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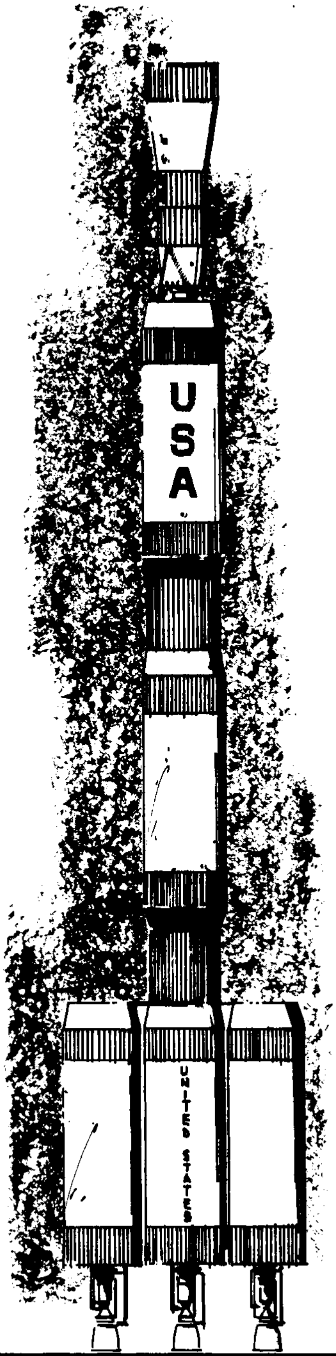

INTEGRATED MANNED INTERPLANETARY SPACECRAFT CONCEPT DEFINITION

Volume I

D2-113544-1

Summary

The BOEING Company • Aerospace Group • Space Division • Seattle, Washington

NASA CR-66558

INTEGRATED MANNED INTERPLANETARY SPACECRAFT CONCEPT DEFINITION

FINAL REPORT

VOLUME I SUMMARY

D2-113544-1

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LANGLEY RESEARCH CENTER

Hampton, Virginia

NASA CONTRACT NAS1-6774

January 1968

AEROSPACE GROUP THE BOEING COMPANY SEATTLE, WASHINGTON

FOREWORD

This study was performed by The Boeing Company for the National Aeronautics and Space Administration, Langley Research Center, under Contract NAS1-6774. The Integrated Manned Interplanetary Spacecraft Concept Definition Study was a 14-month effort to determine whether a variety of manned space missions to Mars and Venus could be accomplished with common flight hardware and to define that hardware and its mission requirements and capabilities. The investigation included analyses and trade studies associated with the entire mission system: the spacecraft; launch vehicle; ground, orbital, and flight systems; operations; utility; experiments; possible development schedules; and estimated costs.

The results discussed in this volume are based on extensive total system trades which can be found in the remaining volumes of this report. Attention is drawn to Volume II which has been especially prepared to serve as a handbook for planners of future manned planetary missions.

▼ Primary Discussion

● Summary or Supplemental Discussion

DOCUMENTATION \ STUDY AREAS	MISSION ANALYSIS	Trajectories and Orbits	Mission and Crew Operations	Mission Success and Crew Safety Analysis	Environment	Scientific Objectives	Manned Experiment Program	Experiment Payloads and Requirements	DESIGN ANALYSIS	Space Vehicle	Spacecraft Systems	Configurations	Subsystems	Redundancy and Maintenance	Radiation Protection	Meteoroid Protection	Trades	Experiment
Volume I/D2-113544-1 Summary Report	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●
Volume II/D2-113544-2 System Assessment and Sensitivities		●	●	●	●	●	●	●				●	●					
Volume III/D2-113544-3 System Analysis Part 1 - Missions and Operations	▼	▼	▼	▼	▼													
Part 2 - Experiment Program			▼			▼	▼	▼										
Volume IV/D2-113544-4 System Definition		●	●	●					▼	▼	▼	▼	▼	▼	▼	▼	▼	▼
Volume V/D2-113544-5 Program Plans and Cost																		
Volume VI/D2-113544-6 Cost Effective Subsystem Selection and Evolutionary Development																		

The final report is comprised of the following documents, in which the individual elements of the study are discussed as shown:

Volume	Title	Part	Report No.
I	Summary		D2-113544-1
II	System Assessment and Sensitivities		D2-113544-2
III	System Analysis	Part 1--Missions and Operations	D2-113544-3-1
		Part 2--Experiment Program	D2-113544-3-2
IV	System Definition		D2-113544-4
V	Program Plans and Costs		D2-113544-5
VI	Cost-Effective Subsystem Selection and Evolutionary Development		D2-113544-6

The accompanying matrix is a cross-reference of subjects in the various volumes.

Space Acceleration Systems	Primary Propulsion --Nuclear	Secondary Propulsion --Chemical	System and Element Weights	IMIEO Computer Program	Earth Orbit Operations and Assembly Equip.	Earth Launch Vehicles	Facilities	System Trades	Space Acceleration--Earth Launch Vehicle	Space Acceleration Commonality	Space Vehicle--Artificial Gravity	SYSTEM AND PROGRAM ASSESSMENT	System Capability	Design Sensitivities	Program Sensitivities	Adaptability to Other Space Programs	Impact on Other Space Programs	Technology Implications	Future Sensitivity Studies	Program Schedules and Plans	Test Program	Facilities Plan	Program Cost	Cost Effective Subsystems
●	●	●	●		●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	
	●	●	●		●	●	●					▼	▼	▼	▼	▼	▼	▼	▼	●	●	●	●	
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ABSTRACT

This volume summarizes the Integrated Manned Interplanetary Spacecraft Concept Definition study. It recommends a common vehicle that is capable of accomplishing the majority of manned mission opportunities to orbit Venus and land on Mars over a Mars synodic cycle from 1975 to 1990. The recommended system is an all-nuclear space acceleration system and a basic spacecraft consisting of a biconic Earth entry module; a mission module which provides the living quarters, vehicle control, and experiment laboratories; and an Apollo-shaped Mars excursion module. The entire space vehicle is placed in Earth orbit by six launches of an uprated Saturn V launch vehicle.

CONVERSION FACTORS

English to International Units

Physical Quantity	English Units	International Units	Multiply by
Acceleration	ft/sec^2	m/sec^2	3.048×10^{-1}
Area	ft^2	m^2	9.29×10^{-2}
	in^2	m^2	6.45×10^{-4}
Density	lb/ft^3	Kg/m^2	16.02
	lb/in^3	Kg/m^2	2.77×10^{4}
Energy	Btu	Joule	1.055×10^{3}
Force	lbf	Newton	4.448
Length	ft	m	3.048×10^{-1}
	n.mi.	m	1.852×10^{3}
Power	Btu/sec	watt	1.054×10^{3}
	Btu/min	watt	17.57
	Btu/hr	watt	2.93×10^{-1}
Pressure	Atmosphere	$Newton/m^2$	1.01×10^{3}
	lbf/in^2	$Newton/m^2$	6.89×10^{3}
	lbf/ft^2	$Newton/m^2$	47.88
Speed	ft/sec (fps)	m/sec	3.048×10^{-1}
Volume	in^3	m^3	1.64×10^{-5}
	ft^3	m^3	2.83×10^{-2}

ABBREVIATIONS

A.U.	Astronomical unit
bps	Bits per second
C/O	Checkout
CM	Command module (Apollo program)
CMG	Control moment gyro
CONJ	Conjunction
CSM	Command service module (Apollo program)
ΔV	Incremental velocity
DSIF	Deep Space Instrumentation Facility
DSN	Deep Space Network
$\oplus$	Earth
ECLS	Environmental control life support system
ECS	Environmental control system
EEM	Earth entry module
ELV	Earth launch vehicle
EMOS	Earth mean orbital speed
EVA	Extravehicular activity
FY	Fiscal year
fps	feet/sec
GSE	Ground support equipment
IBMC	Inbound midcourse correction
IMIEO	Initial mass in Earth orbit
IMISCD	Integrated Manned Interplanetary Spacecraft Concept Definition
I_{sp}	Specific impulse
IU	Instrument unit
KSC	Kennedy Space Center
λ'	Ratio of propellant weight to overall propulsion module weight
LC	Launch complex
LC-34 & -37	Launch complexes for Saturn IB
LC-39	Launch complex for Saturn V
LH_2	Liquid hydrogen
LO	Long
LO_2 or LOX	Liquid oxygen
LRC	Langley Research Center

ABBREVIATIONS (Continued)

LSS	Life support system
LUT	Launch umbilical tower
♂	Mars
MEM	Mars excursion module
MIMIEO	Minimum initial mass in Earth orbit
MM	Mission module
MODAP	Modified Apollo
MSC	Manned Spacecraft Center (Houston)
MSFC	Marshall Space Flight Center (Huntsville)
MTF	Mississippi Test Facility
NAC	Letters designate the type of acceleration systems First letter--Earth orbit depart Second--planetary deceleration Third--planet escape Example: NAC = Nuclear Earth depart/aerobraker deceleration at planet/chemical planet escape
OBMC	Outbound midcourse correction
OPP	Opposition
OT	Orbit trim
P/L	Payload
PM-1	Propulsion module, Earth orbit escape
PM-2	Propulsion module, planet braking
PM-3	Propulsion module, planet escape
RCS	Reaction control system
SA	Space acceleration
S/C	Spacecraft
S-IC	First stage of Saturn V
S-II	Second stage of Saturn V
SH	Short
SOA	State of art
SRM	Solid rocket motor
S/V	Space vehicle
SWBY	Swingby

ABBREVIATIONS (Continued)

T/M	Telemetry
TVC	Thrust vector control
VAB	Vehicle assembly building
♀	Venus
V_{HP}	Hyperbolic excess velocity

1.0 INTRODUCTION

One of the major questions before the nation is what should the planning be after the Apollo program? This area involves the currently approved Apollo Applications Program integrated with possible follow-on space station programs and with interplanetary exploration programs. It is important to accomplish this planning as soon as possible in order to derive optimum commonality and therefore benefit from the NASA Research and Advanced Technology Program.

Relative to the interplanetary portion of the National Space Program plan, certain basic data are needed. The three most important categories of these basic data lie in the areas of (1) definition of the scientific and engineering measurements, (2) definition of the proper mix of unmanned and manned missions for a logical acquisition of the desired scientific and engineering data, and (3) the selection of an integrated approach to designing the hardware for a flexible manned interplanetary mission system. It is principally toward the last of these three areas of planning data development that this study was directed, with a considerable contribution to the first also being involved.

OBJECTIVES

The broad objective of the Integrated Manned Interplanetary Spacecraft Concept Definition Study was to examine the possibility of accomplishing a variety of manned space missions to the near planets using a common set of mission hardware. The specific objectives of the study were:

- To conceptually design interplanetary space vehicle systems suitable for accomplishing manned missions to land on Mars and orbit Venus and to define the missions and mission modes that can be accomplished with such space vehicle capabilities;
- To establish realistic performance requirements and operating characteristics for the spacecraft and its subsystems and to identify critical development and performance problems for technology advancement;
- To define experiments and performance parameters that will guide the planning for the Apollo applications and possible follow-on space station programs to ensure that they will contribute in an optimum manner to the evolution of the manned planetary capabilities;
- To define from the viewpoint of conceptual space vehicle design desirable characteristics for possible Saturn V uprating, post-Saturn launch vehicle requirements, and other advanced technology programs as well as for any orbital operations development and precursor planetary probe activities;

- To define the possible development schedules and estimated costs associated with the space vehicle design.

BACKGROUND

Over the past decade several manned interplanetary mission studies have been performed. The early studies were concentrated on particular missions or classes of missions such as the high-energy opposition class missions and the low-energy long duration conjunction class mission. In general, these studies investigated mission requirements, mission modes, systems, and spacecraft designs suitable for accomplishing specific missions. The more recent studies indicated that it is possible to select mission modes and mission opportunities for both Mars and Venus in such a manner that the range of performance requirements is small enough that it is practical to accomplish a large number of missions with a single space vehicle concept.

It was logical then that this study be performed to examine the various types or classes of missions for the opportunities over a typical Mars synodic cycle in relationship to the hardware alternatives to develop a single space vehicle concept.

2.0 SCOPE AND APPROACH

The initial study guidelines were provided by the NASA at the beginning of the contract. The typical Mars synodic cycle for mission analyses was specified for 1975-1990.

To respond properly to the study objectives, consideration was given to the entire interplanetary mission system and not just to the development of a spacecraft conceptual design. Figure 2-1 identifies the major elements of the overall interplanetary mission system. Each element was examined through analyses to make design selections and develop recommended procedures, program schedules, costs, and reliability and safety provisions.

INTERPLANETARY MISSION SYSTEM

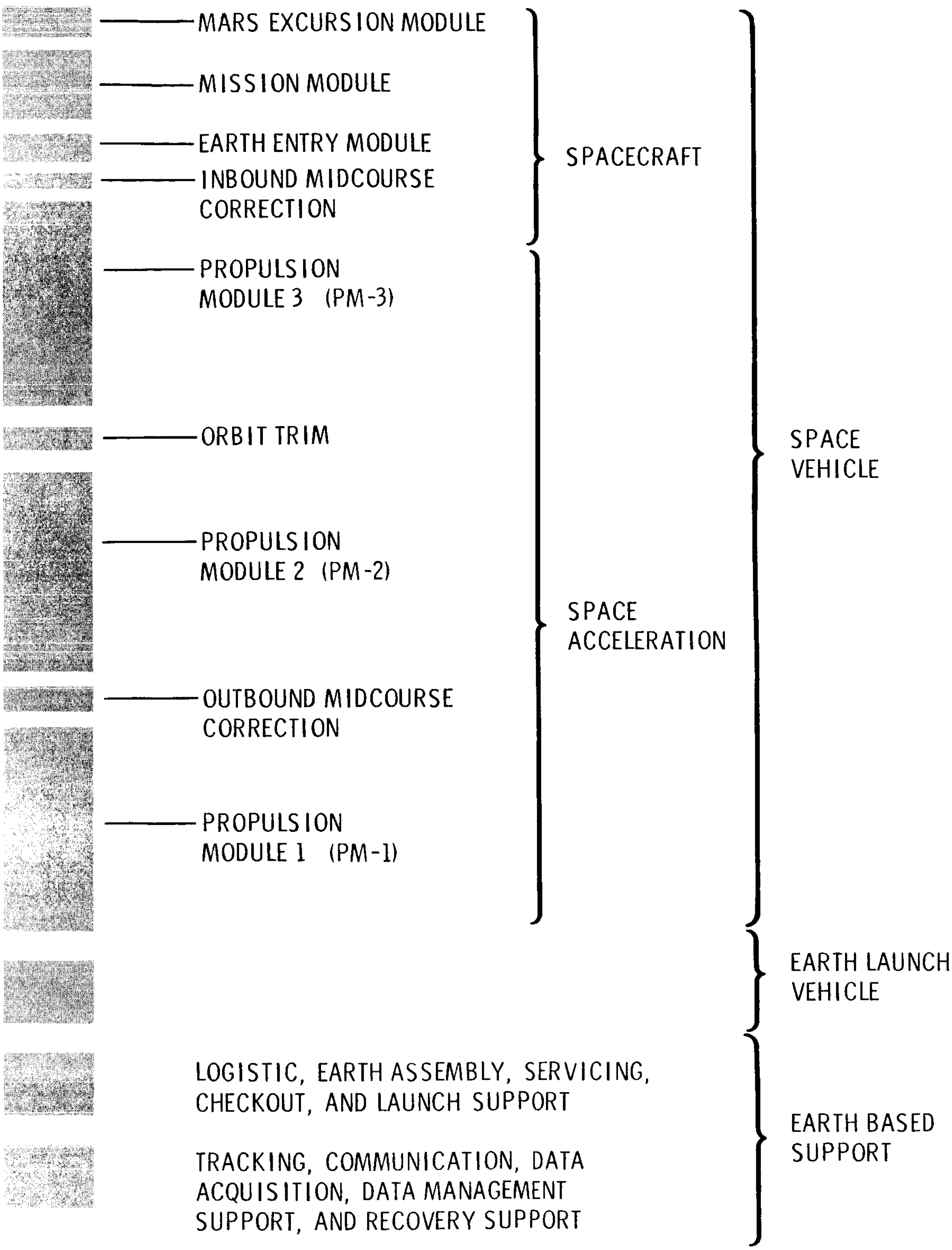


Figure 2-1

A science experiment program was developed around the basic question of the origin and evolution of the solar system. This basic question was expanded into five experiment categories which were then defined in terms of 49 specific classes of observations and measurements. For each of these, the present knowledge base was established.

An unmanned precursor program, through 1975, was laid out for those measurements that can be made with an unmanned system. Their contribution was estimated and added to the present knowledge base.

The science program was then defined for the manned missions and specific payload requirements were generated for the transit phase, the planet orbit phase and the Mars surface phase. A number of unmanned probes and orbiters were included to complement the manned activities. In addition specific probes were included to provide information for final site selection for landing on Mars, both to ensure a productive mission and the safety of the crew. Finally, an assessment was made of the expected scientific return for three manned flights to Mars and two to Venus. The results for Mars are shown in Figure 2-2, which displays the present knowledge base for each of the five categories of experiments, the estimated contribution of the unmanned precursor program and the knowledge return from three manned missions to Mars.

