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Final Report

MANNED MARS SYSTEM STUDY (MMSS)

(Mars Transportation and Facility Infrastructure Study)

for

NASA Marshall Space Flight Center

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FOREWORD

This is the Final Report under Contract NAS8-37126. The purpose of the Study was to design and analyze systems for conducting human missions to Mars and the moon, with special emphasis on the transportation and facility infrastructure.

This program was conducted by Martin Marietta Astronautics Group under the direction of **Dr. B. C. Clark**. An important teaming role by Science Applications International Corporation (SAIC), led by **J. C. Niehoff**, included trajectory analyses and contributions to mission design.

Our Contract Officer's Technical Representatives (COTR) at NASA/MSFC were extremely helpful and encouraging in all aspects of these endeavors. For this we must thank **R. H. Durrett, C. F. Huffaker,** and **B. M. Wiegmann**. We wish also to thank **J. M. Butler, C. C. Priest,** and **R. E. Austin** of MSFC and **I. Bekey** of NASA/ Headquarters for their interest, encouragement, and contributions.

Numerous individuals played key roles in the conduct of this effort, which at times assumed a scope of major proportions, under extremely constrained timelines. We wish especially to recognize major contributions by **D. A. Baker, S. A. Geels, R. S. Murray, W. D. Plaster, L. Redd, P. S. Thompson, W. H. Willcockson, R. M. Zubrin** of Martin Marietta and **J. McAdams** and **A. L. Friedlander** of SAIC.

Subcontractors with significant inputs to this work included Life Systems, Inc. (LSI), led by **F. T. Powell**, and Eagle Engineering, Inc., with important work accomplished by **L. Guerra** and **J. M. Stovall** under the direction of **W. R. Stump.** Individually contracted contributions by **E. W. Cliffton, A. A. Harrison,** and **H. H. Schmitt** are greatly appreciated.

We also wish to acknowledge the efforts of the following Martin Marietta personnel: **M. S. Allen, D. Bentley, H. Braun, R. T. Gamber, J. P. Gille, C. M. MacLeod, C. Marshall, L. M. Mason, R. McMordie, J. Molino, R. S. Murray, R. Obermeyer, R. Simms, A. B. Thompson, M. G. Thornton, D. Sosnay, T. Sulmeisters, B. Tobey,** and **B. Woodis.** Personnel from among this group provided much of the in-depth capabilities that were needed at various times, even though the resources available were not such that they could be involved on a continuous basis with the studies. Their efforts and interest were of major contribution to the soundness of the analyses conducted.

Work reported on lunar liquid oxygen (LLOX) utilization and aerocapture sensitivity to Mars atmospheric density variations were developed in part under Martin Marietta internal development projects D-46S and D-33S, respectively.

Finally, it is a pleasure to acknowledge the work accomplished by a group which spontaneously formed to work in parallel to this effort to design a Mars mission (see Appendix C of this report). This group received no compensation because of overall funding limitations, but included over 25 persons during the course of their studies. Led by **D. Seitz**, major contributors have included **J. Danelek, J. Filbert, W. McCarthy, D. Philipp, M. Schloesslin, J. Schulz, G. Thomason,** as well as **D. Greeson, H. Rackely, B. Tuell,** and **J. Zerr**. Early-on , this group selected nuclear thermal propulsion and artificial gravity for their baseline, two technologies that have recently begun to receive more serious consideration in the official studies. Their purely voluntary effort is testimony to the intense grass-roots support for human exploration missions to the planets.

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Tables

1.0 INTRODUCTION

The Manned Mars System Study (MMSS) was conducted for the Marshall Space Flight Center (MSFC) over the 35 month period between May 15, 1987 and April 30, 1990. During the course of the study, the NASA Office of Exploration (OEXP; Code Z) was created and MSFC was subsequently designated the Transportation Integration Agent (TIA) for support of the OEXP Mission Analysis and Systems Engineering (MASE) team. As a result of this action, modifications to the contract redirected the efforts to be consistent with NASA's overall objectives, including lunar transportation system design. A large number of written submittals were required in order to provide TIA support to MASE. The following list summarizes the documents which have been prepared and delivered by Martin Marietta under this contract during the course of this work. In nearly all cases, full sets of view-graphs were also provided to the MSFC COTR, and in several cases magnetic media were provided as well.

Document	Delivered at/to	Date Submitted	Length
1st Quarterly Progress Presentation	MSFC	Aug. 20, 1987	136 pp
2nd Quarterly Progress Report	MSFC	Dec. 1, 1987	208 pp
FY88 TIA Presentation, CS 1-3	MSFC, JSC	July 11, 1988	160 pp
TIA Input to OEXP FY88 Annual Report	MSFC, JSC	Sept. 8, 1988	37 pp
TIA Implementation Plan for FY88	MSFC, JSC	Sept. 30, 1988	400 pp
Special Assessment Study	MSFC, Hq	Sept-Nov, 1988	70 pp
Gateway Quick-turn Study	MSFC, Hq	Nov. 11, 1988	53 pp
Input to Aerospace America Articles	MSFC	Dec. 1988	10 pp
Working Group Week #1 Presentation	LaRC	Dec. 13, 1988	40 pp
Working Group Week #2 Presentation	St. Louis	Feb. 21, 1989	50 pp
Working Group Week #3 Presentation	Orlando	April 24, 1989	327 pp
Cycle 2, "June 2nd Drop"	MSFC, JSC	June 2, 1989	564 pp
TIA Input to Technology Plan	MSFC, Hq	June 14, 1989	85 pp
Working Group Week #4 Presentation	St. Louis	July 13, 1989	64 pp
FY 89 OEXP Annual Report, Vol. II	MSFC, JSC	Aug. 18, 1989	192 pp
FY 89 Vol. II, Final	MSFC, JSC	Nov. 1, 1989	197 pp

To incorporate all of these materials (more than 2,000 pages) into the present report would obviously produce an extremely unwieldy and confusing document. Therefore, a summary of key findings are presented in this Final Report, supplemented by other material produced under this contract but not already available in the widespread literature.

2.0 MISSION OVERVIEWS

2.1 MARS MISSION CASE STUDIES

The Mars human exploration and transportation scenarios that have been studied under this contract are listed in Table 2.1-1. A synopsis of their individual requirements and the numbers of Heavy Lift Launch Vehicles (HLLV) determined to be required to accomplish each scenario are given in Tables 2.1-2 through 2.1-6. Case Studies 1 and 2 were conducted for the NASA OEXP Fiscal Year 1988 Annual Report, and are documented in NASA Technical Memorandum 4075, dated December 1988. Case Studies 2.1 and 5.0 were conducted for the NASA OEXP Fiscal Year 1989 Exploration Studies Technical Report, and are documented in NASA Technical Memorandum 4170, Volume II, dated August 1989.

Table 2.1-1 Mars Human Exploration Scenarios Studied

Scenario	Date Completed
Case Study 1	7-11-88
(Phobos Flags and Footprints; FY88)	
Case Study 2	7-11-88
(Mars Expeditionary trip; FY88)	
Phobos Gateway	11-88
Mars Evolution	6-2-89
(FY89 Case Study 5.0)	
Mars Expedition	6-2-89
(FY89 Case Study 2.1)	

Table 2.1-2 Synopsis of Requirements—Case Study 1

- **Phobos Emphasis**

(mission does not land on Mars)

- **Flags and Footprints (on Phobos only)**
- **One Split Sprint/Conjunction Mission Set**

(Separate vehicles cargo and manned; 440 day roundtrip for humans)

- **All-propulsive (No Aerobrakes)**
- **Return Propellant Staged at Mars**
- **4 crew, 10 days at Phobos**
- **Humans at Phobos in 2003**

(Phobos Excursion Vehicle and MRSR)

- **Zero-g**

(Required 25 HLLV Launches at 91 t each [200 klb_m])

Table 2.1-3 Synopsis of Requirements—Case Study 2

- **Three Split Sprint/Conjunction Missions**

(Six separate vehicles, 3 cargo and 3 manned)

- **Phobos Emphasis**

(first mission does not land on Mars)

- **8 crew total; 4 crew to surface for 20 days per mission**
- **First launch in 2003**

(cargo mission)

- **Zero-g**
- **Cargo loads**

15 t per cargo flight

12.5 t per manned flight (but includes hab?)

(Required 95 HLLV Launches at 91 t each [200 klb_m])

Table 2.1-4 Synopsis of Requirements—"Phobos Gateway"

- **All-up (No split missions)**
- **One Opposition Mission, followed by three Conjunction Missions**
- **Phobos Emphasis**

(first mission does not land on Mars; subsequent missions visit)

- **3 crew, first mission; 5 subsequent**
- **First launch in 2004**
- **Tether Artificial Gravity**
- **Cargo loads**

15 t per cargo flight

12.5 t per manned flight (but includes hab)

(Required 5 HLLV Launches for first mission, using Shuttle-Z's at 140 t each [308 klb_m])

Table 2.1-5 Synopsis of Requirements—Mars Evolution (FY89 CS-5.0)

- **Use Martian Gateway (Phobos or Deimos) for propellant production and tether momentum exchange**
- **Mars Surface Base**
- **Artificial gravity**
- **Mostly re-usable vehicles**
- **Pipeline constraints: 570 t (wet), 90 t (dry) per year to LEO**
- **Nuclear Propulsion on-line in later missions**

NEP for cargo

NTR for personnel flights

(Required 60 ETO Launches for 7 missions, at 140 t for each HLLV [308 klb_m])

Table 2.1-6 Synopsis of Requirements—Mars Expedition (FY89 CS-2.1)

• **Single split/sprint (minimum science) mission**
2002 piloted launch (2004 backup)
• **3 crew, 30 d at Mars;**
entire crew to surface, 20 d staytime on surface (landing to +5 km alt)
• **All vehicles expendable, and Launched intact with no on-orbit assembly (propellant transfer allowed)**
• **Cargo vehicle is MDV plus 10 t additional equipment**
• **Precursor missions for orbiters & landing beacons.**
Use 1995 technology
• **Aerocapture at Mars, ≤ 5 gee. Use high L/D aerobrake (0.9-1.2) Direct entry at Earth (ECCV)**
• **ETO: Crew 2 launches/yr HLLV 4 launches/yr @ 140 t/launch. 12.5 m dia x 25 m P/L shroud.**
(Required 7 HLLV Launches at 140 t each [308 klb$_m$])

Case Study 1 was constructed as a high energy trajectory (sprint), but without aerobraking for aerocapture at Mars. These two factors combined to result in an extremely large load of hydrogen/ oxygen (H/O) propellants for the trans-Mars Injection System (TMIS) and the all-propulsive Mars Orbital Capture System (MOCS). The vehicle stack is shown in Figure 2.1-1. Multiple cryopropellant tanks are launched, individually as Siamese Twin tank sets, Figure 2.1-2. The rationale of this approach is to have one tank set be physically robust for the launch vibration and acceleration loads during boost into orbit. This tank set is filled with the hydrogen and oxygen propellant loads on the ground, and therefore employ spray-on foam insulation (SOFI) to thermally insulate the contents. The second tank set of the Siamese pair is empty during launch, but may need to be pressure-stabilized to protect it from acoustic and other launch loads. This tank set is extremely lightweight and employs different thermal isolation approaches, appropriate to in-space conditions of low-loads and high ambient vacuum. Mechanical retractable isolators are used, as are vapor cooled shields, multilayer insulation (MLI), and a thermodynamic vent for boiloff. The two tank sets are plumbed together before launch. Soon after exiting the atmosphere, valves are opened and an automatic propellant transfer is begun to remove the cryogens from the ruggedized tanks into the lightweight and better thermally protected tanks. Remotely operated pyrotechnic-release fittings then cause jettison of the heavy tanks from the tanks to be used in the flight. Each Siamese stack is 6.4-m in diameter, 15-m long, and contain a total of 69.3 t of cryopropellant. On-orbit assembly is required to combine the 16 tanks into the TMIS and MOCS. The TMIS employs a very high expansion ratio Space Shuttle Main Engine (SSME) derivative engine, whereas the MOCS (and the TEIS on the cargo vehicle) use RL-10 derivative engines. Although not in the baseline requirements, two alternatives were considered: a nuclear thermal rocket (NTR) propulsion system and a Mars aerocapture brake. Figure 2.1-3 demonstrates the major reductions in initial mass in LEO (IMLEO) that are possible if either or both of these approaches could be adopted for this mission scenario.

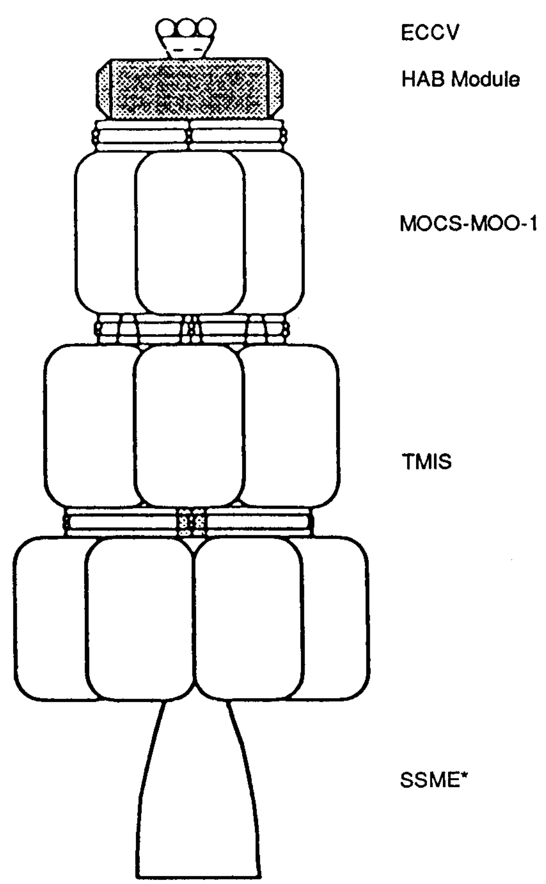


Figure 2.1-1 Case Study 1 Mars Transfer Vehicle (MTV) for Manned Mission

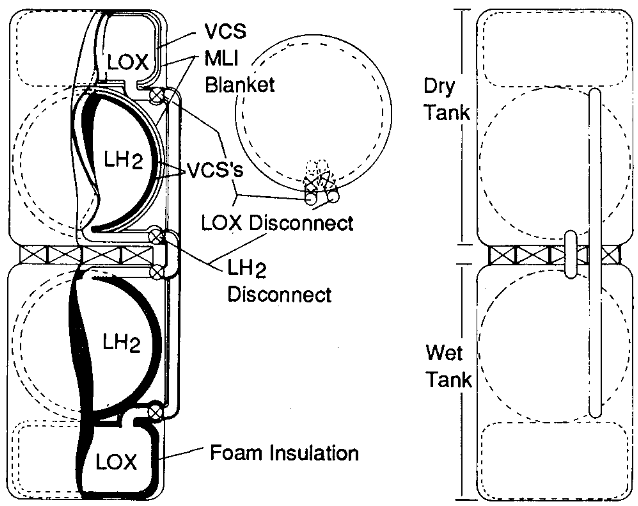


Figure 2.1-2 Siamese Twin Tank Concept

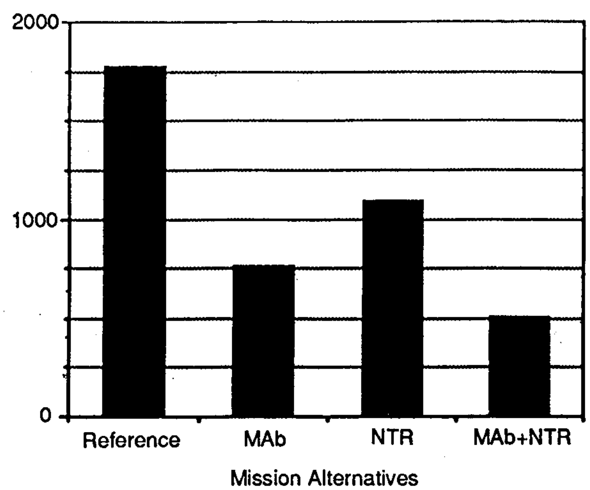


Figure 2.1-3 IMLEO-Savings of Alternative Approaches for Case Study 1

Case Study 2 involved a series of three split sprint/conjunction missions to Mars with the much larger crew complement of eight astronauts per manned mission. Because aerocapture is allowed for both cargo and human flights, it was decided to select a common-sized aerobrake for the designs to force a commonality that could result in reduction in implementation costs. The cargo vehicle is shown in Figure 2.1-4 and the manned interplanetary vehicle in Figures 2.1-5 and -6. During the rendezvous in Mars orbit, the two craft dock as shown in Figure 2.1-7, allowing a shirt-sleeve transfer of four crewmembers into the lander. In this "clamshell" docking configuration, the TEIS is also transferred from the cargo vehicle to the manned vehicle. This TEIS is fully self-contained, with power system, avionics, and an all-up propulsion system. This was a departure from previous concepts, which provided for propellant transfer between the two vehicles at Mars. In this case, the TEIS is mechanically released from the cargo ship and then latches into a receiving framework on the return vehicle. No plumbing or electrical connections are required; firing of the TEIS is accomplished by remote command and control. Propulsion for the TMIS and TEIS follow the same approach for engines and tanks as for Case Study 1.

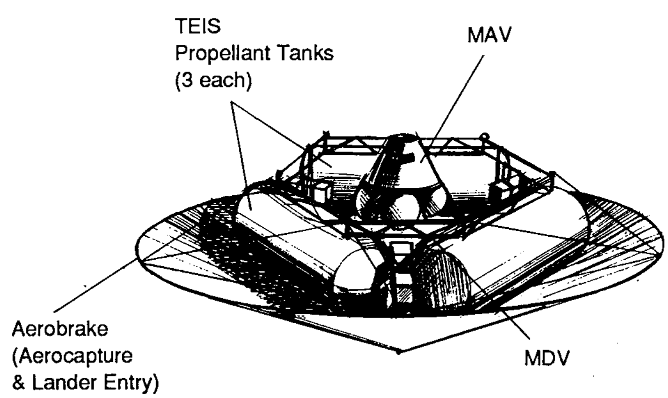


Figure 2.1-4 Cargo Vehicle for Case Study 2

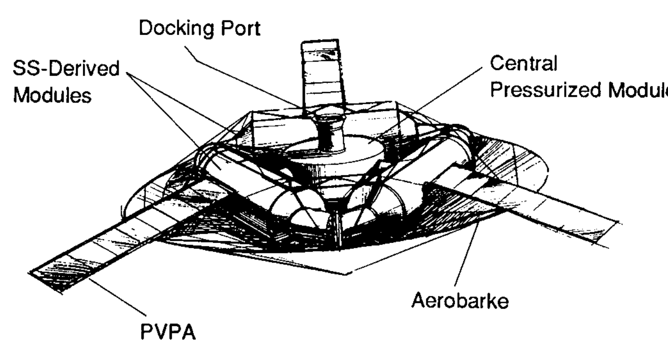


Figure 2.1-5 Manned Vehicle for Case Study 2

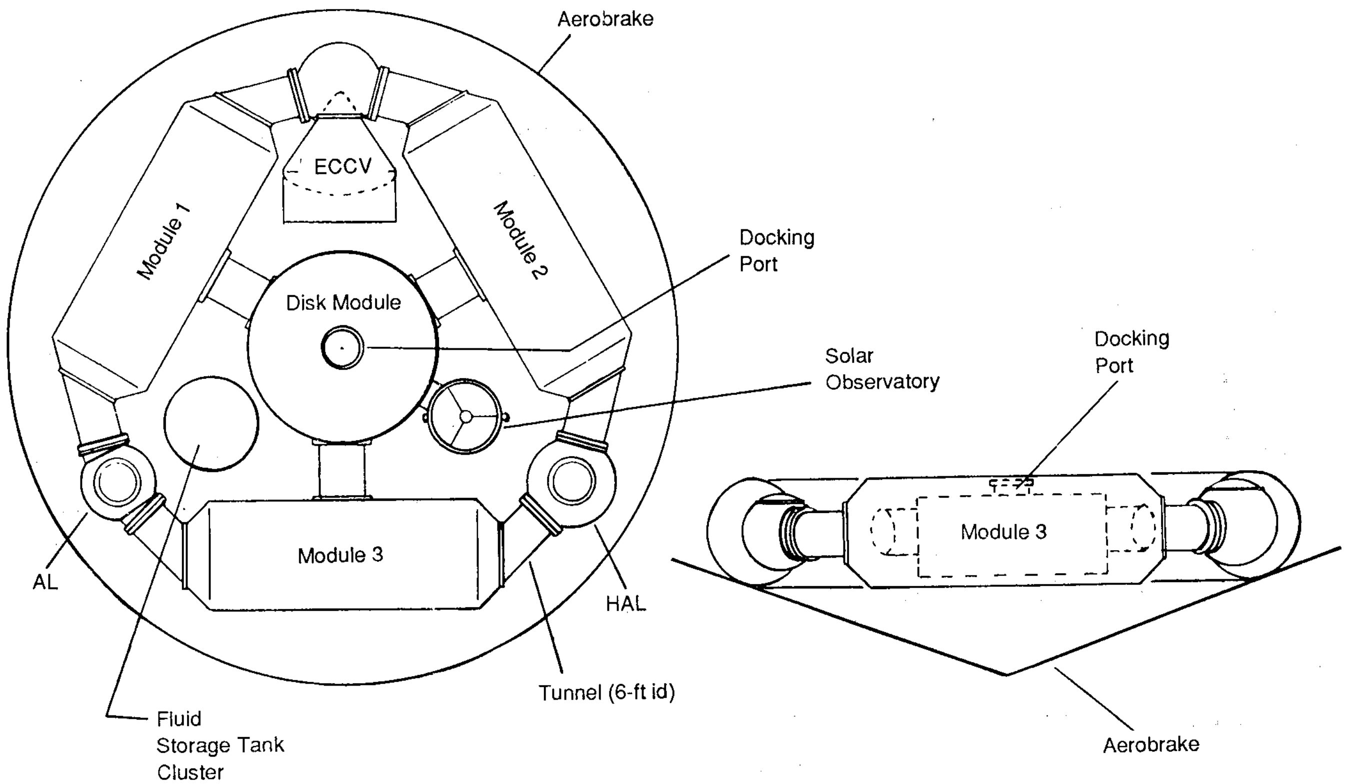


Figure 2.1-6 Detail of Manned Vehicle

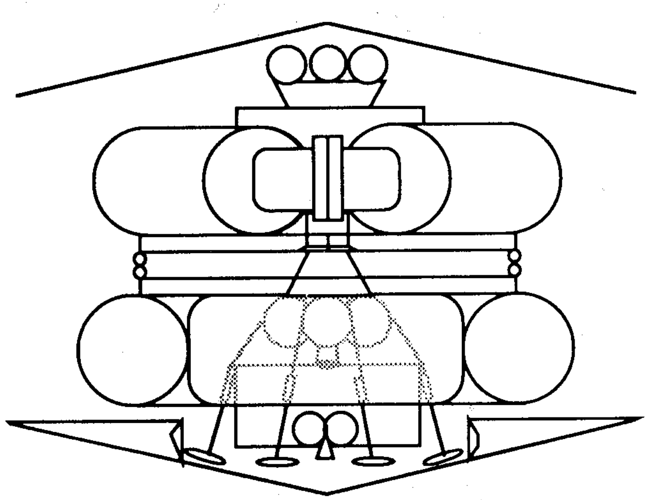


Figure 2.1-7 Docking Configuration in Mars Orbit

The MDV includes deorbit propulsion, an aerobrake, parachutes, and terminal descent propulsion. It contains a single disk module habitat, Figure 2.1-8, connected by a shirt-sleeve tunnel to the small conical cabin of the Mars Ascent Vehicle (MAV). During the complex landing sequence, the crew is located inside the MAV and can accomplish a fly-away abort-to-orbit if a critical fault event occurs. Pressure-fed engines burning storable bipropellants (monomethylhydrazine and nitrogen tetroxide) are used for all descent and ascent propulsion.

