	2						
			Environmental Protection	AR	5	5						
			Structure	AR	17	17						
			TOTAL			63						

6. DATA PROCESSING

6.1 SUBSYSTEM REQUIREMENTS

The data processing requirements were examined for the 29 missions that were selected for study. The data were either extracted from References 3 and 4 or estimated where data were not provided. A summary is presented in Table 6-1.

6.2 DATA BUS

With the advent of space-serviceable satellites it becomes imperative to change the design of the classical telemetry processing subsystem [usually called the digital telemetry unit (DTU)] in order to reduce the number of pin connections. The data bus approach to telemetry processing becomes the preferred approach in a space-serviceable type satellite due to the low number of connections which must be made between the telemetry processor and the subsystem points being sampled. In the classical DTU design there is one connection between the DTU and the subsystem for each analog point to be sampled and as many as 8 to 10 connections for each digital point sampled. In the case of relatively large satellites (e.g., DSP Link II) there are as many as 750 to 1000 connections into the DTU. TRW Systems, in their study of DSP, (Ref. 10) proposed a data bus system which reduced the number of connections to the ultimate, two lines. The penalty for this drastic reduction is the rate at which the data bus must operate.

Figure 6-1 shows a generalized approach to a data bus design. One SRU is devoted to the data bus processor and a remote terminal unit (RTU) is included in each SRU which communicates with the data bus. It should be noted here that although the major driving need for a data bus is telemetry processing, command processing can be included at minimal added cost and further reduce the number of connections into and out of the SRU s. Hence, communication with the data bus processor includes both uplink and downlink functions. The generalized approach shown in Figure 6-1 does not detail the number of connections required between the

Table 6-1. Data Processing System Design Parameters

Payload Code		Payload Name	Number of SRUs	Data Rate Bus (bit/sec)	Number Channels Per Terminal	Number Commands Per Terminal	Controller Program Memory (8 bit bytes)	Controller Data Memory (TLM words)
Mission Model	SSPDA							
AST-1B	AS-03	Cosmic Background	13	256 K	32	16	2048	1024
AST-1C	AS-05	Adv Radio Astronomy	13	256 K	32	16	4096	1024
AST-3	SO-03	Solar Physics Mission	21	64 K	32	16	2048	512
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	24	384 K	16	8	1024	2048
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	24	384 K	16	8	1024	2048
PHY-1A	HE-07	Small High Energy Observatory	13	256 K	32	16	2048	1024
PHY-1B	AP-01	Upper Atmospheric Explorer	17	64 K	32	16	2048	512
PHY-1C	AP-02	Medium Altitude Explorer	13	96 K	32	16	2048	1024
PHY-2A	AP-04	Gravity & Rel - Earth Orbit	14	64 K	32	16	2048	512
EO-3A	EO-8	Earth Observatory Satellite	22	5.12×10^7	16	8	512	4096
EO-4A	EO-9	Sync Earth Observatory Satellite	14	5.12×10^7	16	8	1024	4096
EO-6	EO-12	TIROS	22	1.024×10^7	16	8	512	4096
EO-7	EO-7	Sync Meteorological Satellite	18	1.024×10^7	16	8	1024	8192
EOP-3	OP-07	SEASAT-B	18	5.12×10^7	16	8	512	4096
EOP-4	OP-01	Geopause	11	32 K	32	16	4096	1024
EOP-07	OP-04	GRAVSAT	12	32 K	32	16	2048	512
NND-1	CN-51	International Comm	19	32 K	32	16	2048	512
NND-2A	CN-52	U.S. Domestic - A	10	16 K	64	32	1024	512
NND-2B	CN-53	U.S. Domestic - B (ADV)	19	32 K	32	16	2048	512
NND-2C	CN-58	U.S. Domestic - C (TDRS)	13	32 K	32	16	2048	512
NND-3	CN-54	Disaster Warning	12	16 K	64	32	1024	512
NND-4	CN-55	Traffic Management	12	16 K	64	32	1024	512
NND-5	CN-56	Foreign Communication	12	32 K	32	16	1024	512
NND-6	CN-59	Communication R&D/Proto	27	64 K	16	8	2048	512
NND-8	EO-56	Environmental Monitoring Satellite	28	2.048×10^6	16	8	1024	4096
NND-11	EO-61	Earth Resource - LEO	16	1.024×10^7	16	8	512	4096
NND-12	EO-59	Earth Resource - Geosync	15	64 K	64	32	4096	1024
NND-13	EO-62	Earth Resource - Foreign	13	32 K	128	64	2048	1024
NND-14	OP-08	Global Earth & Ocean Monitoring	18	32 K	128	64	2048	1024

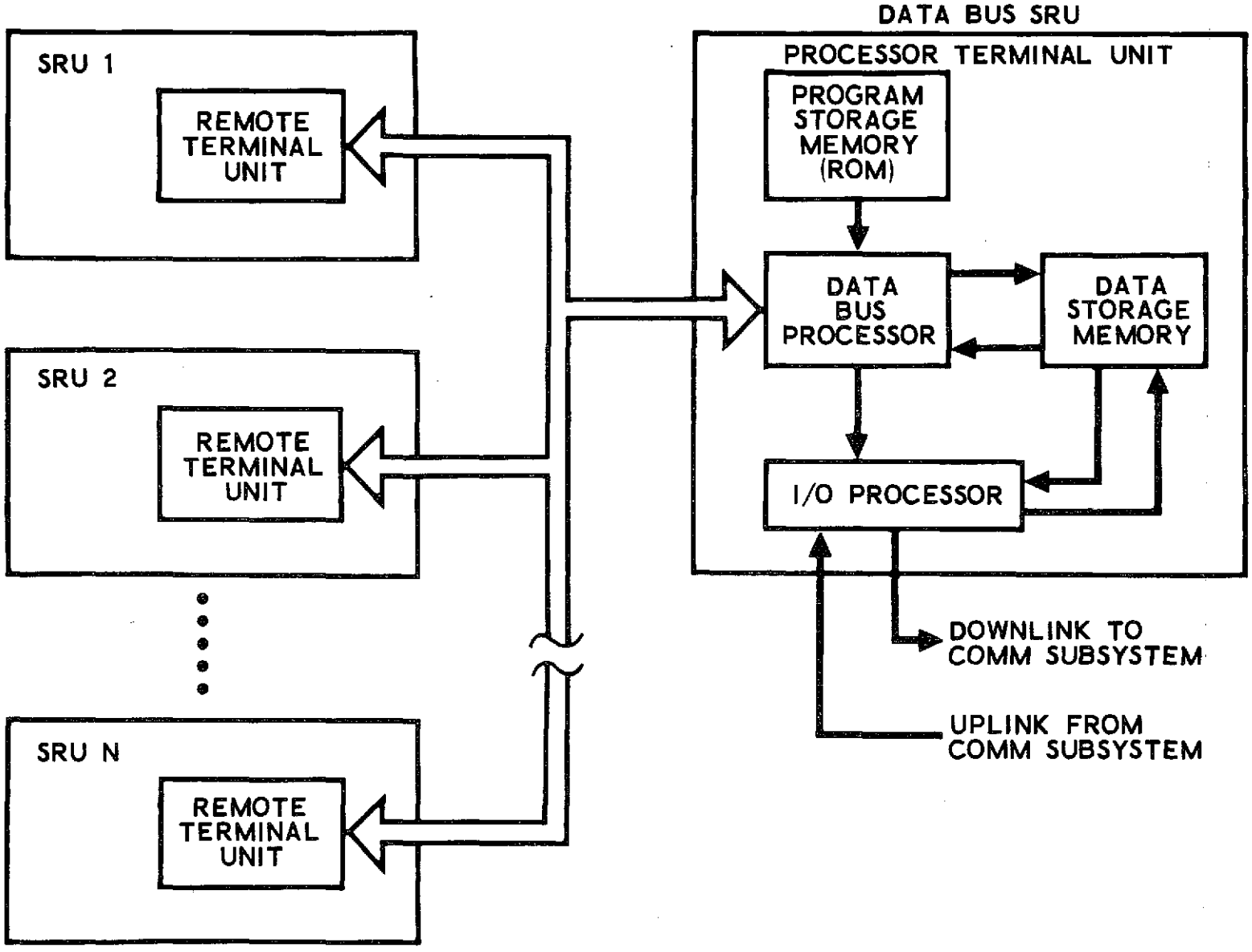


Figure 6-1. Data Bus Block Diagram

processor terminal unit (PTU) and the RTUs but merely shows a data path. The number of wires in that path can vary from the minimum, two (data and clock) to near 50 in a full duplex, parallel transmission type system. It is obvious that a system with 50 connections, while considerably better than present designs, would not solve the problem.

In general, the fewer lines there are in the system the greater the data rate on the bus. In the case of a two-line system, many overhead functions must be included on the bus in order to set up the addresses of the RTUs , effect either a transmission to the RTU or a reception from it, provide synchronizing and provide error checking. Then the required data can be passed. In order to reduce the connections to the minimum (two), approximately 90 percent of the data bus time must be devoted to overhead functions, or stated differently, the data bus rate must be at least an order of magnitude greater than the data rate of the link transmissions. While this may be acceptable to a relatively low data rate system like DSP Link II (128 kbps), it is not acceptable in a general sense where telemetry rates may go as high a 10 mbps per second.

In order to reduce the data rate on the bus, the overhead functions can be relegated to separate lines thus freeing the bus from overhead functions at the expense of added connections. This philosophy can be extended to the point where all functions have a separate line, data is transmitted in full parallel form, and separate input and out buses are provided. This, however, could require up to 50 lines and the data rate can only be reduced to that of the telemetry data at best.

6.3 REMOTE TERMINAL UNIT

Figure 6-2 shows a relatively conservative approach to the design of the RTU which designates special lines for transmit and receive to eliminate the need for data synchronizing and has discrete address selection for up to 31 SRU s. The PTU generates the control signals, activates the bus, and interfaces with the communication subsystem as

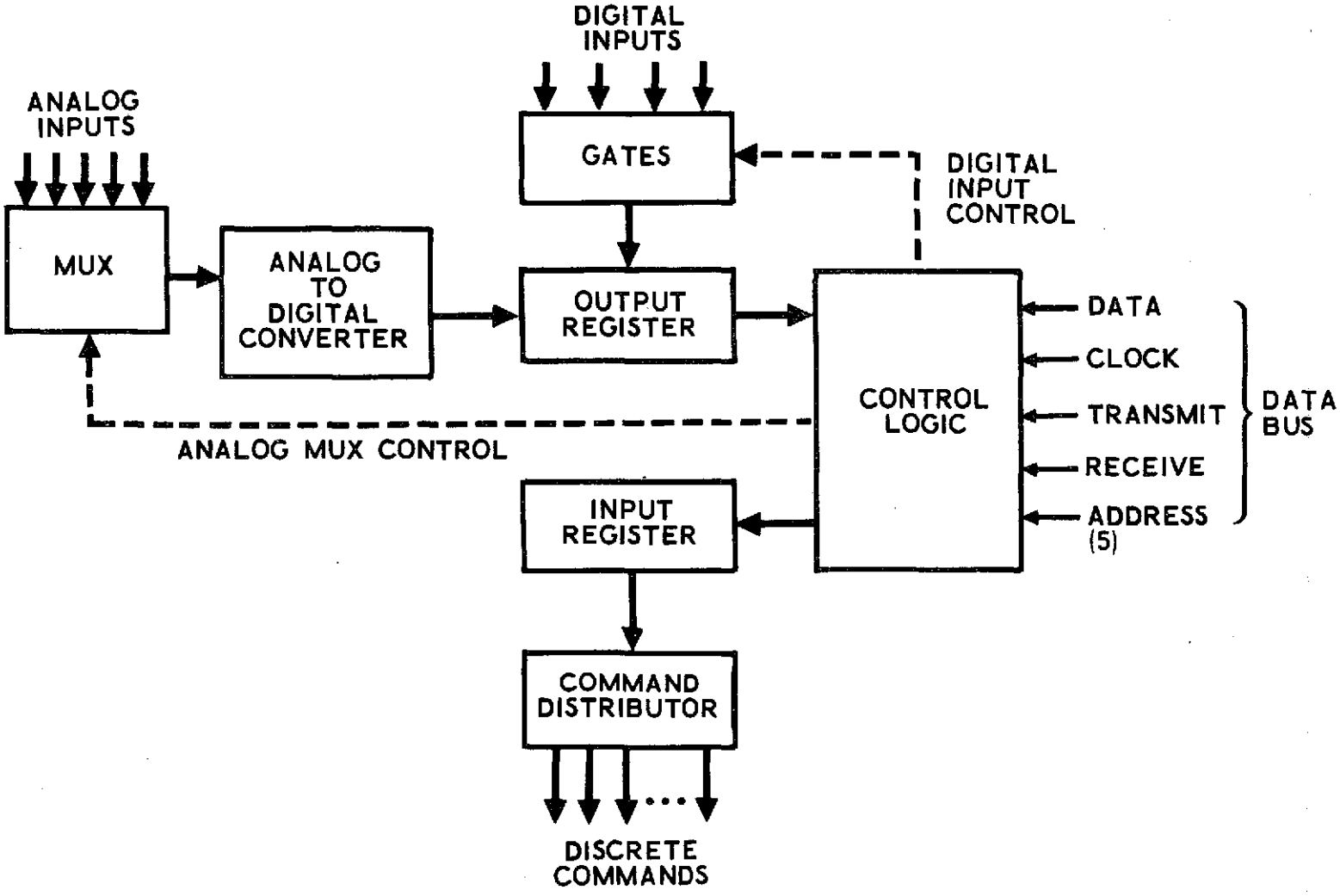


Figure 6-2. Remote Terminal Unit

shown in Figure 6-2. This general approach can be used as a standardized architecture for data bus design and, while it would be inefficient to design one unit to fill all applications, the architecture could be standardized and three or four separate designs, using different technology, could be developed to accommodate the full spectrum of requirements. That is, a high-speed design could be developed (which would also require high power) for the high data rates, a low-power version for low data rates and, if required, a hardened design could be developed.

The approach presented here requires nine connections into each of the SRUs and should reduce the data bus rate to no more than three times the aggregate link rates (if more than one uplink or downlink is to use the data bus). The PTU controls the timing, supplies requests for data to the SRUs (telemetry), sends data to the SRUs (commands), accomplishes error checking and retransmission, buffers data, and supplies the interface to the communications subsystem. The RTUs multiplex the points to be sampled, convert analog to digital (if necessary) and provide the interface with PTU which exercises executive control of all data bus functions.

6.4 OBSERVATIONS

The main advantage attributed to this type of system is, of course, its small number of connections between data processing subsystem and all the other subsystems. It must be recognized that there are disadvantages. These are lower reliability, higher power, and higher cost. The higher power and lower reliability are associated with the need for a greater number of components, especially in that each RTU requires a multiplexer and analog to digital converter rather than a single unit as is now the case. Higher cost is also a result of decentralizing the RTUs so that each not only requires its own multiplexer and ADC but also its own power supply. The PTU must also be more extensive in the data bus approach. The data bus remains, however, the only viable

alternative to the problem of making and breaking connections in space and in addition offers much more flexibility to change and redesign of the basic spacecraft.

6.5 STANDARDIZED SUBSYSTEM MODULES

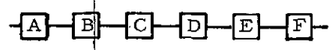
A single (programmable) subsystem module satisfies the data processing requirements of the 29 satellite configurations of interest. Its characteristics are listed in Table 6-2.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 6-2. Standardized Subsystem Modules - Data Processing

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
DP-1	Data Processing	A	Program Storage Memory	1	2.0	2.0	2000	10	.590	18.93	1.0	
		B	Data Bus Processor	1	4.0	4.0	500					
		C	Data Storage Memory	1	3.0	3.0	2500					
		D	I/O Processor	1	4.6	4.6	500					
		E	Power Conditioning	1	3.2	3.2	500					
		F	Cabling & Connectors	1	8.2	8.2						
			Connectors	1	2.3	2.3						
			Environmental Protection	1	4.5	4.5						
			Structure	1	15.0	15.0						
			TOTAL			46.8						



7. ELECTRICAL POWER

7.1 SUBSYSTEM REQUIREMENTS

In order to design the electrical power system, payload orbit characteristics, power requirements, and power system data were extracted from References 3 and 4, or estimated where data were not provided. These data are summarized in Table 7-1.

7.2 BATTERY SIZING

Battery sizing was derived from the power requirements, orbital characteristics, and design life data shown in Table 7-1. The usable battery capacity required for a given payload was based on the maximum possible eclipse period corresponding to that payload's orbital characteristics. The allowable maximum depth of discharge for the Ni-Cd batteries is a function of the total number of charge/discharge cycles occurring during the payloads' design life. The maximum battery depth of discharge selected for each payload is shown in Table 7-1, and was based on the cycle life data of Reference 9. The approximate installed battery capacity requirements were then obtained by dividing the usable battery capacity requirements by allowable depth of discharge (expressed as a fraction). The approximate installed battery amp-hr capacity requirements are shown in Table 7-2, assuming a 28V system.

7.3 BATTERY MODULARIZATION

It was required to define standard battery modules for the 29 satellite configurations being considered. Such modules would be self-contained and replaceable during in-space servicing operations. The following general guidelines were used for battery modularization:

a. A module should contain at least two independent batteries to provide parallel redundancy in case of total failure of one battery.

b. The number of different module sizes should be small.

c. Batteries should be of standard sizes used in current spacecraft programs.

FOLDOUT FRAME

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Table 7-1. Electrical Power System Design Parameters

Payload Code		Payload Name	Average Power Requirement (watts)	Maximum Eclipse (hrs)	Eclipses/Year	Orbital Period (yrs)	Design Life (yrs)	Selected Depth of Discharge (%) (1)	Installed Energy Storage Requirement (watt-hr) (2)
Mission Model	SSPDA								
AST-1B	AS-03	Cosmic Background	150	0.6	6000	1.5	5	10	900
AST-1C	AS-05	Adv Radio Astronomy	150	1.2	90	24	5	50	360
AST-3	SO-03	Solar Physics Mission	250	0.6	6000	1.5	5	10	1500
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	1200	0.6	6000	1.5	3	-	4800
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	1200	0.6	6000	1.5	3	-	4800
PHY-1A	HE-07	Small High Energy Observatory	150	0.6	6000	1.5	2	20	450
PHY-1B	AP-01	Upper Atmospheric Explorer	200	0.6	2400	2.0	2	-	1200
PHY-1C	AP-02	Medium Altitude Explorer	200	0.9	700	11.3	2	40	450
PHY-2A	AP-04	Gravity & Rel - Earth Orbit	110	0.6	3400	1.7	2	25	264
EO-3A	EO-8	Earth Observatory Satellite	958	0.6	5200	1.7	3	10	5750
EO-4A	EO-9	Sync Earth Observatory Satellite	402	1.2	90	24	3	60	800
EO-6	EO-12	TIROS	1400	0.6	4500	1.9	3	15	5600
EO-7	EO-7	Sync Meteorological Satellite	650	1.2	90	24	5	-	2400
EOP-3	OP-07	SEASAT-B	620	0.6	4000	1.6	7	10	3720
EOP-4	OP-01	Geopause	326	1.1	50	19	7	40	900
EOP-07	OP-04	GRAVSAT	325	0.6	4900	1.5	3	15	1300
NND-1	CN-51	International Communication	4400	1.2	90	24	10	30	17,600
NND-2A	CN-52	U.S. Domestic - A	305	1.2	90	24	10	30	1220
NND-2B	CN-53	U.S. Domestic - B (ADV)	4400	1.2	90	24	10	30	17,600
NND-2C	CN-58	U.S. Domestic - C (TDRS)	400	1.2	90	24	7	40	1200
NND-3	CN-54	Disaster Warning	200	1.2	90	24	7	40	600
NND-4	CN-55	Traffic Management	1100 (day) 250 (eclipse)	1.2	90	24	7	40	750
NND-5	CN-56	Foreign Communication	514 (day) 197 (eclipse)	1.2	90	24	7	40	600
NND-6	CN-59	Communication R&D/Proto	9300	1.2	90	24	7	40	27,500
NND-8	EO-56	Environmental Monitoring Satellite	589	0.6	4400	2	5	10	3530
NND-11	EO-61	Earth Resource - LEO	550	0.6	5100	1.7	5	10	3300
NND-12	EO-59	Earth Resource - Geosync	780	1.2	90	24	5	50	1870
NND-13	EO-62	Earth Resource - Foreign	780	1.2	90	24	5	50	1870
NND-14	OP-08	Global Earth & Ocean Monitoring	1000	0.6	6000	1.5	5	10	6000

Notes: (1) Selected Depth of Discharge based on required number of charge-discharge cycles/mission = (Eclipses/yr) x (Design Life)

(2) Installed Energy Storage Requirements = (Average Power Requirements) x (Maximum Eclipse) x 100/(Selected Depth of Discharge)

Table 7-2. Installed Energy Storage Requirements

Payload Code		Payload Name	Design Life (yrs)	Depth of Discharge Selected (%)	Installed Energy Storage Req. (watt-hr)
Mission Model	SSPDA				
AST-1B	AS-03	Cosmic Background	5	10	900
AST-1C	AS-05	Adv Radio Astronomy	5	50	360
AST-3	SO-03	Solar Physics Mission	5	10	1500
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	3	-	4800(1)
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	3	-	4800(1)
PHY-1A	HE-07	Small High Energy Observatory	2	20	450
PHY-1B	AP-01	Upper Atmosphere Explorer	2	-	1200(1)
PHY-1C	AP-02	Medium Altitude Explorer	2	40	450
PHY-2A	AP-04	Gravity and Rel - Earth Orbit	2	25	264
EO-3A	EO-8	Earth Observatory Satellite	3	10	5750
EO-4A	EO-9	Sync Earth Observatory Sat	3	60	800
EO-6	EO-12	TIROS	3	15	5600
EO-7	EO-7	Sync Meteorological Sat	5	-	2400(1)
EOP-3	OP-07	SEASAT-B	7	10	3720
EOP-4	OP-01	Geopause	7	40	900
EOP-07	OP-04	GRAVSAT	3	15	1300
NND-1	CN-51	International Comm	10	30	17,600
NND-2A	CN-52	U.S. Domestic - A	10	30	1220
NND-2B	CN-53	U.S. Domestic - B (Adv)	10	30	17,600
NND-2C	CN-58	U.S. Domestic - C (TDRS)	7	40	1200
NND-3	CN-54	Disaster Warning	7	40	600
NND-4	CN-55	Traffic Management	7	40	750
NND-5	CN-56	Foreign Communication	7	40	600
NND-6	CN-59	Communication R&D/Proto	7	40	27,500
NND-8	EO-56	Environmental Monitoring Satellite	5	10	3530
NND-11	EO-61	Earth Resource - LEO	5	10	3300
NND-12	EO-59	Earth Resource - Geosync	5	50	1870(2)
NND-13	EO-62	Earth Resource - Foreign	5	50	1870
NND-14	OP-08	Global Earth and Ocean Monitoring	5	10	6000(2)

Notes: (1) Energy storage requirements specified by NASA SSPDA.
(2) Power requirements estimated, no NASA data supplied.