Similar data developed for Venus shows somewhat less total accomplishment, due principally to the fact that surface exploration was restricted to the use of unmanned landers.

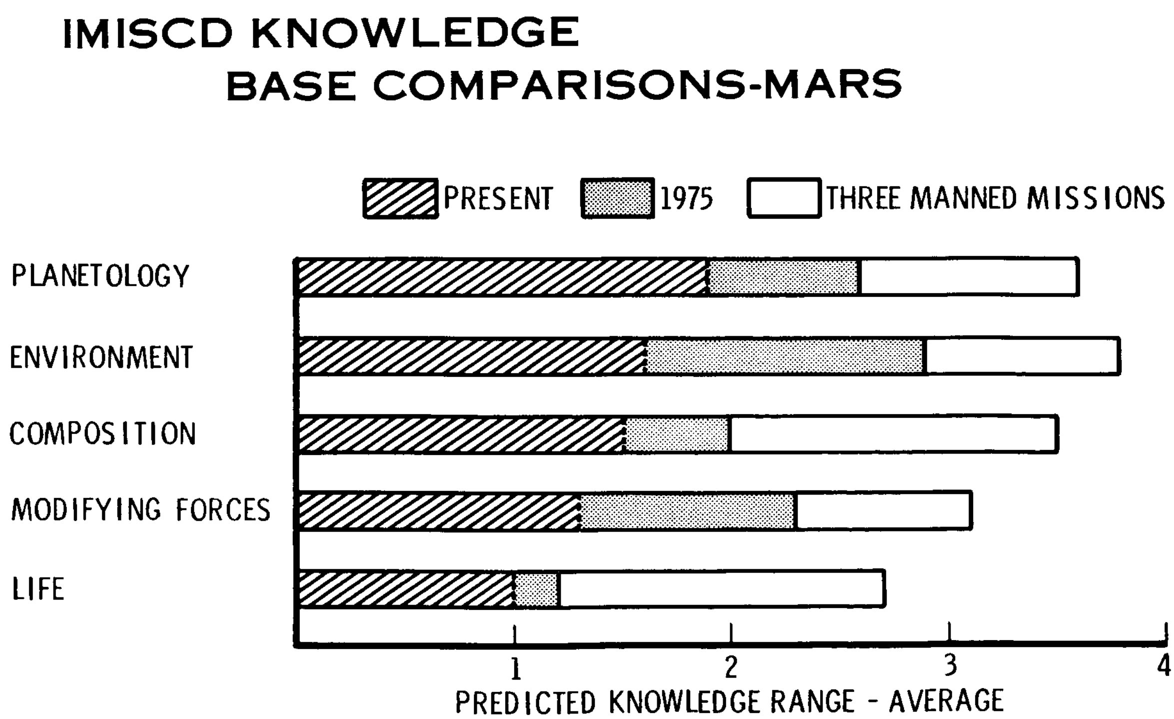


Figure 2-2

A group of missions was selected that spanned the range of energy requirements of the various Mars and Venus opportunities during the typical Mars synodic cycle from 1975 to 1990. The energy requirements for the opposition class missions between 1975 and 1980 are excessive; therefore, only opposition class missions between 1982 and 1990 were considered. The planetary stay time was fixed at 40 days for all of the mission trajectories except for the Mars conjunction class and Venus long stay time missions which have stay times of about 500 days. Requirements shown in Table 2-1 were derived by developing trajectories that provided the minimum initial mass in Earth orbit for each mission in the representative group.

Table 2-1: SELECTED MISSION PARAMETERS

ΔV_1	11,700 - 16,700 fps (3565-5080 m/sec)
ΔV_2	6,950 - 17,400 fps (2120-5300 m/sec)
ΔV_3	6,320 - 19,000 fps (1925-5790 m/sec)
Earth Entry Velocity	38,000 - 60,200 fps(11,600-18,320 m/sec)
Total Mission Time	460 - 1,040 days
Planet Stay Time	40 - 580 days
Minimum Sun Distance	0.50 - 1.00 A.U.
Maximum Earth Distance	0.67 - 2.70 A.U.

Those requirements with the greatest impact on the spacecraft design were the Earth entry velocity and mission time. The distribution of the ΔV requirements and their wide variation between mission opportunities had the strongest influence on the space acceleration system.

In light of the wide variation of mission requirements coupled with such uncertainties as the environment, experiment system requirements, and possible requirement for artificial gravity, it appeared highly desirable to have a system that was relatively insensitive to unpredictables, tolerant to changes, and flexible enough to accomplish the majority of the missions. This suggested that the system should be designed either for the worst-case parameters and take the penalties in the system or designed for the minimum requirements and provide for incremental capabilities as necessary.

With these thoughts in mind, the *spacecraft*, which is made up of a mission module (the living and work area for the crew), an Earth entry module, a Mars excursion module, and experiment accommodation, was broken down to identify those elements basic to all missions and associated requirements. The best overall configuration was then developed by examining all incremental requirements to establish whether their provisions should be a part of the basic vehicle, common to all missions, or whether they should be added for specific missions.

In the development of the *space acceleration system*, various types and combinations of propulsion system-Earth launch vehicles were studied to determine the best combination when considering safety, success, cost, weight in Earth orbit, program risk, complexity, and utilization in other programs. During this study, 105 space vehicle and launch vehicle combinations were examined through design analysis and cost estimation. The space acceleration candidates considered were nuclear propulsion, chemical propulsion, and aerodynamic braking (at the target planet).

Earth launch vehicles ranging from an uprated Saturn V (300,000-pound payload into a 262-nautical-mile orbit---136,000-kilogram payload into a 485-kilometer orbit) to a Post-Saturn capable of placing the entire space vehicle into Earth orbit (approximately 2 to 4 million pounds---0.9 to 1.8 million kilograms) were considered. A detailed analysis was made to determine the impact and necessary modification to the KSC facilities for each of the space acceleration-Earth launch vehicle combinations.

The requirement for gravity to maintain the crew for extended time periods is an unknown. Whether this requirement is valid or not may not be answered before this system is committed. The approach to this problem was to design the basic system for zero-gravity operation and to provide for simple modification for artificial gravity if it is found a necessary requirement.

3.0 RESULTS AND CONCLUSIONS

Previous studies and initially this study used the initial mass in Earth orbit, which can be grossly equated into cost as a figure of merit. It proved useful for identification of desirable candidate systems, but even a detailed cost analysis could not provide a means for the selection of the recommended system. The final selection could only be made when the remaining candidates were evaluated for their relative insensitivities to change and tolerance to unknowns.

The spacecraft, space acceleration system, Earth launch vehicle, and system impact on the launch facilities were found to be the major elements that had to be studied in some depth for the selection of the recommended interplanetary mission system.

It was found that the spacecraft could be analyzed relatively independently of the other system elements. Figure 3-1 shows the evolution of this element. The Apollo shape reaches its Earth entry velocity capability around 45,000 fps (13,720 m/sec); therefore, it must be augmented with a retropropulsion system to accommodate the higher entry velocities. It was found by examining other vehicle concepts that the biconic* design can successfully enter the Earth's atmosphere for velocities up to 65,000 fps (19,800 m/sec). It was also determined that the development and recurring costs for this vehicle were competitive with a six-man Apollo-type shape with a retropropulsion system.

The Apollo shape and a lifting body shape were compared for Mars entry and landing. The Apollo shape was the lighter of the two and was more tolerant to crew size and experiment variations. This shape also lends itself to unmanned precursor probes while providing valuable engineering data for the development and testing of this configuration for the manned system.

It was determined that the environmental control system and power radiators and the communication system should be designed for the most stringent mission requirements because of the difficulty of incorporating increment capabilities. The effects of increased meteoroid shielding, expendables, and system spares were considerable. Therefore, there are provisions to accommodate incremental loading of these items when necessary for the longer duration missions.

Basic experiment laboratories and sensors common to all missions are integrated into the mission module. Those experiment sensors and probes peculiar to certain missions are located in a separate module that can accommodate a wide variation of requirements.

It was found that the space acceleration system and the Earth launch vehicle were interdependent and proper selection could only be made when they were considered together.

*LMSC Document 4-05-65-12, *Study of Manned Vehicles for Entering the Earth's Atmosphere at Hyperbolic Speeds,* NASA Contract NAS2-2526, Lockheed Missiles and Space Co., November 1965.

An initial selection of an all-nuclear acceleration system in conjunction with an uprated Saturn V Earth launch vehicle with a capability in the range of 500,000 to 800,000 pounds (227,000 to 363,000 Kg) in Earth orbit was made. Although this selection was judged best from the initial evaluating criteria, it was considerably off optimum when the individual stages were sized to accommodate the maximum impulses found for the various missions. There were several trajectory modifications that provided some relief, particularly the Mars mission that swings by Venus, but this improved the situation only for a few missions. As shown in Figure 3-2, a modular approach was then considered where three different sized propulsion modules were used in various combinations and multiples to semitailor the acceleration system for each mission. This showed considerable promise but required increased testing for the development of three modules and their arrangements. This approach also had the problem of matching the three different modules to a single Earth launch vehicle. A search to circumvent these shortcomings revealed a concept

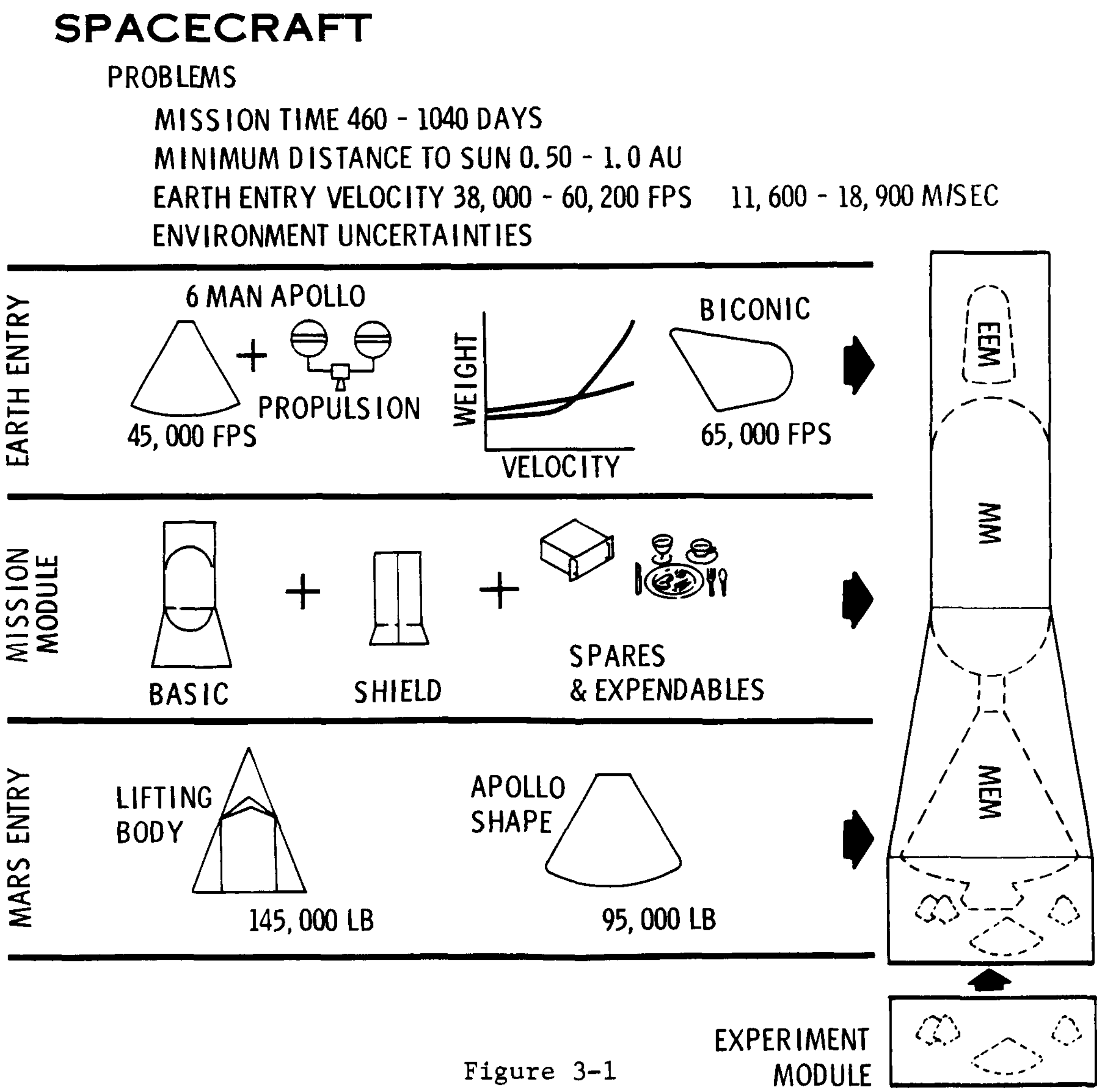


Figure 3-1

that was insensitive to the wide variation of ΔV requirements between impulse events and could be tailored to always take advantage of the Earth launch vehicle's capability. This space acceleration concept utilizes common propulsion modules with a capability of transferring propellant between the modules to accommodate the varying impulse requirements for the different missions. The total mission ΔV requirement was examined for the various missions, and it was found that an acceleration system is made up of identical modules--three for injecting into the interplanetary trajectory (PM-1), a single module for braking into the planet orbit (PM-2), and a single module for injecting into the trans-Earth trajectory (PM-3). The propellant is transferred from PM-2 to PM-1 during the Earth departure acceleration. If additional propellant is required for deceleration into the planet, propellant is transferred from PM-3 to PM-2.

By transferring propellant from the upper to the lower propulsion module, the majority of the desired missions can be performed. For many of the missions, capability for additional payload is provided.

A cost analysis of an acceleration system that uses the common propulsion module versus a less flexible one that uses semitailored modules showed that the slightly greater recurring costs for the common module approach were offset by the additional testing and development costs for the semi-tailored concept.

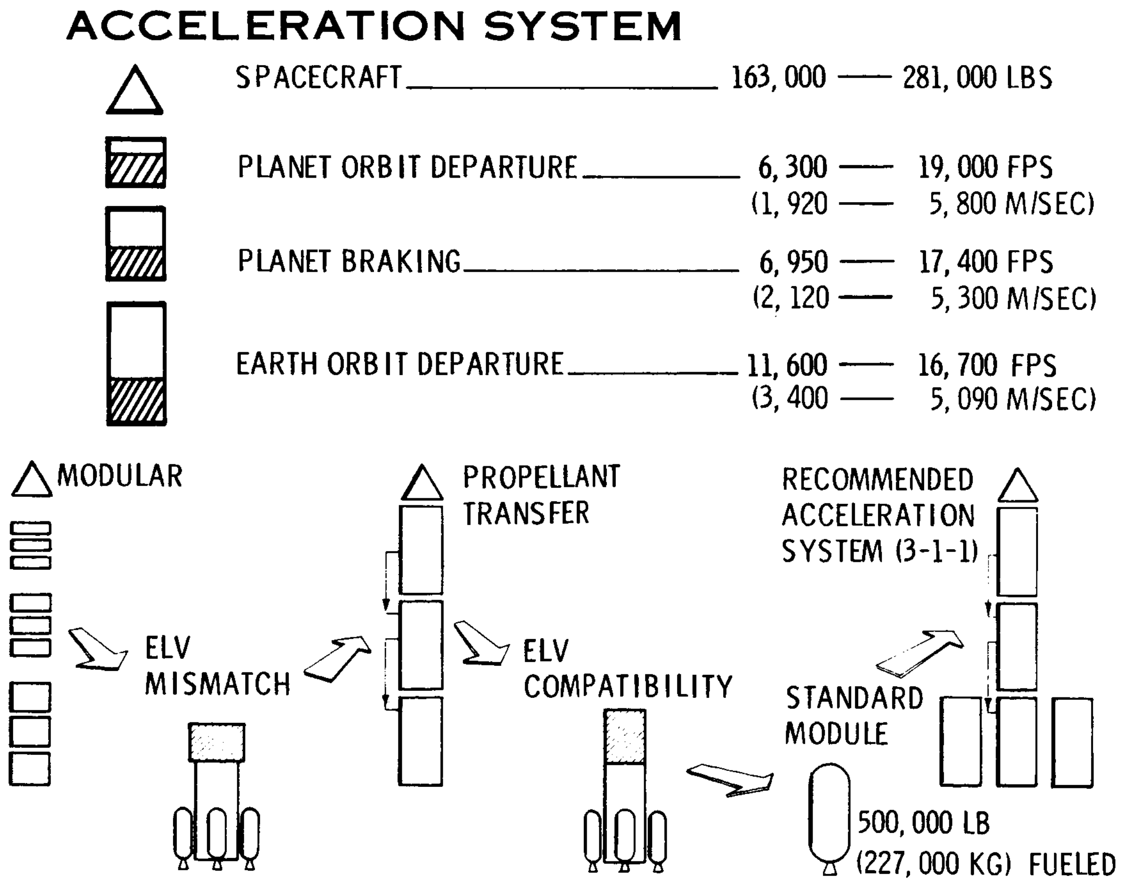


Figure 3-2

Table 3-1 shows that the recommended space vehicle with a 3-1-1 space acceleration train can accomplish missions to Mars and Venus during each opportunity over a Mars synodic cycle. Missions to Venus can be repeated in the 1990's, but are not tabulated in the Figure. Also, Mars conjunction missions with stay times of about 500 days can be repeated at each Mars opportunity and Venus long stay time missions are available at each Venus opportunity.