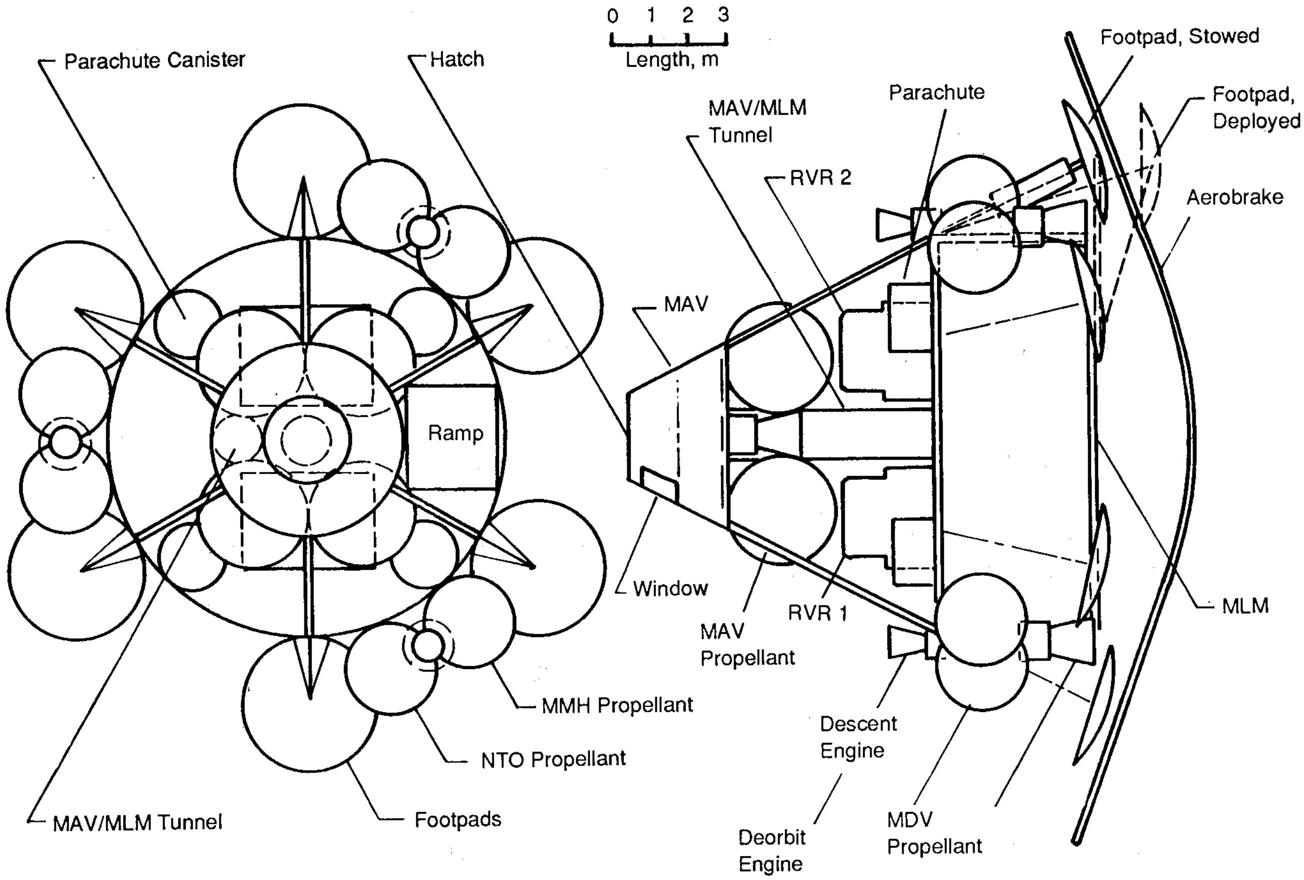


Figure 2.1-8 Mars Descent Vehicle (MDV)

Nuclear thermal rocket (NTR) propulsion was also examined as an alternative for this case study. Major savings can be made in IMLEO if NTR is employed for the TMI. However, as seen in Figure 2.1-9, the leverage of NTR to produce reductions to IMLEO is much, much less for the TEI stage. This is because the vehicle mass drops very significantly in Mars orbit (deployment of MDV), and in this case the ΔV for TEI is small. For this reason, in many mission scenarios, it may not be necessary to attempt the application of NTR to all-propulsive systems. In general, the use of NTR for Earth escape (i.e., TMI) will be the highest leverage application, and in many respects the safest use of this technology.

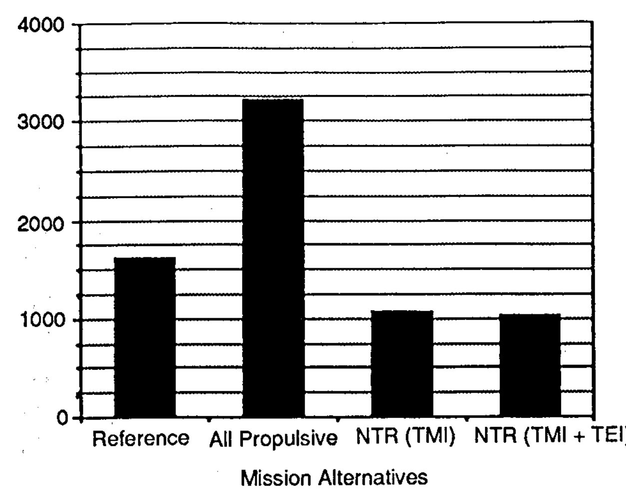


Figure 2.1-9 IMLEO-Savings of Alternative Approaches for Case Study 2

In addition, an "all-up" mission (i.e., single flight rather than splitting cargo and crew between separate flights) flown on a *conjunction* class trajectory was proposed as an alternative. Employing a spinning aerobrake, this concept is discussed in more detail under Section 3.3 on artificial gravity concepts. A comparison of IMLEOs shows that this alternative mission could be performed with almost one-third less mass, yet provide over 70% more habitable volume for the crew, be science-enriched (two pressurized rovers versus one unpressurized; a much larger user payload; more than a ten-fold increase in Mars surface staytime), can be recovered into Earth orbit because of the relatively lower encounter velocity, and arrives with humans at Mars at an earlier date, without requiring a change in programmatics. This alternative, if flown in 2005, requires an interplanetary flight time of 395 days compared to 409 days for the "sprint", but because of the necessarily longer staytimes at Mars has a total mission time of less than 32 months, while the sprint roundtrip is 14.5 months.

The Phobos Gateway was a special study conducted separate from the MASE case studies. This was the first scenario to be baselined with (1) a requirement for artificial gravity (using tethers), (2) non-split (i.e., all-up), and (3) non-sprint (i.e., to use opposition and conjunction class trajectories for crew members). The spaceship design is shown in Figures 2.1-10 and -11. Because the habitation module cluster is detachable for establishing artificial gravity, it can be moved out of the way and the MDV can be flown into and out of the cavity in which it is stowed. For the four missions, starting with the favorable opposition class mission in 2004, the total IMLEO masses are between 575 and 705 t. A novel aspect of the first mission design is that the MDV is landed unmanned and with no MAV on top to demonstrate successful entry and landing of a very large vehicle on the martian surface. Two of the three astronauts on-board the main spaceship for this first mission enter the MAV, before MDV separation and the demonstration landing, to fly an exploration mission to the martian moon Phobos. The ΔV for this maneuver is 3169 m/s, well within the 5400 m/s capability of the MAV (sized for ascent to elliptical orbit from the martian surface). This approach allows landing system verification as well as the use of the MAV to make major orbital changes and rendezvous and docking in Mars orbital space. On subsequent conjunction class missions, the crew complement is increased to five and the MDV is used to land four astronauts each time on Mars.

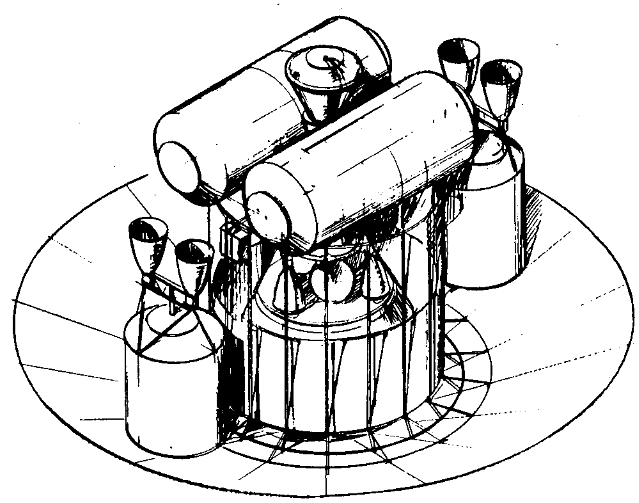


Figure 2.1-10 Phobos Gateway Mars Spaceship Design

The Phobos Gateway study also resulted in the concept of a Shuttle-Z HLLV, more fully described in Section 5.4.1. The Shuttle-Z version selected provided 140 t of useful payload into LEO if the spent upper stage is counted as reusable for the TMIS or other propulsion system; otherwise, the usefully delivered payload was 127 t. From this precedent, virtually all subsequent case studies have specified the LEO delivery capability of the HLLV at 140 t, although Earth-to-orbit vehicles other than the Shuttle-Z are also under consideration for providing this lift capability. A 10-meter diameter launch shroud has also been adopted for all of these Mars mission HLLVs.

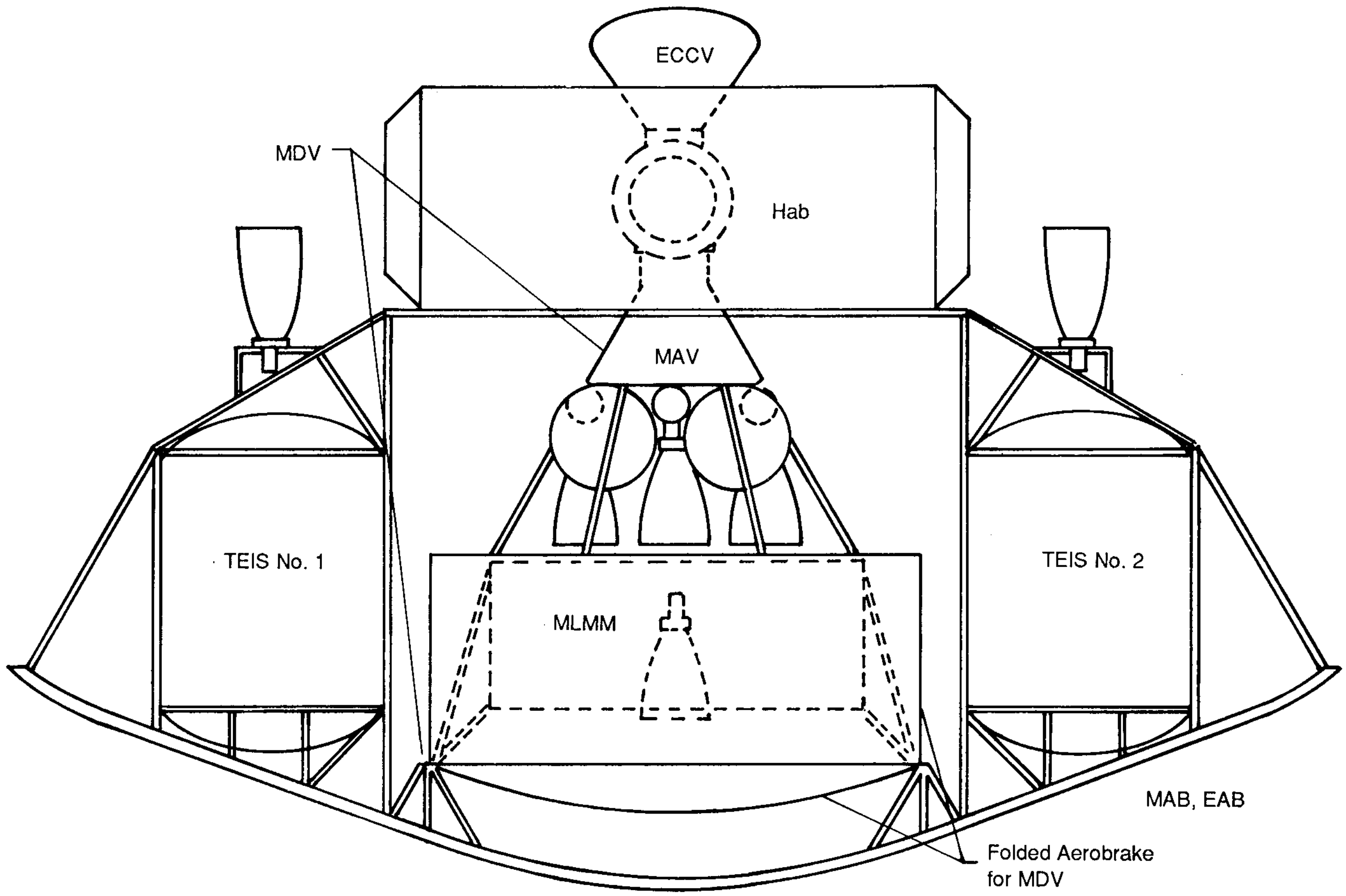


Figure 2.1-11 Phobos Gateway Spaceship, Functional Layout

A sequence for launch and assembly of the Phobos Gateway spaceship is shown in Figure 2.1-12. The first flight launches the spaceship without the MDV. The habitat module cluster is stored in the cavity where the MDV will eventually reside. The hybrid rigid/flex aerobrake is deployed remotely and automatically, folding out umbrella style from its launch configuration. The spent Shuttle-Z upper stage is maneuvered to the front of the aerobrake and docked to ceramic attach points on the rigid core's leading surface. The second flight brings up the MDV, whose aerobrake is also a hybrid rigid/flex design, but will not be deployed until reaching Mars orbit. Also included in the payload are the two TEIS propulsion modules (wet). Man-tended telerobotic assembly operations install the TEIS and the MDV into the cavity after "flying" the habitat cluster out. A second TMIS stage is now available for stacking onto the front-side of the aerobrake. Three more Shuttle-Z launches bring up the TMIS propellant as cargo and more TMIS stages—actually, an excess is available since the trans-Mars injection for this mission requires only three stages. A Shuttle launch brings up the flight crew, the remaining consumables to be carried, and the Earth Crew Capture Vehicle (ECCV), which is flown only as a backup system because the entire Phobos Gateway vehicle is planned to be recovered at Earth by aerocapturing. Compared to the earlier case studies, where 25 to 30 launches of HLLVs were needed for each human flight to Mars, the Phobos Gateway approach, using only 5 HLLVs, was considered a breakthrough into feasibility/plausibility for major manned Mars missions.

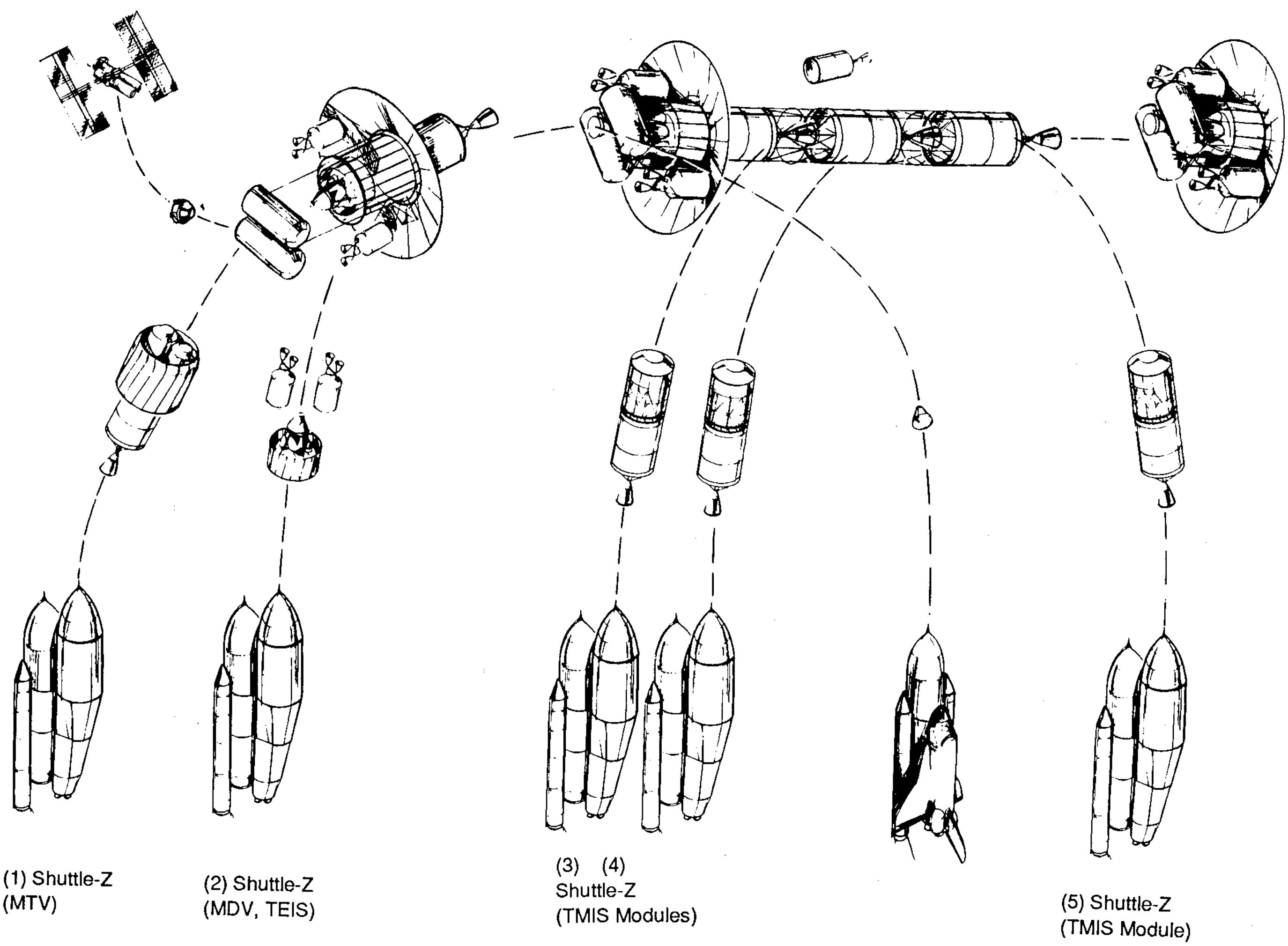


Figure 2.1-12 Launch and Assembly Sequence for Phobos Gateway Mission

The Mars Evolution Case Study 5.0 conducted as part of the fiscal year 1989 MASE effort was actually an ambitious collage of missions with Phobos mandated as an operating centrum for on-site production of propellant and anchoring of tethers for momentum transfer. Artificial gravity was mandated, as was recovery and reuse of some vehicles. For this case study, a large diameter aerobrake was designed, similar to the artificial gravity alternative proposed for Case Study 2. However, to examine another possibility, the cylindrical habitat modules were arranged radially rather than tangentially. To avoid building an aerobrake larger than actually dictated by aerodynamics considerations, it was found necessary to retract the habitats behind the aerobrake envelope during the aerocapture maneuver. A swivel joint was conceived to accomplish this action. The layout of the spaceship is shown in Figures 2.1-13a and -13b. More discussion on this spaceship design is given in Sections 3.3, 4.1, and 7.0. A special lander capable of flying roundtrip to the surface without staging was designed, with the designation of Mars Crew Sortie Vehicle (MCSV). This vehicle is shown in Figure 2.1-14. All propulsion for these vehicles was H/O cryopropellant based.