The distribution of installed amp-hr requirements is shown in Figure 7-1. On the basis of the above guidelines and the approximate installed amp-hr requirements for each payload shown in Table 7-2, two standard battery assemblies were derived, one of 6 amp-hr capacity and one of 50 amp-hr capacity. Each assembly contains battery charge controllers and heaters.

7.4 SOLAR ARRAY SIZING

Solar array sizing was based on the power requirements, design life, and radiation environment of each payload. Payloads requiring more than 500W were arbitrarily assumed to have sun-oriented arrays, while those requiring less were assumed to have fixed arrays. In some cases the array size and/or type were specified in Reference 3. Array area and type for each payload are shown in Table 7-3 which provides a power system summary.

A standardized replaceable solar array drive assembly was designed and is assumed for those satellites for which it is deemed appropriate.

7.5 STANDARDIZED SUBSYSTEM MODULES

The design characteristics of the electrical power system modules are listed in Table 7-4. The allocation of the modules to the satellite of interest is shown in Table 10-2. It will be noted that, although Table 7-4 identifies three electrical power system modules and four variants, these modules actually include only three different electrical power elements, a 6 amp-hr capacity battery, a 50 amp-hr capacity battery, and a solar array drive assembly.

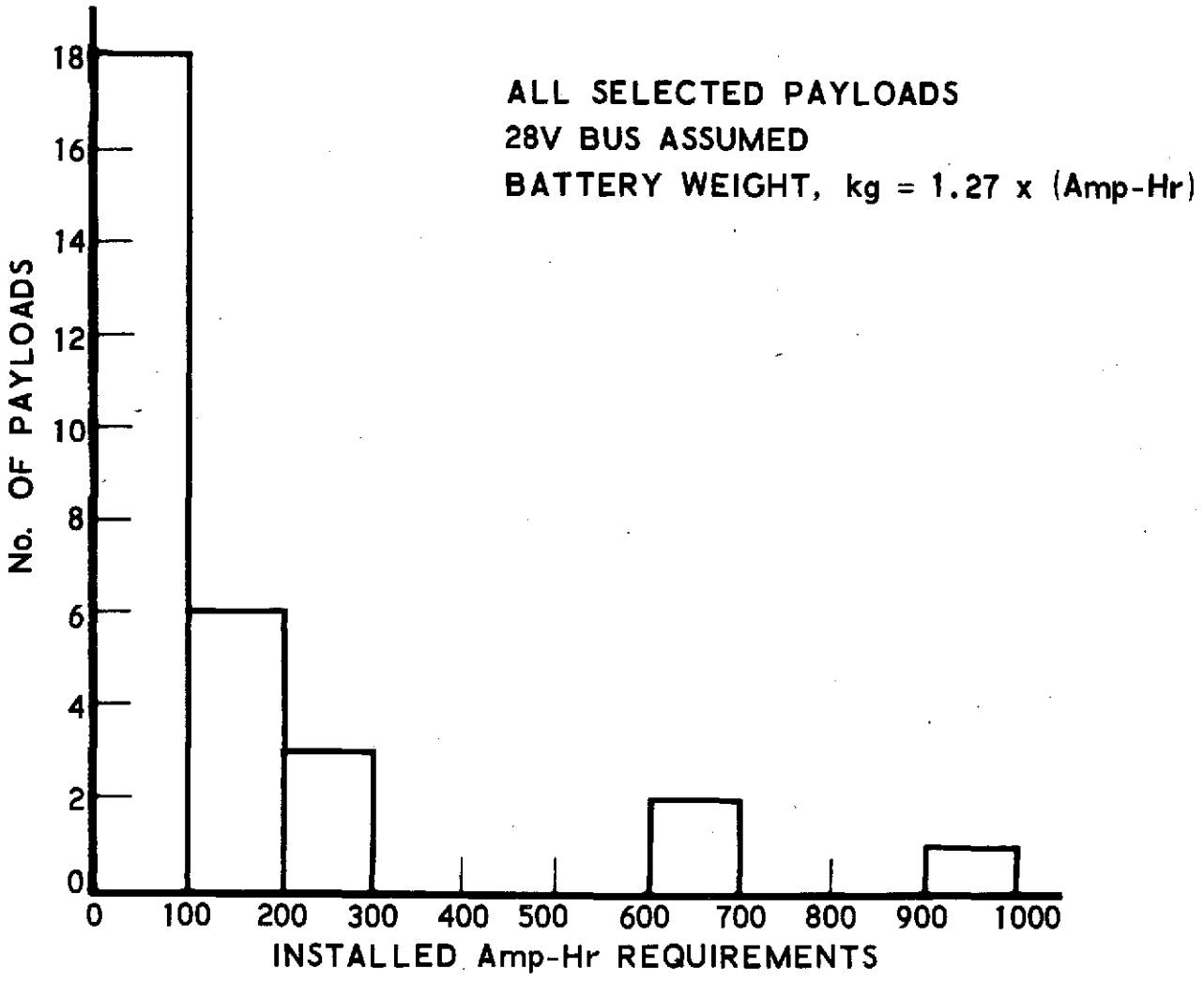

Figure 7-1. Distribution of Installed Amp-Hr Requirements

Table 7-3. Power System Summary

PAYLOAD	BATTERIES			SOLAR ARRAY	
	AH REQ (28 V)	No. MODULES		AREA REQ (sq ft)	TYPE**
		12AH	100AH		
AS-03 COSMIC BACKGROUND EXPLORER	32.1	3	-	75	U
AS-05 ADV RADIO ASTRONOMY	12.9	1	-	75	U
SO-03 SOLAR PHYSICS	53.5	4	-	125	U
HE-11 FOCUSING X-RAY TELESCOPE - 1.2M	171.2*	-	2	170	O
HE-01 FOCUSING X-RAY TELESCOPE - 3.0M	171.2*	-	2	170	O
HE-07 SMALL HI-ENERGY OBS	16.0	2	-	75	U
AP-01 UPPER ATMOSPHERE EXPLORER	42.8*	4	-	100	U
AP-02 MEDIUM ATMOSPHERE EXPLORER	16.0	2	-	100	U
AP-04 GRAVITY PROBE B_2	9.4	1	-	55	U
EO-8 EOS	205.0	-	2	190*	O*
EO-9 SYNCH EOS	28.6	3	-	60	O*
EO-12 TIROS	200.0	-	2	200	O
EO-7 ADV SYNCH METSAT	85.7*	-	2	100*	O*
OP-07 SEASAT	133.0	2	1	90	O

* Specified in NASA data sheets

** O - Sun-oriented; U - Unoriented (fixed)

Table 7-3. Power System Summary (Continued)

PAYLOAD	BATTERIES			SOLAR ARRAY	
	AH REQ (28 V)	No. MODULES		AREA REQ (sq ft)	TYPE**
		12AH	100AH		
OP-01 GEOPAUSE	32.1	3	-	160	U
OP-04 GRAVSAT	46.5	4	-	160	U
CN-51 INTERNATIONAL COMM	629.0	-	6	750*	O*
CN-52 DOMSAT A	43.5	5	-	123*	U*
CN-53 DOMSAT B	629.0	-	6	750*	O*
CN-58 DOMSAT C (TDRS)	42.8	4	-	45*	O*
CN-54 DISASTER WARNING	21.4	2	-	700	O
CN-55 TRAFFIC MANAGEMENT	26.8	2	-	150	O
CN-56 FOREIGN COMM	21.4	2	-	70	O
CN-59 COMM R&D PROTOTYPE	982.0	-	10	1300	O
EO-56 ENVIRONMENTAL MONITORING	126.0	2	1	80	O
EO-61 EARTH RESOURCES -LEO	118.0	2	1	80	O
EO-59 EARTH RESOURCES - SYNCH	66.9	6	-	110	O
EO-62 EARTH RESOURCES - FOREIGN	66.9	6	-	110	O
OP-08 GLOBAL MONITORING	214.0	-	2	140	O

* Specified in NASA data sheets

** O - Sun-oriented; U - Unoriented

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 7-4. Standardized Subsystem Modules - Electrical Power System

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY (1)	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
EPS-1A	Battery	A	6 AH Battery	2	7.2	14	2700	3.5	.847	21.51	.991	
		B	Charge Controller	2	4.5	9	100					
			Cables	AR		3						
			Connectors	AR		2						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			50						
EPS-1B	Battery	A	6 AH Battery	4	7.2	29	2700	5.0	.623	10.65	.993	
		B	Charge Controller	4	4.5	18	100					
			Cables	AR		5						
			Connectors	AR		4						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			78						
EPS-1C	Battery	A	6 AH Battery	6	7.2	43	2700	3.5	.608	7.10	.991	
		B	Charge Controller	6	4.5	27	100					
			Cables	AR		8						
			Connectors	AR		5						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			105						

Notes: (1) AR = As required

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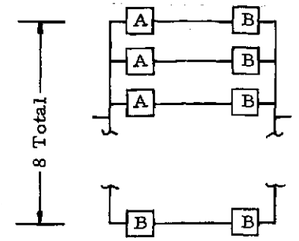
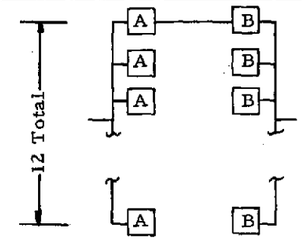
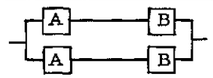
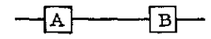
Table 7-4. Standardized Subsystem Modules - Electrical Power System (Continued)

MODULE CODE	MODULE NAME	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	FAILURE RATE (10^{-9}/hr)	MODULE DESIGN LIFE (yrs)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (yrs)	WEIBULL PARAMETERS β	BLOCK DIAGRAM
EPS-1D	Battery	A	6 AH Battery	8	7.2	58	2700	5.0	.388	5.30	.993	
		B	Charge Controller	8	4.5	36	100					
			Cables	AR		11						
			Connectors	AR		7						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			134						
EPS-1E	Battery	A	6 AH Battery	12	7.2	86	2700	3.0	.426	3.53	.991	
		B	Charge Controller	12	4.5	54	100					
			Cables	AR		16						
			Connectors	AR		11						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			189						
EPS-2	Battery	A	50 AH Battery	2	47.6	95	2700	4.5	.808	21.43	.992	
		B	Charge Controller	2	4.5	9	100					
			Cables	AR		3						
			Connectors	AR		2						
			Structure	AR		17						
			Environmental Protection System	AR		5						
			TOTAL			131						
EPS-3	Solar Array Dr.	A	Motor	1	9.0	9	500	5.0	.978	228.31	1.0	
		B	Engage Mechanism	1	14.5	15						
			Cables			3						
			Connectors			2						
			Structure			17						
			Environmental Protection System			5						
			TOTAL			51						

8. REPLACEABLE MISSION EQUIPMENT

As a baseline for this study, it was decided to allocate one mission equipment sensor per module. The reasoning for this decision was: (1) it seemed desirable to have the flexibility of incorporating sensor block changes on orbit as improvements in the various sensors become available, (2) the reliability of the mission equipment is historically lower than that of housekeeping type subsystems; hence, the failure and subsequent replacement of an individual sensor would not necessitate the replacement of other sensors along with it, and (3) some sensors require active cooling such as solid cryogens and, therefore, requiring replacement of expendables on a regular basis. This should be accomplished without having to replace other healthy sensors. The obvious disadvantage to the concept of one sensor per module is that it results in additional weight since each module requires a fixed weight of baseplate and support structure. Most likely, the optimum number of sensors per module will be a function of the particular mission, service lifetimes and reliabilities, and the need for a specific spacecraft to reduce the total spacecraft weight. Operation of the space-servicing simulation program using the input data generated here should indicate the sensitivity of operational costs to this ground rule and serve possibly to point out a different line of reasoning.

A review of the mission equipment listed in References 3 and 4 for each of the candidate spacecraft resulted in their assignment to either space replaceable mission equipment modules as shown in Table 8-1 or to the nonreplaceable part of the satellite as described in Section 9. A weight breakdown at the component level is provided for each module.

In general, very little is known about the mission equipment since the mission model is a projection of expected activity in the future rather than firm program commitments. Consequently, the only available avenue to arrive at reliability and lifetime characteristics is through the use of engineering judgment of qualified personnel. This was performed by contacting qualified experts, familiar with both DOD and NASA programs, who could provide an objective assessment of the type of equipment that would, in general, be required for future missions. Three areas were

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Table 8-1. Space Replaceable Mission Equipment Modules

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY (1)	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
AST-1B	Cosmic Background Explorer	AST-1B-1		IR Telescope	1			3	4	0.5	.560	.86	1.0
				Detectors	1	104.6	104.6						
				Liquid Helium Dewar	1								
				Aspect Camera	1								
				Baseplate	1	10.5	10.5						
				Mechanism	1	4.5	4.5						
				Environmental Control	AR	26.5	26.5						
				Electrical Connectors	AR	2.3	2.3						
				Elec. Dist. & Power Conditioning	AR	11.0	11.0						
				TOTAL			159.4						
AST-1C	Advanced Radio Astron. Explorer	AST-1C-1		Variable Length Crossed Dipole up to 225 M (1 MH_z to 10 MH_z)	1			2	2	5	.700	14.02	1.0
				Radio Receiver/Spectrum Analyzer Assy, 1 to 10 MH_z	2		116.0						
				Aspect Sensor	1								
				Baseline Correlation Radio Trans.	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			170.8						
AST-3	Solar Physics Mission	AST-3-1		Solar X-ray Polarimeter	1		100.0	1	2	2	.94	32.32	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			154.8						
		AST-3-2		X-ray/EUV Spectroheliograph	1		34.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		AST-3-3		White Light/XUV Coronagraph	1		50.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			104.8						
		AST-3-4		X-ray/Gamma Ray Spectrometer	1		11.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			65.8						

Notes: (1) AR = As required

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

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SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		AST-3-5		X-ray Spectrometer	1		18.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			72.8						
		AST-3-6		UV Spectrometer/Spectrohelio.	1		54.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			108.8						
		AST-3-7		X-ray Spectroheliograph	1		39.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			93.8						
		AST-3-8		H-Alpha/X-ray Spectrohelio.	1		54.0	1	4	2	.840	11.47	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			108.8						
		AST-3-9		X-ray Telescope	1		68.0	2	1	2	.720	6.09	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			122.8						
AST-9A	Focusing X-ray Telescope - 1.2 M	AST-9A-1		Field Monitor Camera & Elect.	1		23.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.8						
		AST-9A-2		Guide Star Trackers & Electronics	2		18.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			72.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		AST-9A-3		Position Sensing Prop. Counter & Elec.	1		34.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		AST-9A-4		X-ray Image Det./Int. & Elec.	1		59.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			113.8						
		AST-9A-5		Crystal Spectrometer & Elec.	1		43.5	2	3	2	.680	5.19	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			98.3						
		AST-9A-6		Max. Sen. Det.& Cryo. & Elec	1		41.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			95.8						
		AST-9A-7		Lithium Hydride Polar. & Elec.	1		40.0	.1	4	2	.840	11.47	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			94.8						
AST-9B	Focusing X-ray Telescope - 3.0M	AST-9B-1		Field Monitor Camera & Elect.	1		23.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.8						
				Guide Star Trackers & Elec.	2		18.0	2	2	2	.700	5.61	
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			72.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

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SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		AST-9B-3		Position Sensing Prop. Counter & Elec.	1		34.0	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		AST-9B-4		X-ray Image Det. /Int. & Elec.	1		59.0	3	4	2	.560	3.45	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			113.8						
		AST-9B-5		Crystal Spectrometer & Elec.	1		43.5	2	3	2	.680	5.19	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			98.3						
		AST-9B-6		Max. Sen. Det. & Cryo. & Elec.	1		41.0	3	4	1	.560	1.72	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			95.8						
		AST-9B-7		Lithium Hydride Polar. & Elec.	1		40.0	1	4	2	.840	11.47	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Control	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			94.8						
PHY-1A	Small High Energy Observatory	PHY-1A-1		Proportional Counter Modules	2			2	3	2	.680	5.19	1.0
				Aspect Sensor	1		104.3						
				Fe^{55} Calibration Source	1								
				Am^{241} Calibration Source	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			159.1						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
PHY-1B	Upper Atmosphere Explorer	PHY-1B-1		Electron Analyzer	1		5.0	2	2	2	.700	5.61	1.0
				Current Collector/Analyzer	1		5.0						
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			64.8						
		PHY-1B-2		VLF Receiver & Antenna	1		22.0	2	3	2	.680	5.19	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			76.8						
		PHY-1B-3		Mass Spectrometer	1		15.0	3	2	2	.600	3.92	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			69.8						
		PHY-1B-4		UV Detector	1		4.0	3	2	2	.600	3.92	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			58.8						
		PHY-1B-5		Electric Filed Detector	1		10.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			64.8						
		PHY-1B-6		Magnetometer	2		6.0	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			60.8						
		PHY-1B-7		Drag Analyzer	1		12.0	2	1	2	.720	6.09	1.0
				Pressure Gage	1		2.0						
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	1		26.5						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

FOLDOUT FRAME 2

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		PHY-1B-7 (Cont)		Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			68.8						
		PHY-1B-8		Ion Spectrometer	1		12.0	3	2	2	.600	3.92	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			66.8						
PHY-1C	Medium Altitude Explorer	PHY-1C-1		Solid State Detector	1		30.3	2	1	2	.720	6.09	1.0
				Ion Current Monitor	1								
				Electron Current Monitor	1								
				Photomultiplier Tube	1								
				Scintillation Counter	1								
				Charge Collector	1								
				Electric Field Detector	1								
				Magnetometer	2								
				Langmuir Probe	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			85.1						
		PHY-1C-2		Mass Spectrometer	1		22.3	3	2	2	.600	3.92	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.1						
		PHY-1C-3		Drag Analyzer	1		9.0	2	1	2	.720	6.09	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			63.8						
PHY-2A	Gravity Probe B_2	PHY-2A-1		Precession Gyroscopes	2		90.8	3	5	0.5	.540	.81	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			145.6						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		PHY-2A-2		Star Tracker & Star Telescope	1		64.7	2	2	2	.700	5.61	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			119.5						
		PHY-2A-3		IR Earth Horizon Sensors	2		22.6	1	1	2	.990	199.00	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.4						
EO-3A	Earth Observatory Satellite	EO-3A-1		Thematic Mapper	1		272.0	3	4	1	.560	1.72	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			326.8						
		EO-3A-2		High Res. Imager (multispectral scan)	1		272.2	3	4	1	.560	1.72	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			326.8						
		EO-3A-3		Data Collection System	1		34.0	1	2	3	.940	48.48	
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		EO-3A-4		Radar Imager Antenna	1		227.0	2	3	3	.680	7.78	1.0
				Radar Imager Electronics	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			281.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		EO-3A-5		Coastal Zone Color Scanner	1		27.2	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			82.0						
		EO-3A-6		Lower Atmospheric Temp. & Composition Experiment	1		76.3	2	4	2	.660	4.81	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			131.1						
		EO-3A-7		Measurement of Air Pollution	1		11.3	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			66.1						
		EO-3A-8		Stratospheric Aerosol Measu.	1		7.3	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			62.1						
EO-4A	Synchronous Earth Observatory Satellite	EO-4A-1		Sensor Assembly	1		145.0	4	4	3	.245	2.13	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			199.8						
		EO-4A-2		Data Collection System	1		34.0	1	2	3	.940	48.48	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
EO-6	TIROS	EO-6-1		Adv. Very High Res. Radiometer	2		27.0	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			81.8						