Since the recommended system is not tailored specifically to each mission, the amount of discretionary performance capability varies on all missions. Figure 3-3 shows how this performance margin can be used to add payload going into the planets or leaving, or both. It shows lines of payload capability which trade payload returned to Earth against payload to orbit at the target planet. Also shown are the design payload points. The difference between the design point for each mission and the capability line when measured along the ordinate gives additional payload that can be delivered into the planet orbit, additional probes for example, if payload returned to Earth remains at the design value. When the difference is measured along the abscissa, the result is additional payload returnable to Earth if the payload into the planet orbit is held at the design value. Table 3-2 shows some examples of additional capability that might be considered in using the available discretionary payload.

MISSION CAPABILITY

LAUNCH DATE	DESTINATION	MISSION TYPE	DURATION (DAYS)
NOV 1978 NOV 1979 MAR 1980	MARS MARS VENUS	VENUS SWINGBY CONJUNCTION SHORT	680 900 460
OCT 1981 OCT 1981 NOV 1981	MARS VENUS MARS	OPPOSITION SHORT VENUS SWINGBY	540 460 600
MAY 1983 NOV 1983	VENUS MARS	SHORT VENUS SWINGBY	540 540
JAN 1984 NOV 1984	MARS VENUS	OPPOSITION SHORT	460 550
APR 1985 MAR 1986 AUG 1986	MARS MARS VENUS	VENUS SWINGBY OPPOSITION SHORT	590 480 470
MAY 1988 JUN 1988 JUL 1988 OCT 1989 DEC 1989	VENUS MARS MARS MARS VENUS	SHORT OPPOSITION VENUS SWINGBY VENUS SWINGBY SHORT	350 460 560 640 350
SEPT 1991 NOV 1994 DEC 1996 JAN 1998	MARS MARS MARS MARS	VENUS SWINGBY VENUS SWINGBY OPPOSITION VENUS SWINGBY	600 560 480 680

Table 3-1

DISCRETIONARY PAYLOAD

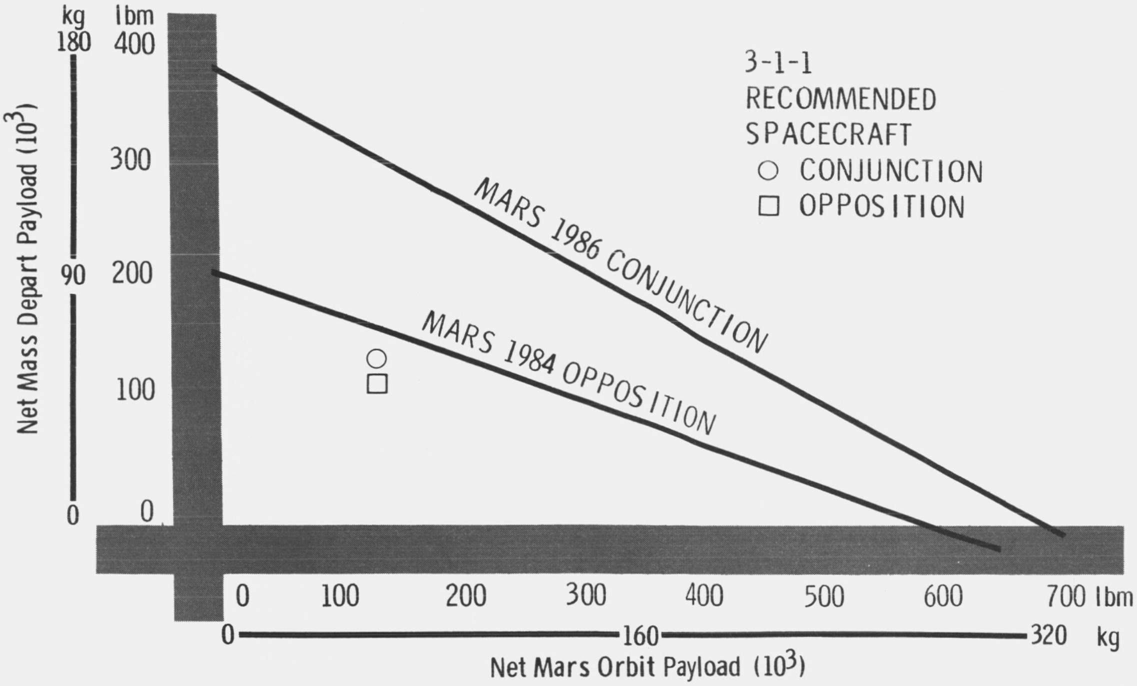


Figure 3-3

DISCRETIONARY PAYLOAD USES

Table 3-2

PARAMETER	OPP'84		CONJ. '86	
	DESIGN	CAPABILITY	DESIGN	CAPABILITY
CREW SIZE (MEN)	6	9	6	12
SPECIFIC IMPULSE (SEC.)	850	815	850	750
METEOROID PROBABILITY (P_0)	0.997	0.9995	0.997	0.9998
METEOROID FLUX $\frac{\text{FLUX}}{\text{(DESIGN FLUX)}}$	1.0	12.4	1.0	20.6
MEM WEIGHT (LB) (KG)	95,000 (43,100)	180,000 (81,700)	95,000 (43,100)	405,000 (204,000)
PROBES WEIGHT (LB) (KG)	24,000 (11,400)	109,000 (49,400)	24,000 (10,900)	334,000 (153,000)
MM + EEM WEIGHT (LB) (KG)	86,000 (39,000)	116,000 (52,700)	119,000 (54,000)	281,000 (127,500)
EXPERIMENTS WEIGHT (LB) (KG)	14,000 (6,350)	44,000 (20,000)	15,000 (6,800)	180,000 (81,700)

An artificial gravity provision can be added to the basic zero-gravity system by assuming it would be reasonable to operate up to 45 days in a nonrotating mode to accommodate experiments and mission impulse maneuvers and to operate in a rotating mode for the remainder of the mission.

- A recommended system with the following elements was conceived: an all-nuclear space acceleration system made up of five identical propulsion modules powered by a single 195,000-pound-thrust Nerva engine on each module; a spacecraft consisting of an Apollo-shaped Mars excursion module, a biconic-shaped Earth entry module, and a mission module designed to accommodate the long conjunction missions with the feature of offloading the unnecessary spares and expendables for the short duration missions; a basic experiment sensor complement and laboratories common to all missions and a module for accommodating the mission-peculiar experiment equipment; a six-man Apollo logistics vehicle launched by a Saturn IB (possible alternates are the Titan IIIC and the intermediate two-stage Saturn V); an uprated Saturn V launch vehicle capable of placing all elements of the space vehicle into Earth orbit with six launches.
- A new launch pad and relatively minor modifications to some of the existing KSC launch facilities are required.
- The total system cost is 29 billion dollars which includes the research and development and the first Mars and the first Venus mission costs.
- A sample schedule based on a go-ahead in 1972 places the first mission to Venus in 1983 and the second mission to Mars in 1986.
- The mission module and its subsystems can be used directly as an Earth-orbiting space station.
- The nuclear propulsion module has alternate applications as an upper stage for the Saturn V and a space acceleration stage for unmanned probes.

CONCLUSIONS

Several general and specific conclusions have been drawn from this study. Where considered appropriate, recommendations in connection with these conclusions are made.

- It is feasible and practical to accomplish a wide variety of missions to Mars and Venus utilizing a common set of hardware.
  Recommendation: NASA should adopt the concept of an integrated manned space vehicle system for future interplanetary mission program planning.
- There is a serious input-data deficiency for space exploration planning. A more detailed definition should be made of the scientific and engineering measurements necessary.
  Recommendation: NASA should initiate, as soon as practical, a study to develop an overall space exploration plan. It should start with the so-called "top-down" approach wherein the broad objectives and goals are defined stepwise through theories and measurements. From these the proper mix of unmanned and manned missions can be defined. Major outputs would identify overall manned program timing, planning for unmanned programs, and planning for future spacecraft and facility requirements.
- Of the space acceleration system candidates examined, the nuclear high-thrust propulsive system appears best because of relative cost and flexibility.

Recommendation: NASA should initiate a program to examine in detail the technological problems of the common module with fuel transfer.

- The payload capability range for an Earth launch vehicle to support the manned Mars and Venus exploration program is approximately 550 to 800 thousand pounds (250 to 370 kg). This capability is achievable through uprating of the present Saturn V launch vehicle; a new ELV development is not required. The uprated Saturn V-25(S)U, an uprated Saturn V core with four 156-inch solid rocket motors strapped on the first stage, was selected from the remaining candidates as the recommended system because it has the least impact on production, logistic, test, and KSC launch facilities.

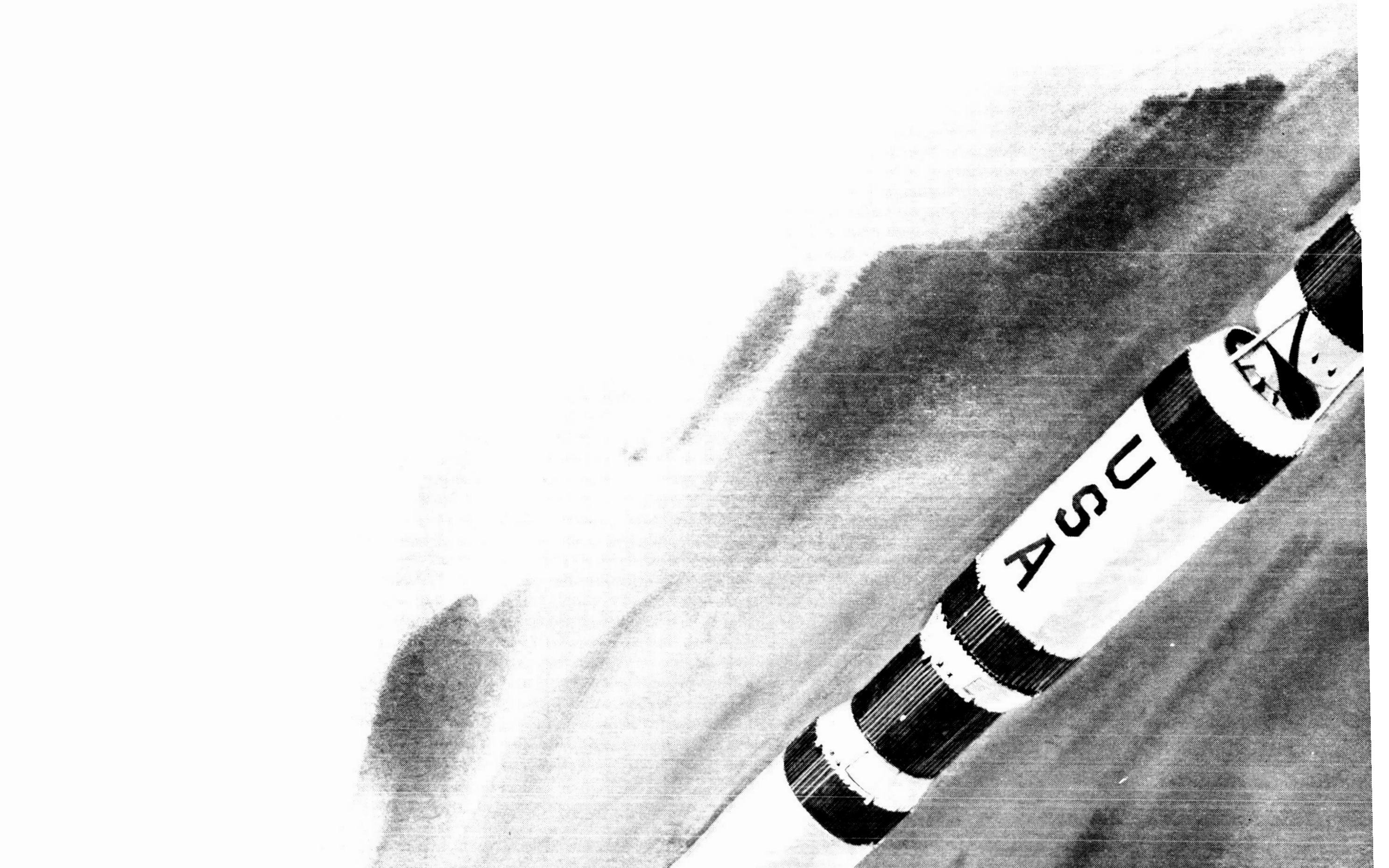
- A minimum crew of six men is necessary to operate the recommended system and perform a reasonable scientific exploration program. Detailed crew time and skill analyses of each mission phase showed that the Mars planet exploration phase was critical with an accumulated average crew requirement of 6.45 men. A high activity schedule and/or limiting the experiment program could reduce this average to six men which was adequate for other mission phases.

- In general, subsystems could not be selected from the desirable candidates when only weight-cost effectiveness trades were considered. Consequently, long-term operation, tolerance to performance requirement changes, and flexibility to adapt to other applications were found to be the most desirable characteristics, and it is recommended that the new technology developments and research should be directed to emphasize these characteristics. *Long-term operation* may be achieved by high reliability, maintenance, repair, or replacement. The proper balance of these must be determined with the subsystem's application in mind.

  Tolerance to performance change and adaptability to other applications suggest that the subsystems should be analyzed to determine their basic element which might be universal to many applications, then increasing its capability by modular additions. Many of the weight savings-cost studies have advocated multiple usage. The subsystem should be decoupled from the other systems in order to be the most adaptable to change or unknowns.

- Hardware sharing with precursor programs helps to provide some of the necessary qualification and test time. The mission module, including many of its subsystems development, could be phased to evolve from the Apollo applications program and a manned orbital space station program. The common propulsion module development could be phased to evolve from Earth orbital or lunar programs requiring a Saturn V nuclear upper stage and/or to evolve from an unmanned planetary space probe program in which the nuclear stage is used as a space acceleration system.

- A high degree of "go-ahead" date flexibility is afforded by the recommended system because the concept (1) allows accomplishment of missions in almost all opportunities for both Mars and Venus and (2) fits with a logical evolutionary development plan of system elements. Development of the propulsion and mission modules could begin today for use in other applications.


USA
PM-1

4.0 RECOMMENDED SYSTEM

The recommended system is a versatile system that is highly tolerant to uncertainties. For most missions, it has considerable discretionary capability to cope with expanded uses yet costs are competitive to a system optimized for a specific application.

SPACE VEHICLE

The recommended space vehicle, Figure 4-1, consists of the spacecraft and a space acceleration system.

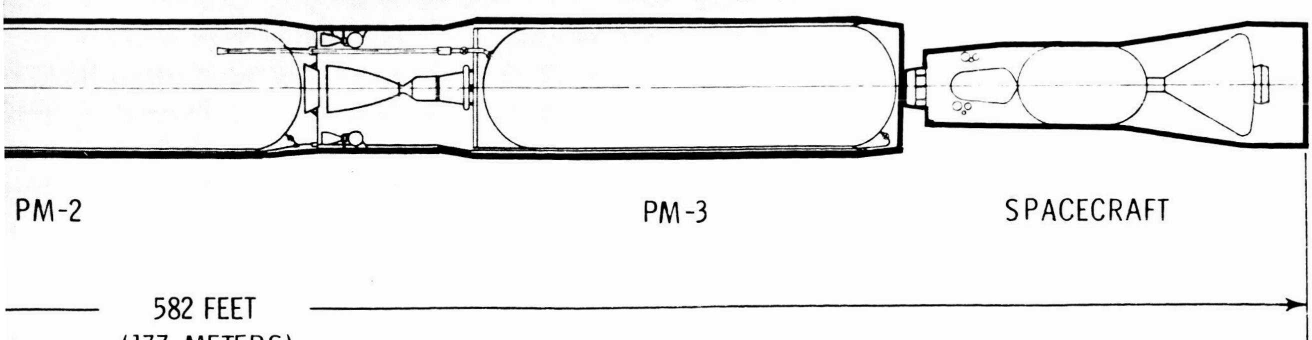


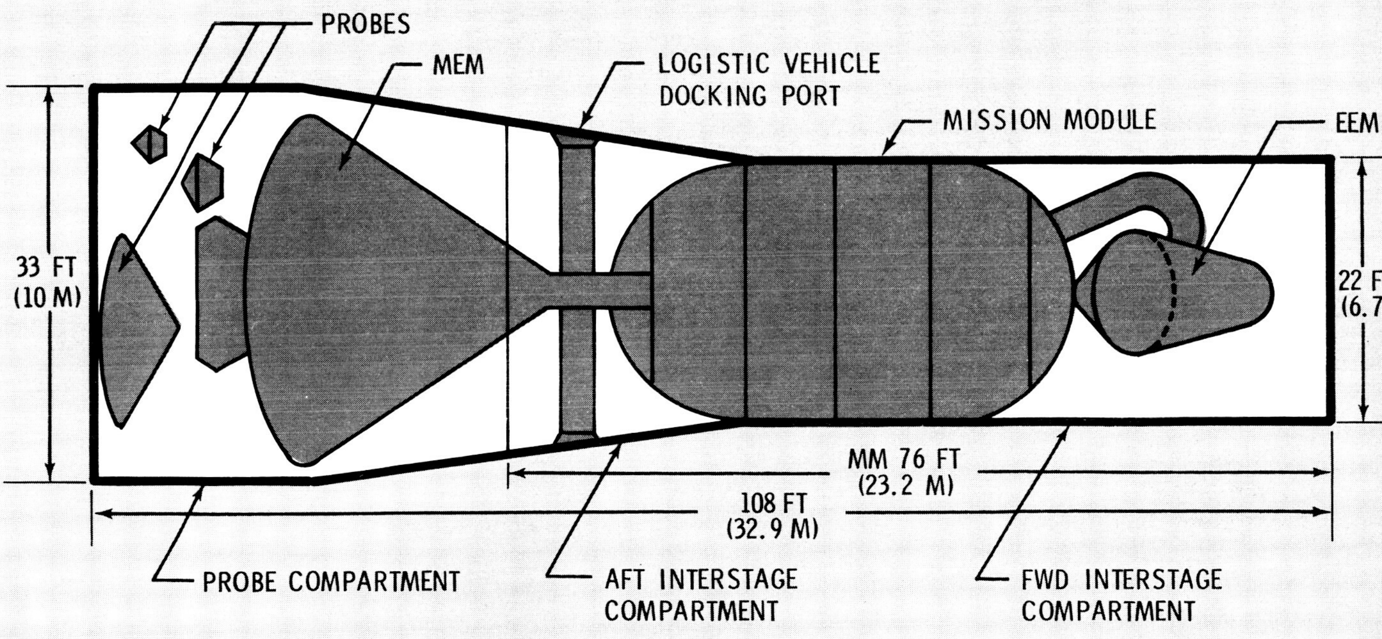
Figure 4-1

SPACECRAFT

Figure 4-2 shows the general arrangement of the spacecraft consisting of the mission module, a Mars excursion module, an Earth entry module, and a probe bay. These major elements are interconnected by pressurized tunnels allowing shirtsleeve passage between them.