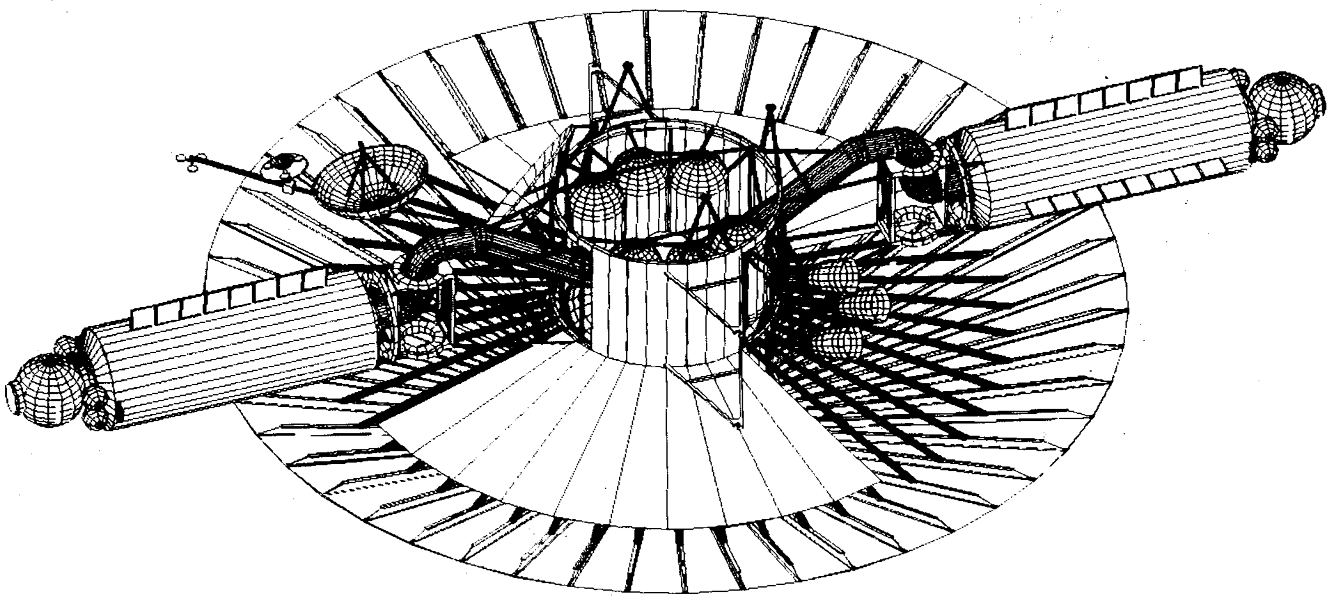


Figure 2.1-13a Mars Evolution Spaceship (Hab Modules Deployed)

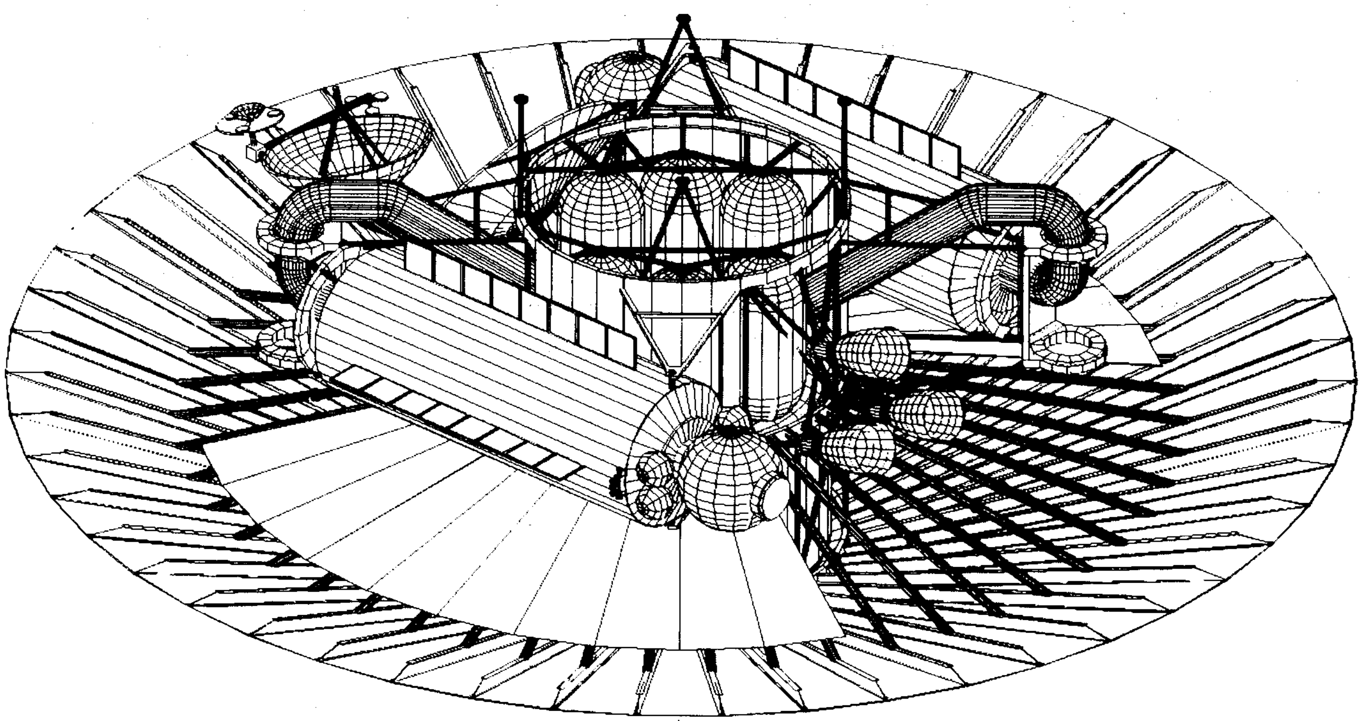


Figure 2.1-13b Mars Evolution Spaceship Hab Modules Stowed for Aerocapture)

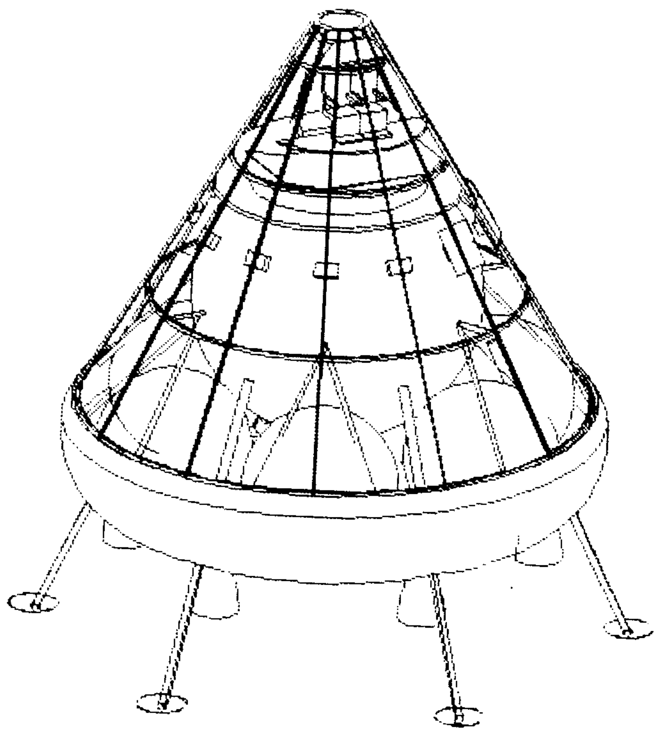


Figure 2.1-14 Mars Crew Sortie Vehicle

The Mars Expedition, Case Study 2.1, is a three crewmember split sprint/conjunction mission to Mars. Unlike Case Study 2, it was decided to eliminate the TEIS from the cargo mission and to incorporate it into the manned vehicle. Furthermore, this case not only required aerocapture at Mars, but actually specified that the design include an aerobrake with a lift-to-drag (L/D) ratio of between 0.9 and 1.2. An L/D of 1.0 was selected. The resulting manned vehicle is shown in Figures 2.1-15 and -16. Because the cold cryopropellant TEIS system was required to be located aft of the crew cabins to satisfy center-of-gravity location envelopes, a highly unsatisfactory thermal situation resulted because the habitats are the centrum of power dissipation and the aerobrake surface can only be breached to allow deployment of radiators by including complex and risky doors penetrating the aerobrake structure and thermal protection system (TPS). Therefore, it was decided to "fly out of the brake" as shown in Figure 2.1-17. In this configuration, the TEIS cryotanks have a much better view factor to deep space, the solar arrays can be readily deployed, astronauts can egress freely, and the habitats can radiate their waste heat to space. Before entering the martian atmosphere for aerocapture, the core vehicle is reeled into the aerobrake and a low-power minimum-sustenance mode of operation is entered so that battery and fuel cell power is sufficient. In case the reel-in operation is unsuccessful, the craft retargets for a Mars flyby and return to Earth. It is highly critical, however, that the core be successfully extracted from the aerobrake after achieving Mars orbit. For this, back-up pyrotechnic release mechanisms are placed at strategic points in the structure. The MDV is brought separately on the cargo flight (Fig. 2.1-18), and re-uses the aerocapture brake for aeroentry. The Mars Expedition case study demonstrated that a fully assembled Mars spaceship is feasible, requiring only propellant transfer to load the TMIS and the mating of the TMIS stages. In fact, the cargo and manned vehicles can be launched on top of their respective HLLVs without need of a payload shroud since the aerobrakes are more than adequate to provide the necessary TPS and structural integrity during the boost into LEO.

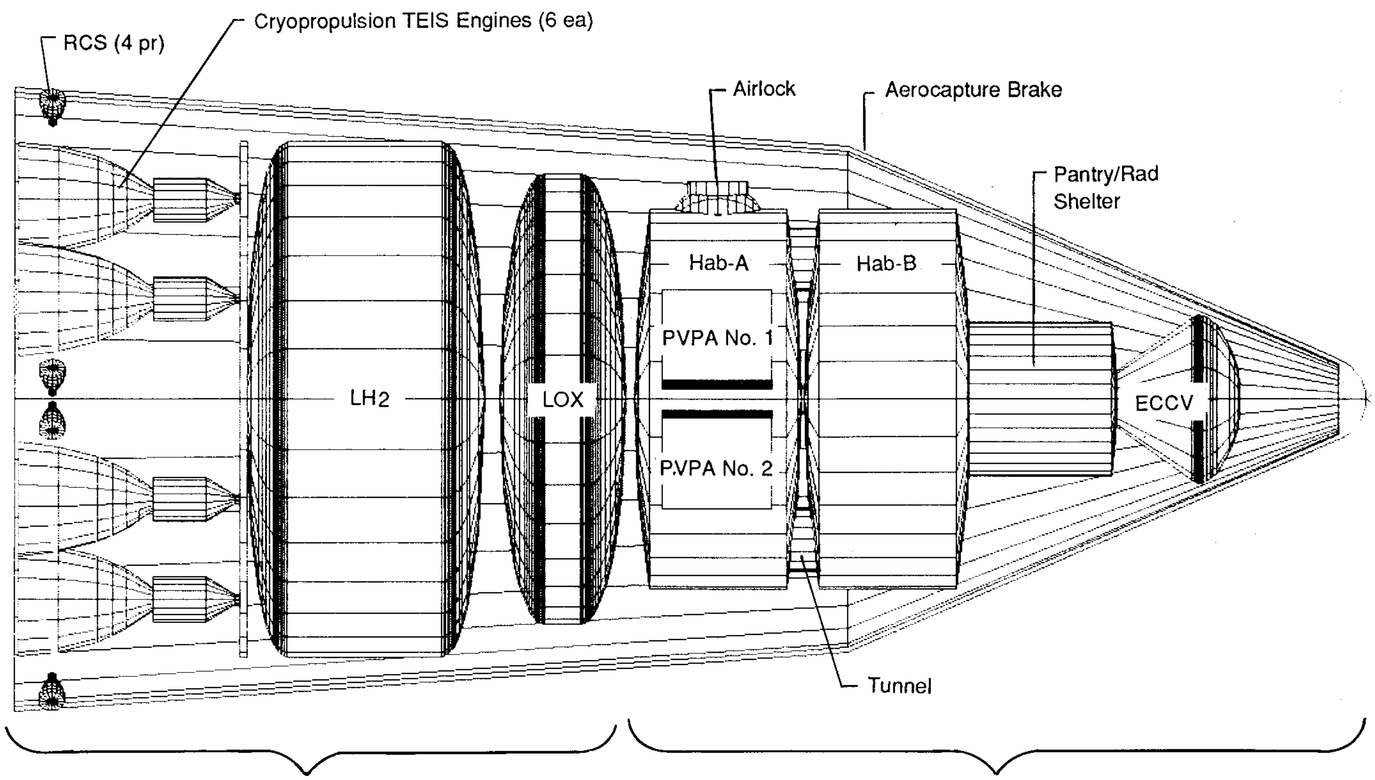


Figure 2.1-15 Mars Expedition Manned Vehicle, Internal Layout

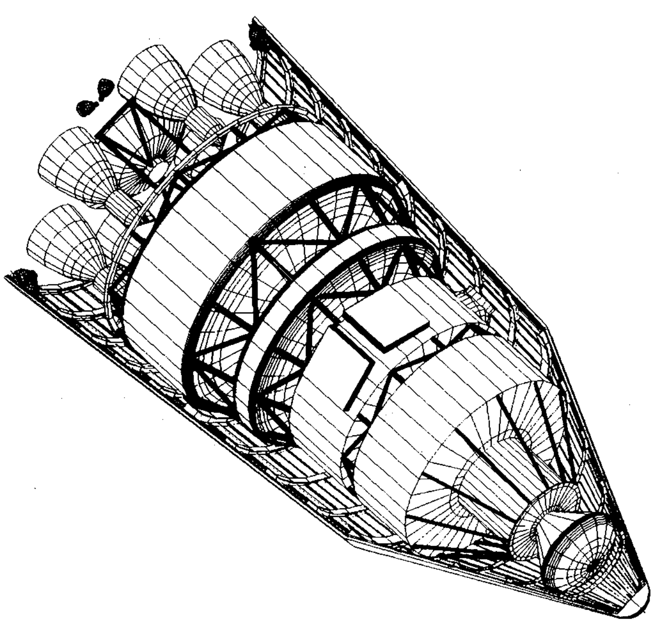


Figure 2.1-16 Mars Expedition Manned Vehicle, Structural Design

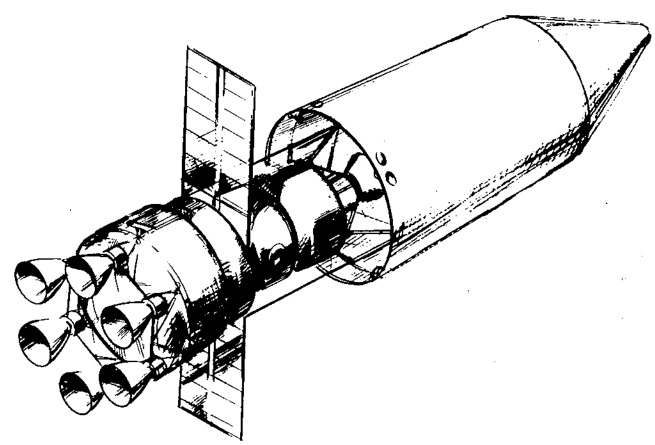


Figure 2.1-17 Trans-Mars Flight Configuration, Mars Expedition Vehicle

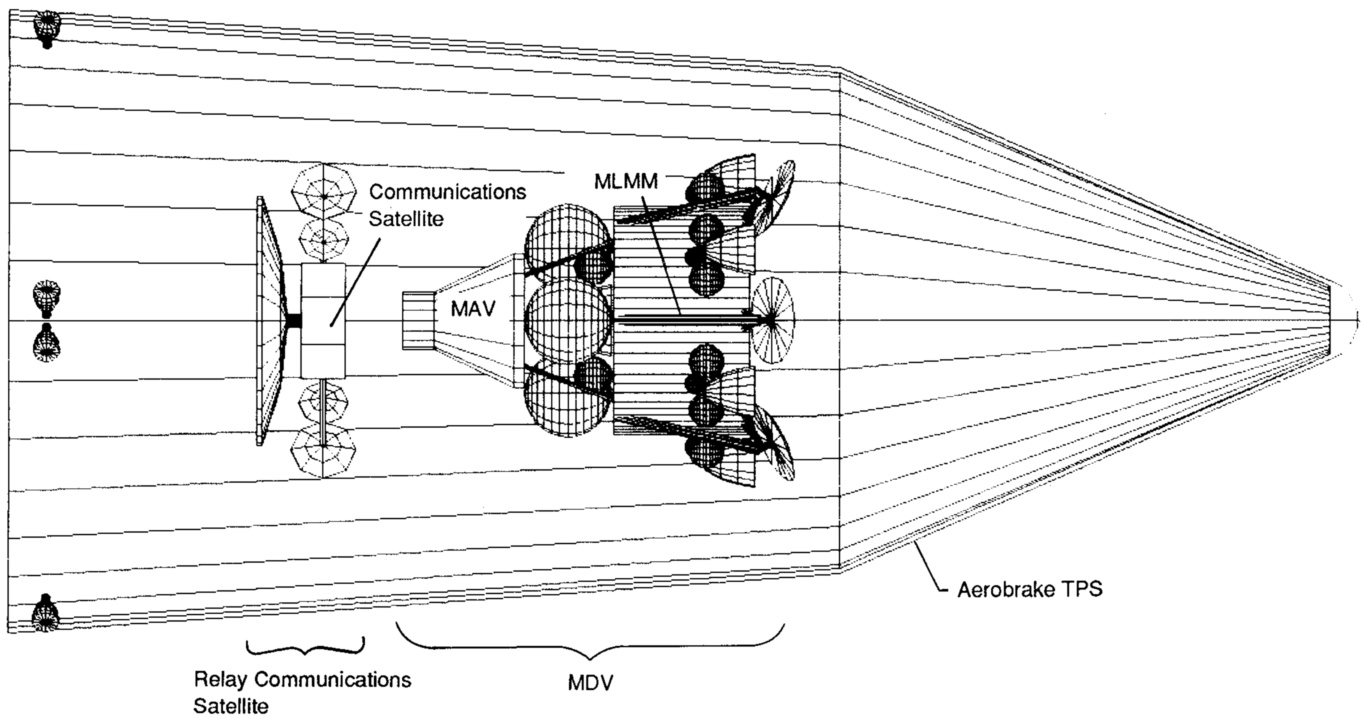


Figure 2.1-18 Mars Expedition Cargo Vehicle

2.2 LUNAR MISSION CASE STUDIES

The lunar human exploration and transportation scenarios that have been studied under this contract are listed in Table 2.2-1. A synopsis of their individual requirements are given in successive Tables 2.2-2 through 2.2-4. Case Study 3 was conducted for the NASA OEXP Fiscal Year 1988 Annual Report, and is documented in NASA Technical Memorandum 4075, dated December 1988. Case Study 4.1 was conducted for the NASA OEXP Fiscal Year 1989 Exploration Studies Technical Report, and is documented in NASA Technical Memorandum 4170, Volume II, dated August 1989.

Table 2.2-1 Lunar Human Exploration Scenarios Studied

Scenario	Date Completed
Case Study 3 (FY88)	7-11-88
Lunar Gateway	11-88
Lunar Evolution (FY89 Case Study 4.1)	6-2-89
Lunar Evolution Synthesis (Modification of FY89 Case Study 4.1)	6-28-89

Table 2.2-2 Synopsis of Requirements—Case Study 3

• **Separate crew and cargo vehicles to moon**
• **Transfers between vehicles in Low Lunar Orbit (LLO)**
• **4 crew, 20 days per mission**
• **Cargo loads**
17.5 t per cargo flight
6.5 t per manned flight
• **First flight (cargo) in 2000**
• **Safe haven on moon for 55 days**
(Required 3 HLLV Launches at 91 t each [200 klb_m], per cargo+human mission)

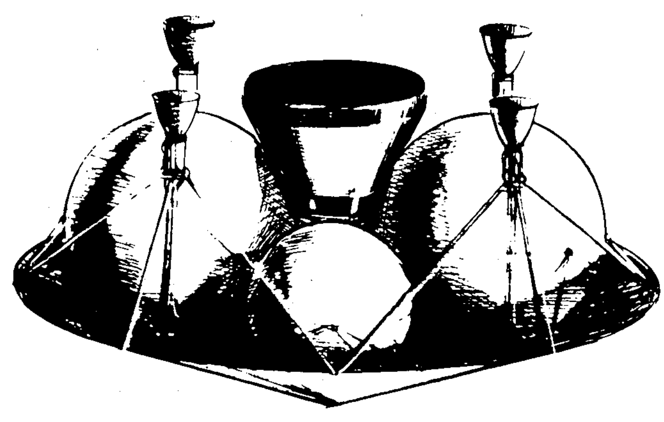
Table 2.2-3 Synopsis of Requirements — Lunar Gateway

• Separate crew and cargo vehicles to moon
• Transfers between vehicles in Low Lunar Orbit (LLO)
• 6 crew, 20 days per mission
• Cargo loads
20 t per cargo flight
• Lunar Descent Vehicle (LDV) is expendable, evolving to re-usable
• LDV has solar flare protection
(Required one HLLV Launch at 140 t each [308 klb_m] and one STS launch with crew and 4.1 t LAV storable biprop; sustained by one Shuttle-C launch at 40 t to replace LDV's)

Table 2.2-4 Synopsis of Requirements—Lunar Evolution (FY89 CS-4.1)

• Three phases on moon:
Outpost (human-tended),
Experimental (to 8-crew),
Operational (to 30-crew base)
• Freedom Station node only
• Re-usable vehicles; 8-crew
• Trajectories: LEO <--> LLO and LLO <--> LSurf
(i.e., LLO is a handover point for all vehicles, but is not a Node)
• Aerobraking at Earth
• Cargo loads
20.0 t per cargo flight (also, 3.6 t LH_2 to LLO)
2.0 t per manned flight
• Pipeline constraints: 570 t (wet), 90 t (dry) per year to LEO
• Chemical propulsion, transitioning to NEP cargo in 2014
• Lunar liquid oxygen (LLOX) propellant produced and used
(Required 2 HLLV Launches at 140 t each [308 klb_m])

Case Study 3 was designed to transfer a crew of four to the moon along with a payload of 6.5 tonnes, but preceded by a cargo mission delivering 17.5 t to the lunar surface. The vehicle staging and architecture builds on the Apollo architecture, but with Earth orbital capture of the Lunar transfer vehicle for reuse on subsequent missions. Lunar Descent Vehicles (LDV; landers) are expendable and a separate Lunar ascent vehicle (LAV) is provided, in analogy with the Apollo Lunar Excursion Module (LEM) ascent stage. Cryogenic H/O propellant systems are used throughout, for both the LTV and LDVs. Engines are RL-10 throttleable derivatives. The LTV includes a 14-meter diameter deployable flex aerobrake for the aerocapture at Earth. Spherical H/O tanks are compact-packaged, taking advantage of the concave form factor of the low L/D aerobrake. A conically shaped return crew cabin is nestled into the four tanks as shown in Figure 2.2-1. The cargo vehicle is designed to hold a centrally located cargo bay, lowered to the lunar surface after touchdown for roll-out of equipment stored in the cargo hold, as shown in Figure 2.2-2. On chosen missions, this cargo hold can be configured as a pressurized disk module habitat. To accommodate this geometry, four propulsion pods are mounted circumferentially. The piloted descent vehicle is modeled after the Mars descent vehicles discussed in the previous section to maximize commonality between the two types of missions. The propellant loads are much different for the two cases. The LAV uses a lightweight crew cab modeled to be similar to the MAV crew cab. Three RL-10 engines are provided for the LAV to enable engine-out recovery.