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FOLDOUT FRAME 2

Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		EO-6-2		Adv. TIROS Oper. Vert. Sounder	2		45.0	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			99.8						
		EO-6-3		Scanning Multichannel Microwave Radio. Elect. & Scanner	1		52.0	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			106.8						
		EO-6-4		Microwave Radio./Scatterometer Electronics & Antenna	2		56.7	3	2	3	.600	5.87	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			111.5						
		EO-6-5		Cloud Physics Radiometer	2		27.2	2	2	3	.700	8.41	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			82.0						
		EO-6-6		Space Environmental Monitor	2		22.8	1	2	3	.940	48.48	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			77.6						
		EO-6-7		Data Collection System	2		45.0	1	2	3	.940	48.48	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			99.8						
EO-7	Advanced Synchronous Meteorological Satellite	EO-7-1		Imaging Radiometer (visible)	1		45.0	3	2	5	.600	9.79	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			99.8						

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FOLDOUT FRAME 2

Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		EO-7-2		Imaging Radi. (IR) (w/passive cooling)	1		34.0	2	2	5	.700	14.02	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
		EO-7-3		Magnetometer Sensor Assy	1		9.2	2	2	7	.700	19.63	1.0
				X-ray Telescope & Detector Assy	1								
				Energetic Particle Detector Assy	1								
				Space Envir. Monitoring Data Handling Assy	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			64.0						
		EO-7-4		Data Collection System	1		4.5	1	2	7	.940	113.13	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			59.3						
		EO-7-5		Vert. Temp. /Moist. Profiler (w/Cooling)	1		27.2	2	3	5	.680	12.96	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			82.0						
EOP-3	SEASAT B	EOP-3-1		Altimeter (K-band pulsed)	1		45.0	2	3	5	.680	12.96	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			99.8						
		EOP-3-2		Scatterometer	1		91.0	2	3	4	.680	10.37	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			145.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		EOP-3-3		IR Scanner	1		43.0	2	2	5	.700	14.02	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			97.8						
		EOP-3-4		Coherent Radar Experiment (155 & 1215 MHz)	1		73.0	3	4	3	.560	5.17	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			127.8						
		EOP-3-5		Transponder C-Band	1		48.0	2	1	7	.720	21.31	1.0
				Transponder S C Band	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			102.8						
EOP-4	GEOPAUSE	EOP-4-1		Tracking Link Precision Transmitter/Receiver/ Transponder System	1		90.6	2	3	5	.680	12.96	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			145.4						
		EOP-4-2		Very Long Baseline Interferometer	1		30.0	3	5	2	.540	3.25	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			84.8						
EOP-7	GRAVSAT	EOP-7-1		Accelerometer Package (3 axis)	1		226.0	4	5	3	.150	1.58	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			280.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
		EOP-7-2		Precision Transmitter/ Receiver/Transponder	1		90.6	2	2	3	.700	8.41	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			145.4						
NND-1	International Communication Sat	NND-1-1		4/6 GH_z Trans., 24 Channel	1		116.2	2	2	10	.700	28.04	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			171.0						
		NND-1-2		12/13 GH_z Trans., 24 Channel	1		193.7	3	4	10	.560	17.25	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			248.7						
NND-2A	U.S. Domestic Satellite	NND-2A-1		Chan. Microwave Repeater - 12 Channel	1		34.0	2	2	10	.700	28.04	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
NND-2B	U.S. Domestic Satellite	NND-2B-1		4/6 GH_z Trans. 24 Channel	1		116.2	2	2	10	.700	28.04	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			171.0						
		NND-2B-2		12/13 GH_z Trans., 24 Channel	1		193.7	3	4	10	.560	17.25	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			248.7						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NND-2D	U.S. Domestic Satellite	NND-2D-1		LDR/TDRS Trans. & Freq. S Source	1		25.3	3	4	7	.560	12.07	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			80.1						
		NND-2D-2		MDR/HDR/TDR Transceiver Freq. Source & Ku Band	1 ea		76.3	4	5	7	.150	3.69	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			131.1						
		NND-2D-3		GS/TDRS Trans., & Freq. Source	1 ea		14.1	3	5	7	.540	11.36	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elect. Dist. & Power Conditioning	AR		11.0						
				TOTAL			68.9						
NND-3	Disaster Warning Satellite	NND-3-1		Communication Transceiver	1		109.0			7	.680	18.15	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			163.8						
NND-4	Traffic Management Satellite	NND-4-1		L Band Transponder	1		49.0	4	3	10	.340	9.27	1.0
				L Band Earth Coverage Antenna	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			103.8						
NND-5	Foreign Communications Satellite	NND-5-1		6 GHz Receiver	2			2	3	10	.680	25.93	1.0
				Input & Output Filter Assy	1		27.0						
				Multiplexer Assy	1								
				TWT's & Power Supply Assy	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			81.8						

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FOLDOUT FRAME 2

Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NND-6	Communication R&D Prototype	NND-6-1		1.25 Kw TWT Assy	2		12.2	4	5	5	.150	2.64	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			67.0						
		NND-6-2		1.25 Kw KLYSTRON Assy	1		13.6	4	5	5	.150	2.64	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			68.4						
		NND-6-3		200 W TWT Assy	1		9.1	4	5	5	.150	2.64	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			63.9						
		NND-6-4		Comm Driver & Receiver	1		17.2	4	5	5	.150	2.64	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			72.0						
		NND-6-5		Experiment Peculiar Equipment	1		99.8	4	5	5	.150	2.64	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			154.6						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NND-8	Environmental Monitoring Satellite	NND-8-1,2		Ozone-Sun Polarimeter	2		30.4	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			85.2						
		NND-8-3,4		Limb Atmosphere Composition Radiometer			63.0	3	3	1	.580	1.84	1.0
				Radiometer	2								
				Cryogen Cooler	2								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			117.8						
		NND-8-5,6		Air Pollution Sensor				2	4	1	.660	2.41	1.0
				Optical Head	2		30.0						
				Electronics	2								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			84.8						
		NND-8-7,8		High Speed Interferometer	2		25.0	2	3	1	.680	2.59	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			79.8						
		NND-8-9,10		Ocean Scanning Spectrophotometer	2		25.0	2	2	3	.700	8.41	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			79.8						
		NND-8-11,12		Coastal Zone Color Scanner	2		27.0	3	3	3	.580	5.51	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			81.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NND-11	Low Earth Orbit Earth Resources Satellite	NND-11-1		Multispectral (5-7 bands) Scanner	1		72.0	2	2	3	.700	8.41	1.0
				MSS Multiplexer	1								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			126.8						
		NND-11-2		Panchromatic Return Beam Vidicon Camera	1		35.0	2	1	3	.720	9.13	1.0
				RBV Electronics	1								
				RBV Control	1								
				Data Collection System Receiver	2								
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			89.8						
NND-12	Geosynchronous Earth Resources Satellite	NND-12-1		Sensor Assy - Visible & IR	1		145.0	4	4	3	.245	2.13	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			199.8						
		NND-12-2		Data Collection System 5000 BPS	1		34.0	1	2	3	.940	48.48	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						
NND-13	Foreign Synchronous Earth Observations Satellite	NND-13-1		Sensor Assy - Visible & IR	1		145.0	4	4	3	.245	2.13	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			199.8						
		NND-13-2		Data Collection System 5000 BPS	1		34.0	1	2	3	.940	48.48	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			88.8						

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Table 8-1. Space Replaceable Mission Equipment Modules (Continued)

SATELLITE CODE	SATELLITE NAME	MODULE CODE	ITEM	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NND-14	Global Earth and Ocean Monitor System	NND-14-1		Radar Altimeter	1		45.0	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			99.8						
		NND-14-2		Multispectral IR Radiometer	1		43.0	2	2	3	.700	8.41	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connectors	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			97.8						
		NND-14-3		Very Long Baseline Interfer.	1		30.0	2	2	3	.700	8.41	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			84.8						
		NND-14-4 & 5		Passive Microwave Radiometer	1		52.0	2	3	3	.680	7.78	1.0
				Baseplate	1		10.5						
				Mechanism	1		4.5						
				Environmental Control	AR		26.5						
				Electrical Connector	AR		2.3						
				Elec. Dist. & Power Conditioning	AR		11.0						
				TOTAL			106.8						

considered, relative to existing equipments and the potential for scientific breakthroughs that could alter future patterns of development. The three areas were earth observation sensors, communications equipment, and scientific experiments, such as solar observations.

For example, the mission reliability of communication satellites has been improving significantly over the past few years. Earlier designs could be expected to have a reliability as low as 0.3 at a design life of five years. Current projections indicate that a reliability at a design life of 10 years, as high as 0.7 is achievable. Although improvements can be expected in the future, it is difficult to imagine any higher values considering the increased complexity which inherently accompanies new concepts.

Earth observations present a somewhat different picture. The current state-of-the-art on multi-spectral scanners is being pushed very hard but inherent limitations exist in optics and photomultiplier sensors. This is a judgment situation, but at this time, it is difficult to expect this type of equipment to survive more than two years of operation. If component failure does not occur, it is still very possible for the equipment to degrade in resolution to the point that spectral regions of interest can no longer be isolated. A further complication exists in that many of these type of sensors require active cooling to maintain the sensor temperature requirements. A tradeoff performed previously (ref. 1) indicates that for these purposes a solid cryogen supply, such as argon, would be required. A two-year supply, weighing approximately 26 kg. would probably be sufficient. In summary, earth observation equipments in general tend to be highly complex and will continue to require development in the future.

Scientific experiments also are inherently developmental and complex, although exceptions can be found, such as a solar X-Ray polarimeter. It can be expected to be very reliable but only has a lifetime requirement of approximately two years. In general, however, much of this equipment is highly complex and lacks a long history of development, whereby a

high reliability might be achieved.

Therefore, the following procedure was employed to arrive at reasonable estimates of mission equipment reliabilities and lifetimes. Except where the lifetime of a particular instrument was in question, the specified satellite design life of reference 3 was used for the mission equipment. For other equipments, a life of one, two, or possibly three years was assumed, based upon the judgment of experienced personnel at the Aerospace Corporation. The mission equipment design life values assumed are shown in Table 8-1.

Reliability estimates are considerably more obscure. Two parameters are considered to be the principal factors which influence equipment reliability; the design complexity and, the estimated state of development. As the complexity of a given design increases it is expected that the reliability will decrease. The form of this relationship is judgmental, based upon prior experience. Also, as the state of development becomes less and less firm, it is expected that the reliability will decrease. This leads to a reliability relationship as shown in Figure 8-1.

A range of reliability values has been estimated for each level of complexity shown. For a simple design, it is expected a range of values from approximately 0.8 to 0.99 is reasonable for the applications of interest. As the higher reliability items increase in complexity, it is anticipated that the drop in reliability will be more pronounced, since there is a rational limit to the techniques which can be employed to maintain a high reliability. Further increases in complexity are assumed to decrease the reliability in somewhat of a linear manner.

On the other side of the spectrum, such as a highly developmental item of equipment, it is anticipated that reliability will be relatively low, decreasing somewhat with increased complexity. That is, until the design pushes beyond the state-of-the-art, in which case it should be expected that a drastic drop in reliability will occur. This provides a band, or region of reliability values which should encompass the majority of the mission equipment anticipated for future programs.

The remaining step is then to divide the range of reliability

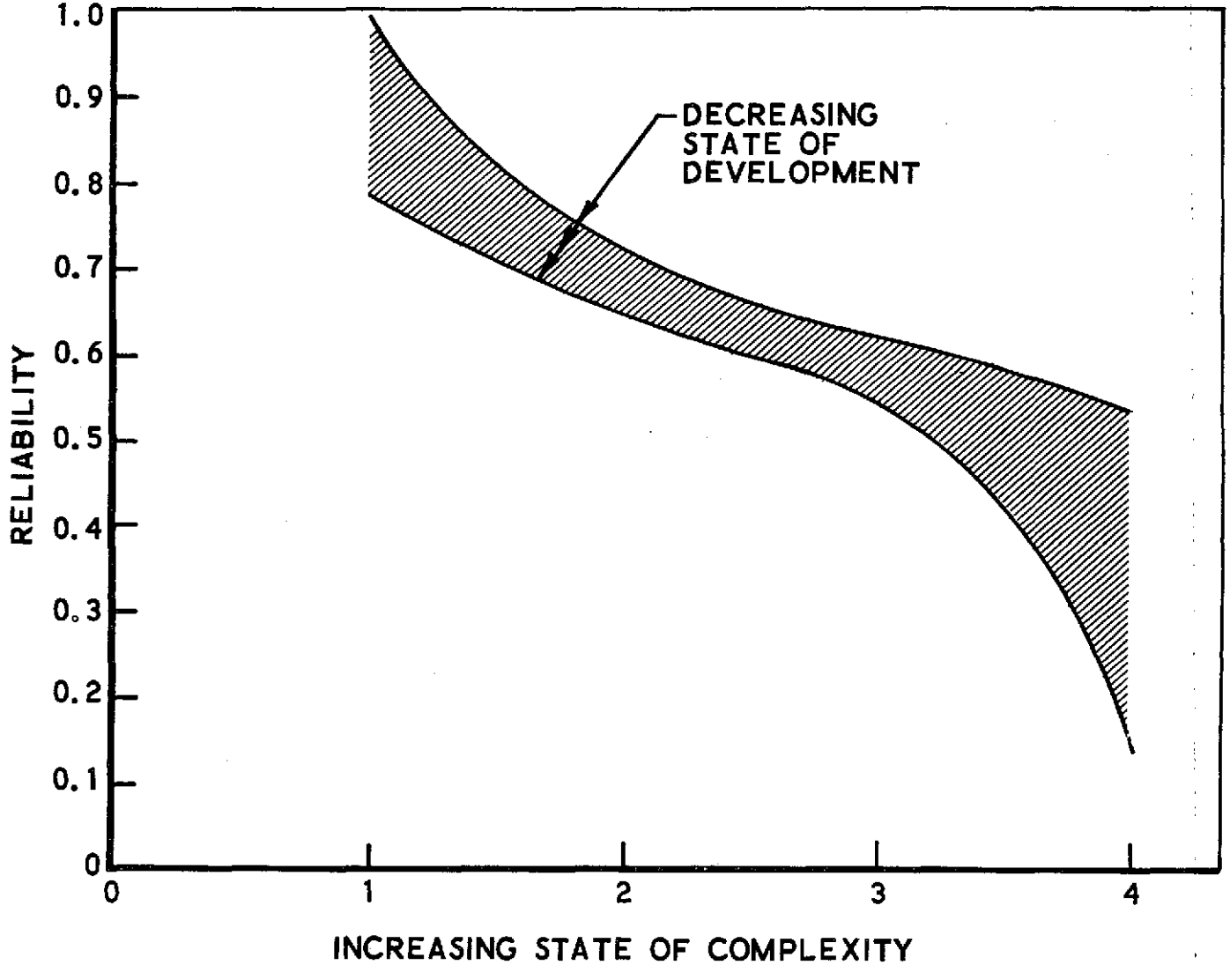

Figure 8-1. Mission Equipment Reliability Factors

at each step in complexity into five increments, representing five states of development. Since the curve is illustrative only (being discontinuous between the complexity states), the values selected are tabulated in Table 8-2. The reliability associated with each combination of the two variable states is provided. This provides an easy reference for future applications wherein different states may be preferred.

Tabulating the results also tends to discourage extrapolation of these gross estimates to more refined interpretations. These values are based upon sound judgment by experienced personnel and should be considered in light of the broad spectrum of application. Any given individual design could easily lie outside the band provided. However, it is intended, through this approach, to stimulate further work in defining mission equipment and placing proper emphasis on design reliability. Until such time as actual design experience is available to improve this table (Table 8-2), it is suggested that the values provided be employed.

A tradeoff between the operational cost of servicing a satellite and the cost of increasing a sensor's reliability should be made for each program application. This will be performed, to a small measure, with the simulation program currently in development at The Aerospace Corporation under the contract. In this way the reliability of each sensor can be varied to determine the sensitivity of operational costs. As a result, the reliability of a given sensor may not have to be as high as would be required if flown on an expendable spacecraft.

Table 8-2. Mission Equipment Reliability Assignment Factors

COMBINED FACTOR *	RELIABILITY	COMBINED FACTOR	RELIABILITY	COMBINED FACTOR	RELIABILITY	COMBINED FACTOR	RELIABILITY
1, 1	0.990	2, 1	0.720	3, 1	0.620	4, 1	0.530
1, 2	0.940	2, 2	0.700	3, 2	0.600	4, 2	0.435
1, 3	0.890	2, 3	0.680	3, 3	0.580	4, 3	0.340
1, 4	0.840	2, 4	0.660	3, 4	0.560	4, 4	0.245
1, 5	0.790	2, 5	0.640	3, 5	0.540	4, 5	0.150

COMPLEXITY FACTORS

1 = LOW
2 = MEDIUM
3 = HIGH
4 = VERY HIGH

STATE OF DEVELOPMENT FACTOR

1 = OFF-THE-SHELF
2 = ADAPTATION OF EXISTING EQUIPMENT
3 = ALL COMPONENTS AVAILABLE
4 = SOME COMPONENTS REQUIRE DEVELOPMENT
5 = EXTREMELY DEVELOPMENTAL

* ORDER OF NOTATION =

COMBINED FACTOR = COMPLEXITY, STATE OF DEVELOPMENT

9. NONREPLACEABLE UNITS

Table 9-1 presents the items of nonreplaceable equipment for each of the candidate space-serviceable spacecraft. All nonreplaceable equipment for a given spacecraft is lumped together and identified as a single module with a design life, weight and reliability. For example, the nonreplaceable equipment for AST-1B, the Cosmic Background Explorer, is identified as NAST-1B and its nonreplaceable equipment, which is typical for all the spacecraft, is listed in Column 3 of Table 9-1. A brief discussion of each of these common items follows:

a. Standardized Structure. The basic satellite cylindrical structure is the same for each satellite and is therefore considered to be standardized for cost estimating purposes. The structure can easily be designed to last the lifetime of the spacecraft.

b. Solar Arrays. The solar arrays are assumed to be rollout arrays with standardized 8-ft long drums. The array area required for each spacecraft is obtained by varying the length of the array. The arrays are sized to account for degradation over the spacecraft life and are therefore nonreplaceable. A replaceable solar array drive unit for oriented arrays is discussed later.

c. Electrical Distribution. This includes the hard wiring on the nonreplaceable structure.

d. Thermal Protection. This includes passive thermal protection needed to maintain temperature balance around the structure, but not the thermal protection required for each individual module.