The forward interstage compartment is an unpressurized area that contains the Earth entry module, the inbound midcourse correction propulsion system, some of the experiment sensors, and mission module equipment. A side hatch is installed in the mission module to the Earth entry module transfer tunnel to provide access to equipment in this area.

The aft interstage compartment is an unpressurized area containing the Mars excursion module, the airlock system, and some of the mission module and experiment equipment. The access tunnel to the Mars excursion module has a side hatch for access to equipment installed in this



WEIGHTS	MARS OPPOSITION 1984	MARS CONJUNCTION 1986	VENUS SHORT 1981
MM (INCLUDES EXPERIMENTS)	82,900 lbm	116,580 lbm	82,900 lbm
MEM	95,290	95,290	--
EEM	17,400	17,400	17,400
PROBES	24,480	24,480	37,610
INTERSTAGES	21,000	21,000	21,000
TOTAL	241,070 lbm (109,349 kg)	274,750 lbm (124,627 kg)	155,910 lbm (70,721 kg)

Figure 4-2

area. There are two connecting tunnels extending from the central transfer tunnel for pressurized transfer from logistic vehicles during Earth orbit operations.

The *mission module* is the control center for the entire space vehicle and provides a habitable living, operations, and experiment center for the mission crew. The basic mission module provides the environmental control and power system radiators and the communications system for the most stringent mission. Provisions are made for incremental loading of meteoroid shielding, expendables and system spares as necessary. The mission module contains all the subsystems necessary for life support, command and control functions, experiments analysis, and information transfer during the course of the mission. It is pressurized to a 7-psia ($48.23(10^3)$ Newton/m^2) oxygen-nitrogen atmosphere, providing a viable, shirtsleeve environment for the crew.

As shown in Figure 4-3, the crew compartment is a 22-foot-diameter (6.7 m) cylindrical pressure vessel with hemispherical heads and is divided into four decks. The upper deck contains the crew's personal quarters, dispensary, personal hygiene, and waste management systems. A pressure hatch and transfer tunnel located in the ceiling provides access to the Earth entry module.

A second deck includes the command and control center and dining and recreational areas. The command and control center includes the displays and controls for all subsystems, environment parameters, and vehicle operations. The control center is manned at all times except during high radiation periods. The dining area includes the food storage and preparation. The wash water/condensate water recovery unit of the waste management system is also located in this area. The recreation area is used for exercise, conferences, leisure activities, and the library and contains a storage area for subsystem spares. Food required for missions in excess of 500 days is also stored in this deck. Miscellaneous electronic equipment is located in a bay between the dining and recreation areas. A pressure hatch located in the floor leads to the radiation/emergency pressure shelter in the third deck. The radiation shelter is a 10-foot-diameter (3.048 m) compartment that provides quarters for the crew during periods of high radiation, and also serves as an emergency pressure compartment if loss of pressure is experienced in the overall mission module. This compartment will be occupied during nuclear propulsion system operation, while passing through the Van Allen belts, and during major solar flares. It also provides a 4-day emergency supply of food, water, and personal hygiene items. The shelter has a separate independent atmosphere supply and atmosphere control loops. The bulk of the radiation shielding is provided by a 20-inch thick (0.057 m) combination food and waste storage. The outer peripheral area contains a food storage compartment and the majority of the environmental control equipment.

The lower deck contains the experiment laboratories. It is divided into five specialized areas: optics, geophysics, electronics, bioscience, and science information center. The laboratory area is connected to the radiation shelter by a pressure hatch in the ceiling. A pressure hatch in the floor leads to the airlock and a pressurized access tunnel. The tunnel leads to the Mars excursion module, logistics spacecraft, or outside for extravehicular activity operations.

CREW COMPARTMENT

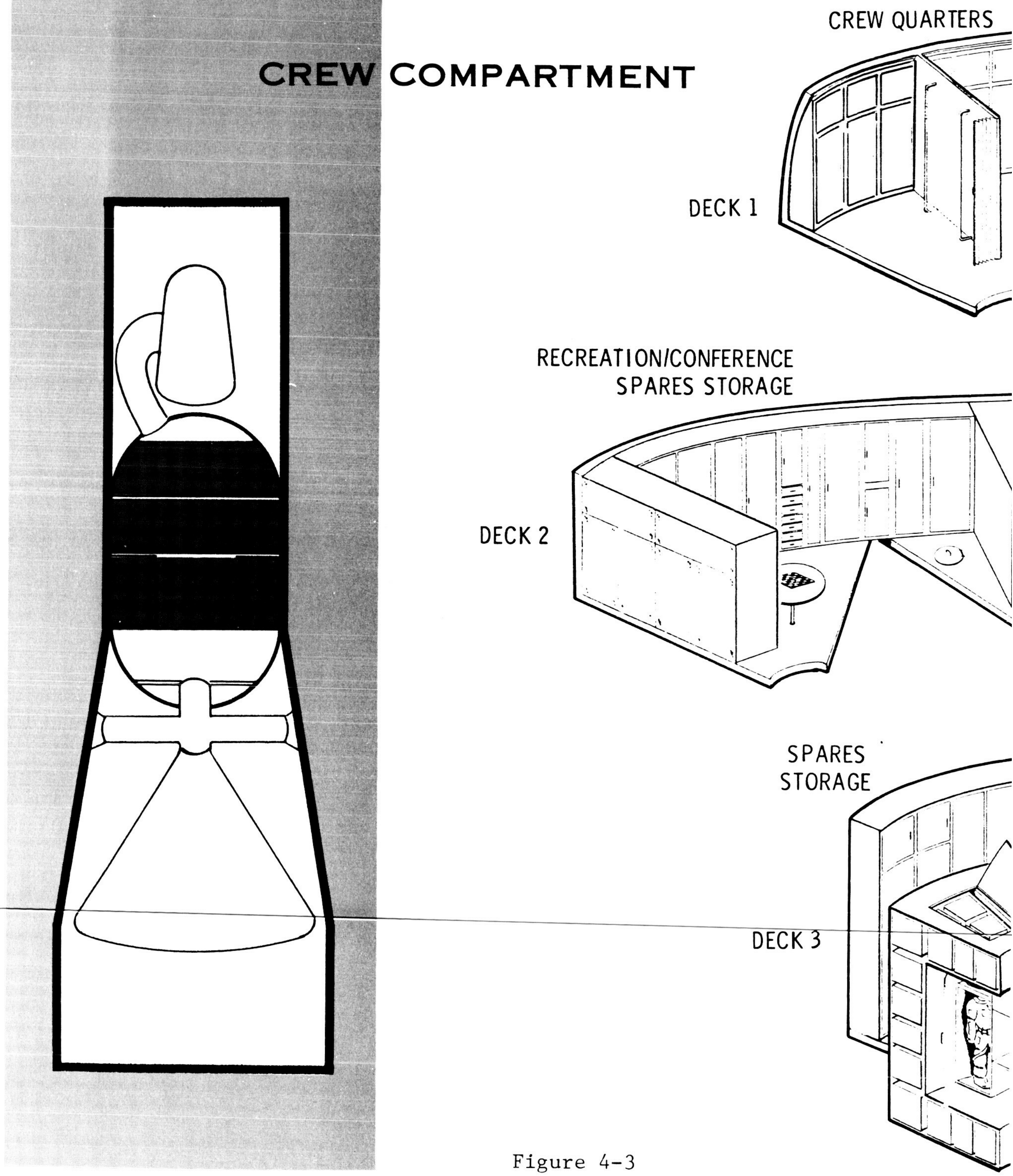
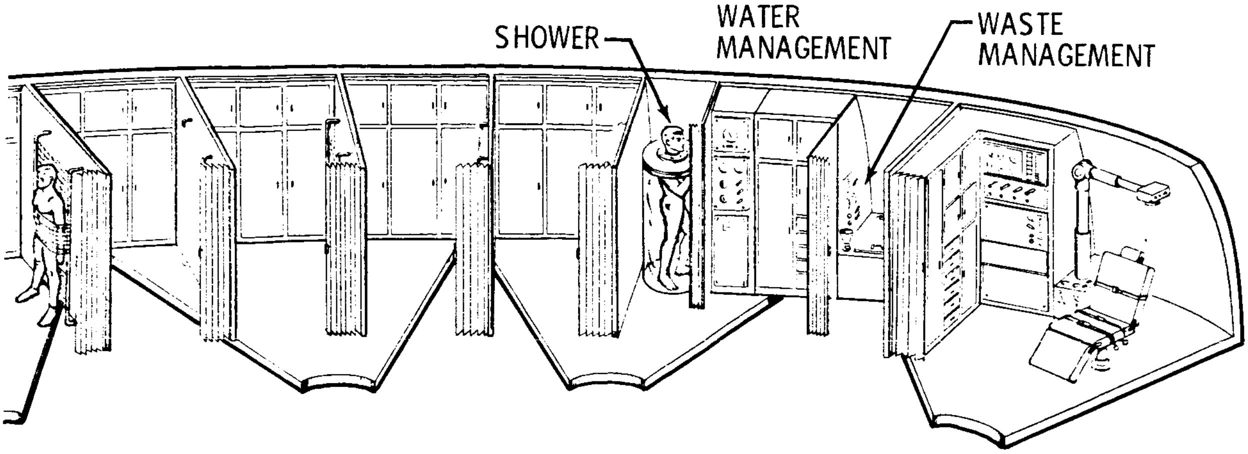
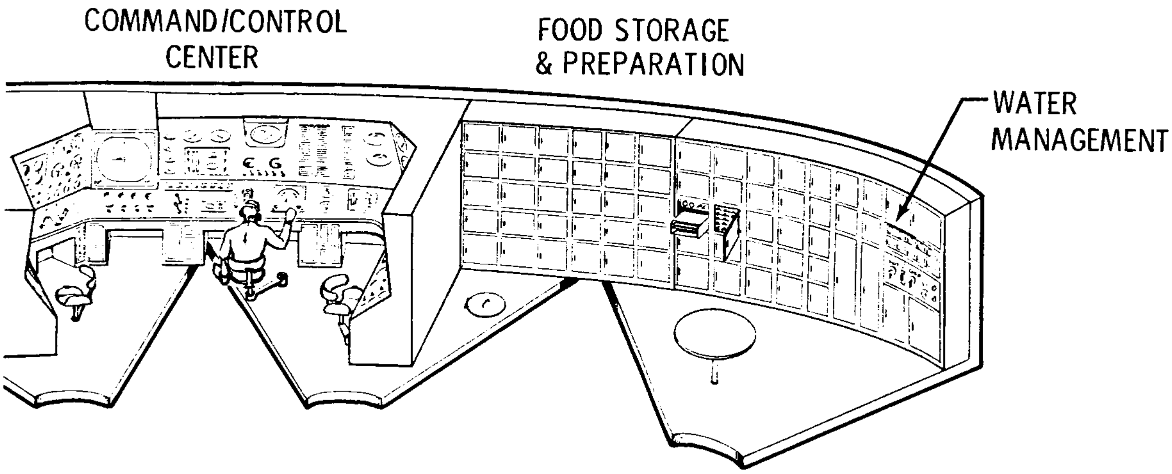
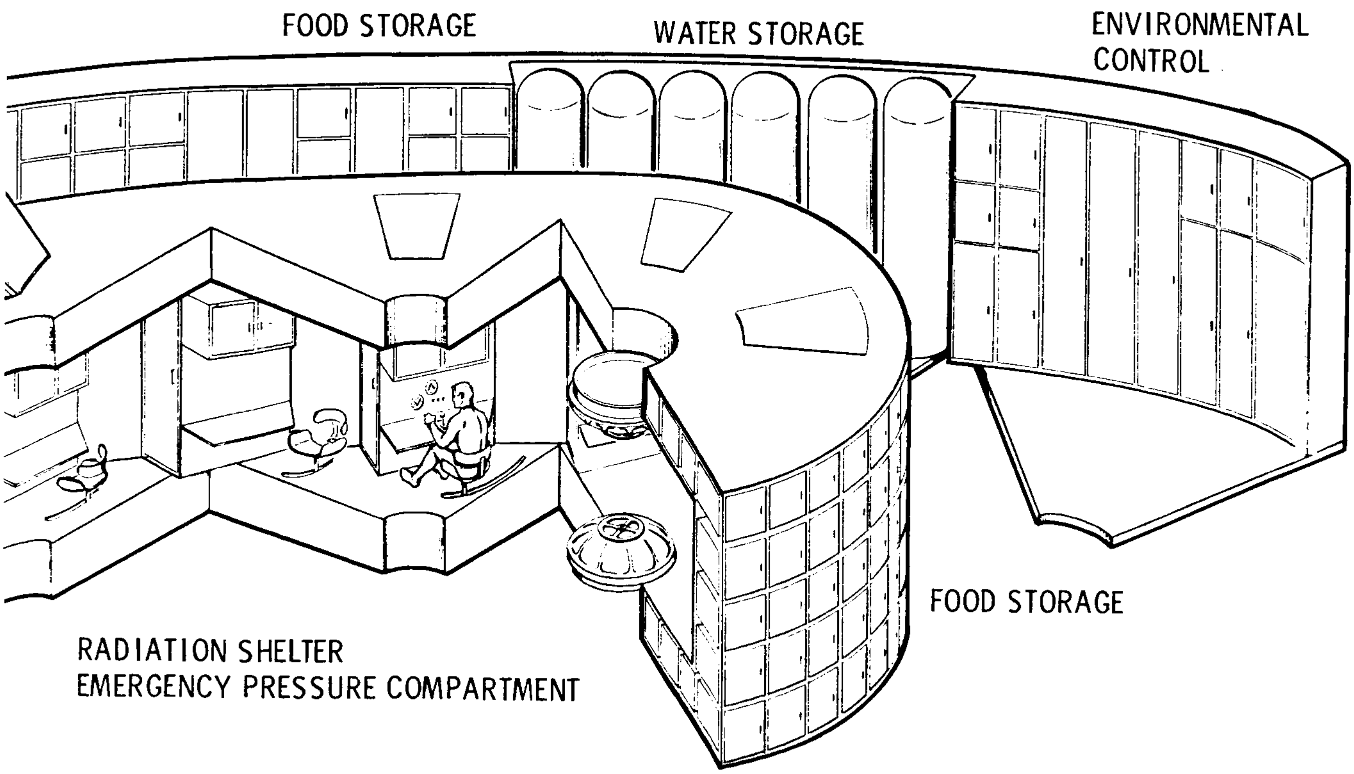