Crew Return to Earth

Figure 2.2-1 Lunar Transfer Vehicle (LTV) for Case Study 3

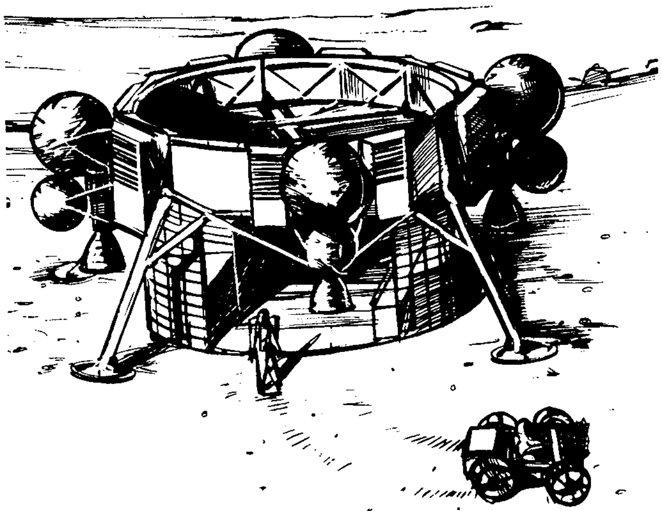


Figure 2.2-2 Lunar Descent Vehicle - Cargo (LDV-C)

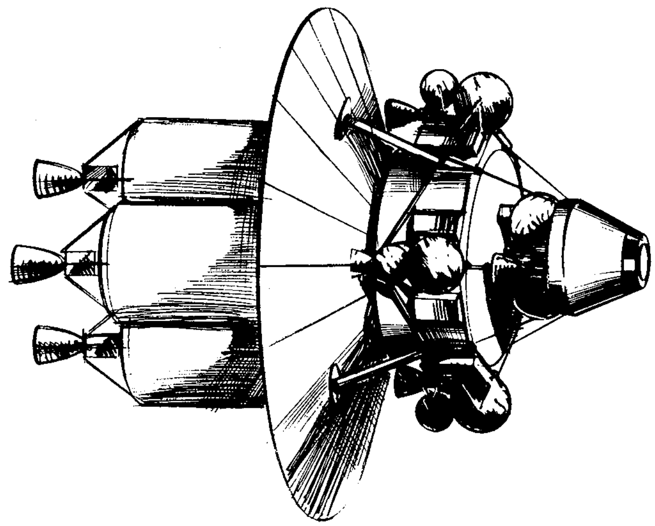


Figure 2.2-3 Lunar Descent Vehicle - Piloted (LDV-P), Mounted to LTV for Lunar Gateway Mission

The Lunar Gateway mission design had a requirement of delivering 20 t of cargo to the lunar surface in unmanned missions. It was adopted that the human missions would deliver a habitat, an LAV, the crew, and an amount of additional cargo that brought the total up to 20 t. With this design stratagem, it was possible to design a large amount of commonality into the two systems—cargo and piloted. Cryogenic H/O was used for all propulsion except lunar ascent; all cryopropellant engines were RL-10 derivatives. The cargo lander was the same as the design concept used in Case Study 3. The LDV-C is transported to low lunar orbit (LLO) in the pusher mode, as shown in Figure 2.2-3. The lander is temporarily attached at ceramic strong-point latches built into the aerobrake's rigid core. The LTV propellant tanks are designed to provide commonality with the Phobos Gateway Mars mission TEIS propulsion units.

The Lunar Evolution Case Study 4.1 conducted as part of fiscal year 1989 MASE effort was an extension of the Lunar Gateway concept, but extended the crew size to eight. The use of lunar liquid oxygen (LLOX) is set as a required factor in transportation system design.

In addition to manned and cargo versions of the transfer vehicles and the landers, there is also specified an eventual lunar propellant tanker to bring LLOX up to LLO. This study employed "through-the-brake" advanced space engines. The Lunar Piloted Vehicle, which is the transfer vehicle from LEO to LLO, is shown in Figure 2.2-4. The reusable lander vehicle, the Lunar Crew Sortie Vehicle (LCSV) is shown in Figure 2.2-5. In LLO, the two vehicles accomplish docking as seen in Figure 2.2-6. These vehicles must also accomplish an in-space propellant transfer to replenish the tanks of the LCSV (note: when LLOX becomes available, the LPT must transfer oxygen into the LCSV and can also re-tank the LPV). Analogous operations occur between the Lunar Cargo Vehicle (LCV) and the Lunar Cargo Lander (LCL) in LLO. When returned for storage and maintenance at Space Station Freedom, the LPV and LCV must be refueled also by in-space propellant transfer. In these design implementations, it was found possible to achieve a high degree of commonality between the LPV and LCV and likewise for the two landers.

Subsequent to completion of the Lunar Evolution study, new requirements for payload delivery were established during the MASE Synthesis procedure, as shown in Table 2.2-5. As a result, the transfer vehicle was redesigned by stretching its propellant tanks (Fig. 2.2-6), to produce the greater carrying capability for payload delivery to LLO and subsequent delivery to the lunar surface.

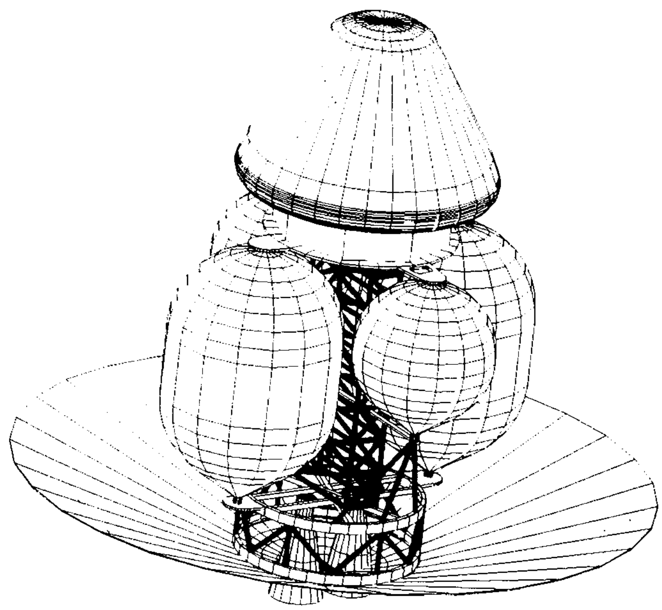


Figure 2.2-4 Lunar Piloted Vehicle (LPV)

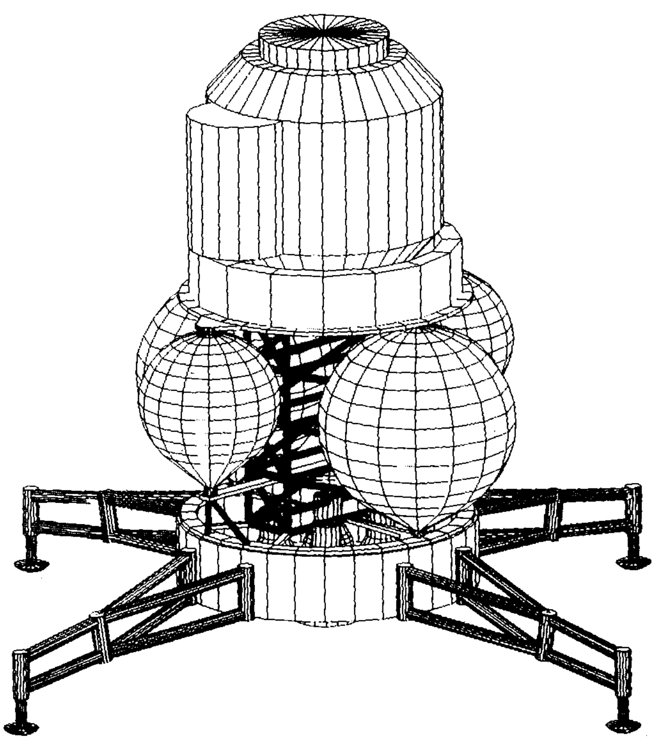


Figure 2.2-5 Lunar Crew Sortie Vehicle (LCSV) (Lander)

Table 2.2-5 Synopsis of Requirements—Lunar Evolution Synthesis

Same as Case Study 4.1 (see Table 2-10), except for
• **Cargo loads**
38.6 t on a one-way cargo mission to LSurf
(No study performed of HLLV Launches required)

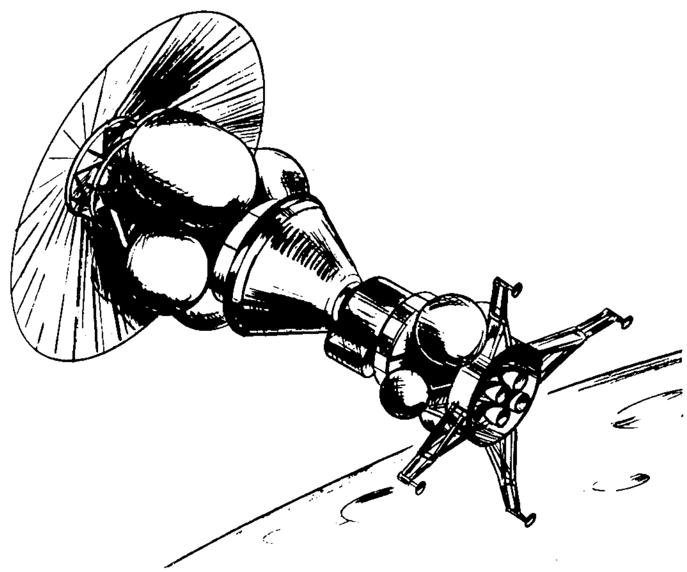


Figure 2.2-6 Docking in LLO for Transfer of Crew

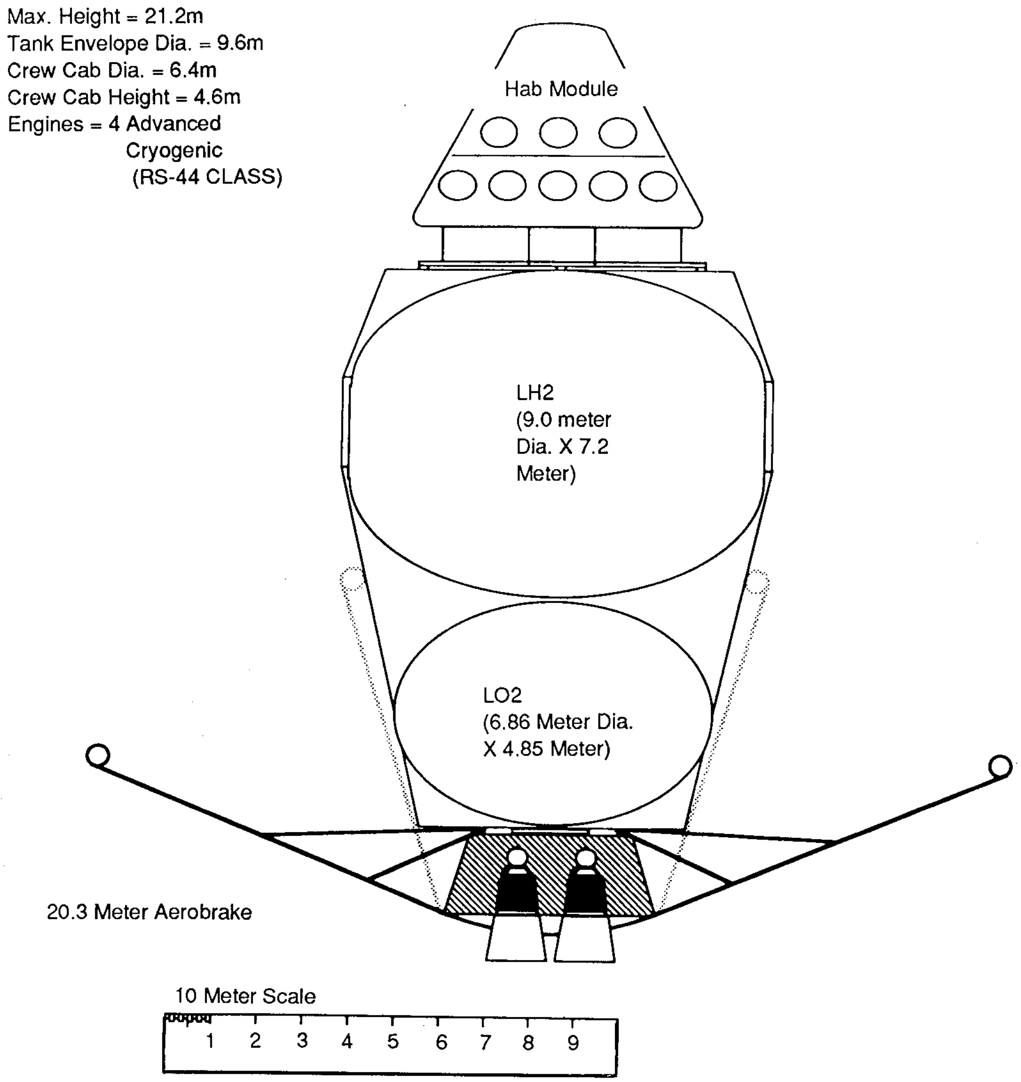


Figure 2.2-7 Lunar Transfer Vehicle (LTV-P), "Synthesis "

3.0 MISSION PARAMETRIC AND SPECIAL TOPICS

With the breadth and level of complexity invoked by human missions to the planets, there are intrinsically a number of topics of significant importance beyond the obvious transportation issues of propulsion, habitability, node support, and aerodynamic surfaces (aerobrakes). These additional topics are discussed in this section, which is placed before the others because in many cases these areas are pivotal in choices or setting requirements that major systems must accommodate.

3.1 ASTRODYNAMICS OF MARS MISSIONS

Missions to Mars vary as the year of the opportunity, with a 26-month interval between opportunities of like type, and an overall near-identical repeat of the synodic cycle every 15 years. To capture the functional characteristics of these opportunities, charted results from the database provided for Mars aerocapture as well as propulsive orbital capture for manned missions by Science Applications International Corporation (SAIC) are shown. These data were optimization runs generated with the use of their trajectory analysis tool, Multiple Impulse (MULIMP). Figure 3.1-1 shows the 17 Mars mission launch opportunities between 2002 and 2013. "Conjunction class" missions are minimum energy Hohmann transfer-type trajectories, the trajectory of choice for all unmanned missions to Mars to date. "Opposition class" missions arrive at Mars when Earth and Mars are at or near astronomical opposition (i.e., closest to one another). These missions involve higher departure and encounter energies than the conjunction class missions. To somewhat alleviate this problem, gravity-assist swingby's of the planet Venus are incorporated when possible. Sprint-type trajectories are a more energetic version of opposition class missions, constrained for very fast roundtrip flights—a total time of 440 days was selected for these calculations.

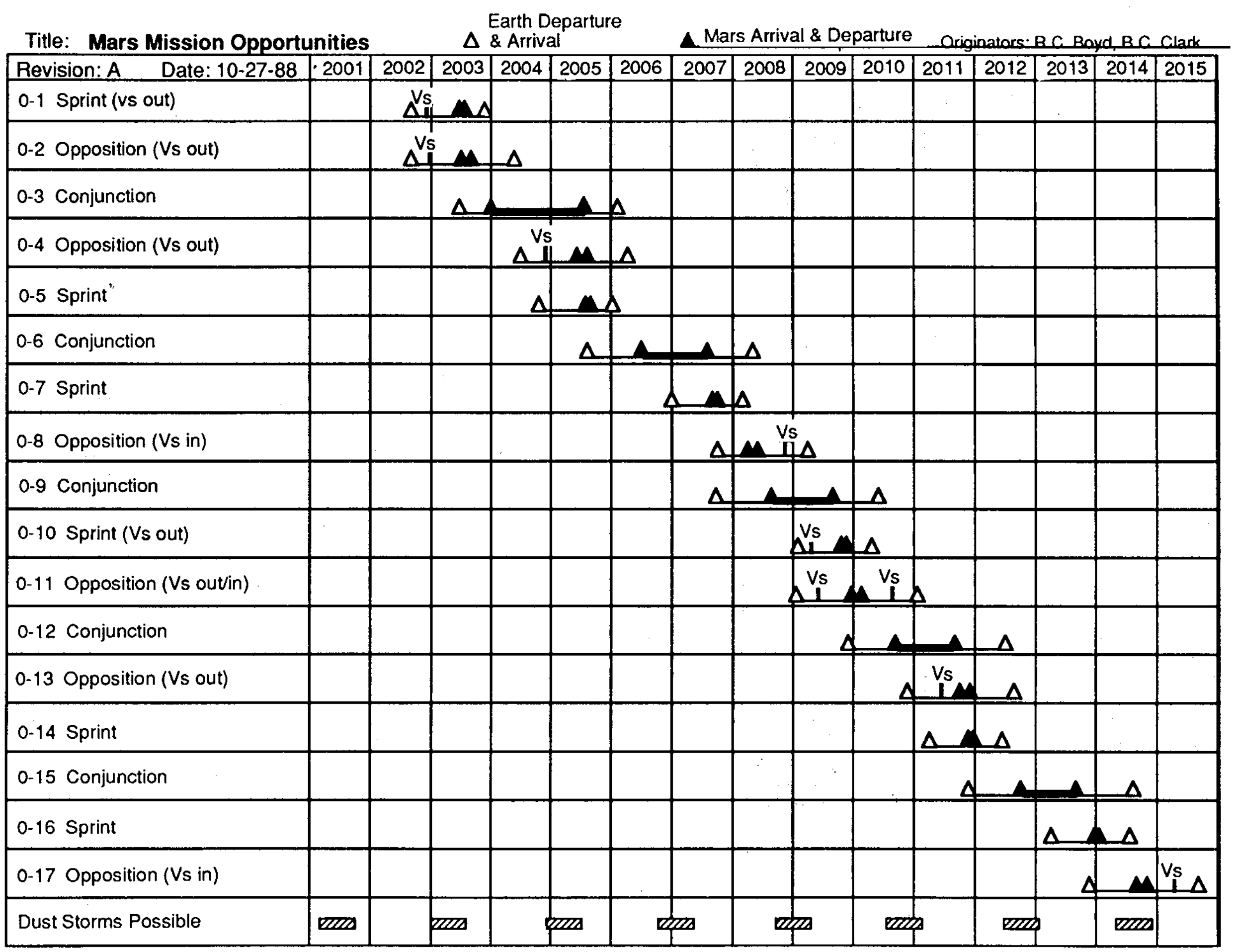
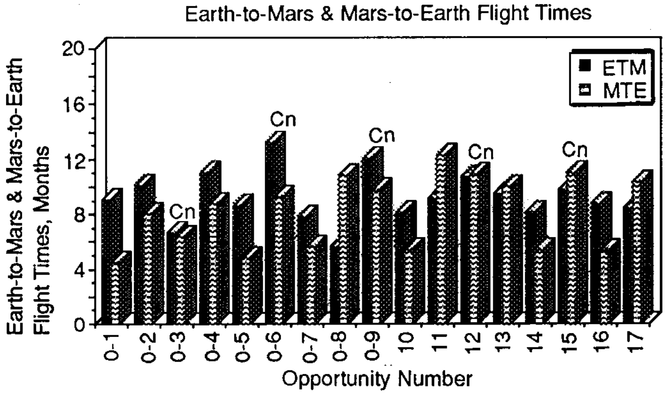


Figure 3.1-1 Mars Mission Launch Opportunities for 2001 through 2013

Interestingly, even though the total trip times vary dramatically among the three mission types (Table 3.1-1), the to-and-from times (i.e., the time spent in interplanetary space) vary less dramatically. Cases even exist where a conjunction class mission can complete one leg of the trip faster than one or more of the sprint class trajectories. On the other hand, the conjunction class missions always produce longer staytimes at Mars, allowing for more time for completion of the mission, as well as the ability to adjust to major martian dust storms, when necessary (see bottom line of Figure 3.1-1 for dust storm seasons). A compendium of trip times are given in detail in Figures 3.1-2 through 3.1-4.