In addition to these standard items of nonreplaceable equipment, some of the spacecraft have items of nonreplaceable mission equipment. This kind of mission equipment can be divided into two categories: (1) telescopes, and (2) antennas. Five satellites (AST-9A, AST-9B, EO-4A, NND-12, and NND-13) have telescopes which are too large to replace. It was decided that the optics could be designed for the satellite lifetime

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Table 9-1. Nonreplaceable Modules

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NAST-1B	Cosmic Background Explorer NRU	Standardized Structure (13)(1)	1	167	167	N/A	N/A	5	.998	2283.10	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (37 ft^2)	2	9.6	19	N/A	N/A				
		Booms	2	2.8	6	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			296						
NAST-1C	Advanced Radio Astronomy Explorer NRU	Standardized Structure (13)	1	167	167	N/A	N/A	10	.996	2283.10	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (37 ft^2)	2	9.6	19	N/A	N/A				
		Booms	2	2.8	6	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			296						
NAST-3	Solar Physics Mission NRU	Standardized Structure (21)	1	227	227	N/A	N/A	5	.998	2075.60	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (63 ft^2)	2	16.3	33	N/A	N/A				
		Booms	2	3.6	7	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			371						
NAST-9A	Focusing X-ray Telescope 1.2M NRU	Stand. Struc. & Tele. Suppts. (24)	1	427	427	N/A	N/A	5	.560	8.61	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (85 ft^2)	2	22	44	N/A	N/A				
		Booms	2	4.1	8	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		X-ray Telescope Assy	1	3670	3670	3	4				
		Aspect Sensor Telescope	1	47.6	48	3	4				
		TOTAL			4301						
NAST-9B	Focusing X-ray Telescope 3.0M NRU	Stand. Struc. & Tele. Suppts. (24)	1	427	427	N/A	N/A	5	.150	2.64	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (85 ft^2)	2	2	44	N/A	N/A				
		Booms	2	4.1	8	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		X-ray Telescope Assy	1	6820	6820	4	5				
		Aspect Sensor Telescope	1	47.6	48	3	4				
		TOTAL			7451						
NPHY-1A	Small High Energy NRU	Standardized Structure (13)	1	167	167	N/A	N/A	5	.998	2283.10	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (37 ft^2)	2	9.6	19	N/A	N/A				
		Booms	2	2.8	6	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			296						

Notes: (1) Number in parenthesis indicates number of SRUs

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Table 9-1. Nonreplaceable Modules (Continued)

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NPHY-1B	Upper Atmosphere Explorer NRU	Standardized Structure (17)	1	227	227	N/A	N/A	5	.997	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (50 ft^2)	2	13	26	N/A	N/A				
		Booms	2	3.3	7	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			364						
NPHY-1C	Medium Altitude Explorer NRU	Standardized Structure (13)	1	167	167	N/A	N/A	5	.997	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (50 ft^2)	2	13	26	N/A	N/A				
		Booms	2	3.3	7	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			304						
NPHY-2A	Gravity Probe B_2 NRU	Standardized Structure (14)	1	167	167	N/A	N/A	5	.997	1756.20	1.0
		Standardized Solar Array Drum	1	33	33	N/A	N/A				
		Solar Cell Array (55 ft^2)	1	14.2	14	N/A	N/A				
		Booms	1	3.4	3	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			255						
NEO-3A	Earth Observatory Satellite NRU	Standardized Structure (22)	1	227	227	N/A	N/A	5	.997	1630.80	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (95 ft^2)	2	24.6	49	N/A	N/A				
		Booms	2	44	9	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			389						
NEO-4A	Synchronous Earth Observatory Satellite NRU	Standardized Structure (14)	1	167	167	N/A	N/A	5	.580	9.17	1.0
		Standardized Solar Array Drum	1	33	33	N/A	N/A				
		Solar Cell Array (60 ft^2)	1	15.5	14	N/A	N/A				
		Boom	1	3.5	4	N/A	N/A				
		Telescope, Cassegrain	1	519	519	3	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			777						
NEO-6	TIROS NRU	Standardized Structure (22)	1	227	227	N/A	N/A	5	.997	1630.80	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (100 ft^2)	2	25.9	52	N/A	N/A				
		Booms	2	4.5	9	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			392						

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Table 9-1. Nonreplaceable Modules (Continued)

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NEO-7	Advanced Synchronous Met Satellite NRU	Standardized Structure (18)	1	227	227	N/A	N/A	10	.994	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (50 ft^2)	2	13	26	N/A	N/A				
		Booms	2	3.3	7	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			364						
NEOP-3	SeaSat B NRU	Standardized Structure (18)	1	227	227	N/A	N/A	10	.987	691.80	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (45 ft^2)	2	11.7	23	N/A	N/A				
		Booms	2	3.1	6	N/A	N/A				
		Retroreflector Array	1	20	20	1	1				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			380						
NEOP-4	Geopause NRU	Standardized Structure (11)	1	167	167	N/A	N/A	5	.997	1426.90	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (80 ft^2)	2	20.7	41	N/A	N/A				
		Booms	2	4	8	N/A	N/A				
		Retroreflector Array	1	18	18	1	1				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			338						
NEOP-7	Gravsat NRU	Standardized Structure (12)	1	167	167	N/A	N/A	5	.997	1630.80	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (80 ft^2)	2	20.7	41	N/A	N/A				
		Booms	2	4	8	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			320						
NNND-1	International Comm Satellite NRU	Standardized Structure (19)	1	227	227	N/A	N/A	10	.700	28.014	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (375 ft^2)	2	97.1	194	N/A	N/A				
		Booms	2	11.4	23	N/A	N/A				
		6 GHz Receive Antenna	2	32	64	2	2				
		4 GHz Transmit Antenna	2	50	100	2	2				
		13 GHz Receive Antenna	3	27	81	3	3				
		13 GHz Receive Antenna Feed	1	9	9	3	3				
		12 GHz Transmit Antenna	3	4.8	14	3	3				
		12 GHz Transmit Antenna	1	11	11	3	3				
		12 GHz Beacon Antenna Feed	1	2	2	3	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			829						

FOLDOUT FRAME

FOLDOUT FRAME

Table 9-1. Nonreplaceable Modules (Continued)

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NNND-2A	U.S. Domestic Sat NRU	Standardized Structure (10)	1	167	167	N/A	N/A	10	.994	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (62 ft^2)	2	16.1	32	N/A	N/A				
		Booms	2	3.6	7	N/A	N/A				
		Parabolic Reflector Antenna	1	8	8	3	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			318						
NNND-2B	U.S. Domestic Sat NRU	Standardized Structure (19)	1	227	227	N/A	N/A	10	.700	28.01	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (375 ft^2)	2	97.1	194	N/A	N/A				
		Booms	2	11.4	23	N/A	N/A				
		6 GHz Receive Antenna	2	32	64	2	2				
		4 GHz Transmit Antenna	2	50	100	2	2				
		13 GHz Receive Antenna	3	27	81	2	2				
		13 GHz Receive Antenna	1	9	9	2	2				
		12 GHz Transmit Antenna	3	4.8	14	2	2				
		12 GHz Transmit Antenna	1	11	11	2	2				
		12 GHz Beacon Antenna	1	1	2	2	2				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			829						
NNND-2B	U.S. Domestic Sat NRU	Standardized Structure (13)	1	167	167	N/A	N/A	10	.561	17.30	1.0
		Standardized Solar Array Drum	1	33	33	N/A	N/A				
		Solar Cell Array (45 ft^2)	1	12	12	N/A	N/A				
		Boom	1	3	3	N/A	N/A				
		K&S Band 3.8M Furlable Ant	2	36.6	73	3	4				
		TDRS/GS Antenna, K Band	1	11	11	3	3				
		TT&C Omni Antenna	4	2	8	1	2				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			345						
NNND-3	Disaster Warning Satellite NRU	Standardized Structure (12)	1	167	167	N/A	N/A	10	.700	28.01	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (350 ft^2)	2	90.7	181	N/A	N/A				
		Boom	2	10.8	22	N/A	N/A				
		Parabolic Antenna	1	59	59	2	2				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			533						

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 9-1. Nonreplaceable Modules (Continued)

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NNND-4	Traffic Management Satellite NRU	Standardized Structure (12)	1	167	167	N/A	N/A	10	.677	25.62	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (75 ft^2)	2	19.4	39	N/A	N/A				
		Booms	2	3.9	8	N/A	N/A				
		Multifeed L Band Antenna	1	17	17	2	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			335						
NNND-5	Foreign Comm Satellite NRU	Standardized Structure (12)	1	167	167	N/A	N/A	10	.701	28.11	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (35 ft^2)	2	9.1	18	N/A	N/A				
		Booms	2	2.9	5	N/A	N/A				
		2.5 GHz Antenna	1	5	5	2	2				
		4 MHz Antenna	1	3	3	2	2				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			302						
NNND-6	Communication R&D Prototype NRU	Standardized Structure (27)	1	227	227	N/A	N/A	5	.562	8.66	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (650 ft^2)	2	168.4	337	N/A	N/A				
		Booms	2	28.2	57	N/A	N/A				
		Antenna	1	27	27	3	4				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			752						
NNND-8	Environmental Monitoring Satellite NRU	Standardized Structure (28)	1	227	227	N/A	N/A	5	.998	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (40 ft^2)	2	10.4	21	N/A	N/A				
		Booms	2	3	6	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			358						
NNND-11	Low Earth Orbit Earth Resources Satellite NRU	Standardized Structure (16)	1	167	167	N/A	N/A	5	.998	1756.20	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (40 ft^2)	2	10.4	21	N/A	N/A				
		Booms	2	3	6	N/A	N/A				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			298						

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 9-1. Nonreplaceable Modules (Continued)

MODULE CODE	MODULE NAME	COMPONENT	QTY	WEIGHT (kg) ITEM	WEIGHT (kg) TOTAL	DESIGN PARAMETERS COMPLEXITY	DESIGN PARAMETERS STATE OF DEVELOPMENT	MODULE DESIGN LIFE (YRS)	MODULE RELIABILITY AT DESIGN LIFE	WEIBULL PARAMETERS α (YRS)	WEIBULL PARAMETERS β
NNND-12	Geosynchronous Earth Resources Satellite NRU	Standardized Structure (15)	1	167	167	N/A	N/A	5	.580	9.17	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (55 ft^2)	2	14	28	N/A	N/A				
		Booms	2	3.4	7	N/A	N/A				
		Telescope, Cassegrain 1.5M	1	519	519	3	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			825						
NNND-13	Foreign Synchronous Earth Observations Satellite NRU	Standardized Structure (14)	1	167	167	N/A	N/A	5	.580	9.17	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (55 ft^2)	2	14	28	N/A	N/A				
		Booms	2	3.4	7	N/A	N/A				
		Telescope, Cassegrain 1.5M	1	519	519	3	3				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			825						
NNND-14	Global Earth & Ocean Monitoring System NRU	Standardized Structure (18)	1	227	227	N/A	N/A	5	.997	1552.10	1.0
		Standardized Solar Array Drum	2	33	66	N/A	N/A				
		Solar Cell Array (70 ft^2)	2	18	36	N/A	N/A				
		Booms	2	3.8	8	N/A	N/A				
		Corner Reflector Array	1	20	20	1	1				
		Electrical Distribution	AR	11	11	N/A	N/A				
		Thermal Protection	AR	27	27	N/A	N/A				
		TOTAL			395						

and the detectors could be designed to be replaceable. Should it turn out that the reliabilities of these nonreplaceable optics cannot be raised to a sufficiently high level to justify their "nonreplaceable" status, then the entire spacecraft should be considered for ground refurbishment.

Seven satellites (NND-1, NND-2B, NND-2D, NND-3, NND-4, NND-5, NND-6) have antennas which either operate at a low enough frequency to justify the use of coaxial cable for their connection to electronics and therefore enable the use of an easily dematable connection, or are shared by a number of different modules. Should problems arise which make it desirable to have replaceable antennas, this could be accomplished but mechanical complexity would ensue and additional weight penalties would have to be paid.

The results of this assessment are shown in Table 9-1. The solar array in general dominates the reliability estimate since the majority of remaining elements are structure or wiring. It should be noted that the structural weight varies depending upon the number of SRUs to be installed. The applicable number of SRUs is identified in parenthesis after "standardized structure" in the component column. The number of SRUs is derived from Table 10-2 in the next section utilizing the design approach shown in Section 11.

10. DESIGN INTEGRATION

10.1 INTRODUCTION

The study effort was viewed primarily as one in which very little original design effort would be performed. In light of the considerable NASA documentation of payload descriptions (Refs. 3 and 4), the many contractor studies of space-servicing techniques (Refs. 11 through 21), contractor studies of subsystem modularization (Refs. 22 and 23), and the point designs developed at Aerospace for space-serviceable satellites (Refs. 1 and 24), the task was considered to be one of review, assimilation, and application of the referenced data, information and experience to suit the needs of the specific effort. The considerable experience gained through the in-house designs of both the Space Serviceable Defense Support Program (SSDSP) satellite and the Space Serviceable Earth Observatory Satellite (SSEOS) was relied upon to make many of the decisions required to achieve the study objectives, select a space-servicing concept, and define the replaceable and nonreplaceable equipment for those satellites.

It should be further stated that, in spite of the iterative nature of the study, the broad scope and limited manpower did not allow optimization of such things as modular and satellite weight, or the number of modules to be used. It was felt that if space servicing could be shown to be economically feasible, based on the results of a nonoptimized study, then follow-on efforts could be initiated to further improve the results.

10.2 DESIGN GROUND RULES

Earlier studies (Ref. 1) established the preliminary ground rules for the study. As the study progressed, minor changes were made. The revised ground rules are as follows:

a. Only automated (unmanned) earth orbiting payloads are considered for space servicing.

b. All satellites to be serviced are designed to be 3-axis stabilized.

c. The satellites are modularized into subsystem modules.

d. The launch system consists of the Space Shuttle and a selection of upper stages.

e. The Space Shuttle payload bay envelope is assumed but service unit and satellite are to provide their own rattle space.

f. The upper stage is reusable.

g. The Solar Electric Propulsion System (SEPS) may be used as part of the launch system for altitudes above 14, 800 km.

h. The servicing system is to be fully automatic and unmanned, and TV assistance, either from space or via a console on the ground, is not to be used in the servicing operation.

i. Teleoperator arms are not to be used for servicing.

j. The service unit will be hard docked to the satellite prior to initiation of the servicing operation. The docking operation shuts down all satellite electrical power.

k. The service unit provides storage volume for stowage of the subsystem modules removed from the serviced satellite.

l. The service unit to be used with the SEPS provides its own stabilization during docking with the SEPS.

m. The upper stage attitude control system controls the docked combination of the upper stage and satellite.

n. The diagnosis of which subsystem modules need to be replaced is made by telemetry prior to the servicing flight.

o. Conventional design techniques and 1973 technology are assumed.

p. Fluid connections across interfaces are to be avoided.

q. Conductive heat transfer across interfaces as a means of thermal control is to be avoided.

r. Physical interfaces are to be self-aligning with simple connect/ disconnect provisions.

s. On-orbit adjustment of subsystem modules or components within subsystem modules is to be minimized.

t. The docking method is to be as simple as possible and not to constrain module transfer.

u. Sensors or mission peculiar equipment are not included in the inventory of standardized subsystem modules.

v. Maximum use is made of existing space servicing and subsystem modularization study results. A minimum of original design effort shall be performed.

w. The number of motions required to exchange a subsystem module shall be held to a minimum.

x. Wherever practical, large components such as solar arrays, antennas, optics, etc., shall be designed for the total mission lifetime and therefore not be replaceable.

y. Data transmission to and from subsystem modules shall be accomplished via a data bus.

z. In general, each mission equipment module shall contain only one sensor.

10.3 SPACE-SERVICING TECHNIQUE SELECTION

It cannot be too strongly stressed that the design of a space-serviceable satellite should not be independent of the design of its servicing unit. The technique selected for exchange of subsystem modules to and from the spacecraft should not impose unnecessarily severe constraints on the design of the spacecraft and yet the technique selected should be as uncomplicated as possible so as to result in a simple, low weight, reliable service unit design. A primary object of the servicing unit design activity was to select a servicing concept which would result in the desirable features described above and would not cause some of the candidate space-servicing payloads selected from the mission model to be eliminated because of constraints imposed by the selected concept (for example, the elimination of payloads with appendages outside the module envelope). It is important to note that the servicing technique selected as a result of this portion of the study is not necessarily the optimum technique and is not the recommended one for all future space-servicing studies. Rather, it is merely one which appeared to impose the least constraints on the spectrum of satellite under consideration and does not cause the elimination of any of the candidate satellites. The selected

concept allowed the derivation of a servicing unit weight, satellite structure weight, and the replaceable module weight to be used as inputs to economic and performance studies. In addition, it provides a baseline concept against which the various sensors and other items of mission equipment may be judged for their adaptability to space servicing.

Due to the large number of servicing concepts advanced by various contractors, it was felt that it would be most advantageous for the purposes of this study to select one from those available rather than attempt an original design effort. Furthermore, the concept selected would be used as is to avoid consuming scarce resources on an optimization study.

Working within the framework of the ground rules established earlier, nine different contractor concepts were investigated. They were grouped into two different categories based on the type of motions required for the exchange of a module. These motions, shown in Figure 10-1, are: (1) index and translate or swing a module from the service unit to the spacecraft and (2) strictly linear motion.

The concepts investigated are listed on Table 10-1 and are shown in Figures 10-2 and 10-3.

Much of this portion of the study was based on the results of a similar study by McDonnell Douglas Astronautics Co. (MDAC). MDAC, under a NASA contract, investigated and evaluated a number of different space-servicing techniques. MDAC personnel transmitted the results of their study to Aerospace through informal discussions held at MDAC.

Table 10-1 lists the nine servicing concepts which were investigated and the parameters which are considered to be important in evaluating servicing units. A brief discussion of these parameters follows:

a. Weight. Light weight is desirable, because any weight savings results directly in increased module carrying capacity and/or capability to service more than one satellite.

b. Size. The payload diameter is constrained to 14.5 ft in order to fit in the Shuttle payload bay and provide adequate rattle space. Short length is desirable in order to have the flexibility

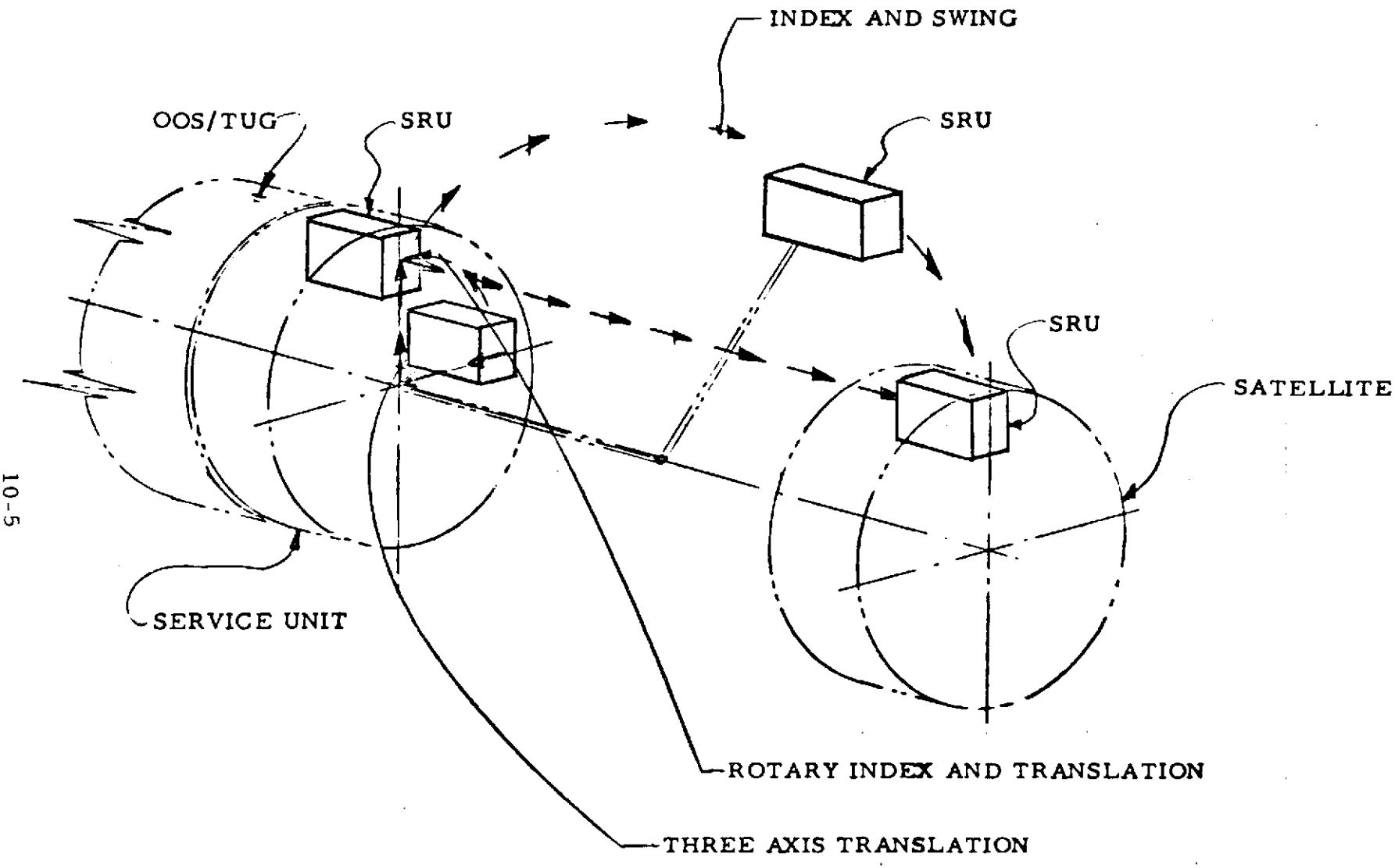


Figure 10-1. SRU Transfer Concepts

Table 10-1. Comparison of Automated Servicing Techniques

SERVICING CONCEPT	WEIGHT (LB)	SIZE	MOTIONS REQUIRED	MAX. NO. OF SRU'S	TV ASSISTANCE	IMPACT ON S/C, SRU DESIGN	DOCKING SYSTEM	HARD DOCK	STORAGE VOLUME	COMMENTS ON SU DESIGN
Aerospace	489	14.5' dia x 10' lg, 5' = service unit incl. docking probe. 5' = adapter	2 reqd: Index + translate	28	None reqd	Deployable eqpt. can be outside SRU envelope during SRU transfer. Allows use of base-plates for SRU.	Apollo type; modified to provide radial indexing.	Yes	Failed SRU's stored in place of new SRU's on service unit (SU).	Concept designed in conjunction with DSP & EOS S/C designs.
MDAC Direct Access	430	14.5' dia x ≈ 3' lg	2 reqd: Index + translate. Trans. motion done in two steps: parking & then into SU or S/C.	Round SU ≈ 28 Sq. SU ≈ 25	None reqd	Use of MDAC docker for module exchange makes it impossible to exchange SRU's w/append-ages outside SRU envelope	MDAC ring type; also used to move S/C to exchange SRU's.	Probably not as rigid as Apollo latch system during exchange of SRU's.	Failed SRU's stored in place of new SRU's on SU.	Sq. shape forced by S/C designs. Orig. MDAC concept quite similar to Aerospace concept. Sq. shape not optimum for load transfer to round tug interface.
MDAC External Manipulator	900	14.5' dia x ≈ 10' lg	7 reqd once manipulator is indexed into pos. & assuming old & new modules are opposite each other. If not, 8 motions reqd.	Allows multiples of approx. 16 by stacking rings since SRU access is from side.	Probably none reqd	Allows more flexibility in SRU design than "direct access type".	MDAC ring type, although Apollo could be used if any advantage found.	Yes	Failed SRU's stored in place of new SRU's on SU.	Control of SRU during exchange not as rigid as index/translate concept. C.g. shift greater than index/translate concept.
GE Internal SRU Exchange	600	14.5' dia x ≈ 3' lg	3 reqd: Index/ translate longitud. & radially.	16 per stack	None reqd	Allows stacking of S/C SRU's, limits SRU equip. to SRU envelope, all equip. (SRU) must be internal to S/C during SRU transfer.	Requires MDAC type	Yes	Failed SRU's stored in place of new SRU's on SU.	Poor use of internal volume.
NASA Polar Arm	700	14.5' dia x 7' lg	4 reqd	Random in no. and size. Max. no. ≈25.	Probably none reqd	All SRU's must be within S/C envelope. SRU's can be random shape.	Apollo or MDAC	Yes	Failed SRU's stored in place of new SRU's on SU.	Large space reqd between S/C & SU to allow pivoting of SRU on arm. Restricts design freedom of SRU appendages.
TRW Travelling Crane (Bell Aerospace design)	600	10' sq x 11' lg	7 reqd + those reqd to remove/ refit 6 SRU attach bolts.	28 ~ 24 std + 4 non-std for DSP.	None reqd	Cannot allow equip outside SRU envelope.	Apollo mod. f/indexing. Also, reqs. telescoping tube.	Yes	Failed SRU's stored in place of new SRU's on SU.	Sq. shape not best for load transfer to round tug I/F. Too many motions reqd for exchange. Telescoping dock probe is heavy.
NASA Module Exchange	1000	14.5' dia x ≈ 10' lg	Min. of 23 reqd. NASA document quotes 80.		Reqd for SAMS	Carrousel type. Storage magazine allows only 1 SRU surface to have protrusions, these would be limited by P/L bay envelope.	Apollo or MDAC type. Could be aided by SAMS.	Yes	Failed SRU's stored in place of new ones. Intermediate transfer to carrousel reqd.	Very complicated due to no. of motions reqd.
Rockwell Boom	900	14.5' dia x 9' lg	6 reqd		TV or stored commands.	Allows relative design freedom for SRU shape.	Apollo	Yes	Failed SRU's stored in place of new ones on SU.	Long boom difficult to achieve rigidity w/o wt penalty. Too many motions for exchange reqd.
Rockwell Crane	800	14.5' dia x 10' lg	5 min. to remove from sat. & store in SU. 3 more to position in front of new SRU on SU.		TV or stored commands.	Allows relative design freedom for SRU shape.	Apollo	Yes	Failed SRU's stored in place of new ones on SU.	Long boom difficult to achieve rigidity w/o wt penalty. Too many motions for exchange reqd.