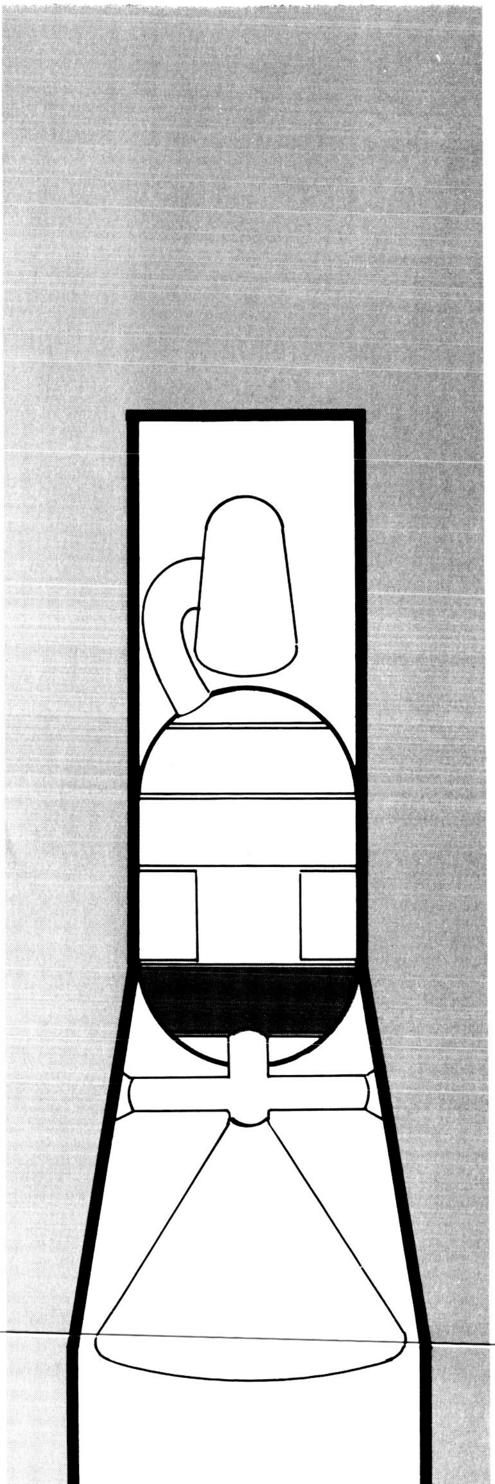
Figure 4-3


SHOWER
WATER MANAGEMENT
WASTE MANAGEMENT


COMMAND/CONTROL CENTER
FOOD STORAGE & PREPARATION
WATER MANAGEMENT


FOOD STORAGE
WATER STORAGE
ENVIRONMENTAL CONTROL
FOOD STORAGE
RADIATION SHELTER
EMERGENCY PRESSURE COMPARTMENT

EXPERIMENT LABORATORIES



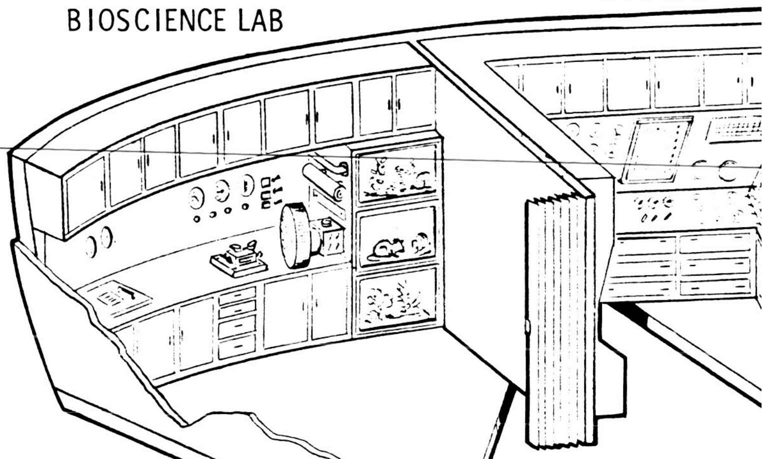


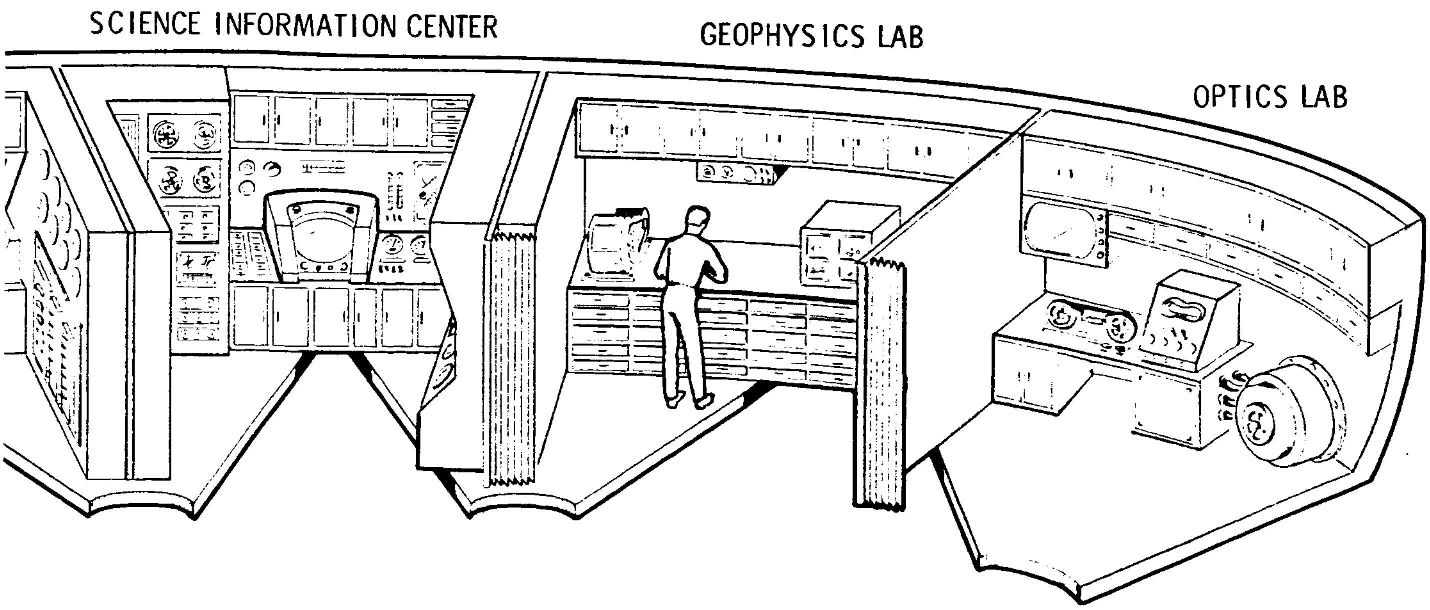
Figure 4-4

The *experiment accommodation* is centered around five laboratories that serve as control centers for conducting all on-board studies and analysis and probe experiments and operation. As shown in Figure 4-4, separate laboratories are devoted to bioscience, optics, geophysics, electronics, and a science information center. These laboratories occupy an entire deck of the mission module and are made up of 7000 pounds (3180 kg) of experiment equipment requiring approximately 2300 watts of power. The major experiment instruments located external to the laboratories are shown in Figure 4-5.

The scientific and engineering probe complement for both Mars and Venus missions are listed in Table 4-1.

Table 4-1: PROBE COMPLEMENT

Probes	Mars	Venus
Occultation Detector	X	
Upper Atmosphere Sounder	X	
Magnetometer	X	
Mars Moons	X	
Mapping Radar	X	X
Atmosphere Drifter Bioprobe		X
Cloud Data		X
RF Window		X
Soft Lander	X	X
Atmospheric (Hard Lander)	X	



UNMANNED PROBE INSTALLATION

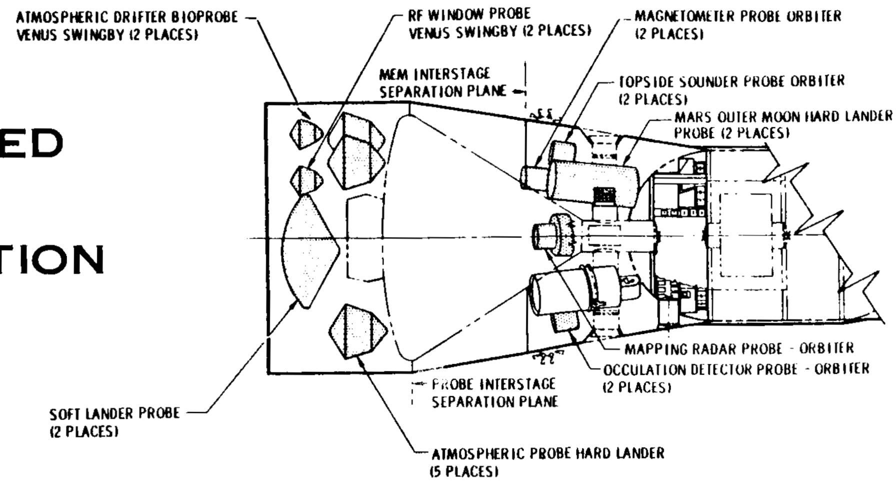


Figure 4-5

Probes for Mars missions weigh 22,255 pounds (9980 kg) and those for Venus 34,190 pounds (15,400 kg).

Installation of the probes for a Mars mission is shown in Figure 4-5. Probes to be launched from Mars orbit prior to the launching of the MEM are located in the aft section of the vehicle. The vacated volume, occupied by the MEM for Mars missions, accommodates the extra probe requirement for the Venus missions. Figure 4-6 shows the experiment accommodation.

EXPERIMENT ACCOMMODATION

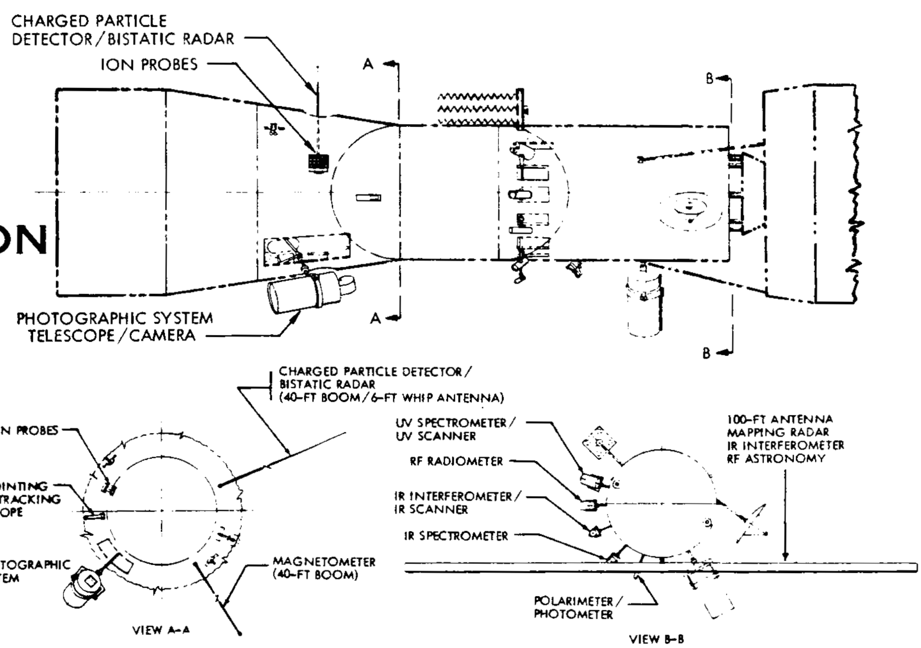


Figure 4-6

The *Mars excursion module* transports three of the crew members and equipment from the space vehicle in Mars orbit to the Mars surface. It provides living quarters and a laboratory during the 30-day stay on the Mars surface and transports the crew and scientific data and samples back to the orbiting space vehicle. An Apollo-shaped module, 30 feet (9.15 m) in diameter, was selected. This design was adapted from work performed by North American-Rockwell Corp.*

Its inboard profile is illustrated in Figure 4-7. The Mars excursion module consists of a descent and an ascent module. The ascent module houses the three-man crew during entry, descent, landing, and ascent. The ascent module consists of the control center, ascent engine, and propellant tanks. The first-stage ascent propellant is stored in eight conical tanks (five for oxidizer and three for fuel) outside the thrust structure. The second-stage ascent propellant is stored in two tanks between the engines and the ascent capsule control center.

The descent stage contains the crew living quarters and laboratory for use while on Mars, the descent engine and propellant tanks, ballutes, landing gear, supporting structure, an outer heat shield/structure, and the various subsystems. The crew quarters and laboratory are formed out of a segment of the toroidal lower part of the vehicle and are connected to the control center of the ascent module by airlocks and tunnels. Seven deorbit motors are arranged in a circle outside the heat shield. The descent propellants are housed in three spherical tanks. The descent and ascent engines are both pump-fed, gimbaled, plug nozzle engines and operate at a chamber pressure of 1000 psi. FLOX- methane propellants are used.

Surface operations include experiments and investigations directed toward increasing knowledge of Mars planetology, effects of modifying forces on Mars, its composition, environment, and possible life forms. The return payload, consisting mainly of samples and data, weighs approximately 900 pounds (408 kg). The requirements placed on the Mars excursion module by this study did not vary from mission to mission and, therefore, the one design is adequate for all missions.

MARS EXCURSION MODULE

THREE-MEN–30 DAYS

ESTIMATED WEIGHT	(LB)	(KG)
ASCENT CAPSULE	5,590	2,540
ASCENT STAGE II PROPULSION	6,860	3,130
ASCENT STAGE I PROPULSION	13,450	6,100
DESCENT STAGE	43,200	19,600
DEORBIT MOTOR	4,200	1,910
GROWTH AND CONTINGENCY (30%)	21,990	9,980
TOTAL	95,290	43,260

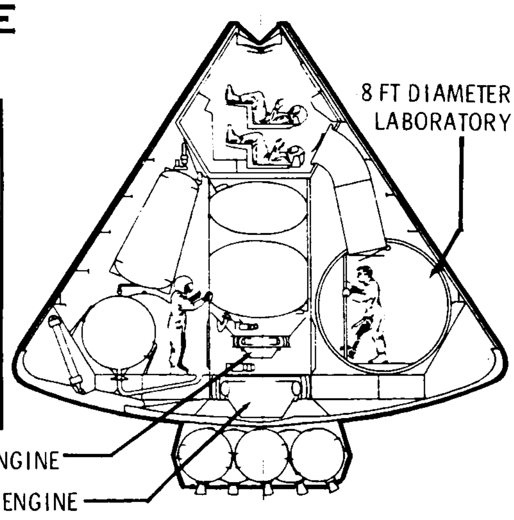


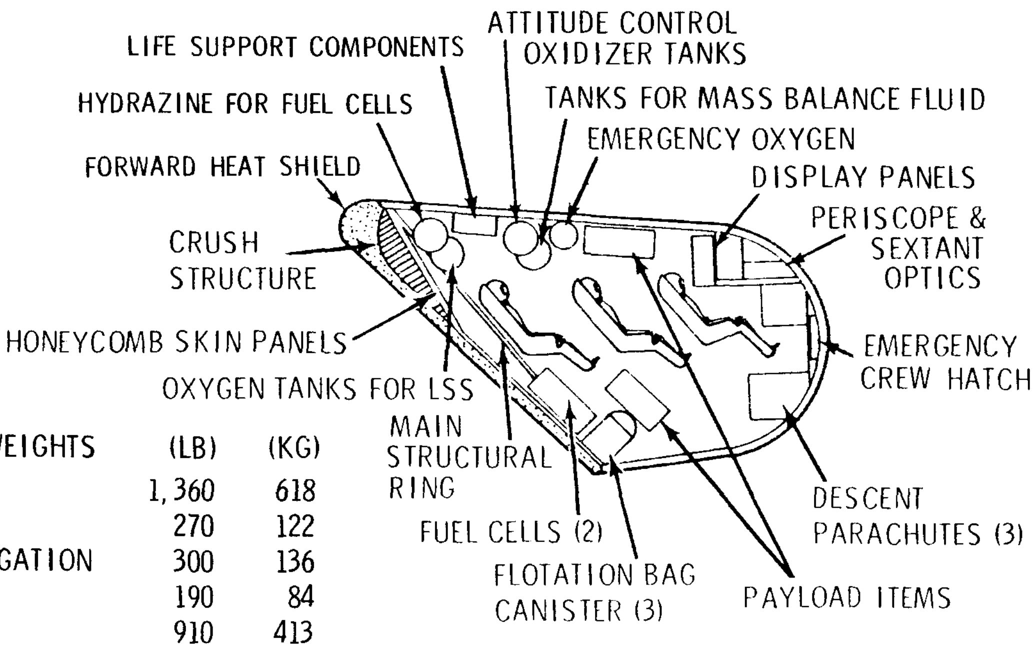
Figure 4-7

*NAA Document SD-67-755, *Definition of Experimental Tests for Manned Mars Excursion Module*, NASA Contract NAS9-6464, North American-Rockwell Corp., January 1968.

A blunted, biconic *Earth entry module*, typical of a high speed entry vehicle, was selected as the recommended Earth entry module. The biconic Earth entry module design was adopted, with some modification, from the work reported under NASA Contract NAS2-2526.* The Earth entry module is designed for a crew of six and a 1-day occupancy time. The Earth entry module is designed for the maximum entry velocity of up to 65,000 fps (19,800 m/sec); thus it is common for all missions.

The Earth entry module performs the vital function of transporting the mission crew and the science data and samples from the mission module on the return hyperbolic trajectory to a safe landing on the Earth's surface. It is designed for water landing, consistent with midcourse adjustment with Earth arrival time.

The biconic Earth entry module configuration is illustrated in Figure 4-8. The crew is arranged in two side-by-side rows of three men. The crew volume allowance is 40 cu ft/man (1.13 cu m/man). The elliptical cross-section of the afterbody, in which almost all the internal subsystems are packaged, dictates the arrangement of most of the large components. These are placed above the heads and below the feet of the crewmen to allow the seats to fill the center portion of the vehicle.