The energetics of Mars missions for the 17 launch opportunities studied are delineated in Table 3.1-2 and Figures 3.1-5 through 3.1-7.



Note:
Data Source: SAIC Database, 10-28- 88

Figure 3.1-2 Earth-to-Mars and Return Flight Times

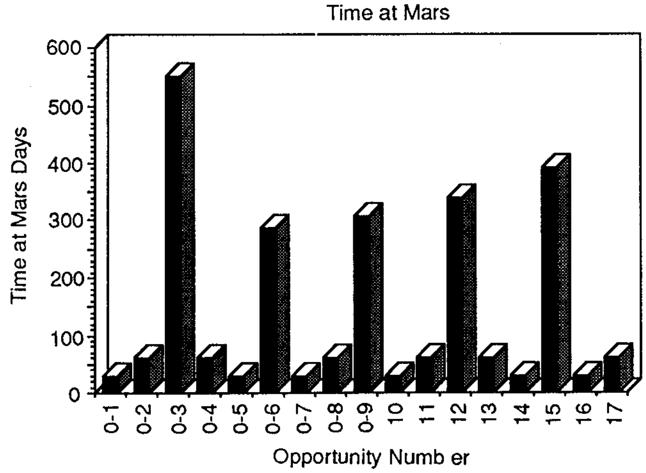


Figure 3.1-3 Mars Staytime for the 17 Launch Opportunities

The arrival and departure declinations for these Mars mission opportunities are shown in Figures 3.1-8 and -9. These parameters are important because these two declinations set the minimum inclination for direct capture (without plane change) and the minimum inclination needed for departure asymptote. For short stay times at Mars, as with the sprint and opposition class missions, it is found necessary to enter very restricted orbits to cause appropriate orbital precession and positioning for the near-term departure. However, for conjunction class missions, the staytime at Mars is sufficient that orbit management can produce the desired effect for relatively minor propulsion penalties. These declination constraints also severely affect the ease of accessibility of Phobos and Deimos because the two moons lie almost nearly in the martian equatorial plane (inclination <1°).

Table 3.1-1 Trip Times (months)

	Earth to Mars (ETM)	Stay Time t Mars	Mars to Earth (MTE)	Total Trip Time
Sprint	8 - 9	1	4.5 - 6	14.5
Opposition	6 - 11	2	8 - 12	19 - 23
Conjunction	6.5 - 13	9 - 18	6 - 11	31 - 34

Note:
(Obtained from Mission Opportunities 0-1 through 0-17)

Table 3.1-2 Encounter Energetics

Opp No.	Trajectory Class	Mars Arrival	C3 (km^2/s^2)	Earth Arrival	C3 (km^2/s^2)
1	Sprint	31-May-03	49.22	12-Nov-03	12.36
2	Opposition	21-Jun-03	42.63	20-Apr-04	38.03
3	Conjunction	29-Dec-03	7.28	13-Jan-06	12.07
4	Opposition	14-May-05	37.98	28-Mar-06	13.94
5	Sprint	01-Jul-05	50.00	23-Dec-05	25.00
6	Conjunction	12-Jun-06	7.10	27-Apr-08	8.56
7	Sprint	06-Aug-07	50.00	24-Feb-08	25.00
8	Opposition	01-Mar-08	38.51	17-Mar-09	20.47
9	Conjunction	10-Aug-08	6.03	23-May-10	8.02
10	Sprint	01-Oct-09	50.00	13-Apr-10	25.00
11	Opposition	07-Dec-09	23.62	21-Jan-11	21.06
12	Conjunction	19-Aug-10	6.07	25-Jun-12	10.25
13	Opposition	27-Sep-11	33.48	29-Aug-12	47.71
14	Sprint	14-Nov-11	50.00	25-May-12	24.97
15	Conjunction	11-Sep-12	7.31	28-Jul-14	14.21
16	Sprint	23-Dec-13	50.00	03-Jul-14	24.96
17	Opposition	18-Aug-14	13.40	15-Aug-15	19.86

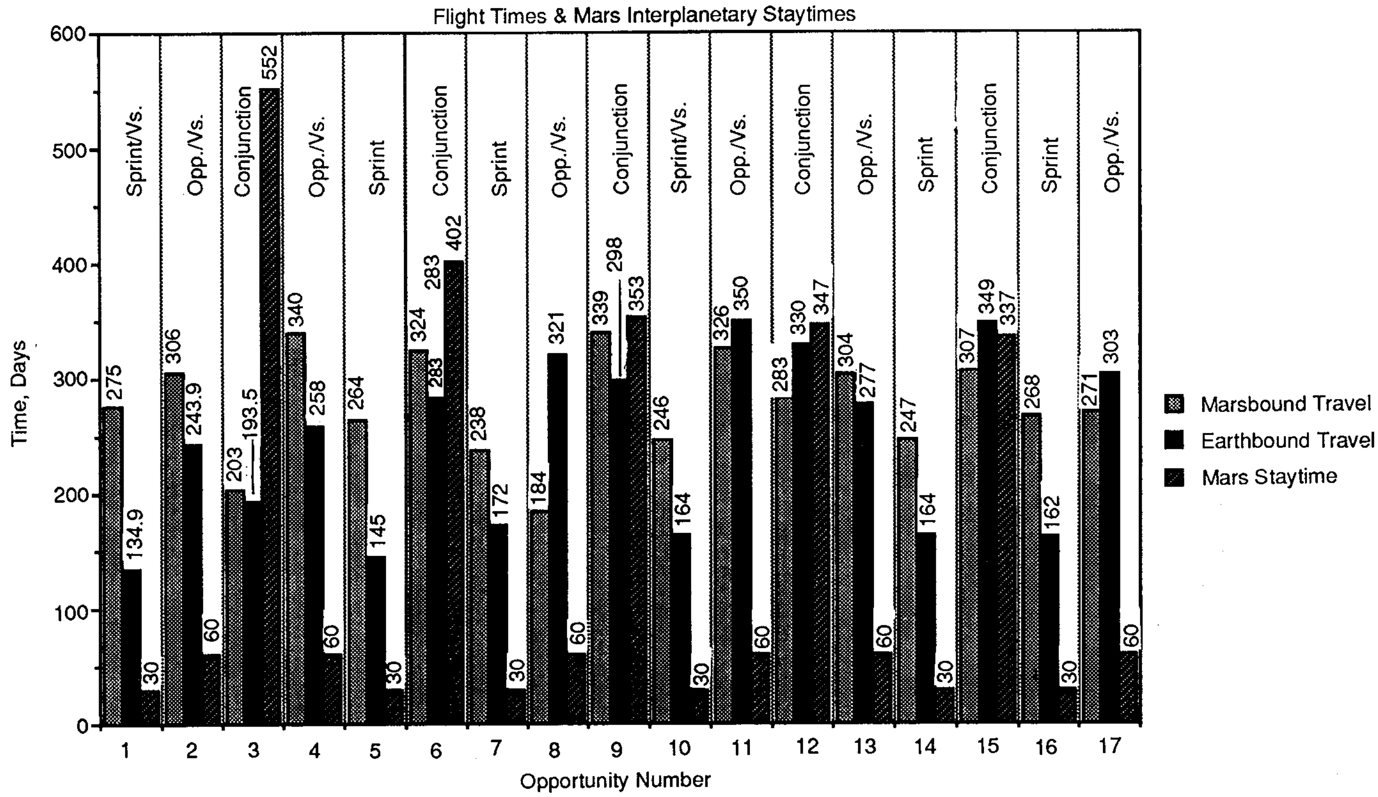


Figure 3.1-4 Full set of 17 Mission Opportunity Flight Times (Time in Days)

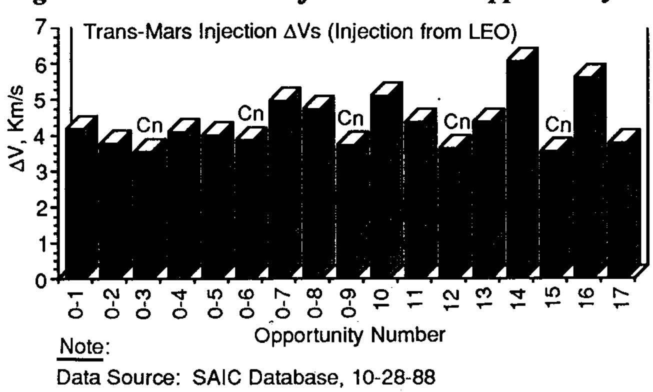


Figure 3.1-5 Trans-Mars Injection (TMI) ΔV

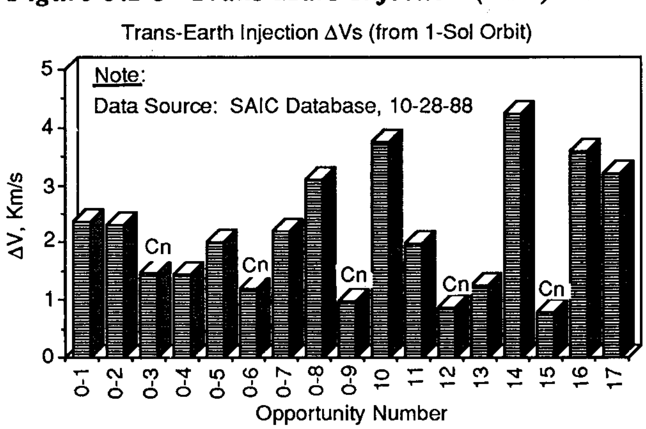


Figure 3.1-6 Earth Return: Trans-Earth Injection (TEI) from 1-sol Orbit

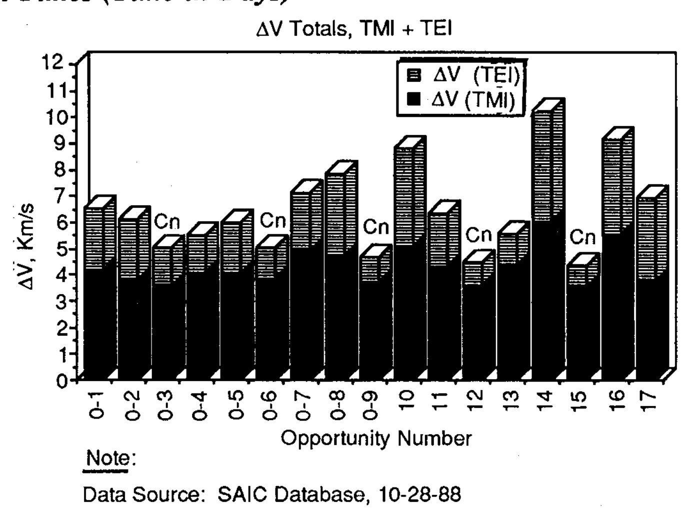


Figure 3.1-7 Summary of Total ΔV's from Previous Two Figures

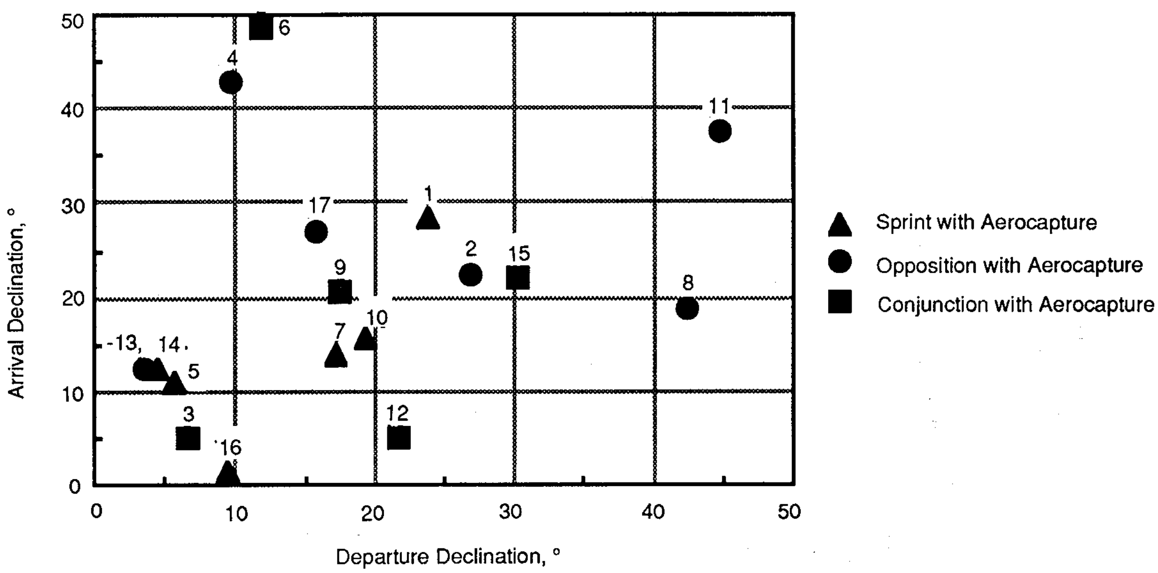


Figure 3.1-8 Declinations at Earth

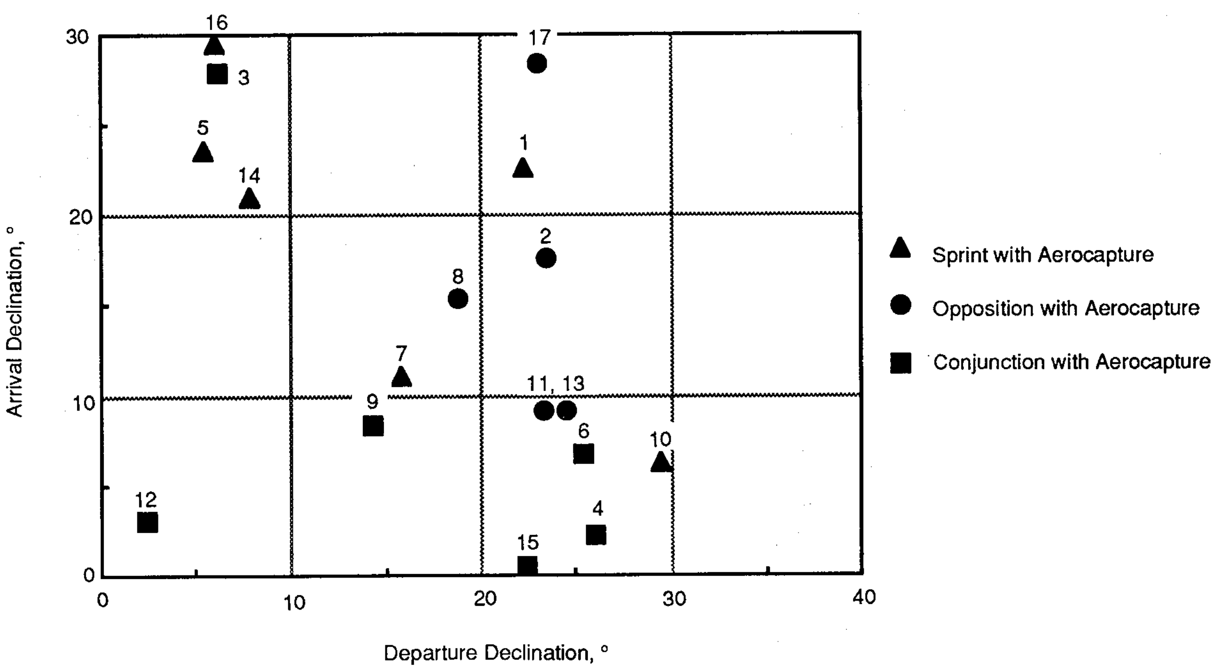


Figure 3.1-9 Declinations at Mars

3.2 RADIATION PROTECTION

Concern for space radiation has been a hallmark of U.S. and Soviet programs since the dawn of the space age. Three types of ionizing radiation must be considered: (1) planetary radiation belts, (2) galactic cosmic rays (GCR), and (3) solar particle events (SPE) from flare activity. The first involves only the trans-Mars injection trajectory through the Earth's Van Allen belts, because Mars has no trapped radiation zones due to the absence of an appreciable magnetic field. The second is ever-pervasive once the partial deflection effect of the geomagnetic field is left behind. The third is highly episodic—for a 3-year mission, the hazard ranges from none at all to potentially lethal.

Shielding optimization is different for all three types of radiation environments. It is impractical to provide the mass in the habitat necessary to totally eliminate the dose that must be taken by GCR. It is likewise unnecessary to provide Mars surface shielding against GCR since the transit times are longer than the surface residence time. Shielding against solar flare radiation can and must be provided. Most SPE energy spectra are such that modest shields are quite effective. However, events rich in relativistic particles can occur and contingencies should provide emergency survival shielding. Food and water provisions, as well as internal flight equipment, can be used for bulk shielding. In addition, an extensive on-board monitoring and warning system is required. Components of the system include active fix-mounted dosimetry, portable active and passive dosimetry, and various continuous solar-monitoring devices. With these approaches, the risk to radiation insult should be reducible to that commensurate with other major mission risk factors. On-board radiation, such as from radioisotope or nuclear fission power sources, may also be a major concern if such sources are required for success of the mission. The doses from these artificial sources generally are placed under different guidelines than the exposures to natural radiation such as the three sources discussed above.

3.2.1 Planetary Radiation Belts

Although the Van Allen belts around the Earth present some hazard, as shown by the Apollo missions, as long as the traversals are short the integrated dose behind even minimal wall thickness are acceptable. Neither the moon nor Mars have radiation belts because of the lack of significant indigenous magnetic fields.

3.2.2 Galactic Cosmic Rays (GCR)

The expected level of hazard from GCR is currently under intense scrutiny and uncertainty. It has been recommended, however, that it might be necessary to provide wall thicknesses or added shielding of up to 25 g/cm^2. When this option was examined for the Mars Expedition (CS 2.1), it was found that the habitat modules increased in mass by more than 55 t and the IMLEO for the mission increased by over 75%. If the cost-to-LEO is taken as 5 M$/t, then the transportation cost of the radiation shielding alone would be of the order of two and a half billion dollars for each Mars mission. From this sensitivity value, it is seen that the actual level of shielding required can be of major consequence to a Mars project.

3.2.3 Solar Particle Events (SPE)

Solar flare eruptions on the sun can produce large fluxes of protons and heavier ions that propagate into deep space. It well known that without shielding, large solar particle events (SPE) can produce radiation absorbed doses in the 10's to 1000's of rad, which encompasses the range of lethality to the human organism (about 500 to 1000 rad). Any missions outside the protection of the Earth's geomagnetic field, including all missions to Mars or the moon (neither of which have any effective magnetic field) will subject astronauts to this potential hazard. For the large majority of the total dose, the particles arrive as an omnidirectional flux rather than a unidirection beam from the direction of the sun. This is because all but the most energetic particles are trapped in a magnetic flux bubble from the sun that completely envelops the vehicle as it passes. During the course of a 3-year conjunction class mission to Mars, approximately 6 SPEs of various size will be encountered. Missions to the moon face a lesser risk, but only because of the reduced trip times. The Apollo missions relied on this lower probability, and somewhat fortuitously avoided these exposures.

Because the SPE ionized particles are constrained in their motion by the spiraling interplanetary magnetic field of the rotating sun, some events will be detected at Earth but not at other locations. Likewise, some SPEs can be experienced on the way to or at Mars with no detectable events at Earth. For this reason, an on-board monitoring and prediction system is needed for Mars missions. For Lunar missions, these functions can be accomplished by one or more appropriately placed unmanned satellites.

Active regions on the sun can be monitored before the eruptions that produce solar flares. Spaceborne instrumentation should map solar phenomena in the visible, UV, and x-ray wavelengths, as these emissions have been shown to be most indicative of impending outbursts and particle releases. Other diagnostics, such as radio wave, IR, magnetic field measurements, and directional relativistic particle streams should also be considered. For Mars missions, a spaceborne computerized Expert System for real-time data analysis, interpretation, and prediction must also be part of this technology because round-the-clock manned monitoring is an unacceptable mission requirement.

The prediction of the possible or probable occurrence of an SPE is necessary to provide warning for safe activities. Examples of such activities include avoidance or termination of extravehicular activity (EVA); reconfiguration of equipment and supplies to augment a radiation shelter; readying partial body shields, activating and placement of dosimetric devices, testing of alarm and monitoring systems; increasing solar monitoring rates; protection of astrophysical instruments with covers or power-off; rescheduling of cabin activities; and activation or increased readout of supplementary spaceborne and ground-based solar activity monitoring systems. This monitoring is not considered optional; it is required for all planetary-class manned missions.

3.3 ARTIFICIAL GRAVITY

Long interplanetary flight times, combined with possibly protracted stays in Mars orbit, would subject crewmembers to up to three years of weightlessness. In view of the known problems with zero gravity, a spinning spacecraft offers many advantages and may indeed be an enabling technology for human travel to Mars. Several concepts have been developed during the course of these studies.

Missions in space as well as Earth-based medical studies have clearly demonstrated a number of human physiological adaptations to the absence of normal gravitational forces. These adaptations might be acceptable were the human subjects never again to require exposure to gravity, but their occurrence must be considered highly detrimental and possibly threatening to the short and long-term health of astronauts when returning to Earth or landing on another planet. Among the effects are the well known progressive losses of skeletal mineral mass; the atrophy of most muscles, including the heart; and the susceptibility to orthostatic intolerance. Potentially serious effects also include alterations in both immunological and pharmacological response. The duration of trips to Mars—which can be from 14 to 36 months for a nominal trajectory, but toward the longer times for abort mode trajectories of many different trajectory classes—is beyond current experience. No assurances can yet be obtained that countermeasures to chronic deprivation of gravitational forces for these lengthy periods can be successfully found, especially if the pacing criterion is that no ultimate health effect will result.