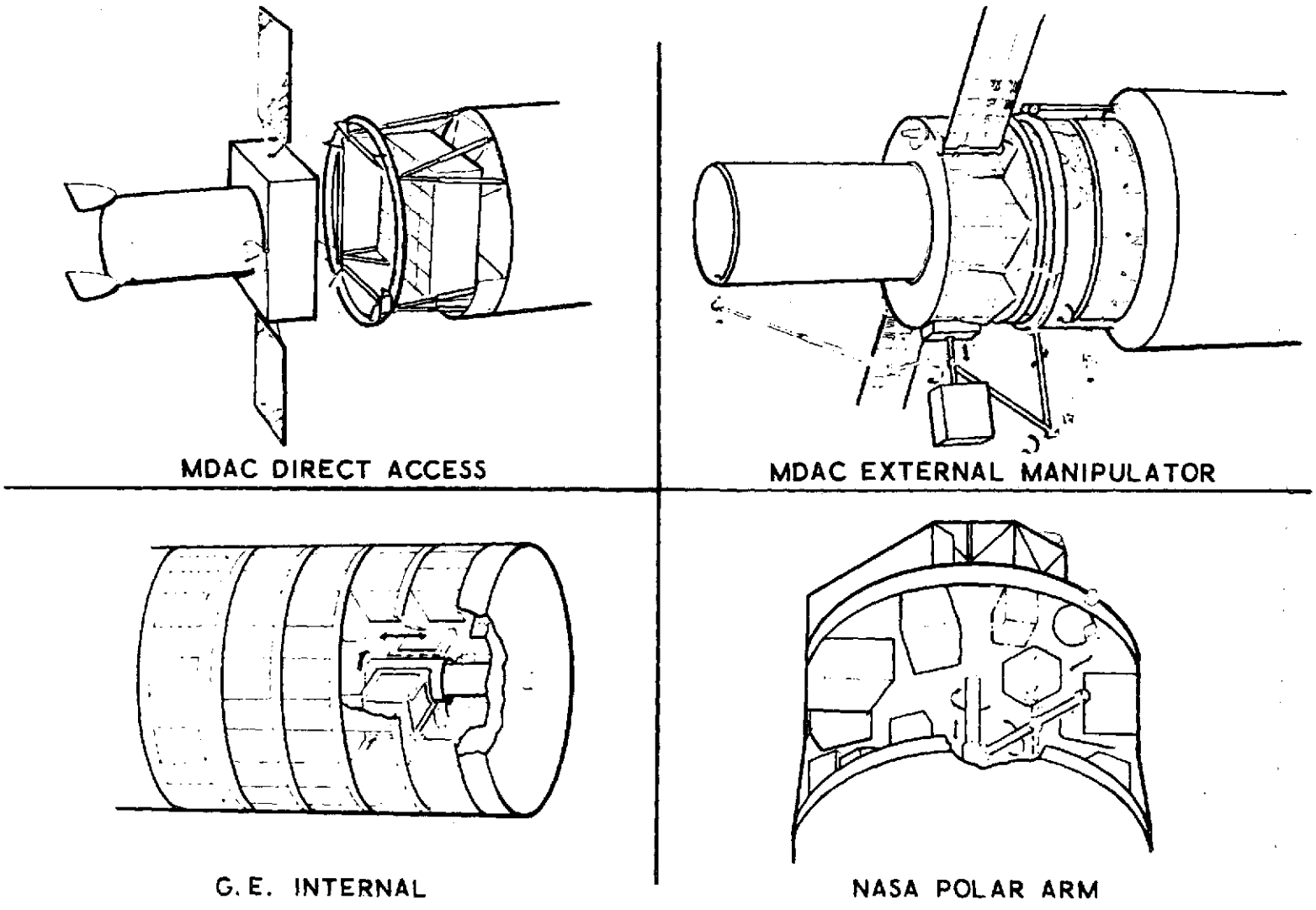


Figure 10-2. Index/Translate or Swing Service Units

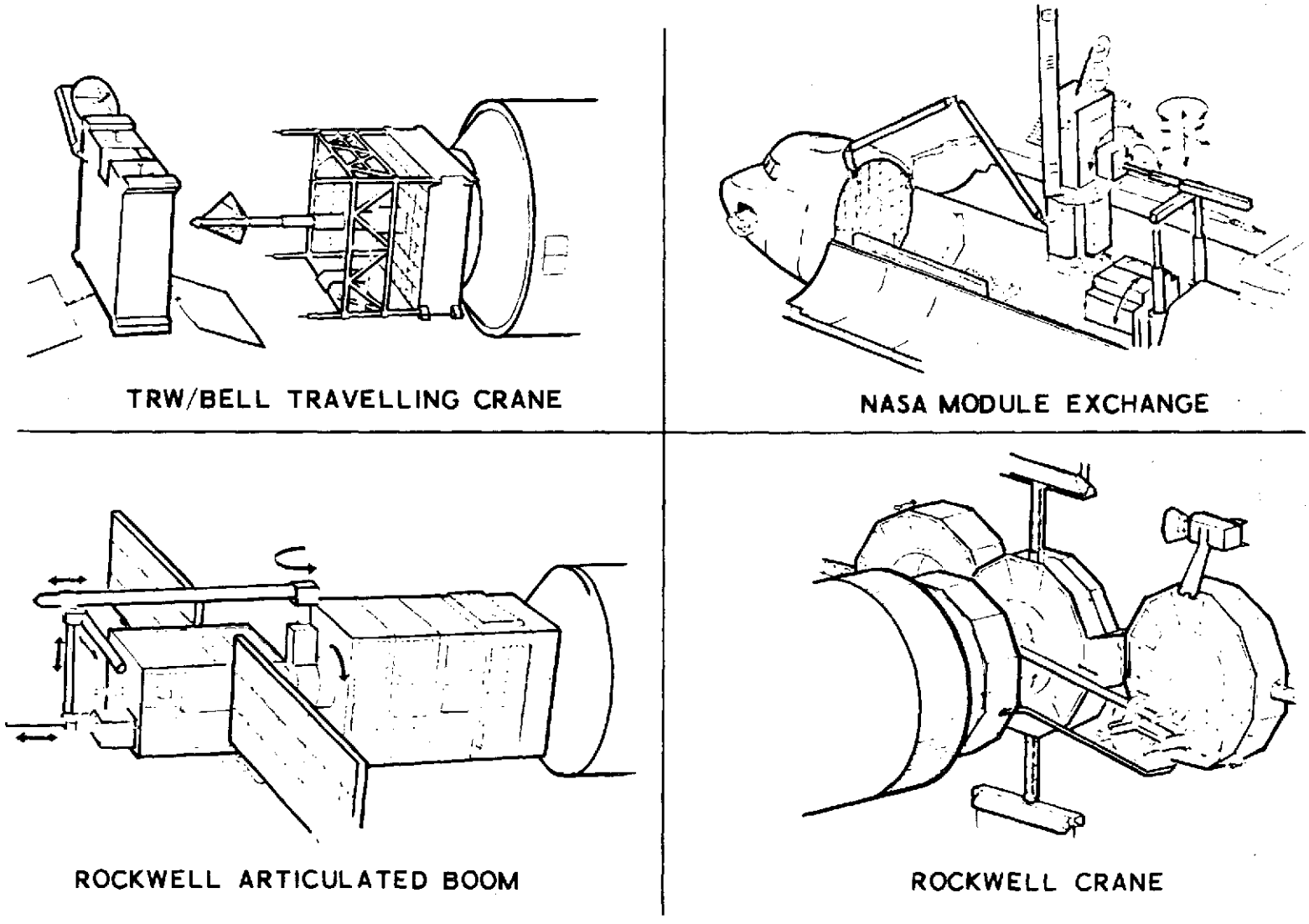


Figure 10-3. Linear Motion Service Units

of stacking satellites in front of the service unit in the payload bay. In addition, some modules may have long appendages (for example the 27-ft long synthetic aperture radar antenna of the EOS) and accommodation of these in the payload bay could be difficult, should the service unit be too long.

c. Motions Required to Exchange a Module. The number of motions required by the service unit to remove and replace a module in a satellite is an indication of the complexity of the mechanisms which might be required to achieve the exchange. A small number of motions is desirable in order to minimize the transfer mechanism complexity.

d. Number of Modules Stored. The module capacity of the service unit should be great enough not to constrain the design of the spacecraft. It is not known at this point what the optimum number of modules should be. Operating under the ground rules established for this study, the number of modules per satellite for the candidates selected varied from 10 to 27. Therefore, any servicing concept which would force the spacecraft to have a small number of modules would be undesirable.

e. Television Assistance. Whether television assistance is necessary or desirable is outside the scope of this effort. For the purposes of this study television assistance was assumed not to be required.

f. Impact on Satellite or Module Design. Table 10-1 lists any constraints that the servicing concept places on the design of the module or on the design of the spacecraft.

g. Docking System. Only two different docking systems were considered: the Apollo probe and drogue type and the MDAC external ring docker. Any constraints placed on the servicing process by the docking system, such as constraining the allowable module envelope, are noted.

h. Hard Dock. It is desirable to have a rigidly docked combination during module exchange in order to minimize the problems of alignment between the module and the spacecraft (or service unit) during insertion of replaceable units.

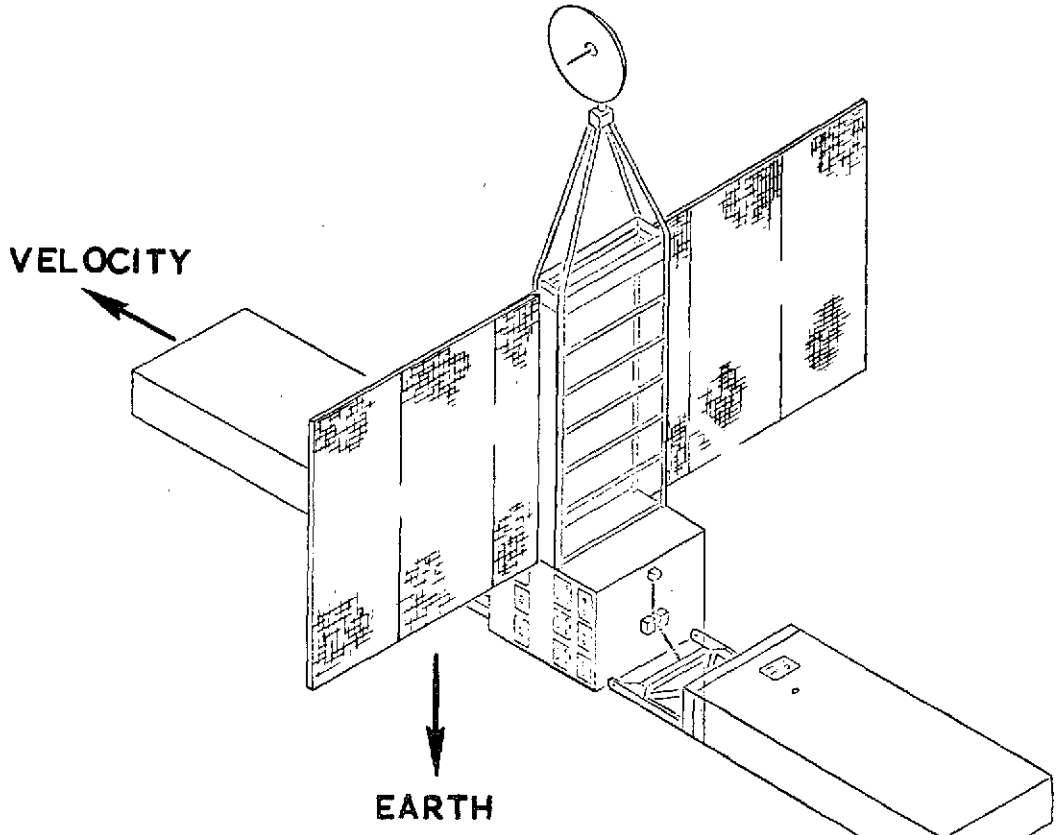
i. Storage Volume. Provision is made for the return of failed modules to earth for refurbishment.

The comparison resulted in the selection of the index and translate technique for module exchange because it offers the lightest weight, simplest concept. Both the Aerospace and the MDAC Direct Access techniques fall into this category. For the purposes of this study, the Aerospace concept was chosen for a number of reasons; the first is that it does not require the elimination of the EOS satellite as a candidate payload, as would the MDAC concept. This is due to the fact that the EOS has four sensors which are 5 ft or more in length including a 27-ft long radar antenna, none of which could be replaced using the MDAC concept. The MDAC concept relies on the use of an external docking mechanism with telescoping struts which are used to translate the entire satellite structure to the service unit as part of the module exchange process. As can be seen from Figure 10-2, these struts attach to the upper stage and extend to the front of the servicing unit. No appendages may protrude from a module beyond these struts since they would then foul the struts during module indexing.

It is important to consider also that many of the sensors of the satellites listed in the mission model have the potential to grow beyond their present definitions and that the service unit should impose no constraints on this potential growth. An example, shown in Figure 10-4, presents a comparison of the EOS payload as described last year in Ref. 1 and the EOS as presently defined in Ref. 3 & 4.

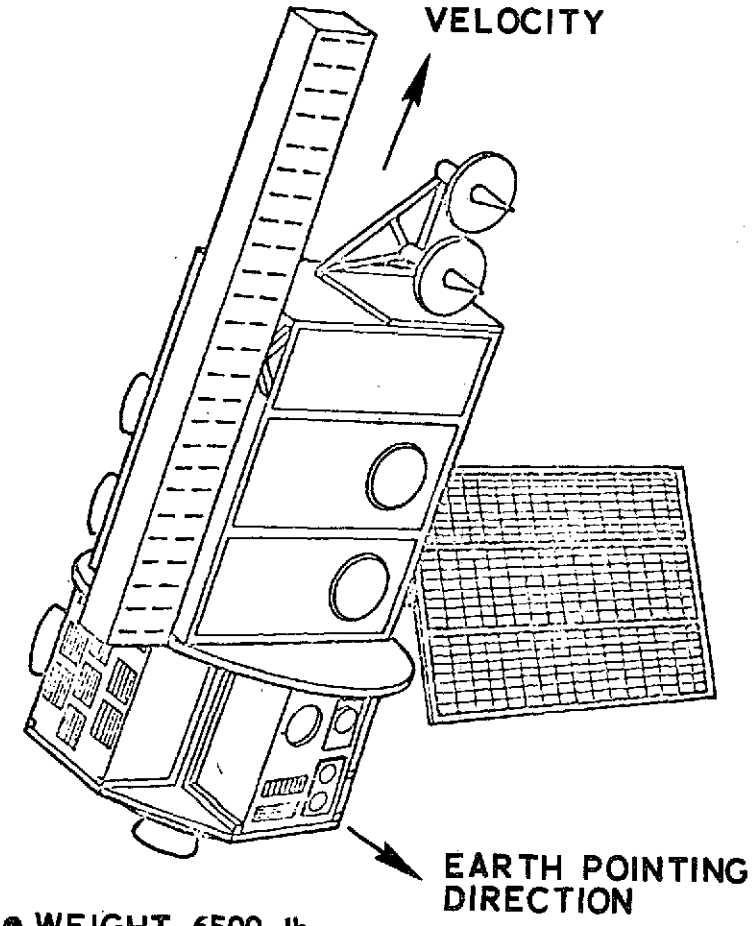
MDAC, in performing the Payload Utilization of the Tug study (Refs. 12, 13, and 14), did not have the advantage that Aerospace has of reviewing an entire spectrum of spacecraft, identifying the individual mission equipment modules for each of the spacecraft in the spectrum, and then identifying a spacecraft and servicing unit concept which would be applicable to as many of those spacecraft as possible. It has been determined through past design efforts performed at Aerospace on space-serviceable satellites (Refs. 1 and 24) that the service unit design activity and the spacecraft design activity should be a single integrated effort.

MDAC, early in their study, decided on a servicing concept which required a round spacecraft almost identical to the Aerospace concept.



- WEIGHT 3800 lb

INITIAL EXPENDABLE DESIGN
(3-Axis Stabilized)



- WEIGHT 6500 lb (2200 lb constitutes the propulsion unit)

CURRENT DESIGN
(Modularized/Serviceable)

Figure 10-4. EOS Payload Reconfiguration

However, they were later presented with three point designs* of square spacecraft generated by the spacecraft contractors and were then forced to redesign the service unit to that shown in Figure 10-2.

While the MDAC servicing concept appears to be adequate for the three point designs considered in the MDAC study, it cannot service other satellites, such as the EOS, for the reasons mentioned earlier.

Other reasons for selecting the Aerospace concept are that it satisfied the requirements of low weight, short length and perhaps more importantly, a thorough knowledge of the design is at hand, thus helping to expedite the study. At the same time, the selection should not be construed as a recommendation of a specific space-servicing concept.

10.4 MODULE ASSIGNMENT

The composite buildup of each payload consisting of nonreplaceable equipment, standard subsystem SRUs and mission equipment SRUs is provided in Table 10-2. This table also indicates the number of SRUs required in each subsystem, i.e., four attitude and velocity control propulsion units. A summation of the weights is also provided. This table can be correlated with Table 3-1 wherein space serviceable payload candidates are identified within the total spectrum of the NASA October 1973 Mission Model. The total number of SRUs for each payload varies from 10 to 26.

For convenience, the number of modules required in each subsystem is provided in Table 10-3. This table indicates the basic number of SRUs plus any associated module variants required to cover the broad span of application to 42 different automated payload programs.

*The three spacecraft imposed on MDAC were the Synchronous Earth Observatory Satellite (SEOS) by General Electric (GE) and two communications satellites by Fairchild: the Communication R and D Satellite (Comm R&D) and the Communication Satellite Corp. Satellite (CSC SAT).

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Table 10-2. Satellite Module Assignment

Satellite			Non-Replaceable Units			Space Replaceable Units																	Satellite Weight (kg)		
						Standardized													Non-Standardized						
Payload Code						Attitude & Velocity Control			Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
AST-1B	AS-03	Cosmic Background	NAST-1B	1	296.0	AVCS-3 AVCS-6A AVCS-7 AVCS-9	1 1 4 1	109.2 72.5 202.4 51.3 435.4	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-7A	1	64.0	DP-1A	1	46.8	EPS-1C	1	105.0	AST-1B-1	1	159.4	1228	36	1264
AST-1C	AS-05	Advanced Radio Astronomy Explorer	NAST-1C	1	296.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-5	1	54.0	DP-1B	1	46.8	EPS-1A	1	50.0	AST-1C-1	1	170.8	1106	24	1130
AST-3	SO-03	Solar Physics Mission	NAST-3	1	371.0	AVCS-3 AVCS-6A AVCS-7 AVCS-9	1 1 4 1	109.2 72.5 202.4 51.3 435.4	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-7A	1	64.0	DP-1C	1	46.8	EPS-1D	1	134.0	AST-3-1 -2 -3 -4 -5 -6 -7 -8 -9	1 1 1 1 1 1 1 1 1	154.8 88.8 104.8 65.8 72.8 108.8 93.8 108.8 122.8 921.2	2093	55	2148
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	NAST-9A	1	4301.0	AVCS-4 AVCS-6A AVCS-7 AVCS-9	3 1 4 1	343.5 72.5 202.4 51.3 669.7	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-7A	1	64.0	DP-1D	1	46.8	EPS-2 EPS-3	2 2	262.0 102.0 364.0	AST-9A-1 -2 -3 -4 -5 -6 -7	1 1 1 1 1 1 1	77.8 72.8 88.8 113.8 98.3 95.8 94.8 642.1	6204	41	6250
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	NAST-9B	1	7451.0	AVCS-4 AVCS-6A AVCS-7 AVCS-9	3 1 4 1	343.5 72.5 202.4 51.3 669.7	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-7A	1	64.0	DP-1E	1	46.8	EPS-2 EPS-3	2 2	262.0 102.0 364.0	AST-9B-1 -2 -3 -4 -5 -6 -7	1 1 1 1 1 1 1	77.8 72.8 88.8 113.8 98.3 95.8 94.8 642.1	9359	34	9393
PHY-1A	HE-07	Small High Energy Observatory	NPHY-1A	1	296.0	AVCS-3 AVCS-5A AVCS-7 AVCS-9	1 1 4 1	109.2 55.9 202.4 51.3 418.8	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-7A	1	64.0	DP-1F	1	46.8	EPS-1B	1	78.0	PHY-1A-1	1	159.1	1184	36	1220

Note:

(1) Sunshade not included. If required, add 400 kg.