ESTIMATED WEIGHTS	(LB)	(KG)
CREW AND SEATS	1,360	618
CONTROLS	270	122
GUIDANCE AND NAVIGATION	300	136
COMMUNICATIONS	190	84
SCIENCE	910	413
LIFE SUPPORT	730	332
ELECTRICAL POWER	660	299
ATTITUDE CONTROL	1,140	508
RECOVERY	880	395
HEAT SHIELD	4,530	1,970
STRUCTURE	4,160	1,890
GROWTH & CONTINGENCY (15%)	2,270	1,130
TOTAL	17,400	7,897

BICONIC EEM CONFIGURATION

Figure 4-8

*LMSC Document 4-05-65-12, *Study of Manned Vehicles for Entering the Earth's Atmosphere at Hyperbolic Speeds*, NASA Contract NAS2-2526, Lockheed Missiles and Space Co., November, 1965.

SPACE ACCELERATION

The acceleration system is made up of identical nuclear propulsion module with three modules for injecting into the interplanetary trajectory (PM-1), a single module for braking into the planet orbit (PM-2), and a single module for injecting into the trans-Earth trajectory (PM-3).

Smaller FLOX-methane secondary propulsion systems are provided for mid-course correction and orbit trim.

The *primary propulsion module* is shown in Figure 4-9. This module is common for all applications except for additional insulation added to the planet-departure module for the long-stay-time Mars and Venus missions. The tank is 33 feet (10.6 m) in diameter by 115 feet (35 m) long and contains 385,000 pounds (175,000 kg) of liquid hydrogen (91,600 cu ft--2,590 cu m). The tank is a pressure vessel supported within a load-carrying structure by fiberglass straps. A 195,000-pound-thrust (88,500 kg) Nerva nuclear engine is attached to the tank head. The engine is 40 feet (12.2 m) long with a nozzle exit diameter of 13.5 feet (4.12 m). The weight of the engine and thrust structure, less the radiation shield, is 25,540 pounds (11,580 kg).

The outer shell around the propellant tank serves as the Earth-launch load-carrying structure and as meteoroid protection barrier during the missions. It is split into four segments and secured by hoop straps. These straps are severed just prior to engine ignition, allowing the shell segments to drop off.

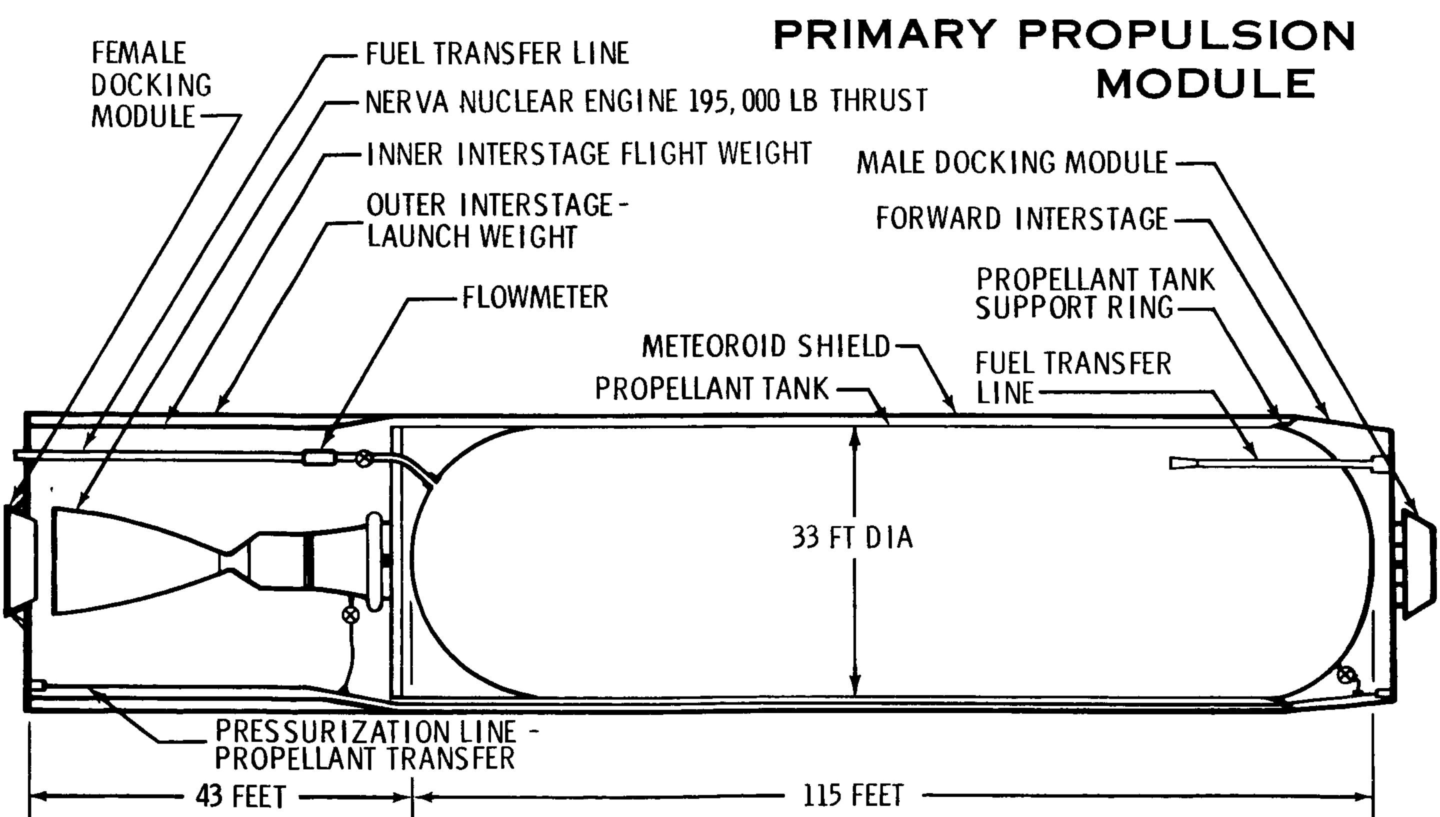


Figure 4-9

Because of the long interstage required for the Nerva engines, a double interstage is provided. The outer interstage serves as a load-carrying structure for the Earth launch. It is jettisoned after the module is docked to its mating module. An inner interstage is provided for the mission flight loads.

The module has an 8-inch-diameter (20 cm) propellant line used to transfer propellant between modules during mission operations.

EARTH LAUNCH VEHICLE

The recommended Earth launch vehicle, shown in Figure 4-10, is an uprated Saturn V. It consists of the Saturn V first stage lengthened 40 feet (12.2 m), five uprated (1.8×10^6 lb thrust/engine--8×10^6 Newton) F-1 engines, a standard-length second stage with five uprated J2S engines, and four 4-segment, 156-inch-diameter (3.96 m) solid rocket motors attached to the first stage. This vehicle can place a payload of 548,400 pounds (249,300 kg) into a 262-nautical-mile (485 km) circular orbit. A LO_2/LH_2 transtage is used to provide the final 475 fps (145 m/sec) for circularization.

EARTH LAUNCH VEHICLE

VEHICLE	GROSS P/L TO 262 N. MI. CIRCULAR ORBIT	
-25(S)U CORE	302,700	(137,300 KG)
CORE + 2 S/O	410,800	(186,300 KG)
CORE + 4 S/O	548,400	(248,750 KG)

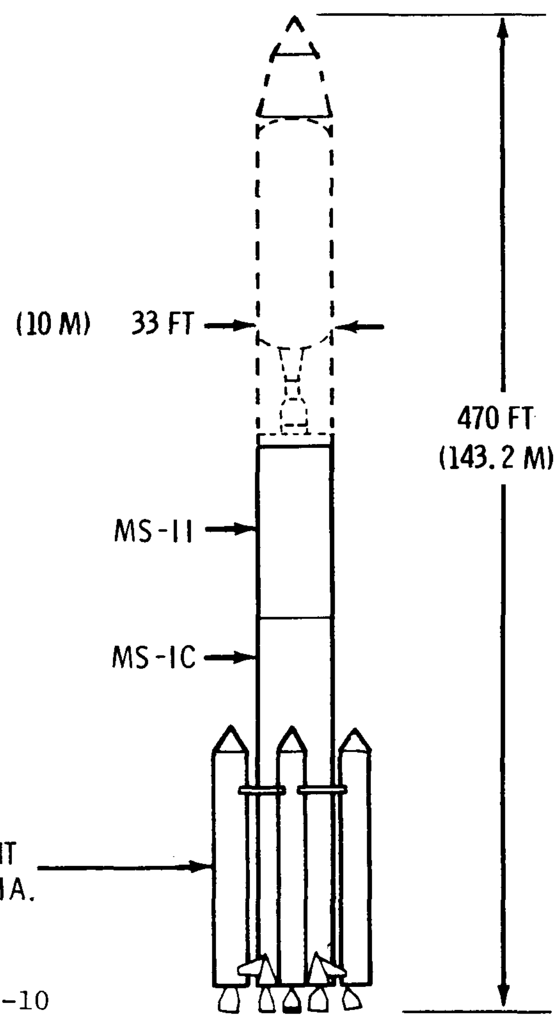


Figure 4-10

FACILITIES SUMMARY

Assembly, checkout, and launch of the Saturn V-25(S)U Earth launch vehicle and the various payloads required for qualification testing and planetary missions will be accomplished through the use of existing, modified, and new facilities at Launch Complex 39 and the industrial area at the Kennedy Space Center. Expansion and modifications of the existing facilities are shown in Figure 4-11. These are primarily to accommodate the increased length of the first stage of the Earth launch vehicle core, the addition of the strapon solid rocket motors, and the increased launch rate required to support the mission.

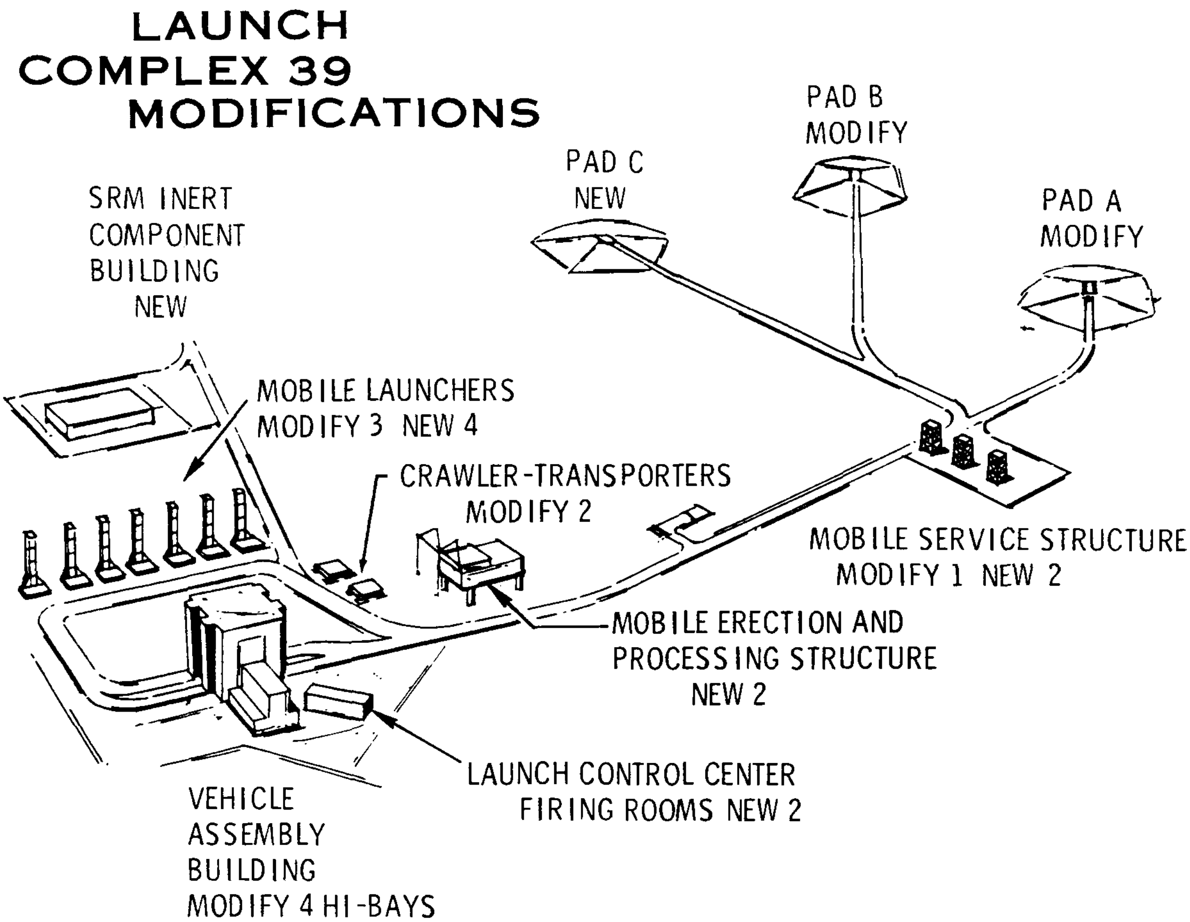


Figure 4-11

SYSTEM OPERATIONS

Operations of the recommended system on a typical mission starts with the arrival of the elements at KSC, and continues through launch processing, launch, orbital assembly and processing, mission performance, and recovery of the Earth entry module.

KSC PROCESSING

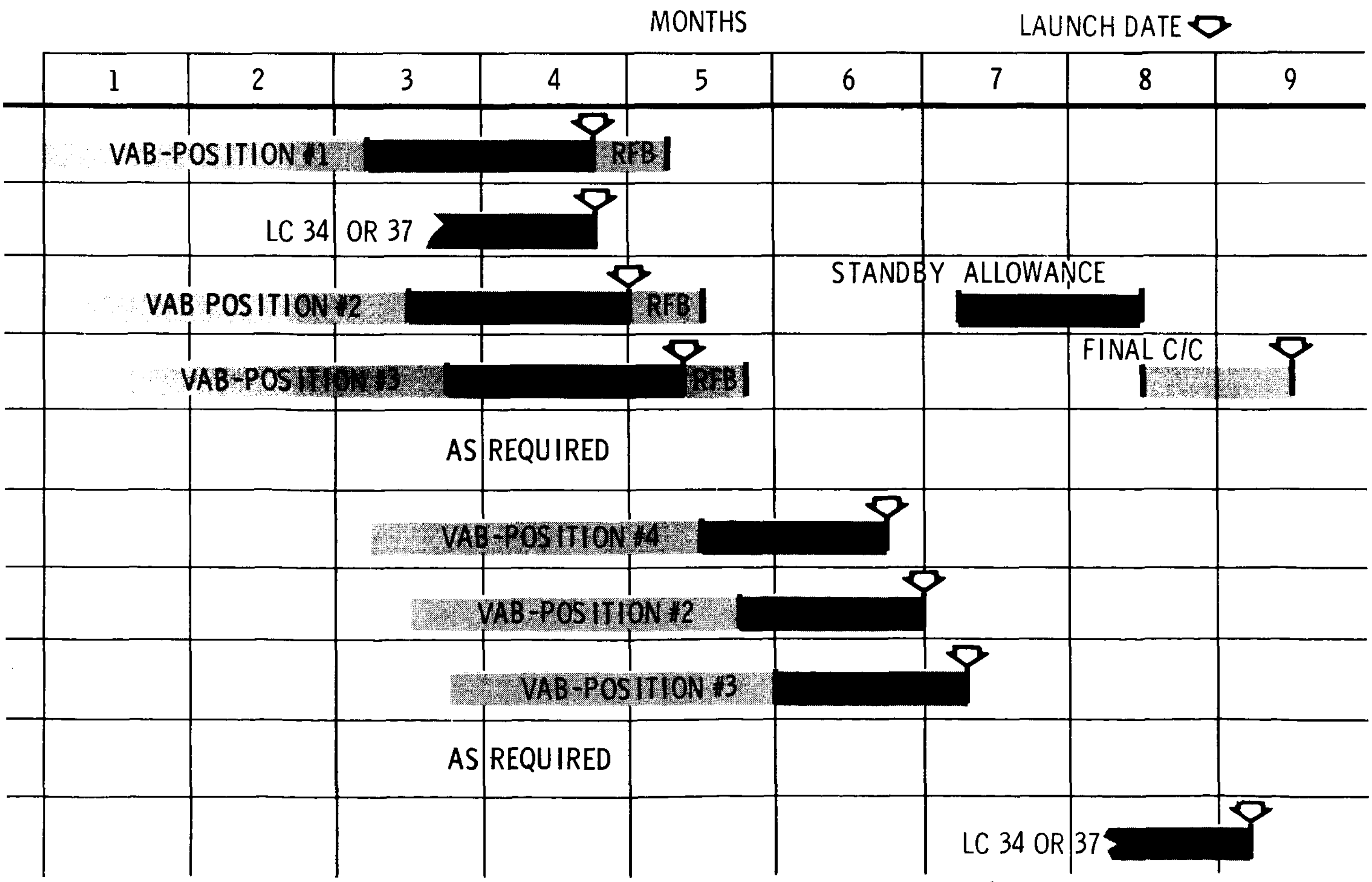
The assembly, checkout, and launch of the Earth launch vehicle and propulsion module payload begin with the arrival by barges at KSC of the MS-IC stage, the MS-II stage, and a propulsion module (PM) tank. The solid rocket motors are also water transported in railroad cars on barges. Due to the increased length of the first stage, a new transportation vehicle will be required to move the MS-IC stage from the unloading dock to the vehicle assembly building. A new vehicle will also be required to transport the propulsion module tank to the nuclear engine/fuel tank mating facility. The railroad cars containing the live rocket motor components go directly to a new open rail car storage area. The inert components are transferred to the new inert components building.