Fluid pooling and alterations in vestibular response may be less serious in the long run, but are certainly in the category of significant annoyances. In addition, Space Adaptation Syndrome (SAS) is known to affect different persons for unpredictable periods of time and levels of severity.

Most importantly, it may be extremely difficult to evaluate crew candidates for their susceptibility to long-term difficulties with the zero gravitational environment based on testing done on Earth alone. This would imply a requirement for a major program for screening of astronaut candidates in space, with all the attendant financial costs as well as losses in mission preparation time. An artificial gravity environment holds the potential of providing levels of protection adequate for most members of the candidate crew population.

There are, however, negative aspects of rotational acceleration to provide artificial gravity. These include coriolis effects, which cause pseudo weight increases when astronauts move in the direction of vehicle motion, weight decreases when moving opposite, and a number of unconventional kinematic effects, especially when objects move transverse to the axis of rotation and the primary direction of motion. A significant gravity gradient occurs vertically along the human body unless the radius of rotation is very large.

Vestibular disturbances caused by head movement are possible until the astronauts adapt to the spinning environment, which is normally within hours or days, as with SAS. EVA is much more hazardous under artificial gravity because of forces tending to separate the crewmember from the vehicle. Many unique advantages of zero gravity are no longer available: e.g., the capability to move very massive objects by human power, the assumption of any orientation, and access to the ceiling storage areas as readily as the floor. The Extravehicular Mobility Unit (EMU), or spacesuit and life-support system, cannot be of the new hard-suit designs because of excessive weight. It hardly requires stating, however, that spinning space vehicles can also be despun to temporarily achieve certain advantages of microgravity.

On the other hand, artificial gravity has many secondary advantages. For example, it establishes a well-defined vertical/horizontal reference system. On another level, pseudogravity can enhance the quality of life, including such amenities as automatic collection of trash and particulates to the floor area; a conventional toilet; more normal eating and grooming practices; and convective motions of atmospheric gases to promote more normal heat transfer. Performance of straightforward as well as complex medical procedures, including surgery, could become difficult and unreliable under zero gravity.

By adjusting the gravitational environment to the Mars-g level (approximately 3.7 m/s^2, or 38% of Earth-g), it will be possible to provide prior adjustment to the Mars surface environment. Training in the Mars Descent Vehicles and with donned spacesuits to simulate operations on the martian surface will also be possible with greater simulation fidelity than can be attained on Earth.

Although some implementations are more intricate than others, there has been no indication of any major engineering obstacles that would impede the development of artificial gravity spacecraft. Remaining issues involve concerns over dynamical effects (including vehicle instabilities, which might be overcome using active reaction systems or through judicious incorporation of passive damping by tailored structure), assessment of the effect on crew operations, and of course the vital tests to verify and quantify the protective efficiency of the rotational acceleration environment against the multitude of deleterious physiological effects of zero gravity on the human organism. In summary, a large number of advantages support serious consideration of the artificial gravity approach to interplanetary space travel.

Many assessments of artificial gravity were performed during the first decade or so of the U.S. space program. These included analytical and theoretical treatments as well as experimental work with slowly rotating rooms (SRR) and other special centrifugal apparatus. This impressive body of work now numbers over 400 references in Dialog's NASA/Aerospace computerized databank. In addition, there exists a less documented literature consisting of unpublished papers prepared for presentation at various symposia and meetings on the subject.

As almost everyone is aware, rapid rotation of the human body can result in temporary disorientation and inaccurate coordination of limb movements. Nausea is common, although not a universal occurrence when a subject is rotated rapidly in a spinning chair. Vestibular disturbances caused by rotations are well known and have formed the basis for extensive scientific investigations. Using the slowly rotating rooms and a variety of experimental subjects, acute rotation effects phenomena have been extensively studied to rates as high as 10 and 12 rpm. By allowing a progressive adaptation through stepwise increases in spin rate and judicious restriction of head motions, it has been found that most human subjects can adjust more quickly to perform well at the high rates and avoid adverse physical symptoms. Early objections to rotational artificial gravity ("rotogravity") above 3 rpm apparently were based partly on the mistaken impression that zero gravity does not cause motion sickness, because of a lack of recognition of the occurrence of SAS malaise. An early suggested ceiling on rotation rate for artificial gravity of 4 rpm was also later revised upwards based on good results at 10 rpm. From analysis of a large number of limiting criteria on human performance, various sources recommended that the radius should be at least 14.6 m and the spin rate be normally limited to 6 rpm.

There unfortunately exists no comparable body of evidence on the effectiveness of acceleration levels of less than one Earth-g in maintaining physiological health. Combined with rigorous periodic training, levels of less than one Earth-g may effectively counteract the normal tendency for skeletal degradation over long periods, with a time scale of several years. Bed rest studies and other experimental approaches such as unweighting and neutral buoyancy could address certain of these clinical responses to provide a measure of confidence in partial gravity effectiveness. Proposed animal experiments on short-radius centrifuges in a microgravity laboratory in space may not allow definitive conclusions vis-a-vis humans in artificial gravity because of the following reasons: animal models of human response are poorly extrapolated in this case and animals can neither exercise nor ambulate in the same manner as humans. Rather, an in-space rotational spacecraft with human occupants, such as an Artificial Gravity Research Facility (AGRF), could directly answer virtually all of the concerns that have been raised for the application of artificial gravity to astronaut health and peak performance maintenance.

A total of five different approaches to rotational artificial gravity have been examined (Fig. 3.3-1). Concept 1, produced during the pre-OEXP study phase of the work, is a ring of eight cylindrical modules on a 41-m diameter aerobrake, intended to house a large crew of 12 to 18 members. The brake is sized for aerocapture at both Mars and again at Earth to allow recovery of the vehicle. Concept 2 is the "Bent-I", a modified version incorporating only 4 modules for smaller crew sizes, but lacking the feature of the full ring of Concept 1 that enables a 100-m circular jogging track for fitness maintenance. This concept was presented as an alternative to Case Study 2 for the Mars Expedition study in FY89. It was shown in this alternative that by selecting a conjunction class trajectory rather than a sprint mission, not only could the IMLEO be reduced by one-third (more than 500 t) but more than 70% additional habitable volume could be provided, the mission could be flown all-up, and the science and exploration payload could be doubled. For sprint missions, of course, artificial gravity may not be called for because of the fact that Soviet Cosmonauts have apparently survived one-year exposures without major detrimental consequences.

Concept 3 is the Mars Evolution (Case Study 5.0), which also employed a large aerobrake, but the cylindrical habitat modules are mounted on swivel joints to allow them to swing out for artificial gravity spinning. For planetary capture, the modules are cranked back inside the protective envelope of the aerobrake. Habitats for this concept are provided with five separate floors (Ref. Section 4.0). The ladder access between decks will provide opportunity for incidental as well as purposeful exercise.

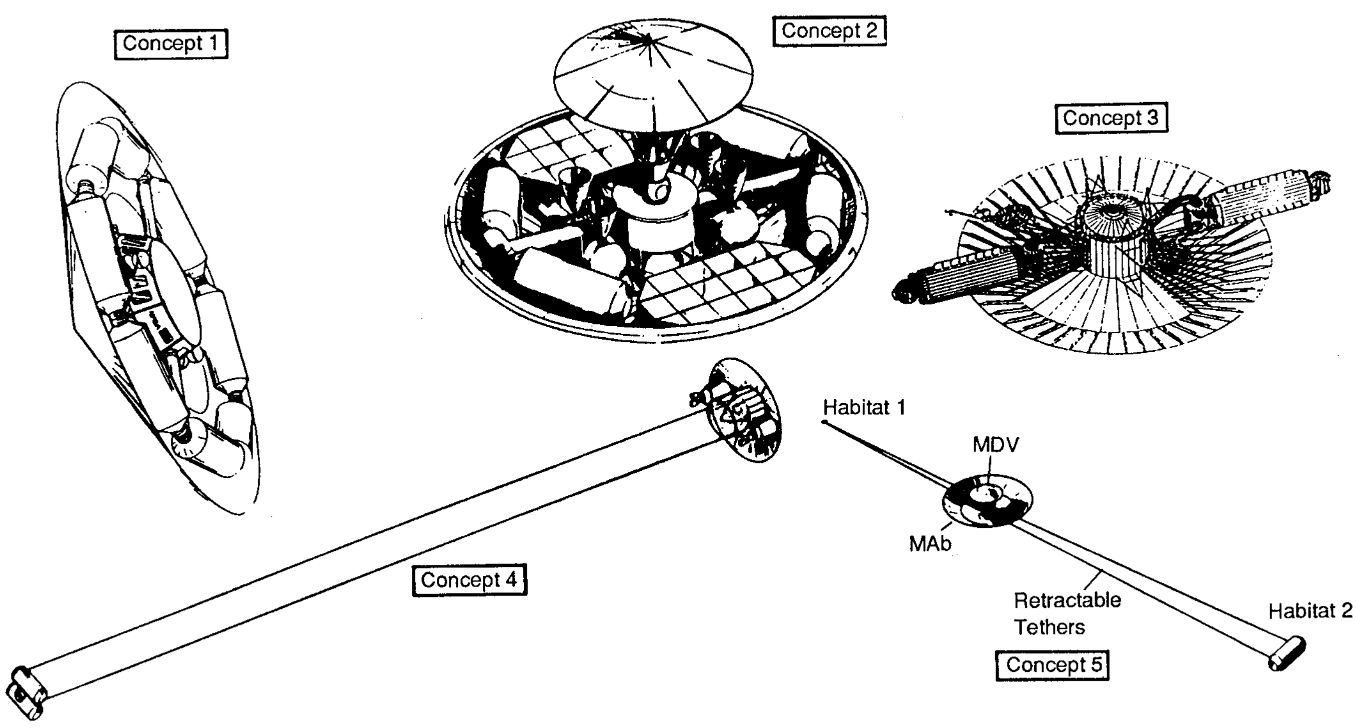


Figure 3.3-1 Artificial Gravity Vehicle Concepts

Concept 4, developed as part of the Phobos Gateway study, uses dual tethers to separate a cluster of cylindrical habitat modules from the aerobrake and primary propulsion systems. Rotation is about the center of mass at some point between the separated objects. With tether lengths of several hundreds of meters, rotation rates as low as 2 rpm can still provide 9.8 m/s^2 (1.0 Earth-g) of centrifugal acceleration for the inhabitants. Concept 5 deploys two habitat modules symmetrically from a central structure. This fixes the center of mass, avoiding the ambiguity of the required tether length in concept 4 that results from variations in mass ratio between the two objects as the propulsion and payload of the aerobrake cluster change at major mission events (propellant burns, MDV deployment, etc). In concept 5, using tether lengths of 226 m each, a rotation rate of 2 rpm produces one Earth-g.

3.4 LIFE SUPPORT SYSTEMS

During the course of this study, Life Systems, Inc. of Cleveland, OH and Martin Marietta worked together to design and illustrate an optimum Environmental Control Life Support System (ECLSS) for manned Mars missions. The study program consisted of the definition of ECLSS requirements for advanced space missions, identification of unique mission drivers, assessment of existing ECLSS technology capabilities, application of these technologies towards the design of an ECLSS for manned Mars vehicles, and identification of ECLSS advanced technology needs.

The main focus of our effort was directed towards the early stages of a manned Mars mission: specifically, short duration and preliminary exploration habitable vehicles. Mass requirements for ECLSS onboard this type of vehicle are detailed later in this section.

Of the characteristics that describe any mission, several, e.g., duration, number of crew, and location/destination, affect the approach to and the extent of ECLSS cycle-closure. Those that do are termed unique mission drivers. However, the criticalities of these drivers vary in different mission scenarios. For example, the availability of local resources is irrelevant to the ECLSS onboard a surface-to-orbit transfer vehicle, while it is a major driver on a Mars base. Table 3.4-1 identifies those applicable mission drivers to unique situations, such as planetary bases or service vehicles.

Table 3.4-1 Identification of ECLSS Related Unique Mission Drivers[a]

*Mission Location[b]	Power Source (including capacity)	Available development funding (1-5 yr)
*Time frame to launch	Propellant source	Sunlight (duration and intensity)
*Crew Size[c] (Final)	*Initial flight capability costs	Shielding (from meteoroid)[d]
*Mission duration	Diet scenario	Equipment packaging density
*Resupply Period	Emergency life support duration	Local gravity (e.g., lunar, Mars)[e]
*Technology risk (Including availability & proven operating life)	Life cycle costs	Local resource (e.g., lunar or Mars "soil")
Local atmosphere composition (e.g., Mars)	Communications delay	

a Not including such other drivers as reliability, maintainability, weight, power, volume, etc.

b *=Major unique drivers.

c Including passengers

d Insofar as regeneration of food requires large areas and exposure of these to sunlight

e Including artificial gravity, e.g., on trip to and from Mars

Table 3.4-2 narrows the field of candidate drivers to those related to a Mars mission infrastructure. Of the eight mission drivers in the table, the communications delay between Mars and Earth is the most universal, affecting the ECLSS design for all envisioned Mars vehicles. This is a result of the need for a certain amount of independence should a life support system emergency occur at or near Mars. Crew size, on the other hand, has little effect on the ECLSS design for early Mars missions. Often, an additional crew member will increase only the amount of consumables, such as food and water. This process is one that aids in the identification of key areas where further ECLSS development is necessary.

Table 3.4-2 ECLSS Mission Drivers Related to a Mars Mission Infrastructure

	Transfer Vehicle	Ascent/Descent Vehicle	Sortie	Outpost	Base	Settlement
Gravity			X	X	X	X
Local resources			X	X	X	X
Crew size					X	X
Mission duration	X				X	X
Crew factors	X				X	X
Propellant system	X		X	X	X	X
Power source	X		X	X	X	X
Communications delay	X	X	X	X	X	X

Performance Requirements

The ECLSS performance requirements define the level at which the life support system must perform. A major consideration in defining these requirements is that deviations from terrestrial conditions poses not only a health risk, but could also increase technology development costs. Therefore, the performance requirements of an ECLSS designed for interplanetary travel are tailored to correspond closely to an Earth-like environment (Table 3.4-3).

Table 3.4-3 ECLSS Performance Requirements

Parameter	Units	Operational	90-day degraded[a]	28-day emergency
CO_2 Partial Pressure	mm Hg	3.0 max	7.6 max.	12 max.
Temperature	°C	18-24	15-29	15-32
Dew Point[b]	°C	4-15	2-21	2-21
Ventilation	m/min	4.6-12.2	3.1-31	1.5-61
Potable Water	kg/person-day	3.1-3.7	3.1 min.	3.1 min.
Hygiene Water	kg/person-day	5.5 min.	2.7 min.	1.4 min.
Wash Water	kg/person-day	12.7 min.	6.4 min.	0
O_2 Partial Pressure[c]	psia	2.85-3.35	2.4-3.45	2.3-3.45
Total Pressure	psia	14.7	14.7	14.7
Trace Contaminants	ppm	SMAC	TBD	TBD
Microbial Count	CFU/m^3	500	750	1000

a Degraded levels meet "Fail Operational" reliability criteria.

b In no case shall relative humidities exceed the range 25-75%.

c In no case shall the O_2 partial pressure be below 2.3 psia, or the O_2 concentration exceed 26.9%.

The ECLSS must therefore perform several functions:

1) Provide O_2 and H_2O,
2) Remove CO_2, H_2O, and trace contamination,
3) Provide N_2, a habitable environment, crew support facilities, bacterial control, and food.

The 90-day and 28-day emergency requirements also shown in Table 3.4-3 are the same as those for the proposed Space Station. These time periods are based on the resupply period and the minimum time required to perform a rescue operation with the Shuttle. Of course, emergency operations for a manned Mars mission will be significantly different, yet regardless of what those operations are (i.e., repair or escape), it is assumed that the necessary equipment will be available to return the spacecraft to operational levels in 28 days.

The configuration of a life support system depends on not only the identification and definition of mission drivers and performance requirements, but also the degree of system closure, or regeneration. The massive complexity of a totally closed ecological system drives most long-duration manned missions to support by a combination of stored and regenerative ECLSS technology.

ECLSS expendables per person-day that are available for regeneration are listed, along with their total mass, in Table 3.4-4. Several techniques are available for the regeneration of these expendables, the most developed of which are for water and carbon dioxide.

Regenerative technology can be incorporated into a life support system when there are mass or volume advantages over simple storage. In the case of water, if one crew member consumes a maximum of approximately 28 kg/day (Fig. 3.4-1), the total mass of water required for a crew of four escalates so rapidly that resupply is no longer an alternative.

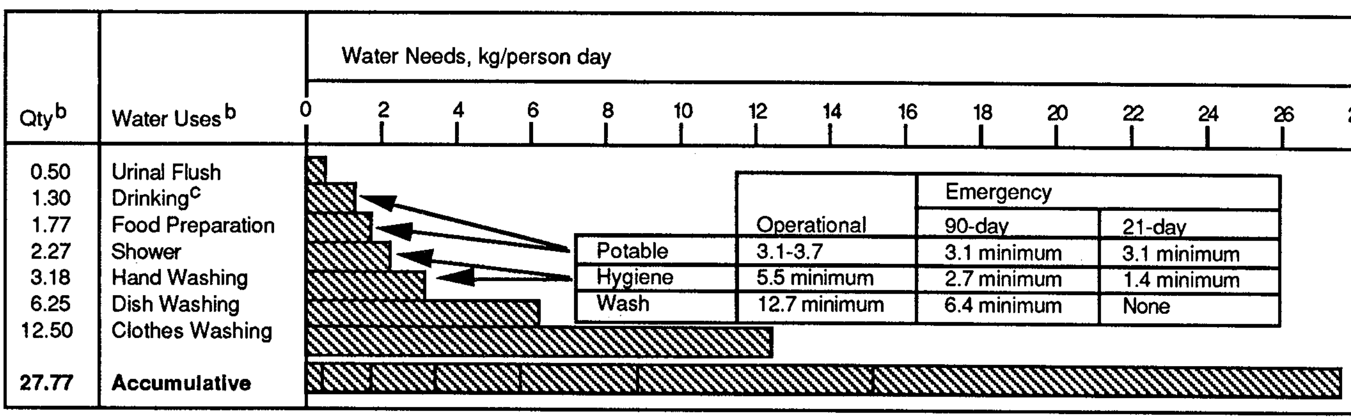
Table 3.4-4 ECLSS Expendables per Person Day

Life Support Needs	Expendables, kg/person-day	Tankage/ Packaging, kg/ person-day(%)	Total, kg/person-day
Water Supply	21.9	4.4 (20)	26.3
Potable	3.7	0.7 (20)	4.4
Hygiene	5.5	1.1 (20)	6.6
Wash	12.7	2.5 (20)	15.2
CO_2 Removal (via LiOH)	1.4	0.7 (50)	2.1
O_2 Supply- metabolic	0.8	0.8 (100)	1.6
Solids Supply	0.7	0.4 (50)	1.1
Food Supply	0.6	0.3 (50)	0.9
Spares, % orig. hardware			
20%/year	0.4	0.2 (50)	0.60
10%/year	0.2	0.1 (50)	0.30
N_2 Supply	0.23	0.23 (100)	0.46
Liquid Supply	0.14	0.03 (20)	0.17
O_2 Supply- Leakage	0.06	0.06 (100)	0.12
Trace Contam. Removal	0.06	0.06 (100)	0.12
Bactericide	0.86	0.17 (20)	1.03
Drug Supply	0.0005	0.0005 (100)	0.001

The case for regenerative CO_2 removal is different than that for water. On previous brief missions, CO_2 has simply been absorbed on a bed of LiOH. One alternative is a regenerative electrochemical cell system (EDC) that removes the CO_2 from the air, concentrates it, and recycles the pure air. Figure 3.4-2 shows that, in a trade study between the mass and volume of the EDC and LiOH bed versus the operating time, for mission durations of 180 days or more, the EDC is advantageous.

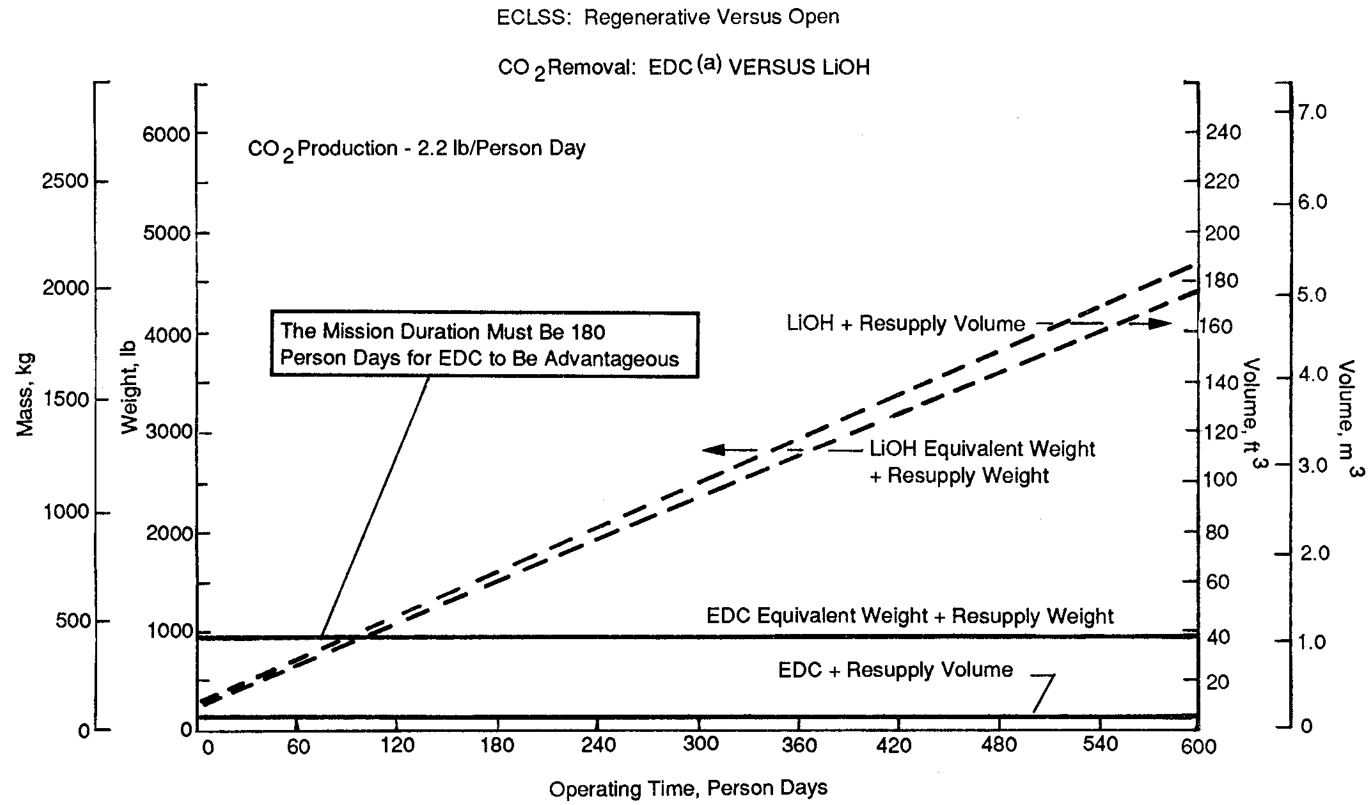
At this time, food regeneration through plant and animal production is largely undeveloped. A Controlled Ecological Life Support System (CELSS) is not yet a cost- or risk-competitive option for 2-4 year missions. Therefore, only the closing of air and water loops is considered beneficial for missions lasting over a few days. Table 3.4-5 illustrates the large mass savings one can achieve by recycling water and oxygen.