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Table 10-2. Satellite Module Assignment (Continued)

Satellite			Non-Replaceable Units			Space Replaceable Units															Satellite Weight (kg)		
						Standardized												Non-Standardized					
						Attitude & Velocity Control			Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Payload Code																									
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
PHY-1B	AP-01	Upper Atmospheric Explorer	NPHY-1B	1	364.0	AVCS-2 AVCS-5A AVCS-8 AVCS-9	1 1 2 1	42.6 55.9 115.2 51.3 265.0	N/A			TTC-9A	1	65.0	DP-1G	1	46.8	EPS-1D	1	134.0	PHY-1B-1 -2 -3 -4 -5 -6 -7 -8	1 1 1 1 1 1 1 1	64.8 76.8 69.8 58.8 64.8 60.8 68.8 66.8 531.4	1406	293	1699
PHY-1C	AP-02	Medium Attitude Explorer	NPHY-1C	1	304.0	AVCS-2 AVCS-7 AVCS-5A AVCS-9	1 2 1 1	42.6 101.2 55.9 51.3 251.0	N/A			TTC-5A	1	61.0	DP-1H	1	46.8	EPS-1B	1	78.0	PHY-1C-1 -2 -3	1 1 1	85.1 77.1 63.8 226.0	967	10	977
PHY-2A	AP-04	Gravity and Relativity Satellite - LEO	NPHY-2A	1	255.0	AVCS-3 AVCS-5A AVCS-8 AVCS-9	1 1 4 1	109.2 55.9 230.4 51.3 446.8	GN-2	1	42.6	TTC-10	1	63.0	DP-1I	1	46.8	EPS-1A	1	50.0	PHY-2A-1 -2 -3	1 1 1	145.6 119.5 77.4 342.5	1247	415	1662
EO-3A	EO-08	Earth Observatory Satellite	NEO-3A	1	389.0	AVCS-3 AVCS-6A AVCS-8 AVCS-9	1 1 4 1	109.2 72.5 230.4 51.3 463.4	GN-1 GN-2	1	78.3 42.6 120.9	TTC-9A	1	65.0	DP-1J	1	46.8	EPS-2 EPS-3	2 2	262.0 102.0 364.0	EO-3A-1 -2 -3 -4 -5 -6 -7 -8	1 1 1 1 1 1 1 1	326.8 326.8 88.8 281.8 82.0 131.1 66.1 62.1 1365.5	2815	227	3042
EO-4A	EO-09	Sync. Earth Observatory Satellite	NEO-4A	1	777.0	AVCS-1 AVCS-6 AVCS-8	1 1 4	38.9 72.5 230.4 341.8	GN-2	1	42.6	TTC-4	1	63.0	DP-1K	1	46.8	EPS-1C EPS-3	1 1	105.0 51.0 156.0	EO-4A-1 -2	1 1	199.8 88.8 288.6	1716	133	1849
EO-6	EO-12	Tiros	NEO-6	1	392.0	AVCS-3 AVCS-6A AVCS-7 AVCS-9	1 1 4 1	109.2 72.5 202.4 51.3 435.4	GN-2	1	42.6	TTC-8	1	62.0	DP-1L	1	46.8	EPS-2 EPS-3	2 2	262.0 102.0 364.0	EO-6-1 -2 -3 -4 -5 -6 -7	1 1 1 1 1 1 1	81.8 99.8 106.8 111.5 82.0 77.6 99.8 659.3	2002	35	2037

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Table 10-2. Satellite Module Assignment (Continued)

Satellite			Non-Replaceable Units			Space Replaceable Units																		Satellite Weight (kg)		
						Standardized															Non-Standardized					
Payload Code						Attitude & Velocity Control			Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
EO-7	EO-07	Synchronous Meteorological Satellite	NEO-7	1	364.0	AVCS-3 AVCS-6 AVCS-7	1 1 4	109.2 72.5 202.4 384.1	GN-2	1	42.6	TTC-6	1	54.0	DP-1M	1	46.8	EPS-2 EPS-3	1 2	131.0 102.0 234.0	EO-7-1 -2 -3 -4 -5	1 1 1 1 1	99.8 88.8 64.0 59.3 82.0 393.9	1519	62	1581
EOP-3	OP-07	Seasat B	NEOP-3	1	380.0	AVCS-2 AVCS-5A AVCS-7 AVCS-9	1 1 2 1	42.6 55.9 101.2 51.3 251.0	N/A			TTC-9A	1	65.0	DP-1N	1	46.8	EPS-1B EPS-2 EPS-3	1 1 2	78.0 131.0 102.0 311.0	EOP-3-1 -2 -3 -4 -5	1 1 1 1 1	99.8 145.8 97.8 127.8 102.8 574.0	1628	50	1678
EOP-4	OP-01	Geopause	NEOP-4	1	338.0	AVCS-2 AVCS-5 AVCS-7	1 1 2	42.6 55.9 101.2 199.7	N/A			TTC-5A	1	61.0	DP-1O	1	46.8	EPS-1C	1	105.0	EOP-4-1 -2	1 1	145.4 84.8 230.2	981	57	1038
EOP-7	OP-04	Gravsat	NEOP-7	1	320.0	AVCS-3 AVCS-5A AVCS-8 AVCS-9	1 1 4 1	42.6 55.9 230.4 51.3 380.2	N/A			TTC-10	1	63.0	DP-1P	1	46.8	EPS-1D	1	134.0	EOP-7-1 -2	1 1	280.8 145.4 426.2	1370	255	1625
NND-1	CN-51	International Communication Satellite	NNND-1	1	829.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	GN-2	1	42.6	TTC-1	1	51.0	DP-1Q	1	46.8	EPS-2 EPS-3	6 2	786.0 102.0 888.0	NND-1-1 -2	1 1	171.0 248.7 419.7	2645	40	2685
NND-2A	CN-52	U.S. Domestic Satellite - A	NNND-2A	1	318.0	AVCS-1 AVCS-5 AVCS-7	1 1 4	38.9 55.9 202.4 297.2	N/A			TTC-1	1	51.0	DP-1R	1	46.8	EPS-1D	1	134.0	NND-2A-1	1	88.8	936	50	986
NND-2B	CN-53	U.S. Domestic Satellite - B	NNND-2B	1	829.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	GN-2	1	42.6	TTC-1	1	51.0	DP-1S	1	46.8	EPS-2 EPS-3	6 2	786.0 102.0 888.0	NND-2B-1 -2	1 1	171.0 248.7 419.7	2645	40	2685
NND-2D	CN-58	U.S. Domestic Satellite - C	NNND-2D	1	345.0	AVCS-1 AVCS-5 AVCS-7	2 1 4	77.8 55.9 202.4 336.1	N/A			TTC-2	1	51.0	DP-1T	1	46.8	EPS-1D EPS-3	1 1	134.0 51.0 185.0	NND-2D-1 -2 -3	1 1 1	80.1 131.1 68.9 280.1	1244	81	1325

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Table 10-2. Satellite Module Assignment (Continued)

Satellite			Non-Replaceable Units			Space Replaceable Units															Satellite Weight (kg)		
						Standardized												Non-Standardized					
Payload Code						Attitude & Velocity Control			Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
NND-3	CN-54	Disaster Warning Satellite	NNND-3	1	533.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	N/A			TTC-3	1	50.0	DP-1U	1	46.8	EPS-1B EPS-3	1 2	78.0 102.0 180.0	NND-3-1	1	163.8	1341	8	1349
NND-4	CN-55	Traffic Management Satellite	NNND-4	1	335.0	AVCS-3 AVCS-5 AVCS-7	1 1 4	109.2 55.9 202.4 367.5	N/A			TTC-1	1	51.0	DP-1V	1	46.8	EPS-1B EPS-3	1 2	78.0 102.0 180.0	NND-4-1	1	103.8	1084	52	1136
NND-5	CN-56	Foreign Communication Satellite	NNND-5	1	302.0	AVCS-1 AVCS-5 AVCS-8	1 1 2	38.9 55.9 115.2 210.0	N/A			TTC-1	1	51.0	DP-1W	1	46.8	EPS-1B EPS-3	1 2	78.0 102.0 180.0	NND-5-1	1	81.8	872	115	987
NND-6	CN-59	Communication R&D Prototype	NNND-6	1	752.0	AVCS-1 AVCS-5 AVCS-8	1 1 2	38.9 55.9 115.2 210.0	GN-2	1	42.6	TTC-5	1	54.0	DP-1X	1	46.8	EPS-2 EPS-3	10 2	1310.0 102.0 1412.0	NND-6-1 -2 -3 -4 -5	1 1 1 1 1	67.0 68.4 63.9 72.0 154.6 425.9	2943	205	3148
NND-8	EO-56	Envioronmental Monitoring Satellite	NNND-8	1	358.0	AVCS-3 AVCS-6A AVCS-8 AVCS-9	1 1 4 1	109.2 72.5 230.4 51.3 463.4	GN-2	1	42.6	TTC-7	1	57.0	DP-1Y	1	46.8	EPS-1B EPS-2 EPS-3	1 1 2	78.0 131.0 102.0 311.0	NND-8-1 -2 -3 -4 -5 -6 -7 -8 -9 -10 -11 -12	1 1 1 1 1 1 1 1 1 1 1 1	85.2 85.2 117.8 117.8 84.8 84.8 79.8 79.8 79.8 79.8 81.8 81.8 1058.4	2338	264	2602
NND-11	EO-61	Earth Resources Satellite - LEO	NNND-11	1	298.0	AVCS-3 AVCS-5 AVCS-7 AVCS-9	1 1 4 1	109.2 55.9 202.4 51.3 418.8	GN-2	1	42.6	TTC-9	1	54.0	DP-1Z	1	46.8	EPS-1B EPS-2 EPS-3	1 1 2	78.0 131.0 102.0 311.0	NND-11-1 -2	1 1	126.8 89.8 216.6	1388	72	1460

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Table 10-2. Satellite Module Assignment (Continued)

Satellite			Non-Replaceable Units			Space Replaceable Units																		Satellite Weight (kg)		
						Standardized															Non-Standardized					
Payload Code						Attitude & Velocity Control			Guidance & Navigation			Telemetry, Tracking & Command			Data Processing			Electrical Power			Mission Equipment					
Mission Model	SSPDA	Payload Code	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	Item	Qty	Weight (kg)	DRY	WP	WET
NND-12	EO-59	Earth Resources Satellite - GEO	NNND-12	1	825.0	AVCS-3 AVCS-6 AVCS-8	1 1 4	109.2 72.5 230.4 412.1	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-4	1	63.0	DP-1AA	1	46.8	EPS-1E EPS-3	1 2	189.0 102.0 291.0	NND-12-1 -2	1 1	199.8 88.8 288.6	2047	349	2396
NND-13	EO-62	Earth Resources Satellite - Foreign	NNND-13	1	825.0	AVCS-3 AVCS-6 AVCS-8	1 1 4	109.2 72.5 230.4 412.1	GN-1 GN-2	1 1	78.3 42.6 120.9	TTC-4	1	63.0	DP-1BB	1	46.8	EPS-1E EPS-3	1 1	189.0 51.0 240.0	NND-13-1 -2	1 1	199.8 88.8 289.6	1997	343	2340
NND-14	OP-08	Global Earth and Ocean Monitoring Satellite	NNND-14	1	395.0	AVCS-3 AVCS-5A AVCS-8 AVCS-9	1 1 4 1	109.2 55.9 230.4 51.3 446.8	N/A			TTC-9A	1	65.0	DP-1CC	1	46.8	EPS-2 EPS-3	2 2	262.0 102.0 364.0	NND-14-1 -2 -3 -4 -5	1 1 1 1 1	99.8 97.8 84.8 106.8 106.8 496.0	1814	329	2143

Table 10-3. SRU Inventory Requirements Summary

SUBSYSTEM TYPE	MODULE/VARIANTS	TOTAL
ELECTRICAL POWER	3/5	8
DATA PROCESSING	1/0 (1)	1
ATTITUDE-VELOCITY CONTROL	9/2	11
GUIDANCE & NAVIGATION	2/0	2
TELEMETRY TRACK/COMM	10/3 (2)	13

(1) SOFTWARE MODS FOR EACH PAYLOAD
(2) STANDARD COMPONENTS EMPLOYED

- TOTAL INVENTORY - 42 PROGRAMS
  - / SUBSYSTEM MODULES 35
  - / MISSION EQUIPMENT MODULES 114

11. POINT DESIGN

11.1 SATELLITE POINT DESIGN

In order to validate the design approach adopted in this study, a specific satellite and its servicing unit were selected for detailed examination. The EOS was chosen for the point design effort because it presented the most difficult case in terms of having a large number of modules, large, complex sensors, and because it would serve to demonstrate the capability of the selected space-servicing technique in exchanging modules which have large or long appendages. In addition, a space-serviceable design of an earlier version of the EOS, with quite different, smaller sensors was done during a previous study (Ref. 1). The selection of the present version of the EOS presented an opportunity to demonstrate typical changes in satellite configurations as the missions become better defined and, as mentioned earlier in this report, demonstrate the capability of the selected servicing technique to accommodate these changes.

The design of the EOS is presented in Figure 11-1. Two views are shown, a plan and a side view. Looking at the plan view, the orbit direction is shown as toward the bottom of the paper and nadir is into the paper. Sixteen replaceable modules with their baseplates fitted into guide rails are arranged around the outside of the non-replaceable structure. Modules occupying 7 of the 12 available locations are arranged around the inside of the structure. The baseplates for the inside modules are slightly wider than those for the outside modules; however, their designs are otherwise identical. A typical baseplate design is shown in Figure 11-2. Also, in the center of the spacecraft is the docking cone designed to mate with an Apollo type docking probe on the service unit.

A better definition of the EOS mission equipment listed in Reference 4 was obtained from References 25 through 28 and also via discussions with cognizant personnel. The "Air and Water Pollution Package" listed in Reference 4 was assumed to be four of the sensors defined in the Nimbus G experiment Summary (Ref. 27). These four sensors are the Stratospheric

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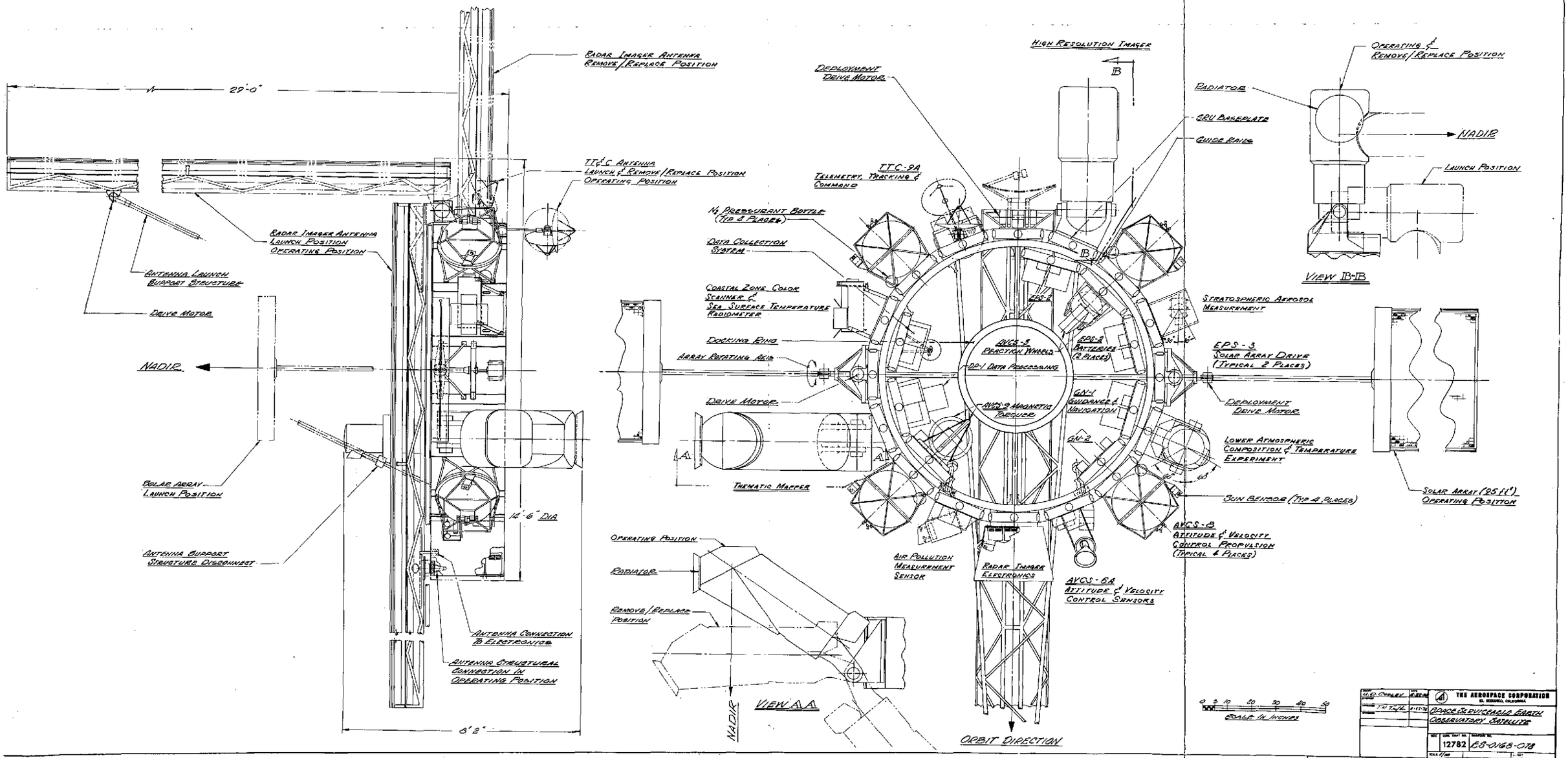

Figure 11-1. Space Serviceable Earth Observatory Satellite

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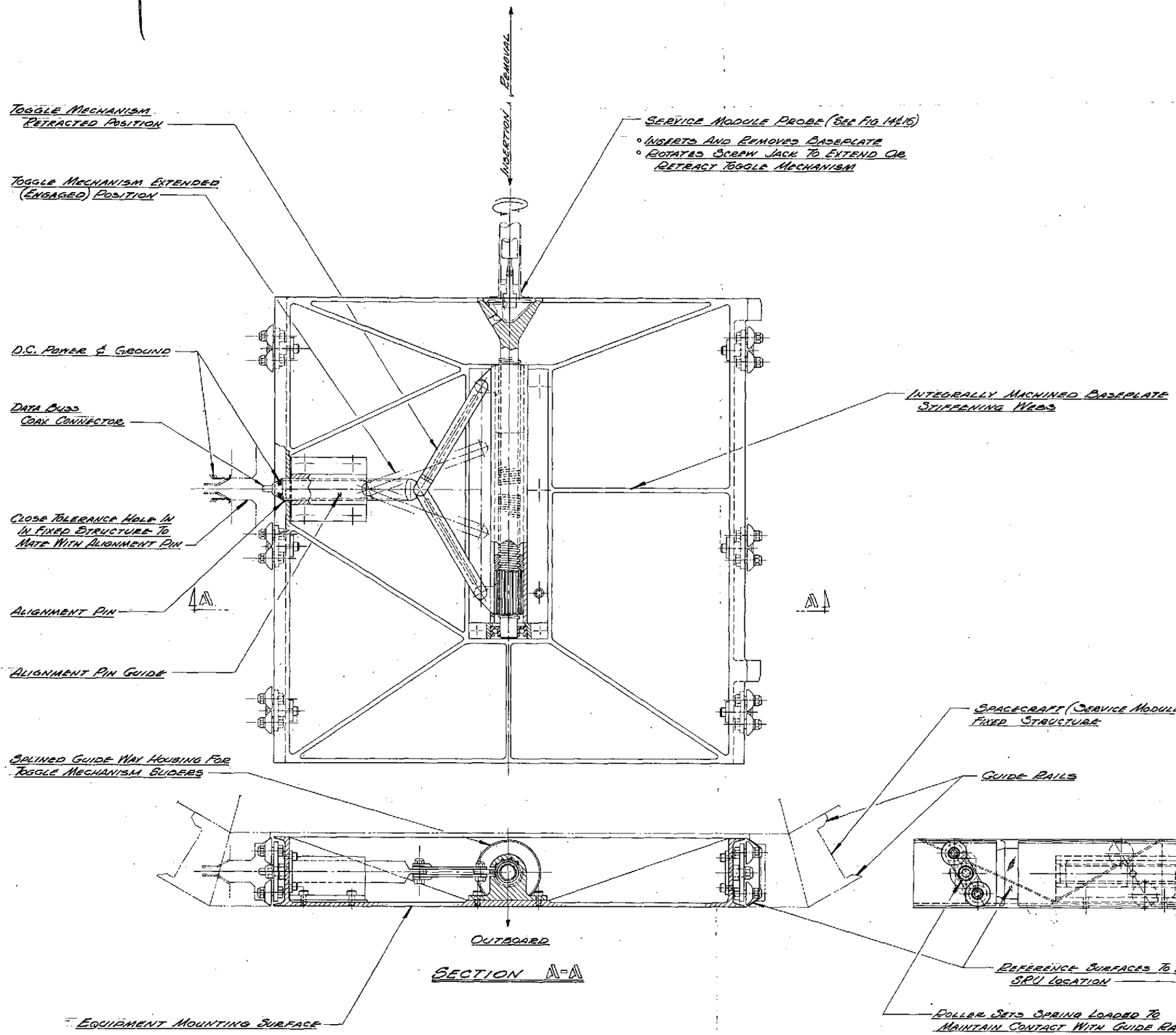

Figure 11-2. SRU Baseplate

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Aerosol Measurement, the Lower Atmospheric Composition and Temperature Experiment, the Air Pollution Measurement Sensor, and the Coastal Zone Solar Scanner and Sea Surface Temperature Radiometer.