LAUNCH NO.	ELEMENT	ELV
1	SPACECRAFT	SAT V-25 (S)U (CORE)
2	APOLLO-ATC	SAT IB
3	PM-3	SAT V-25 (S)U
4	PM-2	SAT V-25 (S)U
5	APOLLO-ATC	SAT IB
6	PM-1, CENTER	SAT V-25 (S)U
7	PM-1, SIDE	SAT V-25 (S)U
8	PM-1, SIDE	SAT V-25 (S)U
9	APOLLO-ATC	SAT IB
10	APOLLO-M/C	SAT IB

Figure 4-12

In the vehicle assembly building, erection of the Earth launch vehicle on the mobile launcher follows the Apollo, Saturn V procedure. Following the integration and checkout of the payload, the vehicle is moved by crawler-transporter to the launch pad. Concurrent with the assembly and checkout of the Earth launch vehicle core, the solid rocket motor components are being processed through the new inert components building and the new mobile erection and processing structure. Upon completion of checkout, the solid rocket motors are transported to the launch pad in the mobile erection and processing structure by use of the crawler-transporter. At the pad, the solid rocket motor segments are assembled and integrated with the core of the Earth launch vehicle. Completion of the pad checkout procedure, fueling operations, and launch follow the Saturn V routine.

Figure 4-12 shows the flow time for assembly and launch operations. The assembly test crew and the mission crew will be launched from Complexes 34 and 37 in six-man logistic vehicles, using the Saturn-IB Earth launch vehicle. Scheduling provides for a minimum logistic launch rate of one every 45 days for assembly test crew turn-around, and replenishment of expendables, special tools, or equipment.



LAUNCH OPERATIONS

An indirect, rendezvous-compatible, circular orbit mode was selected for the assembly operation. This mode provides an intermediate phasing orbit to compensate for launch time errors. The rendezvous-compatible orbit permits two coplanar launch opportunities per day. Launch occurs at or near the coplanar launch opportunity, and the Earth launch vehicle will provide sufficient yaw steering to accommodate at least a 10-minute ground launch window. The Earth launch vehicle will burn out supercircular at 100 nautical miles (185 km) to achieve an apogee orbit altitude of 262 nautical miles (485 km) coincident with the assembly orbit. A LOX-LH_2 transtage propulsion unit on each payload is used to provide the necessary ΔV to circularize the orbit and accomplish the docking maneuver.

Figure 4-13 shows the Earth launch and assembly sequence for the preparation of the space vehicle in Earth orbit.

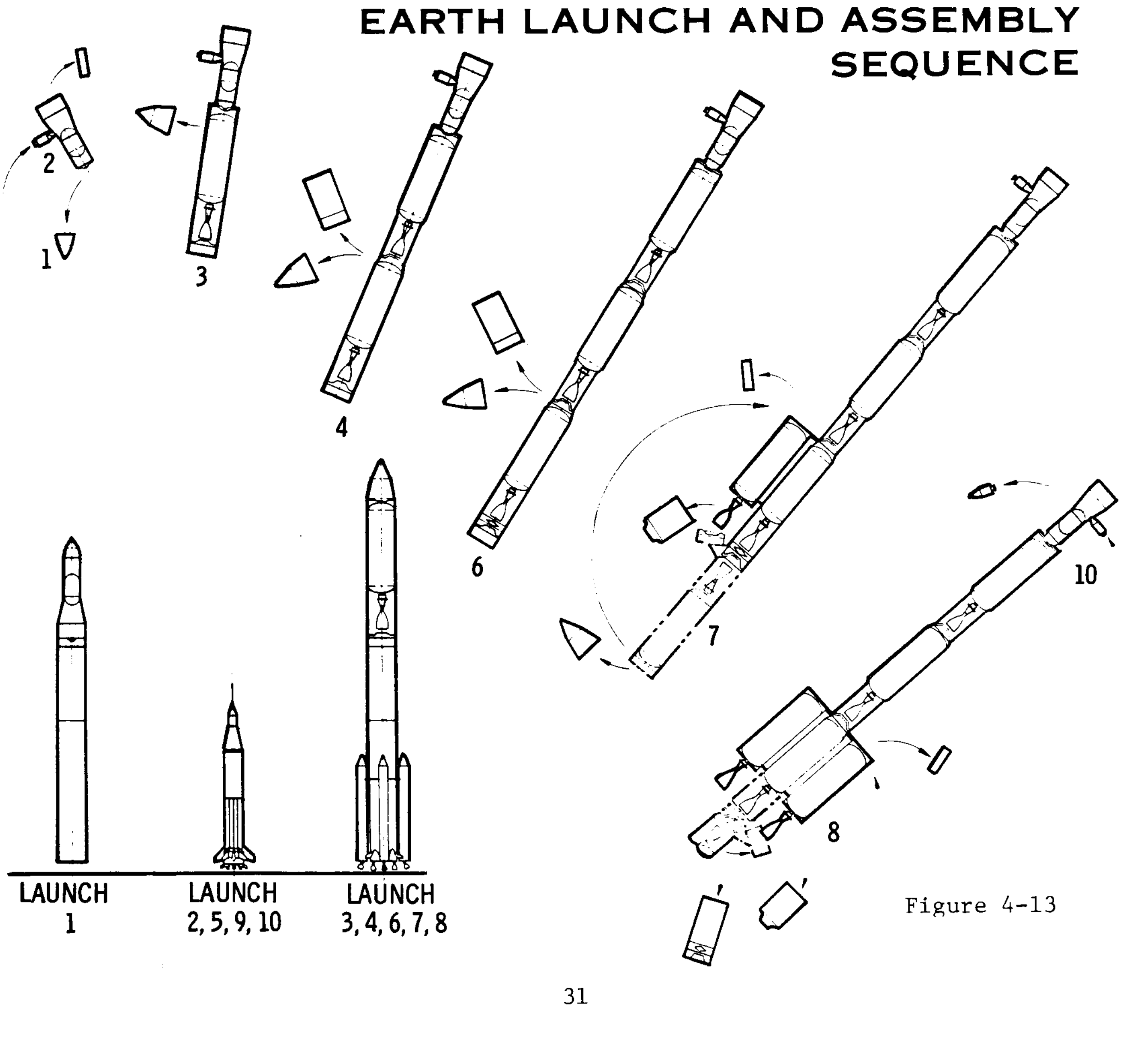


Figure 4-13

In Launch No. 1, the Saturn V-25(S)U core vehicle without the solid motor strapons launches the spacecraft unmanned. The transtage and instrumentation unit interfaces with both the ELV and the spacecraft.

In Launch No. 2, the assembly test crew of six men is launched by a Saturn IB. At rendezvous the logistic vehicle docks into the side of the spacecraft and the crew transfers into the mission module. The assembly test crew immediately activates and checks out all systems and inspects for damage that might have occurred during launch. This includes inspection for structural damage using extravehicular activity.

In Launch No. 3, the first propulsion module, PM-3, is launched by a Saturn V-25(S)U with four solid rocket motors. When the transfer to the assembly orbit is completed and the nose cone jettisoned, the rendezvous radar system within the mission module is activated and provides the range, line-of-sight, and rate data to close the distance between the spacecraft and PM-3 to within approximately 10 feet (3.05 m). The spacecraft now stabilizes itself, as all docking operations consider the orbiting elements as passive and the ascending elements as active. At this close distance, a television camera in the male cone (ascending element) provides visual control required to make the final alignment for mating with the docking cones. The two elements are halted by an energy-absorbing system within the docking mechanism. Umbilicals are then automatically engaged, permitting the assembly test crew to remotely check out the PM-3.

In Launch No. 4, propulsion module PM-2 is launched and the procedure described for the third launch is repeated. The elements are drawn together by a hydraulic system in the docking mechanism until the automatic aligning and latching mechanism on the interstage structure secures the elements. In addition to the electrical umbilical connection, the fuel transfer duct and the pressurization line must be connected automatically, requiring EVA inspection only.

Launches No. 5 and 9 are reserved for additional transportation of crew and parts as necessary.

In Launch No. 6, the first of the three PM-1 propulsion modules is injected into assembly orbit.

In Launches No. 7 and 8, the remaining two PM-1 modules are injected into assembly orbit. Their configuration and operations are the same, but they are launched in sequence after the assembly of the first is complete. A swinging mechanism at the engine end of the center module is used to position the outer modules in place. The Earth launch interstages of all three PM-1 modules are jettisoned just prior to PM-1 engine burn.

Launch No. 10 replaces the assembly test crew with the mission crew for final checkout prior to Earth departure of the space vehicle.

MISSION OPERATIONS

Upon completion of assembly and final checkout, the space vehicle is injected into a heliocentric orbit targeted to Mars or Venus. The sequence of events associated with a typical Mars landing mission is shown in Figure 4-14. The sequence of events for a Venus orbiting mission is similar except for the deletion of the Mars excursion module activities and substitution of Venus unmanned probes.

The meteoroid shield is jettisoned from the PM-1 modules just prior to stage firing.

Firing of the nuclear PM-1 modules injects the space vehicle into the transfer trajectory.

Three midcourse corrections are assumed for each interplanetary leg of the trip, the first occurring 5 days after launch from orbit, the second about 20 days later, and the third at about 20 days before arrival at the destination planet.

For Mars missions with a Venus swingby on the outbound trip, probes are launched prior to Venus encounter, and data return is recorded and monitored during the swingby and as long as communications can be maintained. Planet capture is accomplished by the PM-2. The meteoroid shielding and outbound midcourse correction system are staged prior to PM-2 burn. The spent PM-2 stage is separated in the higher initial orbit

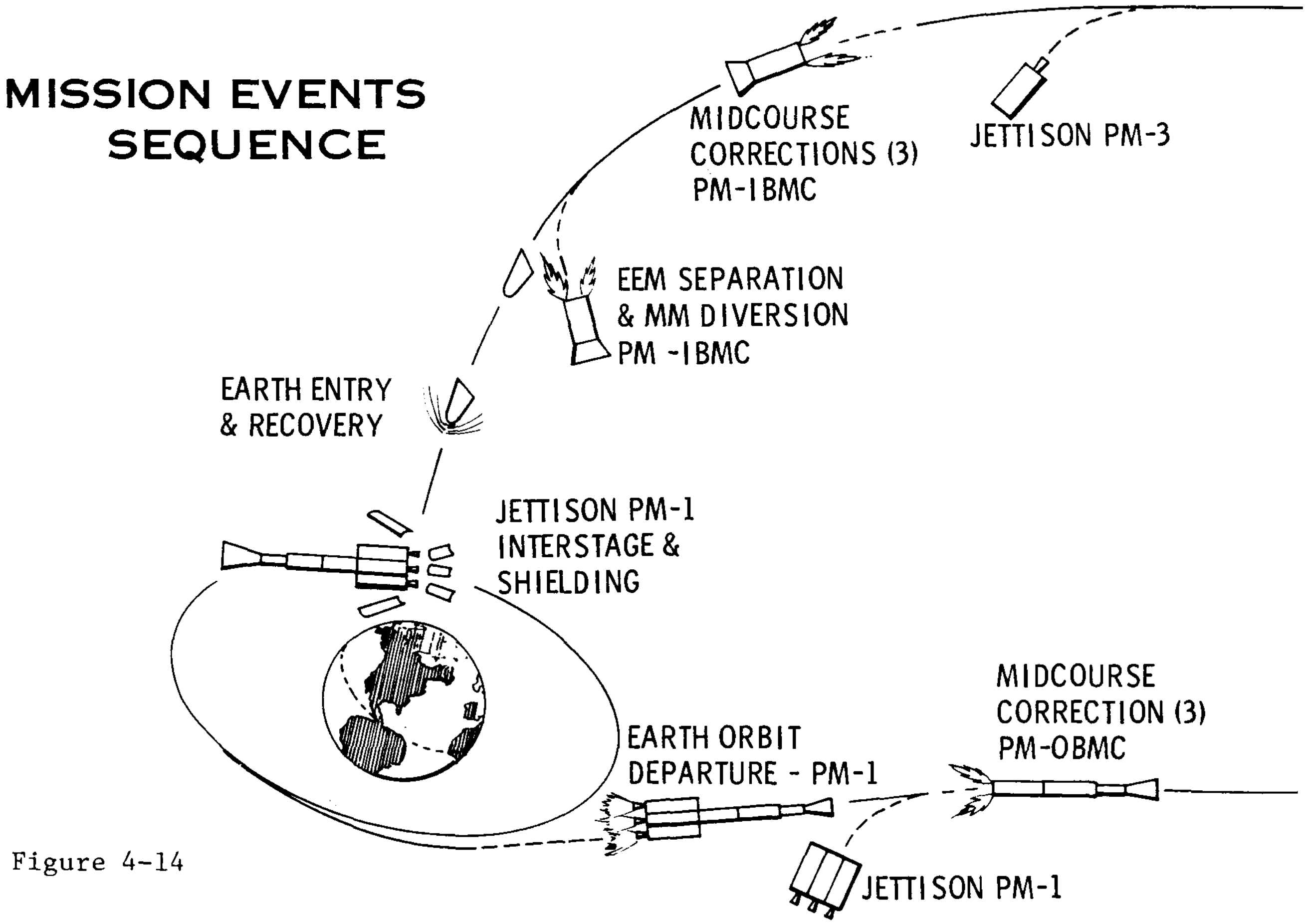
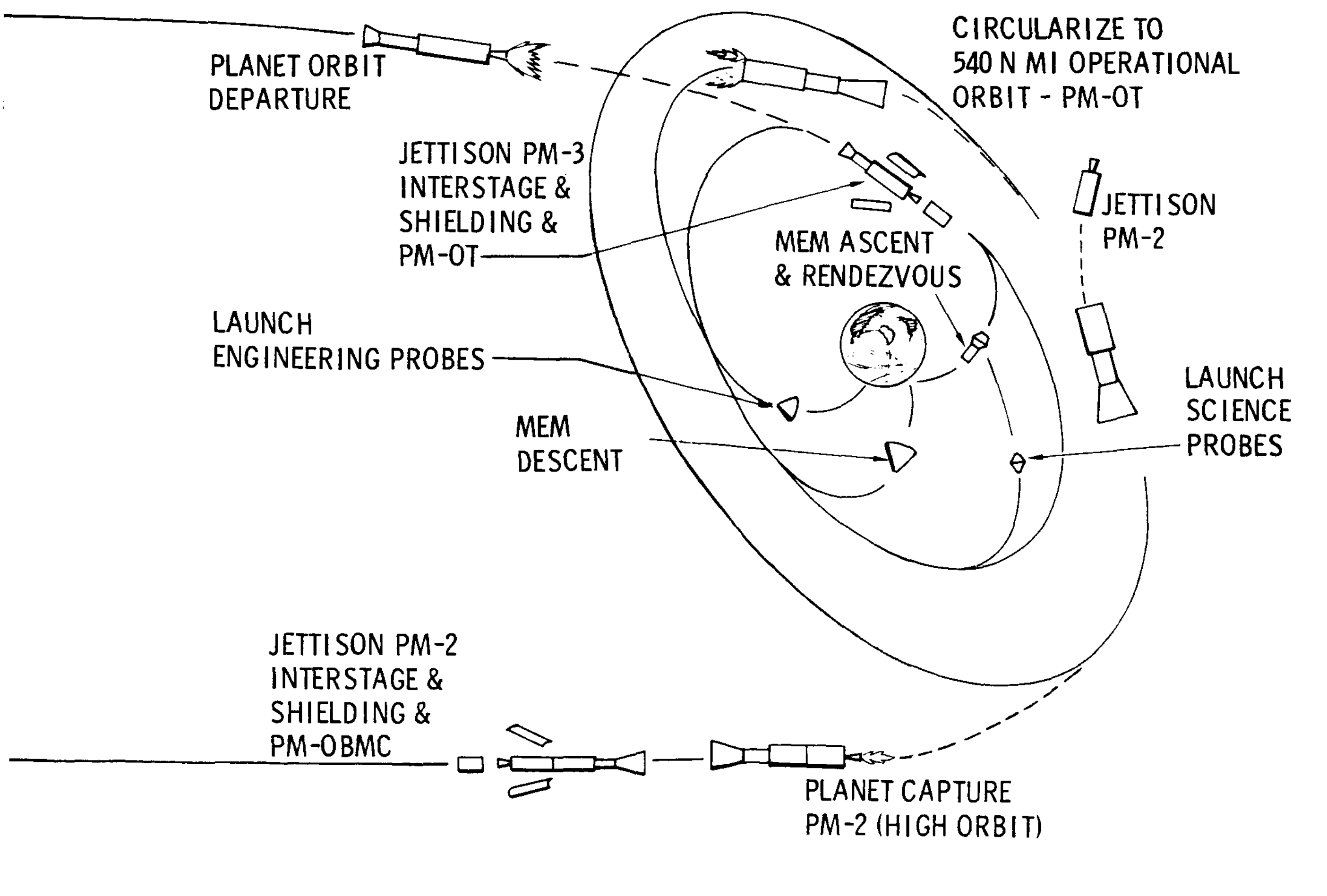


Figure 4-14

to prevent radiation and contamination problems, and the space vehicle transfers to a 540-nautical-mile (1000-km) operational orbit using the chemical trim propulsion system.