The system that evolves from the need to recycle water and air is highly interdependent. For example, water recovered form humidity condensate can be stored for emergency use, recirculated through the atmosphere revitalization system, used for drinking and food preparation, or as an oxygen supply.



(a) Kg/Person Day
(b) EVA Cooling Water Requires Additional 0.68 kg/Day Based On 1,000 EVA Hours Per 365 Days At 2 kg Water/8 Hour EVA (Unless Nonventing Thermal Approach Is Used)
(c) These Water Uses Require Potable Quality Water

Figure 3.4-1 Water Needed: Supplied or Reclaimed?



Legend:
(a) EDC — Electrochemical CO 2Concentrator
(b) Weights & Volumes At Zero Include the Hardware Equivalent Weight & Spares Which Will Be Launched Initially. Values after Zero Days Include Expendables.

Figure 3.4-2 ECLSS: Regenerative Versus Open

Table 3.4-5 LSS Consumables Estimate

Requirement	Amount of Supply Required, tonnes/person-year: No Recycling	Recycling
1. Potable Water	1.4 (1.0-2.4)	0.2 (0.1-0.3) (90% eff.)
Drinking, Food Preparation		
2. Grey Water		
a. Personal Hygiene	1.6-2.7	0.14 (0.1-0.2) (95% eff.)
Hand Wash 28-53%		
Shower 37-62%		
Toilet 7-10%		
b. Utility	0.8-6.7	0.1 (0.05-0.4) (95% eff.)
Dishes 10-85%		
Clothes 15-90%		
3. Food	1.4 (0.5-1.4)	1.4 (1.4-0.35) (0-75% eff.)
Solids (Dehyd.) 20-50%		
Water Content 15-35%		
Packaging 25-62%		
4. Breathing Oxygen	0.68 (0.3-0.7)	0.05 (0.03-0.1) (90% from CO_2)
Totals	5.9 (4.2-14.0)	1.8 (0.5-2.3)

Figure 3.4-3 illustrates the complex interrelationships between the crew and an ECLSS with air and water loops. The seven subsystems that comprise the core of life support must monitor and control as many as eight parameters, along with maintaining redundant back-up systems and emergency stores.

The result of the above work was applied towards the design of an ECLSS for a Mars six-month surface mission vehicle. Key to the system are the air revitalization and water recovery loops. The air revitalization loop consists of three main components: the EDC, Bosch, and Static Feed Electrolyzer (SFE), which concentrate CO_2, reduce it, and generate oxygen, respectively. Figure 3.4-4 shows the flow of reactants from one component to the next. Hydrogen is initially stored and fed to the system, and the pure carbon byproduct of the Bosch system is the only element that cannot be reused.

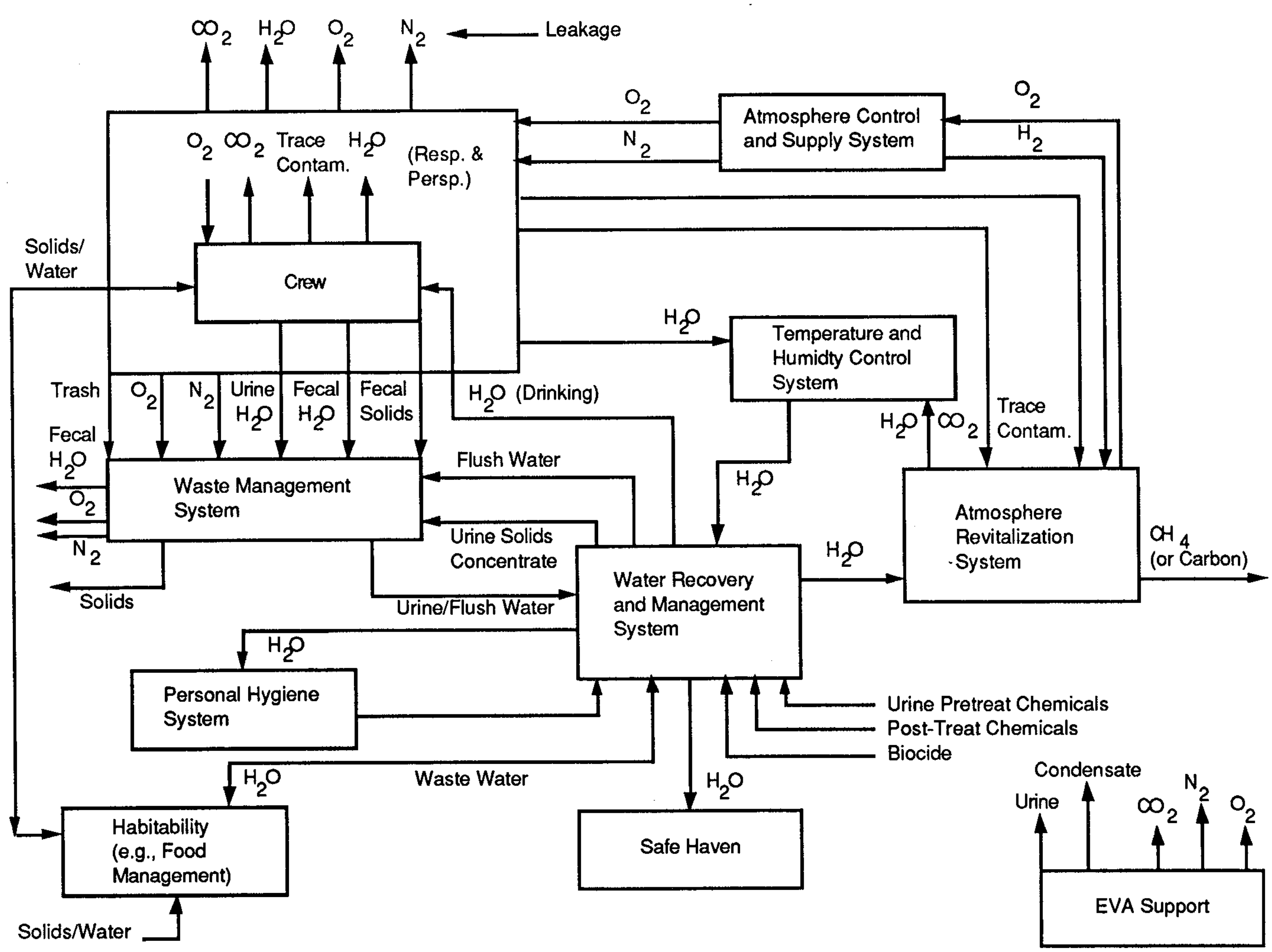


Figure 3.4-3 ECLSS Block Diagram

The water recovery system is initially supplied with 178 kg of water, and through a system of vapor compression distillation, is regenerated from hygiene facilities (wash, urine, and urine flush) and humidity condensate. Variable quality (potable versus hygiene) is permitted in the interest of power conservation. Table 3.4-6 provides a summary of the mass, volume, and power characteristics of this four-person ECLSS.

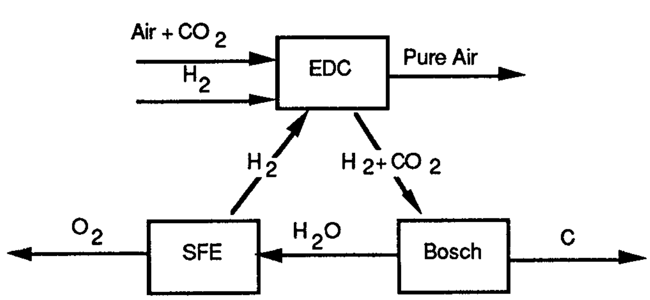


Figure 3.4-4 Atmoshpere Revitalization

When considering advanced manned Mars exploration missions and the life support systems needed to sustain them, several technology milestones must be reached. These goals are listed below and can serve as a guide to a progressive program of ECLSS development.

1) Eliminate residual expendables from post Space Station technologies,
2) Eliminate return of waste liquid and solid waste,
3) Reduce water loop components to one,
4) Reduce level of spares needed,
5) Enable use of local resources,
6) Complete system level integration testing,
7) Carry out basic & applied tech advancements,
8) Develop large crew size ECLSS processes,
9) Effective use of artificial gravity,
10) Advance controlling and monitoring systems,
11) Determine the impact on a partial gravity process.

Table 3.4-6 Mars Six-Month Surface Mission ECLSS Characteristics (four person crew)

ECLSS Function/Technology	Mass, kg	Volume, m^3	Power, W
CO_2 Removal-			
Electrochemical CO_2 Concentrator	39	2.3	-48
CO_2 Reduction-			
Bosch	70	11.0	324
O_2 Generation-			
Static Feed Electrolyzer			
Metabolic	46	1.5	833
CO_2 Conc. & Reduction Requirements	8	0.3	432
Leakage & Airlock Requirements	-		57
N_2 Leakage Make-Up-			
N_2 Generation	21	1.5	89
Trace Contaminant Control-			
Expendable Bed	49	1.9	113
Temp & Humidity Control-			
Heat Exchanger	63	5.6	32
Ventilation	107	16.4	4
Water Recovery-			
Vapor Compression Distillation	51	2.8	34
Water Storage and Distribution	178	83.3	5
Toilet and Urinal Unit	63	13.8	132
Trash Collecting & Processing	10	2.0	0
General Housekeeping	15	1.0	0
Total	720	143.3	1918

3.4.1 Rover Life Support System

The fundamental elements of human life sustenance must also be provided in the spacesuit and conditioned compartment of a shirt-sleeve rover. This includes removal of exhaled CO_2 and provision of breathing oxygen, drinking water, and food. The latter may be a low-residue diet to minimize the problems with waste management in the confined compartment. In principle, closed-cycle chemical processing could provide oxygen and water. However, the equipment to accomplish these processes is, within the current state-of-the-art, very power intensive. The intrinsic power problem attendant with a Mars rover is not consistent with recycling, unless a major power source, such as the compact nuclear reactor cited above, is available. Hydrogen peroxide, a propulsion candidate, has the virtue that its decomposition provides not only energy but also oxygen and water.

The life support system (LSS) must also provide for control of the temperature (30° C versus the average external temperature of -55° C), humidity, and pressure. To avoid the pure oxygen prebreathe necessary to avoid decompression sickness when entering a spacesuit, it is highly desirable that the compartment be a low-pressure atmosphere, 5 to 10 psi, rather than the standard of 14.7 psi adopted for the Shuttle and Space Station (note: all previous U.S. spacecraft—Mercury, Gemini, Apollo, Lunar Excursion Module, and Skylab—employed low-pressure cabin atmospheres).

3.5 ROVER TRANSPORTATION

A major objective of missions to Mars and the moon will be to accomplish reconnaissance and exploration of the surface. To realize the potential of cognition, serendipity, generalization, opportunism, and those other uniquely human attributes that importantly contribute to the exploration of uncharted territory, it will be necessary to provide systems that allow astronauts to operate freely in the planetary environment. These systems include transportation, life support, environmental control, and portable equipment. Strategic planning of objectives and means is of utmost necessity in the design of this infrastructure of equipment to maximize efficient use of the invaluable time on Mars.

3.5.1 Transportation Modes

On Mars, the only beast of burden will be man himself. Walking and hiking will be relatively arduous simply because of the extra weight and flexing resistance of the spacesuit. Before missions to Mars, the only uses of spacesuits have and will have been either in the microgravity of free space or the low gravity of the lunar surface. On Mars, gravity is less than half that of Earth. Nonetheless, it is almost three times higher than on the moon and the weight of the equivalent spacesuit will be that much heavier. Although considerable engineering efforts could and should be placed on improvements to decrease mass, the need for boots, helmet, multilayered suit materials, and the life-support system backpack will result in an irreducible minimum of burdensome weight during EVA on Mars.

The distance an astronaut will be able to walk on Mars in a 4-hour period of time will be limited, perhaps to about 10 km. The hopping-skipping motion so successful on the moon will not be possible on Mars. For an excursion time of 8 hours total, the maximum likely distance for safe travel from the home lander will be 10 km. Mechanical means of locomotion, such as a rover, will accordingly be most welcome. Indeed, it may be judged more or less mandatory because exploration of an area bounded by a mere 10-km radius will often preclude the possibility of covering more than one geologic unit or setting.

Unmanned rovers, although they will be much slower, could cover the same area, albeit requiring a much longer period of time, with the expectation of making many useful explorations and discoveries. To support the extraordinary human capacity for explorations into the unknown, the machines for transportation must provide this much needed extension of range and speed of movement. Artificial intelligence will be replaced with native intelligence, allowing faster driving and reconnoitering. The ability of the human intellect for observation, which simultaneously synthesizes a model consistent with the contextual setting and searches for the deviations from the model, will greatly magnify the chances for major breakthroughs in sample selection and discovery scenarios.

The range of a man-transporting vehicle (i.e., a rover), should be at least some tens of km, preferably well in excess of 100 km. This will allow access to only roughly 0.02% of the martian surface, but if the centerpoint (lander base) is strategically placed, many different terrain types and geologic units of interest should be within this area.

Assuming a top speed of about 30 km/hr for safe driving on the martian surface, and an average speed of about 10 km/hr, a total driving time of 3 hours per sol (1 martian day = 1 sol = 24.6 hr) will cover a 100-km distance in about 3 sols. Assuming the same amount of time for returning to the lander base (most likely, along a different route), the duration of the traverse will be of the order of 7 sols.

Reliability and safety issues must be major concerns in any such endeavor. It is incumbent on mission designers to assure provisions for safe return to the lander in the event of malfunction of the rover. This will include repair/corrective procedures and equipment, as well as backup modes for operating and for powering the vehicle. In addition, in the event of an unrecoverable failure, some alternative must be available, which could include one or more of the following: (1) the rover must be within walk-back range (as was specified for the Apollo lunar surface explorations); (2) another means of transportation must be provided, such as a small all-terrain vehicle (scooter or tricycle); (3) a second rover staged at the base or an intermediate location ready to come to rescue the stranded members; (4) caches of life-support supplies (food, water, air) deposited at intermediate locations to extend the walk-back range in case of dire emergency; or (5) some other stratagem.

For ambitious coverage of a selected region, a permanent transportation system could be installed, segment by segment, just as railroads and roadways are built on Earth. One example would be a suspended cable system, with intermediate low-mass support posts. Such a highwire could provide the basis for transport of small payloads of materials and persons, using a mechanical crawler to move along the cable. The advantage of such an installation would be to avoid a rough and perhaps hazardous surface terrain.

Assuming, however, the rover approach to the transport scenario, it must be considered that several alternative concepts are available, as summarized in Table 3.5.1-1. The simplest rover is perhaps that typified by the lunar roving vehicle. An open-frame, bare bones rover is shown in Figure 3.5.1-1. An important drawback must be recognized. This is that the maximum time for spacesuit operation is typically 8 hours, accounting for the limiting supplies that can be carried. Additional air and water could be provided from built-in reserves carried on the rover, as shown in Figure 3.5.1-2. However, because of the difficulty of eating during this period of time, the astronaut may need to live mostly from liquid nutrients. Furthermore, removal of human excrement is difficult, if not totally impractical, in a spacesuit. For these reasons, it is unlikely that astronauts will be required to remain in the suits for more than 4-8 hours at a time.

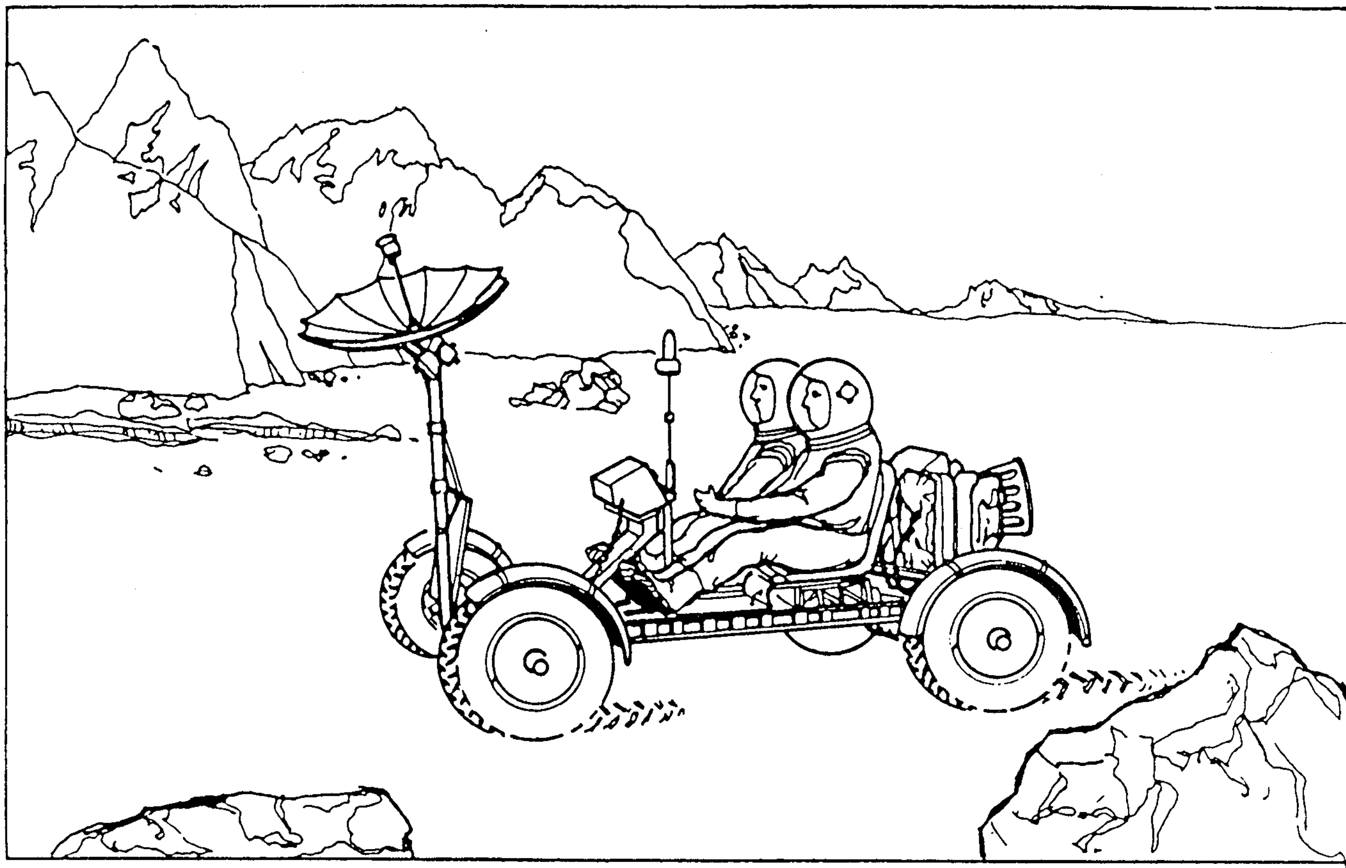


Figure 3.5.1-1 Minimum Rover

The next level of rover would be a "shirt-sleeve" approach, wherein the human explorer would be encapsulated in a pressurized capsule within the vehicle. In this case, shown in Figure 3.5.1-3, remote manipulator arms would be employed for sample inspection and retrieval. This mode is not unlike manipulators that have been used in nuclear-hazardous situations for over 40 years. Or, it could be compared to the operations of deep submersibles, such as the Alvin, which have an enviable record of successful exploration and sample retrieval at geologic and biologic sites deep in the ocean.

If it is decided that manipulators do not adequately support the exploration capacity of the geologist/ astronauts, an approach such as is portrayed in Figure 3.5.1-4 could be followed. This concept is a hybrid approach, providing both a shirt-sleeve environment and a "rumble seat" for a second astronaut who remains fully suited during periods of intensive investigation of selected areas. Working together, the "driver" can transport the "suitman" to the next desirable area. Once in such an area, the suitman can demount and conduct typical field geological exploration. Meanwhile, the driver can continue to rove, to reconnoiter the area, and/or conduct independent sampling sorties using manipulator arms, as shown in the lower portion of the figure. The rover can also provide a refresh capability for the suitman to prolong his outside time. After an area has been adequately covered, the suitman mounts the rover for transport to the next area of interest. During the traverse, he or she can also be a spotter for the driver and suggest a halt for additional sampling. After a hard sols work, the suitman can ingress (enter) the rover for eating, sleeping, sample analysis, and planning of the next exploration segment. Another advantage of the hybrid approach is that the buddy system is automatically invoked, providing two persons who can interact and help one another in especially challenging situations, including dangerous circumstances and life-threatening emergencies.