There are four sensors which are too large to be packaged entirely within the module envelope; these are the High Resolution Imager, the Lower Atmospheric Composition and Temperature Experiment, the Radar Imager, and the Thematic Mapper.

The High Resolution Imager and the Thematic Mapper are fitted to the spacecraft in a similar manner. They are stowed in such a way that they fit in the positions shown and for exchange they are positioned so that they do not protrude above or below the thickness dimension of the satellite.

The Lower Atmospheric Composition and Temperature Experiment is exchanged in a similar manner but does not require deployment from the launch position for operation.

It is recognized that the requirement to deploy optical sensors from a stowed to an operating position introduces added complications in the design of the supporting structure. Optical supporting structure is generally designed to a stiffness rather than a strength criteria, and the introduction of deployable structure will result in a reduction in the structural rigidity of the sensor support system. In these cases, a weight penalty will have to be paid in order to achieve the required rigidity. In the case of very high resolution sensors operating in the visible light range, the weight penalty could possibly be excessive. Tradeoffs for each individual sensor would have to be made before final decisions can be reached. In the specific cases of the Thematic Mapper and High Resolution Imager carried on the EOS, it is felt that they can be accommodated in the manner shown for the weight estimates listed. By comparison to other sensors which have been flown, these are not considered to be very high resolution sensors.

The Radar Imager requires two modules. The electronics are

contained in one module and in the module on the opposite side of the satellite is contained the antenna mounting structure and deployment drive motors. The antenna can be in any of three positions as shown. In the launch position, additional support structure is used. Prior to deployment to the operating position, the antenna support structure is disconnected from the satellite and moved to a position alongside the antenna via a drive motor on the antenna. Then the antenna and its structure are driven to the operating position where connection between the electronics and the antenna is accomplished.

The modules containing the housekeeping subsystems are located around both the inside and outside of the spacecraft. The solar arrays are designed to rotate about a single axis to accommodate the sun synchronous orbit. While the arrays are nonreplaceable, the drives may be replaced through the scheme shown in Figures 11-3 and 11-4. The housekeeping modules around the inside were located to achieve satellite balance.

11.2 SERVICE UNIT POINT DESIGN

The service unit designed to transfer the replaceable modules to and from the satellite is shown in Figure 11-5. The unit is designed to service a spacecraft with up to 28 replaceable modules (16 modules on the outside and 12 modules on the inside). It consists of a fixed structural ring of approximately 3.0m diameter which provides a mating interface with the spacecraft at one end and connects to the upper stage via a conical adapter at the other end. Supported from this ring, and centrally placed, is an Apollo docking probe modified to provide radial indexing with the mating cone on the spacecraft. Once docking is achieved, the satellite and service unit are firmly latched together with 6 latches around the structural ring.

Around the outisde and inside of the structural ring are indexable rings, driven by motors through gear drives and supported by rollers. These indexable rings have guide rails attached in order to contain the baseplate of the replaceable modules. The modules are transferred

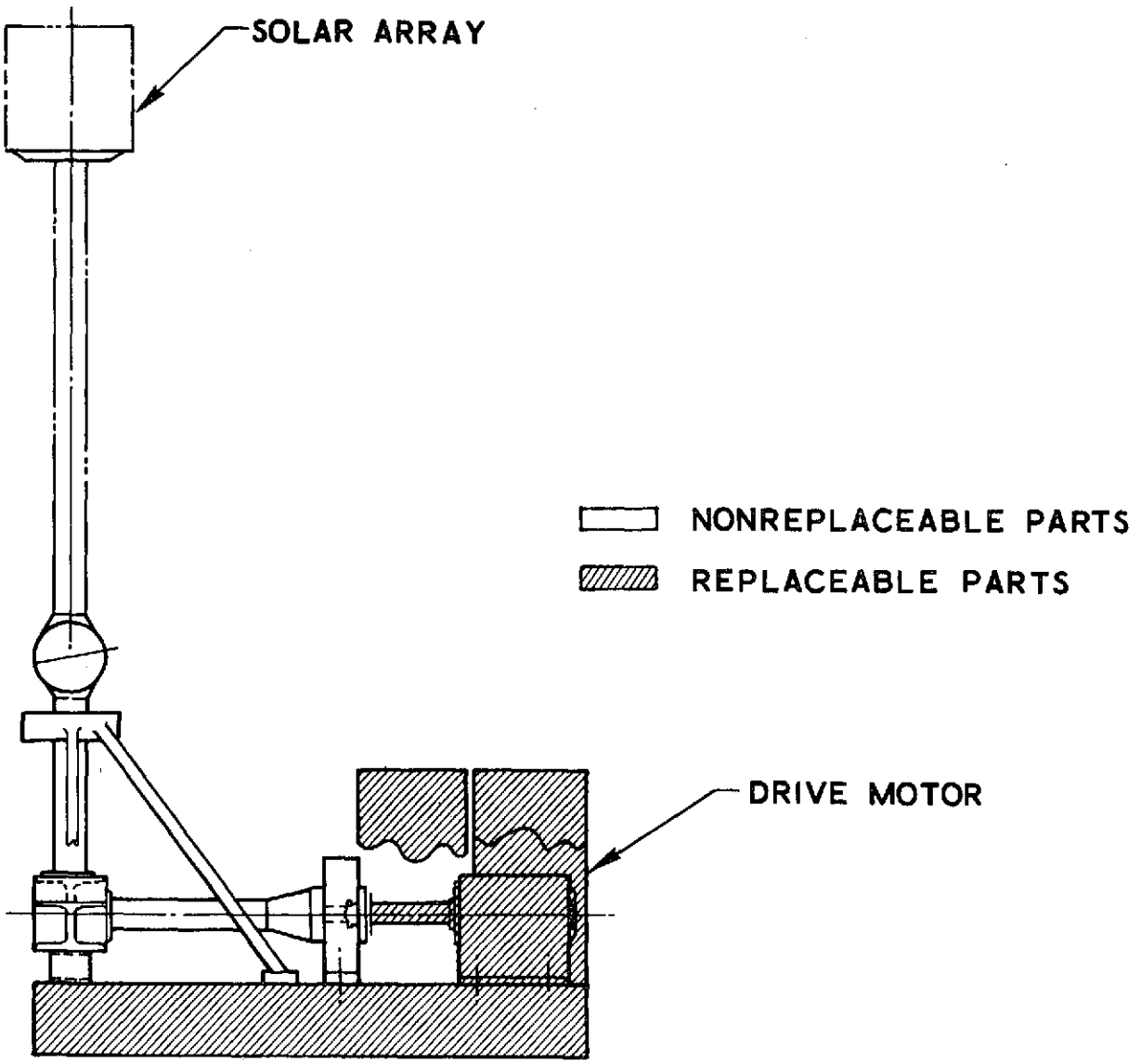

Figure 11-3. Replaceable Solar Array Drive

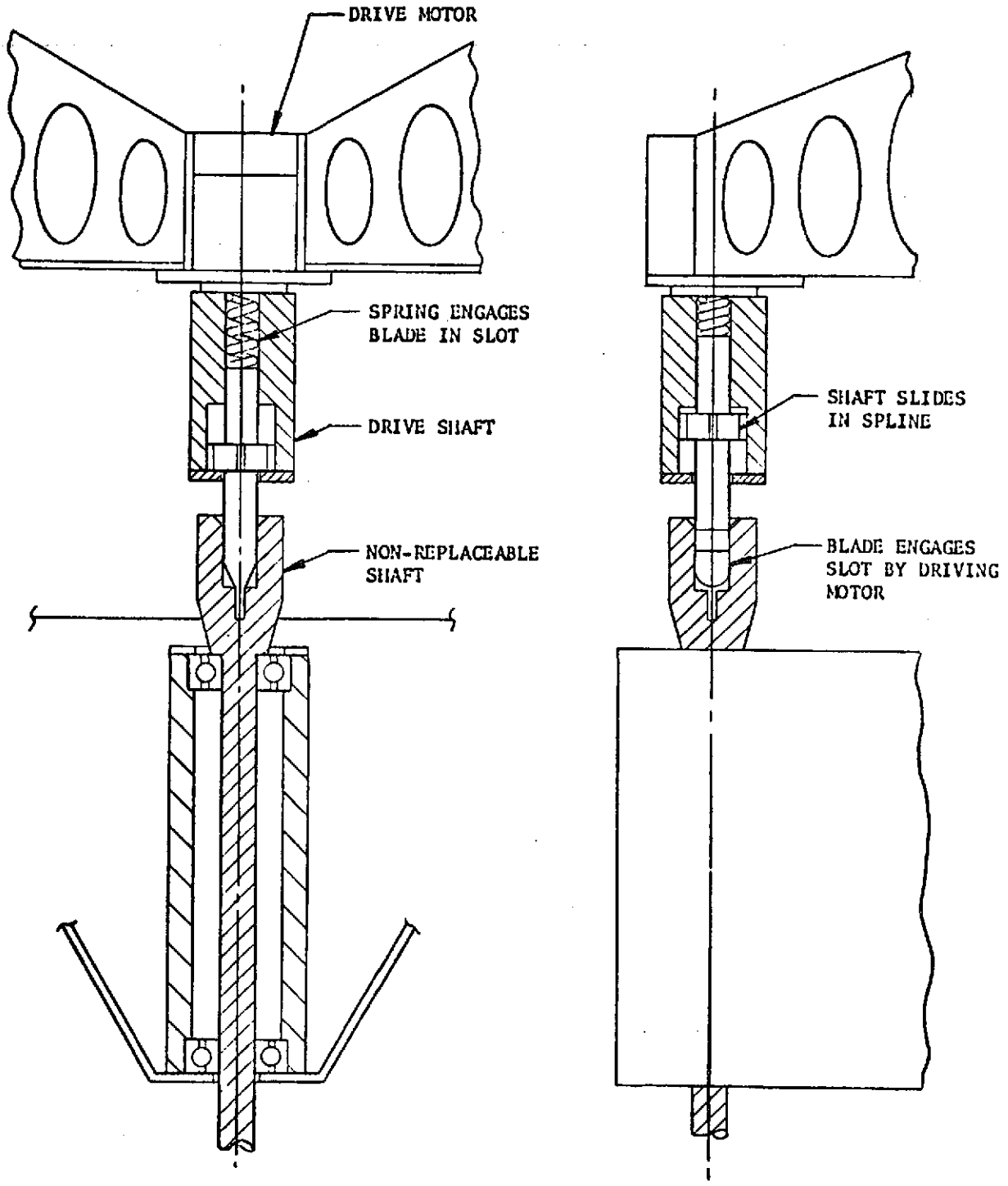

Figure 11-4. Replaceable Drive

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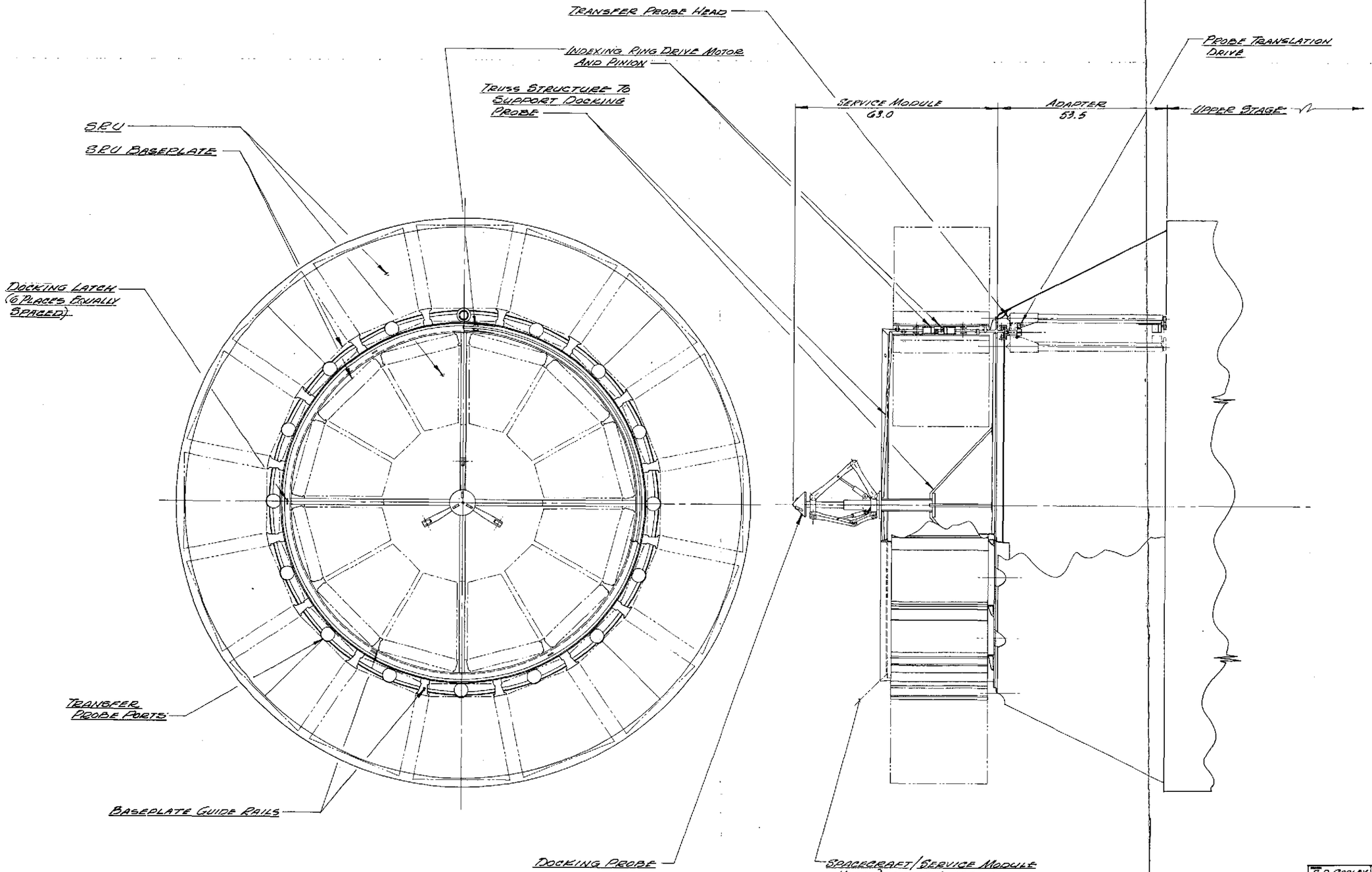


Figure 11-5. Service Unit

DRAWN R.D. COOLEY	DATE 9-27-76	THE AEROSPACE CORPORATION, EL SEGUNDO, CALIFORNIA
CHECKED		
APPROVED Tim Tuttle	9-27-76	SERVICE UNIT
APPROVED		SIZE 12782 · DRAWING NO. ES-0168-073 · SCALE · SHEET

to or from the spacecraft via a transfer probe in the adapter. This transfer probe also actuates the baseplate locking mechanism (Fig. 11-2).

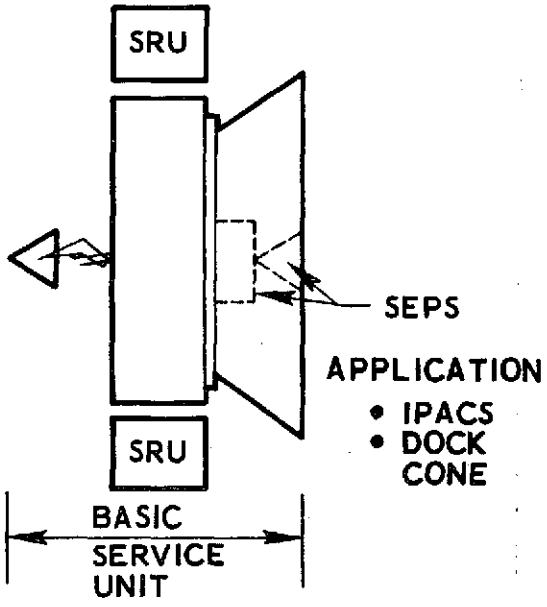
The transfer process is achieved by: (1) indexing an empty service unit slot opposite the old module on the satellite, (2) indexing the transfer probe behind the empty slot, (3) translating the probe to the satellite and locking into the baseplate, (4) unlocking the baseplate from the satellite by rotation of the probe, (5) withdrawing the module into the service unit, and (7) translating the module to the satellite.

The service unit shown in Figure 11-5 is one which might be used to service satellites in low earth orbit (LEO) and which would contain from 17 to 28 replaceable modules. Actually, four models of the service unit are envisioned in order to save weight and to be compatible with use of the SEPS. These four models are described in Figure 11-6. Models A and B are to be used in low earth orbit. Model A is used with satellites which have 16 modules or less, all on the outer diameter of the satellite. Model B is used with satellites which contain additional modules up to a maximum of 28 modules. The unit illustrated in Figure 11-5 is a model B.

Models C and D are used for satellites in orbits higher than 14, 816 km and which utilize SEPS. Use of SEPS requires that the service unit undock from the chemical upper stage and dock with the SEPS for transfer to orbits up to geosynchronous. The transfer to SEPS requires that the service unit be a stable platform during docking. This is accomplished by the addition of an Integrated Power and Attitude Control System (IPACS), a unit proposed by Rockwell International (Ref. 29). The docking procedure with SEPS also requires the addition of a docking cone on the service unit and a probe on the upper stage, as well as a docking probe on the SEPS and these differences are reflected in the weight of the units. It should be recognized that all four models are derived from the basic unit by simple kit modifications, thereby requiring only one developmental program.

Table 11-1 shows the assignment of these four different variations of the service unit to the candidate spacecraft.

SERVICE UNIT ITEM	MOD A (lb) (16)	MOD B (lb) (28)	MOD C (lb) (16)	MOD D (lb) (28)
APOLLO PROBE	85	85	85	85
DOCK STRUCT	35	35	35	35
MODULE STRUCT	150	150	150	150
BASIC TRACK (16) @ 10 lb	160	160	160	160
ROTARY MECH	100	100	100	100
12 TRACKS @ 10 lb	-	120	-	120
ROTARY MECH	-	50	-	50
SEPS EQUIP	-	-	55	55
TOTAL	530	700	585	755



NOTES: SEPS APPLICATION REQUIRES ADDITIONAL DOCKING PROBE ON UPPER STAGE

120 lbs

Figure 11-6. Operational Concepts Service Unit Design (Preliminary)

Table 11-1. Satellite Servicing Unit Requirements

Payload Code		Payload Name	Number of SRUs	Service Unit			
Mission Model	SSPDA			A(1)	B(2)	C(3)	D(4)
AST-1B	AS-03	Cosmic Background	13	x			
AST-1C	AS-05	Adv Radio Astronomy	12			x	
AST-3	SO-03	Solar Physics Mission	21		x		
AST-9A	HE-11	Focusing X-ray Telescope - 1.2 M	24		x		
AST-9B	HE-01	Focusing X-ray Telescope - 3.0 M	24		x		
PHY-1A	HE-07	Small High Energy Observatory	13	x			
PHY-1B	AP-01	Upper Atmospheric Explorer	19		x		
PHY-1C	AP-02	Medium Altitude Explorer	14			x	
PHY-2A	AP-04	Gravity & Rel - Earth Orbit	16	x			
EO-3A	EO-8	Earth Observatory Satellite	21		x		
EO-4A	EO-9	Sync Earth Observatory Satellite	9			x	
EO-6	EO-12	TIROS	22		x		
EO-7	EO-7	Sync Meteorological Satellite	17				x
EOP-3	OP-07	SEASAT-B	16		x		
EOP-4	OP-01	Geopause	9			x	
EOP-07	OP-04	GRAVSAT	12	x			
NND-1	CN-51	International Comm	19				x
NND-2A	CN-52	U.S. Domestic - A	10			x	
NND-2B	CN-53	U.S. Domestic - B (ADV)	19				x
NND-2D	CN-58	U.S. Domestic - C (TDRS)	13			x	
NND-3	CN-54	Disaster Warning	12			x	
NND-4	CN-55	Traffic Management	12			x	
NND-5	CN-56	Foreign Communication	10			x	
NND-6	CN-59	Communication R&D/Proto	24				x
NND-8	EO-56	Environmental Monitoring Satellite	27				x
NND-11	EO-61	Earth Resource - LEO	16	x			
NND-12	EO-59	Earth Resource - Geosync	15			x	
NND-13	EO-62	Earth Resource - Foreign	14			x	
NND-14	OP-08	Global Earth & Ocean Monitoring	18		x		

Notes: (1) A: 16 SRU/without SEPS
(2) B: 28 SRU/without SEPS
(3) C: 16 SRU/with SEPS
(4) D: 28 SRU/with SEPS

12. RELIABILITY

12.1 INTRODUCTION

Fundamental to this study was the derivation of reliability characteristics for each entity subject to failure. This involved the development of functional flow block diagrams for each module and the assignment of failure rates.