Two to five days are spent surveying the planet for landing sites, performing orbital experiments (including deployment of probes), and preparing the Mars excursion module for operation. Three of the six-man crew then descend to the planet surface in the Mars excursion module. After aeroballistic entry, the Mars excursion module is slowed by a ballute retardation system, and, using propulsion descends to the surface. After a 30-day stay on the planet, the ascent module of the Mars excursion module brings the three men and scientific payload back to the space vehicle. During planetary operations, the men in the space vehicle continue the orbital experimentation, monitor the planetary operations, and maintain the space vehicle operations. The ascent vehicle is discarded in the planet orbit after the crew has transferred to the mission module.

Preparations for planet departure include staging of the orbit trim propulsion system, PM-3 aft interstage, and PM-3 meteoroid shield. Departure from Mars orbit is accomplished by the nuclear PM-3. Approximately 1 day prior to Earth entry, the crew and scientific payload transfer to the Earth entry module and separation from the mission module is accomplished. The trajectory is adjusted for entry and water landing at the desired location on Earth.



5.0 SYSTEM DEVELOPMENT

A basic program plan example was developed for the recommended interplanetary mission system. Development plans and costs are based on initial missions being of short duration. The first mission in the example is a 1983 Venus Short and the second, a 1986 Mars Opposition. Program plans include the schedules, the test plans, and the costs associated with the basic program plan example.

Schedules, test plans, and yearly funding rates were developed in detail to the module level, while program costs were developed in detail to the subsystem level.

A flexible planning method was also devised whereby the detailed scheduling and costing data, which was developed during the study, can be applied to various desired mission programs to yield overall program schedules, costs, and yearly funding rates.

SCHEDULE

The schedule for the recommended system example program is depicted in Figure 5-1. Development go-ahead is January 1972 with 11-1/2 years flow time to the first Venus 1983 mission launch date. The development go-ahead date for the second mission is mid-1976, with 9-1/2 years to the Mars 1986 launch date. Development and integration of the Mars excursion module is the major effort for the Mars mission. The development go-ahead for mission probes and experiments is 1976 with 6 years and 2 months flow time to meet the initial Venus mission.

The overall flow times noted above are one of the basic ingredients used in the flexible planning method.

Schedules for the example program begin with early design and development, which is followed by ground qualification and progresses successively to orbital qualification of the modules, to system qualifications, through orbital demonstration, and finally to mission operations. A significant milestone is the launching of the mission module for orbital qualification. It is launched and tested early since it can be used in orbit for interface testing instead of expensive simulation equipment. Furthermore, its early availability provides a habitat for orbital test personnel during the 3-year module testing phase prior to orbital demonstration. It was assumed that logistic space vehicles would be developed in association with early Earth orbital programs and would be available to support the testing and mission orbital operations.

A space vehicle qualification and orbital demonstration test is scheduled. All launch, orbital, and mission operations will be accomplished

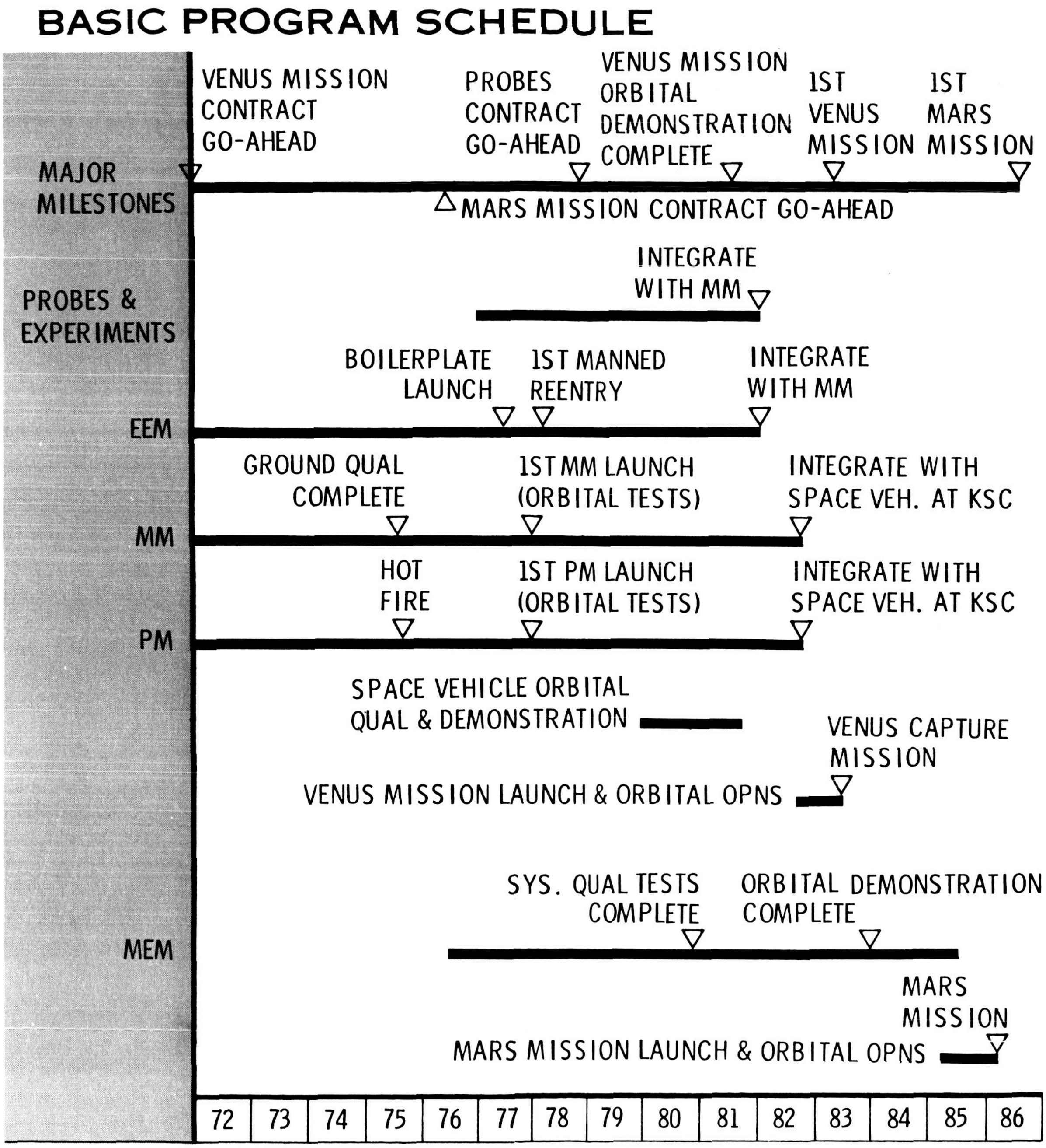


Figure 5-1

in the vicinity of the Earth, but mission in-transit times will be foreshortened. Standby units for each module, including a Saturn V-25(S)U are planned for the orbital demonstration test. Standby units not used will be refurbished and transferred to the first mission as standby or operational units. A 2-year period is provided between completion of the demonstration tests and the first mission to allow final design improvements to be incorporated into the operational hardware.

PROGRAM COSTS AND FUNDING

Costs for the example two mission program were estimated in detail to the subsystem level. Technical configuration data such as structural weights, electrical power requirements, etc., for each module, were related to parametric cost curves for basic R&D and unit costs. Costs of flight tests were estimated on an individual basis.

Yearly funding requirements were calculated using a computer program which printed out funding curves from the module level to the total system level. The cost and funding data developed in detail for the two mission example was expanded to include any type of mission and for any space vehicle configuration combination. Summaries of the costing and funding information were one of the basic ingredients used in the flexible planning method.

Total costs for the example two mission program are approximately $29.0 billion consisting of $23.7 billion non-recurring and $5.3 billion recurring.

A peak funding rate of approximately 3.5 billion dollars per year occurs during the 1975-1977 time period.

PRICE LIST

(NON-RECURRING)

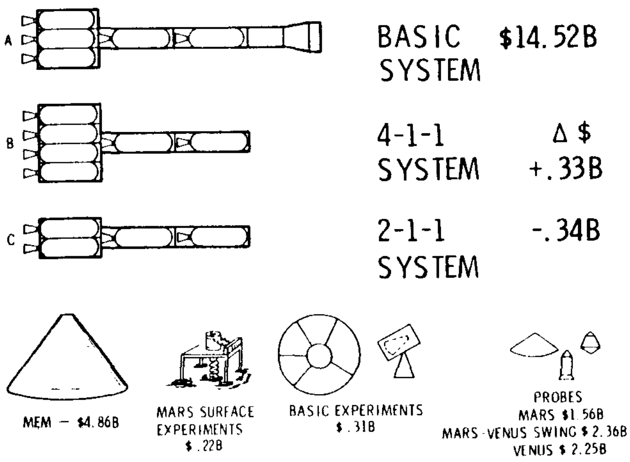


Figure 5-2

FUNDING DISTRIBUTION

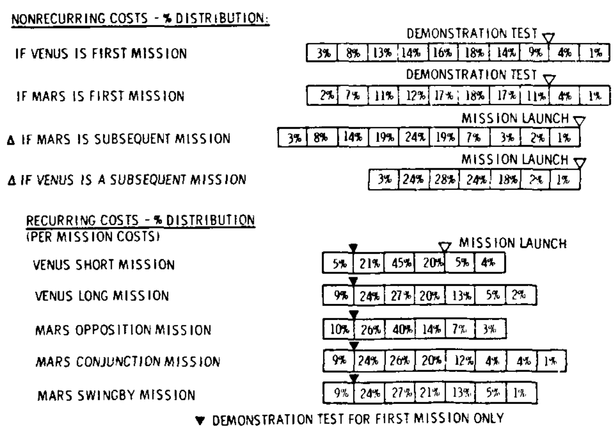


Figure 5-3

FLEXIBLE PLANNING METHOD

The flexible planning method utilizes the summary data from the detailed scheduling and costing effort to provide a rapid means of devising and evaluating alternate mission program schedules and costs. The basic ingredients to the flexible planning method are:

- A price list (nonrecurring);
- A price list (recurring);
- A funding distribution list which also provides the scheduling information.

Figures 5-2 and 5-3 are examples of the basic ingredients.

The price list (recurring) includes costs for each type mission for each space propulsion system combination. Costs range from approximately 2.4 billion dollars per mission for a Venus Short mission with a 2-1-1 space propulsion system combination, to 2.9 billion dollars for a Mars Conjunction mission with a 4-1-1 space propulsion system combination.

Figure 5-4 illustrates how the flexible planning method would be used to arrive at program schedules, costs, and funding rates. The basic mission program example of a 1983 Venus Short as the first mission and the 1986 Mars Opposition mission as the second is used. The first step would be to line up the two bars representing the two missions with the

COST & FUNDING LEVEL EXAMPLE

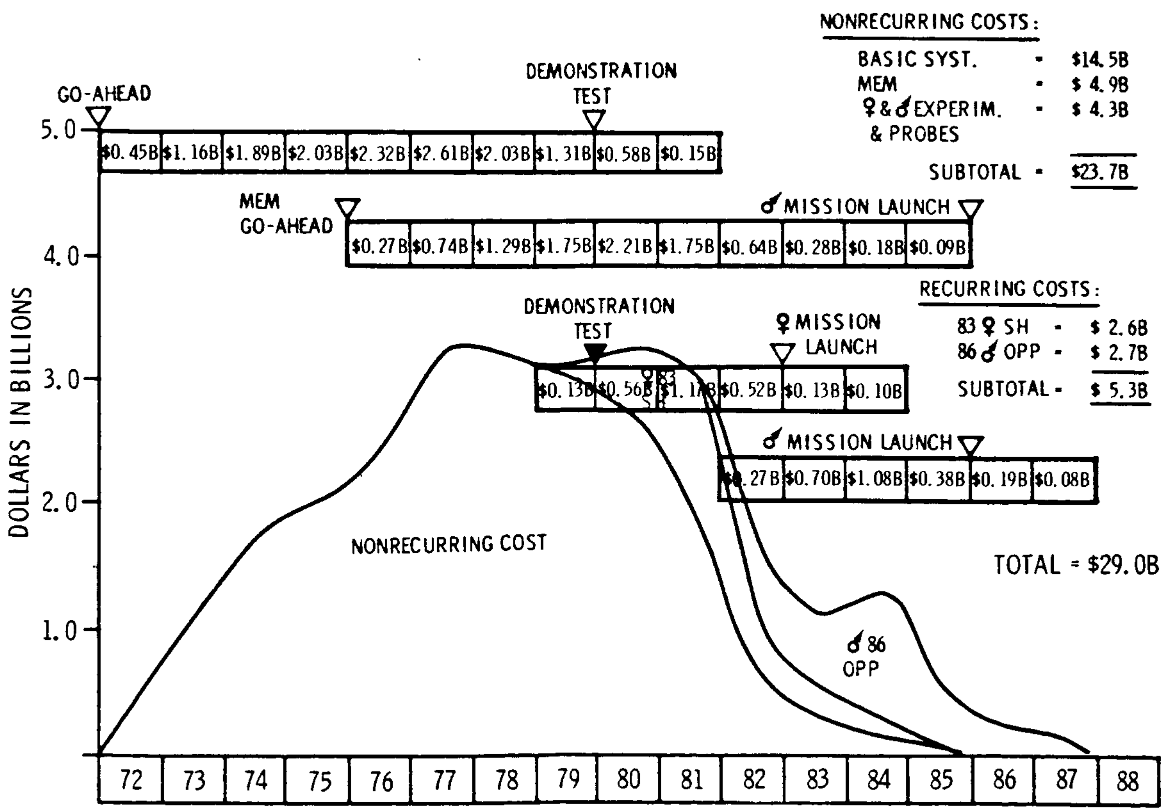


Figure 5-4

mission launch milestones on the bars corresponding to the desired launch dates. Mission costs would be derived from the price list (recurring) and distributed on a percentage basis per the funding distribution list, Figure 5-3. The next step would be to line up the bars for nonrecurring costs and distribute the yearly costs in a similar manner. The final step would be to add up the total costs per year and draw in the yearly funding rate.

6.0 TECHNOLOGY IMPLICATIONS

During the course of this study certain technology research and development requirements were identified. They are shown in Table 6-1. A further discussion may be found in Volume II of this report.

Table 6-1: TECHNOLOGY RESEARCH AND DEVELOPMENT REQUIREMENTS

FLIGHT MECHANICS

- Launch Window Energy Requirements
- Abort Trajectories and Requirements

SPACE PROPULSION

- Propellant Loading and Storage
- Highly Efficient Thermal Control (Insulation and Supports)
- Propellant Heating--Engine Radiation and Zero-G Space Flight
- Radiation Shielding
- Propellant Transfer
  - Propellant Pressurization
  - Liquid Hydrogen Flow Metering
  - Liquid Hydrogen Zero Leak Shutoff Valve
- Nuclear Engines
  - Engine Burn Life $\geq$ 60 Minutes
  - Specific Impulse $\geq$ 825 Seconds
  - Shorter Engine Length
  - Zero Net Pump Suction Head LH_2 Pump
  - Better Definition of Engine Radiation Environment

MATERIALS AND STRUCTURES

- Meteoroid Environment and Shielding
- Low Thermal Conductance Supports
- Nonflammable Materials
- Self-Sealing Pressure Structures

MISSION MODULE

- Subsystems--General
  - Long Life Equipment
  - Maintainable Equipment
  - Zero Gravity Operation
- Attitude Control
  - Large Momentum Storage Devices
- Electrical Power
  - Isotope Encapsulation
  - Isotope Safe Recovery
  - Rotating Components Performance and Life
  - Isotope Availability
  - Large Retractable Solar Arrays
- Communications
  - High Power and Efficiency Amplifiers
  - Large Light Weight Antennas
  - Laser Systems
- Environmental Control and Life Support
  - Molten or Solid Electrolyte CO_2 Reduction
  - Electrodialysis CO_2 Separation
  - Water Electrolysis Cells
  - Water Recovery Systems
  - Contamination Effects (Man and Equipment)
  - Self-Regulating Thermal Radiators
- Crew Systems
  - Substitution of Pressure Forces and Exercise for Physical Conditioning

MARS EXCURSION MODULE

- Mars Atmosphere and Constituents
- Large Hypersonic Ballutes
- Heat Shield Performance
- Full Scale Testing
- FLOX-Methane Propulsion
- Long Duration Space Storage

EARTH ENTRY MODULE

- Boundary Layer Transition, Afterbody Heating, Gas Behavior and Heat Shield Performance at Entry Velocities of 50 to 60,000 fps
- Test Techniques and Full Scale Testing
- Long Duration Space Storage

EARTH LAUNCH VEHICLE

- Effects of Solid Rocket Motors on ELV Aerodynamic Coefficients
- Wind Environments and Vehicle Response
- Acoustical Environment
- Safety and Overpressure Environment