Traditionally, most failure analyses have been conducted assuming constant failure rates at the component level. By assigning a failure rate value, λ, to each component of a module, the reliability of that module can be analytically determined in various ways depending upon how the module components are connected.

The use of a constant failure rate at the component level assumes the occurrence of only random failures which follow an exponential distribution. Hence, it is assumed that infant mortality failures must be eliminated from consideration by requiring that the components have been burned in before use and that wear-out failures have been eliminated from consideration by requiring that the design life be very much less than the wear-out life.

In the case of a module consisting of a network of components containing redundancy, the reliability versus time curve can be approximated by a Weibull distribution, and this is a major assumption in this study.

12.2 DATA SOURCE

Either reliability or functional flow block diagrams for the individual modules were obtained from subsystem specialists. For those modules where only functional flow diagrams were available, some interpretation was required to develop the corresponding reliability block diagrams.

Constant failure rate values for the module components were obtained from References 1 and 23 from consultation with cognizant personnel.

12.3 WEIBULL FIT CALCULATIONS

12.3.1 Derivation

Assuming that the random variable module time-to-failure, t, follows the Weibull probability density function

$$f(t) = \frac{\beta}{\alpha} \left(\frac{t}{\alpha}\right)^{\beta-1} e^{-\left(\frac{t}{\alpha}\right)^{\beta}}$$

it then follows that module reliability is given by the expression

$$R(t) = e^{-\left(\frac{t}{\alpha}\right)^{\beta}}$$

where α and β are parameters to be estimated. The procedure used to estimate α and β was to calculate the analytical module reliability, $R'(t)$, using constant failure rates for individual components, at one-month intervals for the design life of a module. These reliabilities were then transformed by

$$\ln\left[\ln\left(\frac{1}{R'(t)}\right)\right] = -\beta\ln(\alpha) + \beta\ln(t)$$

i.e.,

$$y_t = a + bx_t$$

where

$$y_t = \ln\left[\ln\left(\frac{1}{R'(t)}\right)\right]$$

$$a = -\beta\ln(\alpha)$$

$$b = \beta$$

and

$$x_t = \ln(t)$$

This is a linear equation in a and b, and estimates of a and b, say $\hat{a}$ and $\hat{b}$, were obtained by the method of least squares, yielding

$$\hat{b} = \frac{\sum_t (x_t - \bar{x})(y_t - \bar{y})}{\sum_t (x_t - \bar{x})^2}$$

and

$$\hat{a} = \bar{y} - \hat{b}\,\bar{x}$$

Having obtained a and b, α and β are transformed back to obtain the estimates

$$\hat{\alpha} = e^{-\frac{\hat{a}}{\hat{b}}}$$

$$\hat{\beta} = \hat{b}$$

12.3.2 Example

Consider the Hot Gas Propulsion (N_2H_4) Module (AVCS-8) of Table 4-3 . Here the analytical reliabilities are given by the expression:

$$R'(t) = \left[1 - \left(1 - e^{\lambda_D t}\right)^2\right]\left[1 - \left(1 - e^{-\lambda_E t}\right)^2\right]\left[e^{-\left(\lambda_A + \lambda_B + \lambda_C + \lambda_D + \lambda_H + \lambda_I + \lambda_J + \lambda_K + \lambda_L\right) t}\right]$$

Computing this reliability at one-month intervals over the ten-year module design life and applying the least squares estimation procedure to these values yields the following estimated Weibull parameters:

$$\hat{\alpha} = 14.3454 \text{ years}$$
$$\hat{\beta} = 1.0218$$

Figure 12-1 illustrates the fit of the estimated Weibull reliabilities

$$\hat{R}(t) = e^{-\left(\frac{t}{\hat{\alpha}}\right)^{\hat{\beta}}}$$

to the analytical reliabilities.

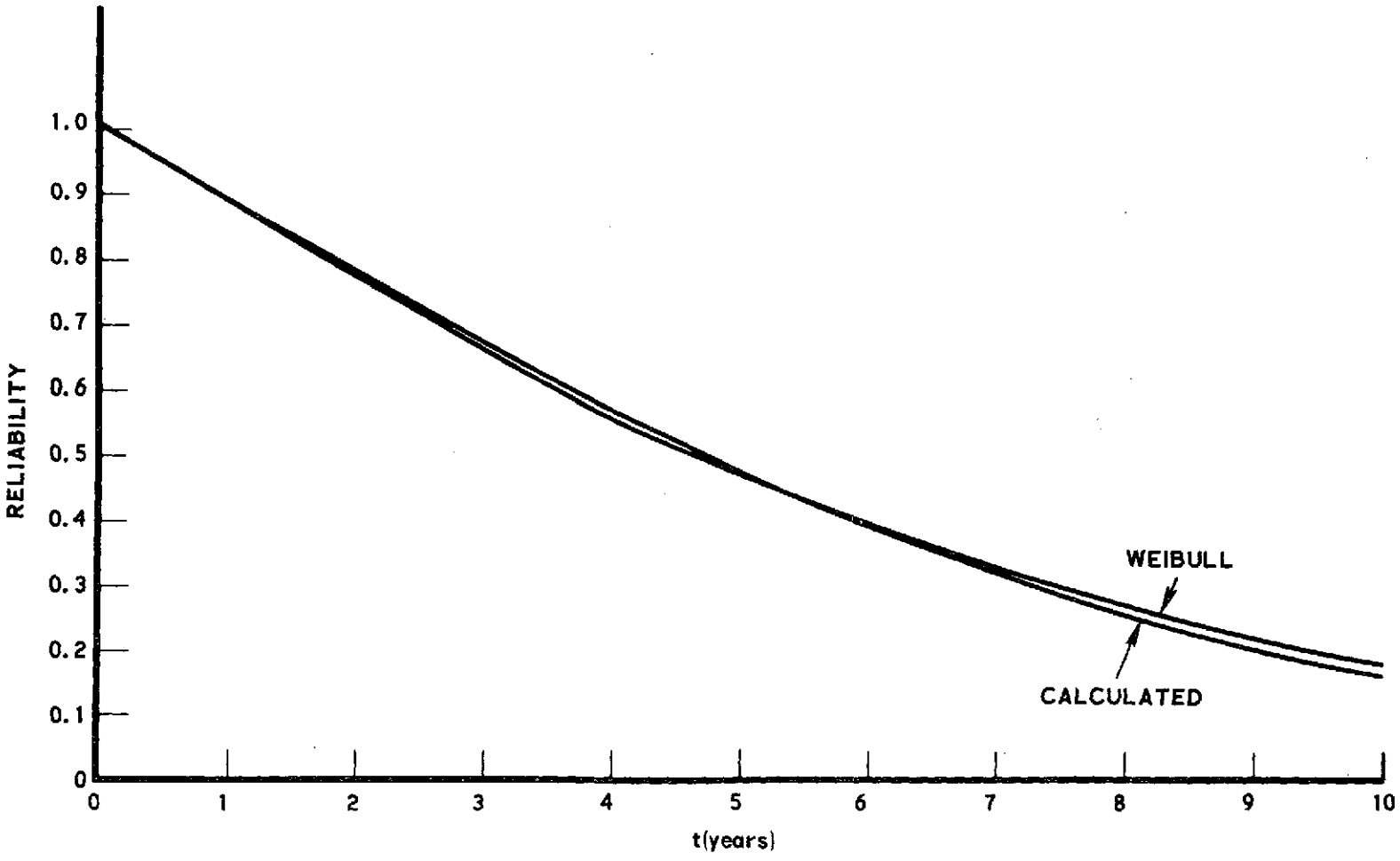


Figure 12-1. Reliability Estimation (Weibull Fit: AVCS-5)

13. WEIGHT ANALYSIS

13.1 WEIGHT REDUCTION

A major guideline for this effort was that maximum use would be made of earlier studies. In particular, it was proposed that the modular structure, temperature control, and electrical distribution weights determined in the Space Serviced Defense Support Program (SSDSP) study (Ref. 5) would be used in an effort to reduce the resources necessary to complete the study. For this reason, the weights listed in Reference 4 were utilized and no attempt was made to modify them to suit the individual characteristics of the new satellites.

Because of the above guideline, the weights quoted are somewhat heavier than necessary and several opportunities exist to reduce the weight. Some of these are listed below:

a. Reduce the Width of the Replaceable Modules. As noted above, the module baseplates were designed for the DSP study. In that program, the average equipment weight carried by each of the eight SRU s was 67 kg. However, the 127 SRU s in the present study carry an average equipment load of 50 kg. This suggests that, either more equipment should be carried on each SRU, or that the size of the SRU should be reduced. The structural and temperature control subsystem weight of each SRU as determined for the DSP satellite is noted below:

Item	Weight (kg)
Baseplate	10.5
Mechanism	4.5
Electrical System	2.3
Temperature Control	4.5
TOTAL	21.8

From the above, it can be deduced that if the width of each baseplate is decreased by 33 percent, the weight could be reduced by 3.5 kg. Since the satellites carry from 10 to 27 SRU s (with an average of 17 SRU s), it appears that this would lower the average weight of a satellite by 3.5 kg/SRU x 17 SRU s or 59.5 kg.

b. Combine Equipment Onto Fewer Modules. A second weight reduction might be realized if the equipment now carried on two SRU s were combined in one SRU. A case in point would be to carry two attitude control units on one module. The AVCS-1 or -2 modules (reaction wheels) carry equipment weights of only 5.9 or 9.6 kg, but require 33 kg of baseplate and electrical integration equipment weight. Also the AVCS-4, -5A, or -6 modules (sensors) carry equipment weights of 11.4, 21.9, or 39.6 kg of equipment. Any two of these sets of equipment could be combined on one SRU (instead of two) to produce a weight reduction of 33 kg per satellite. An examination of the weight list will suggest other weight reducing combinations such as data processing and TT&C or guidance and control. This procedure tends to increase the resupply weight however, since a failure in one subsystem will require a new module containing both subsystems. Also the effect on reliability and total cost effectiveness cannot be assessed at this time.

c. Re-size Electrical System on Each SRU to Suit Equipment Carried on that SRU. In the DSP study the following electrical items were carried on each SRU:

Item	Weight (kg)
Electrical Harness & Integration	8.2
Circuit Breakers	0.5
Converters	1.8
Automatic Load Control	0.5
Connectors	0.4
TOTAL	11.4

This equipment was required on each SRU because the electrical bus from the solar array supplied unregulated power to each SRU. Obviously, with the large variation in equipment carried on the various SRU s in this study and particularly in light of the fact that some SRU s carry a very low amount of equipment, it appears that if each SRU were treated on an individual basis, some weight reduction would be possible. If a 25 percent reduction could be obtained in the electrical harness and integration weight, then 2 kg x 17 SRU s or 34 kg could be saved on an average satellite.

d. Reduce Number of Attitude Control Modules. It may be possible to lower vehicle weight by reducing the number of attitude control modules from four to three on some satellites. Standard sized propellant tanks are used on all vehicles, i.e., 61 cm diameter on the larger units and 38cm diameter on the

smaller. Because of this standardization procedure, some tanks which are larger than necessary are off-loaded and in a number of cases, the required weight of propellant could be carried in three tanks. This would make it possible to eliminate one attitude control module with a weight saving of 50 kg per satellite. There are disadvantages to this plan, such as a reduction in the ability to control the satellite since one set of thrusters has been eliminated and the lever arm for the remaining thrusters has been deceased. However, if the benefits to be obtained from the possible weight reduction are great enough, it should be considered.

e. Conduct a Weight Reduction on Subsystem Weights. The subsystem weights were derived to a large extent by the cognizant personnel participating in the study; therefore, a final weight reduction should be possible if a careful review is made with the emphasis being on lower redundancy and decreased weight. The most lucrative field appears to be the attitude control subsystem since this area represents the greatest amount of weight.

13.2 RESULTS

The results of the weight analyses are summarized in Figures 13-1 and 13-2. The items discussed above will clearly reduce satellite weight. However, their implementation requires more time and effort than is available at this time. Even so, it is recommended that they be considered in any future studies.

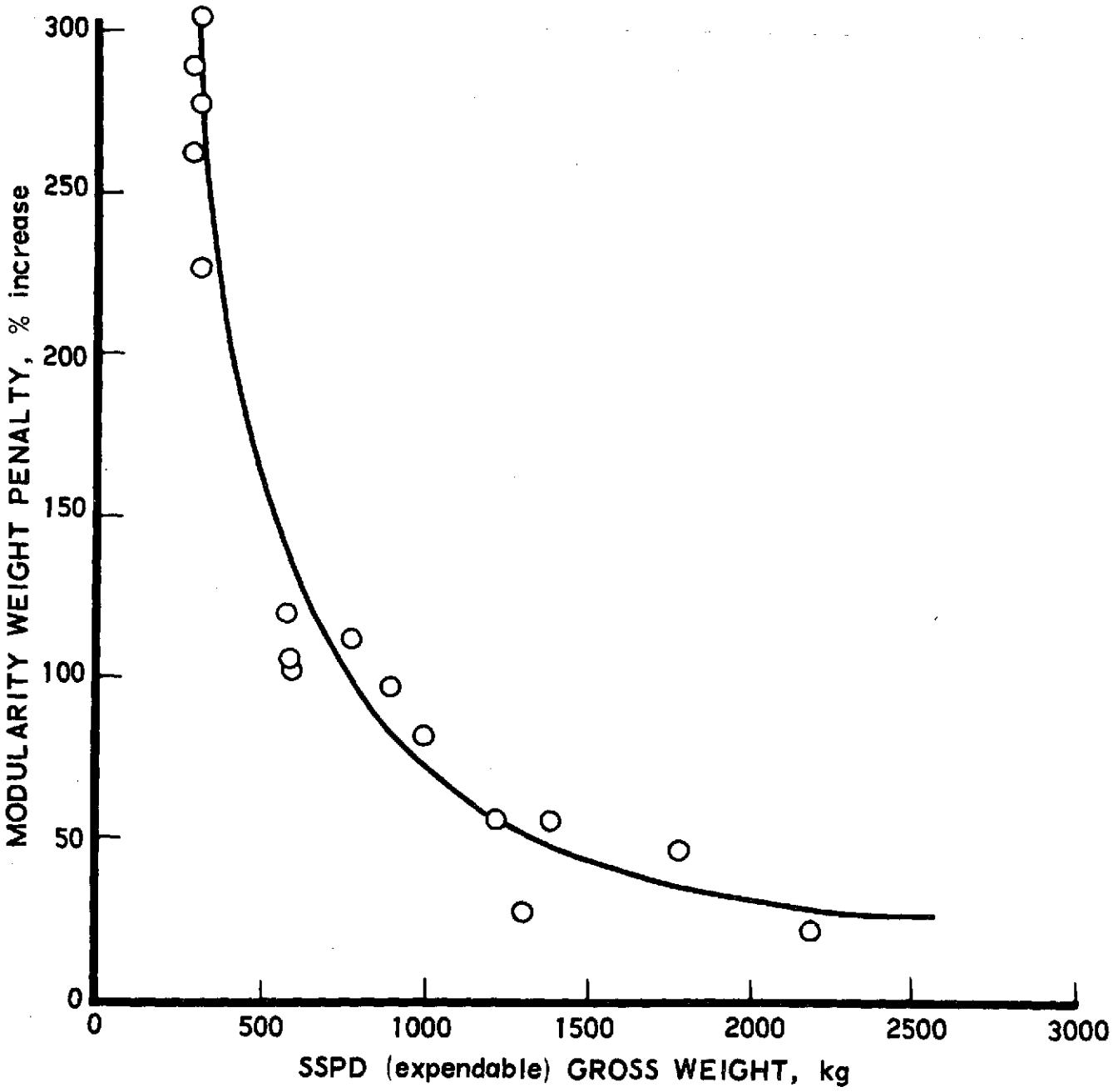


Figure 13-1. Satellite Modular Weight Penalty vs Expendable Gross Weight

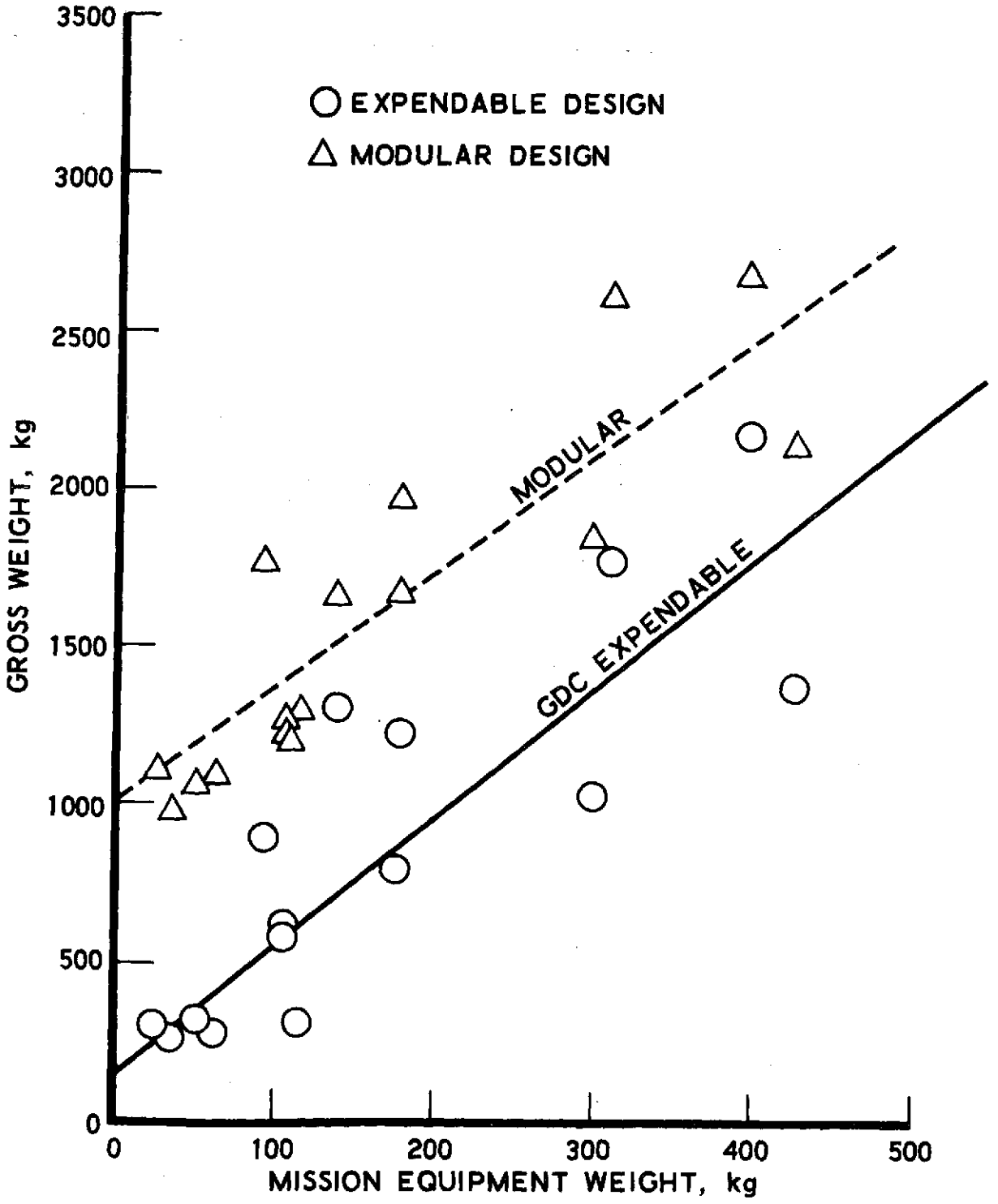


Figure 13-2. Gross Weight vs Mission Equipment Weight

14. CONCLUSIONS AND RECOMMENDATIONS

Final conclusions concerning the utility or cost effectiveness of space servicing cannot be drawn until the data developed in this report is used to exercise the space operations computer simulation program. However, the following tentative conclusions can be drawn:

a. Space servicing of satellites in orbits ranging from low earth to geosynchronous appears to be applicable to a broad spectrum of NASA automated payloads. In this study, 42 NASA payloads were identified as candidates for space servicing.

b. All 42 payloads are adaptable to the same basic space-servicing concept.

c. A basic (modularized and standardized) service unit is adaptable to service all the 42 payloads.

d. Servicing of payloads offers flexibility in replacement of mission equipment with short lifetimes, replacement for block changes and allows demonstration of experimental equipment on orbit rather than in ground simulation.

e. The point design of the EOS demonstrates the feasibility of adapting all the candidate missions to the space-servicing concept utilized in this study.

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