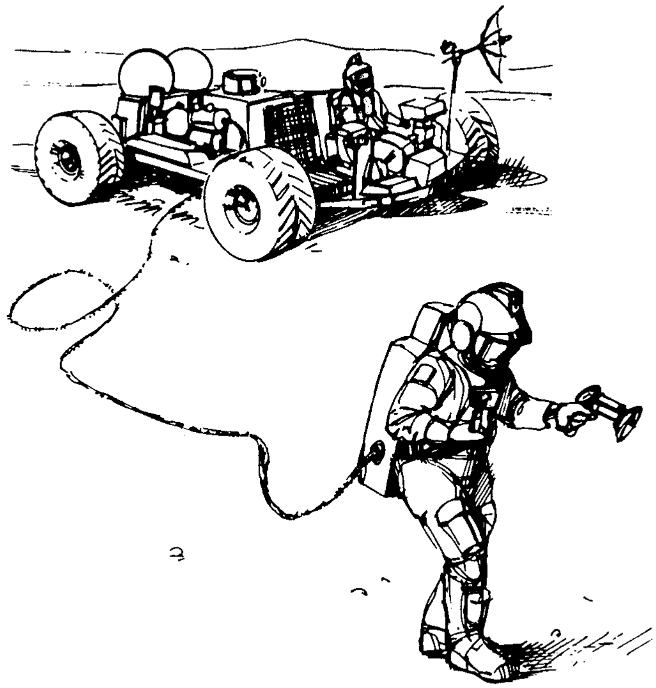


Figure 3.5.1-2 Rover with Augmented Life-Support Services

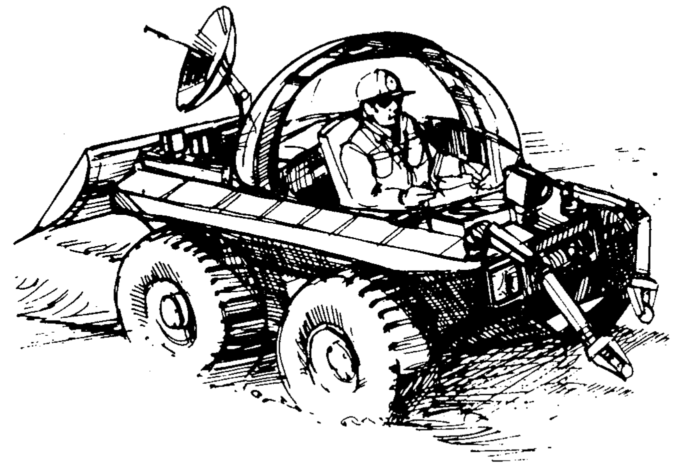


Figure 3.5.1-3 Shirt-Sleeve Rover, One-Person

Table 3.5.1-1 Rover Concepts and Issues

* **Minimum Rover** (ala Apollo Lunar Rover) Minimum weight Limited to 8-hr sortie (suit time)
* **Moderate Rover** With plug-in life support, but no shirt-sleeve Hab Could have umbilical Could have LSS Cart
* **Maximum Rover** Shirt-sleeve, one or two-person (2 preferred) Remote manipulators (telepresence) Rumble seat for suited astronaut
* **Augmented** Any of the above, with Wanigon
Safety Issue: If rover becomes immobile, walk-back range is <10 km.
Science Issue: Range of 10 km is much less than automated Mars robotic rover exploration is expected to accomplish, especially with 1-3 year projected lifetime for unmanned rover.
Solution: Dual long-range rovers to provide backup safe return.

Figure 3.5.1-5 shows a wheeled-rover concept, as contrasted to the tracked-vehicle of Figure 3.5.1-4. Again, an outboard suitman is provided for, but clearly shown is the two-seat capacity of the interior conditioned environment. Egress/ingress could be, as shown in this example, out the front-bottom. This rover has a high clearance capability to allow driving in rock- and boulder-strewn terrain, a capability that may be necessary in a rough Martian landscape. This rover has four-wheel independent drive, with each wheel having separate steering command, clutch release, and powered vertical extensions and retractions. Over smooth terrain, especially where major slopes may be encountered, the rover can be lowered (Fig. 3.5.1-6) so the center of gravity relative to the wheelbase protects against the possibility of tip-over. In such a configuration, higher speeds would remain safe and would permit rapid access to the next geological area of interest. Ideal rover concepts are summarized in Table 3.5.1-2.

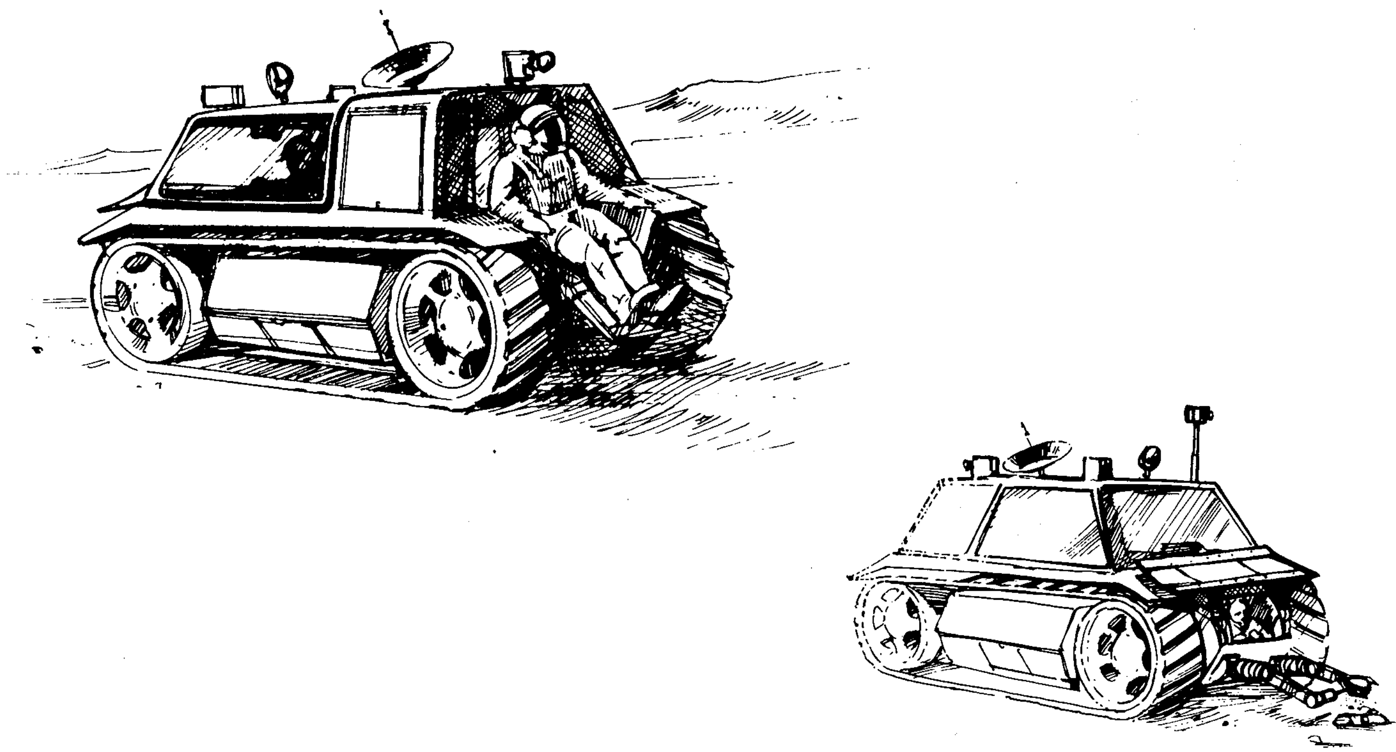


Figure 3.5.1-4 Hybrid Rover Concept

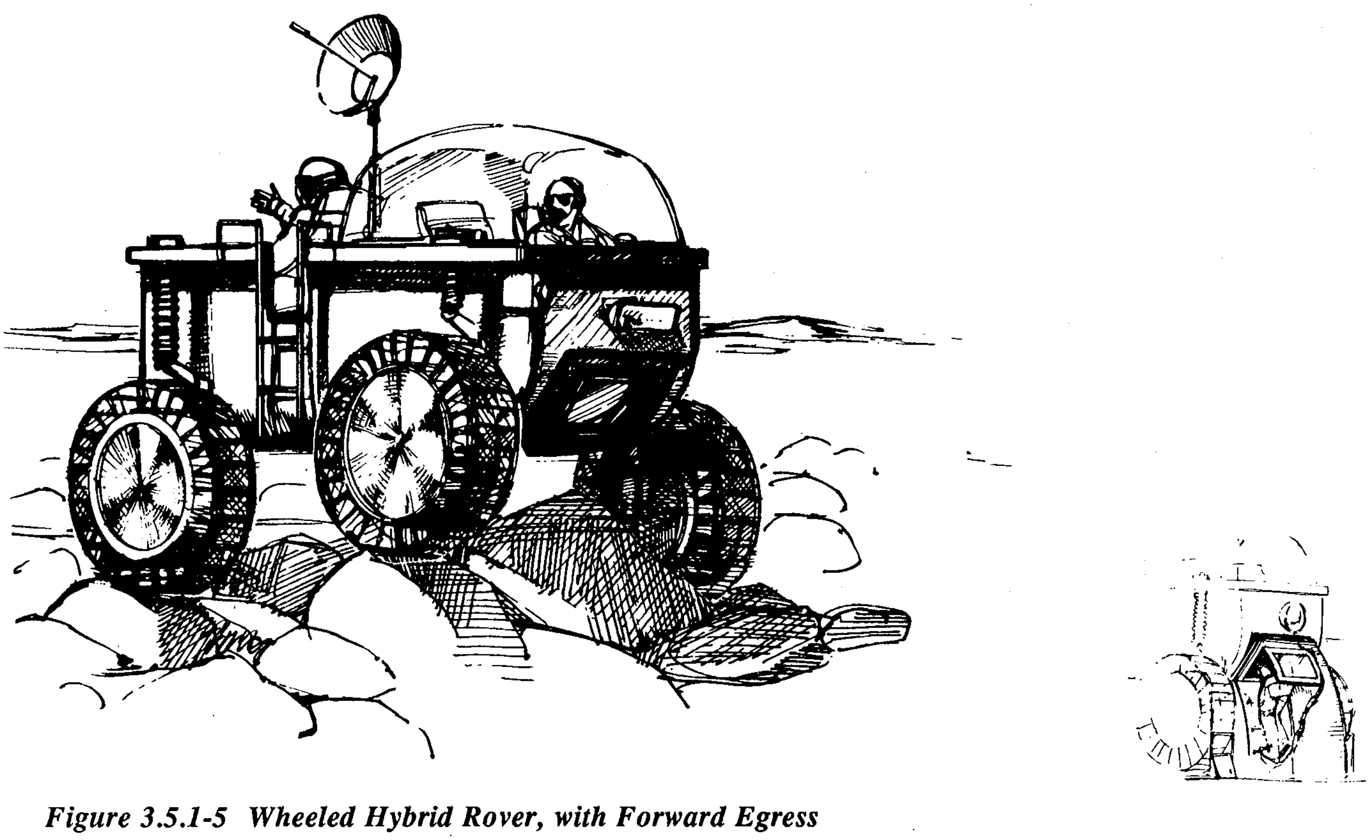


Figure 3.5.1-5 Wheeled Hybrid Rover, with Forward Egress

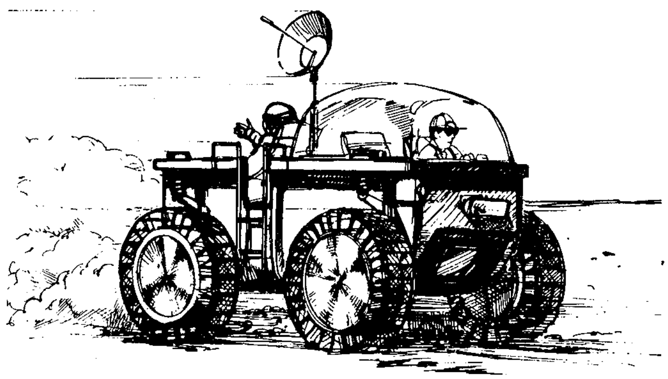


Figure 3.5.1-6 Wheeled Hybrid Rover, Lowered for Fast Transport

Table 3.5.1-2 Optimized Rover Concept Summary

To do "good" science and exploration
Shirt-sleeve rover, with two-man capsule
LSS for 20 person-days (7 sols traverse, with emergency reserves)
Mobility and fuel for 100 km travel radius
Suited-astronaut rumble seat
Plug-in backpack revitalization for suited-astronaut
Instruments:
Stereo video and film camera; microscope
Sampling equipment and storage
rock hammer, saw, coring drill, trencher, soil bags
Diagnostic analytic instrumentation
IR, x-ray, alpha backscatter, DSC, EGA
Deployable science equipment
seismic (passive, active), meteorology, heat flow, gamma spectrometer, UV radiometer, VLBI, balloons
To provide for safety
Two units, with rescue capability for return sprint
Follow-the-leader protocol
Continuous communication capability
Automated warning system, with speed-cap (governor) autopilot

3.5.2 Transport Power

A major concern for a long-range roving vehicle is the motive force providing drive-power. The lunar rover used batteries to propel it over the relatively smooth and low-gravity surface. A martian rover will require much more power to cover the same distance; furthermore, the ranges are expected to be many times farther. Electric power is ideal in the sense that recharge of the batteries by solar cells could provide an unlimited capability. The problem is that Mars' greater distance from the sun, the diurnal cycle, the difficulty in achieving optimum array orientations all the time, attenuation by the dusty atmosphere, and the danger of dust clouds kicked up by the rover coating the cell surfaces all mitigate against any dependence on solar energy as a sole power source.

Nuclear power sources can be considered. These include radioisotope thermoelectric generators (RTG) and small nuclear reactors. The former can provide on the order of 0.5 kW_e (and 20 times this amount of thermal power) without an intolerable radiation dosage over the one-week exposure period if kept at least one meter from the astronauts. A dynamic isotope power system (DIPS) can provide several times the electric power for the same radiation dose. A nuclear reactor would require massive shielding, significantly affecting the size and weight of the rover. It could, however, provide 5 to 10 kW_e of power.

Another alternative is chemical propulsion. Because the martian atmosphere has an extremely low partial pressure of oxygen, the generation of power through combustion of hydrocarbon fuel with atmospheric gas, as is almost universally used in transportation systems on Earth, is not possible. Some consideration has been given to the use of ambient martian CO_2 atmospheric gas to react with a selected fuel. For example, either pure carbon, calcium cyanamide, or carbon disulfide might be "burned" in CO_2 to produce energy. Or, an oxidizer could be carried along with an appropriate fuel. The oxidizer could be liquid oxygen, but storage problems resulting from boiloff would be encountered. Nitrogen tetroxide is a proven oxidant for space use, but is hazardous to handle. Another candidate oxidizer is hydrogen peroxide, which has also been proposed as a monopropellant for rover propulsion. Fuels could include various organic compounds or hydrazine. The latter could also be used as a monopropellant.

3.5.3 Rover Communications

Suitman-to-driver communications are a must. In case of failure of the radio link, hand signals provide a workable backup, as amply demonstrated by underwater diving teams. Rover-to-base communications will be complicated by the fact that the horizon on Mars is much closer than on Earth. Terrain obscuration will also be a factor at most geologic sites of interest. For these reasons, the mainline link will be through the manned orbiter and possibly also a dedicated communications satellite. The orbiter should also be capable of ultra-high-resolution camera coverage of the terrain surrounding the rover to assist in exploration sorties.

3.5.4 Rover Requirements

Exploration and science supporting equipment for rover transport systems are described in Section 3.6. Design of a manned rover for the exploration of the martian surface invokes many disciplines and tradeoff analyses. To conduct a wide-ranging exploration, a sortie time of at least 7 sols and adequate fuel/power source is needed for at least 100-km range radius. Life support systems for this type of transportation system were described in Section 3.4.3. A hybrid of spacesuit and shirt-sleeve environments are highly desirable and could be implemented within the engineering state-of-the-art with a proper, but substantial development effort.

3.6 EXPLORATION AND SCIENCE

The most visible purpose for human exploration of the planets, other than the adventure of exploration and the political and social benefits, will be to make scientific discoveries and enhance our knowledge of the universe. Of great potential direct benefit is learning new aspects about Mars and possibly other planets that will improve our understanding of key processes on Earth, such as geological episodic events, climatological trends, and protection of the environment. Surrounding these missions will be a multitude of opportunities for a wide variety of endeavors, which has the potential to encompass nearly every major scientific discipline.

3.6.1 Interplanetary Science

An Interplanetary Science Experiment (ISE) set of packages can include all of the disciplines listed in Table 3.6.1-1.

A solar monitoring (SolMon) experiment cluster, needed for predicting solar particle events, will provide major increases in knowledge of the dynamics of the sun, with expected spin-offs in understanding solar-terrestrial relationships with regard to ionospheric activity and its effects on communications, weather, and other important processes on Earth.

Table 3.6.1-1 Science: Objectives During Interplanetary Transfers

Human Physiology
Bone demineralization
Cardiovascular deconditioning
Muscle atrophy
Vestibular dysfunction
Immune system, drug efficacy
Human Psychology/Sociology
Isolated, confined, and hazardous environment (ICHE)
Stress assessment, consequences
Microsocietal interactions
Astronomyhazardous
Astrophysics: stellar, galactic, extragalactic sources
(Vis, IR, UV, X-ray, gamma-ray observations, VLBI)
Planetary science: Earth, moon, Mars, Venus, Jupiter
(Vis, IR, UV)
Solar research: sunspots, flares, corona
(Vis, IR, UV, radio)
Space Environment Effects/Manufacturing
Microgravity, variable-g
Ultra-high Vacuum
HZE Particle Irradiation
Space Agriculture
CELSS Demonstrations

3.6.2 Remote Science at the Planets

A Mars Orbiting Science Experiment (MOSE) package is included in all mass allocations for these missions. In addition, it has been shown that dropping off probes during Venus swingby trajectories is of very minor IMLEO impact and provides a unique opportunity for major new investigations of this other intriguing planet.

3.6.3 Landed Exploration and Science

An enormous range of scientific objectives could be addressed at Mars, as listed in Table 3.6.3-1. A subset of these objectives could be addressed as well at the moon, but obviously mainly the geology objective.

Table 3.6.3-1 Science: Objectives at Mars

Geology
Volcanism, many styles; active volcanism?
Seismic activity?
Eolian activity
Water: channels, permafrost, water-laid sediments?
Atmosphere
Weather systematics
Photochemistry
Climatology; analogous ice ages?
Life on Mars?
Endolithic organisms
Sulfur-based metabolism
Beneath the superoxidized zone
Oases (warm, wet spots from volcanic, impact processes)
Fossils (microfossils, unique structures and signs)
Survival of terrestrial organisms on Mars
Moons (Phobos, Deimos)
Composition, resource potential
Age and Origin
Effects on Martian surface?

Rovers can obviously benefit Mars and lunar exploration enormously. To achieve the long ranges needed for systematic and thorough exploration, a pressurized rover will need to be designed. Issues affecting this system were discussed in Section 3.5.

A large variety of tools and analysis equipment will be needed if the maximum potential for scientific exploration is to be realized. This is because science investigations are iterative, adaptive, and exploitative of the results of each step taken, while knowledge builds on previous knowledge. Simple grab sampling by an unqualified person would be highly counterproductive and wasteful of the resources that must be expended. The Mars Landed Science Equipment (MLSE) complement should include sample acquisition and processing tools, such as rock hammer, soil scoop, drive tubes, rake, scribe, saltating grain sampler, dust collector, regolith core drill, rock coring minidrill, cleaver, crusher, grinder, siever, and thin-sectioning apparatus. For use outside, prospecting aids such as portable element analyzers (x-ray fluorescence and Rutherford scattering spectrometers), mineral detectors (infrared reflection spectrometer), water/ice detectors (neutron thermalization sensor, differential scanning calorimeter, hygrometer), and organics detectors (evolved gas analyzer) could be extremely valuable in locating specimens of high scientific importance.

In addition, a high-quality geological and chemical analysis laboratory should be provided inside the lander. Equipment that should be strongly considered include petrographic microscope, electron microscope, wet chemistry set, x-ray fluorescence and diffraction units, gas chromatograph, mass spectrometer, thermal analyzers/pyrolyzers, and various physical properties analyzers. A considerable amount of development will have to be placed in miniaturizing many of these laboratory techniques, although the unmanned planetary spacecraft projects have already accomplished much of this effort or have proposed plans for such developments.

3.7 TETHERS

Significant potential exists for the use of tethers as length adjustable, non-rigid linear tensile members in possible lunar and Mars expeditions.

Tethers are an integral part of some designs for generating artificial gravity while traveling to Mars, as discussed in Section 3.3. Concerns about the physiological and psychological effects of prolonged weightlessness on the crew have prompted the development of workable systems for creating artificial gravity. One approach would be to divide the spacecraft in two (separating the habitation modules from the main spacecraft) and linking the parts with a pair of 222 m tethers, spinning them about each other to create artificial gravity (Fig. 3.7-1). A second possibility is to separate the two habitation modules from the main spacecraft, reel them out in opposite directions, and spinning them around a main hub (Fig. 3.7-2).

Another ambitious tether concept for the Mars Mission is a plan to lower a sortie vehicle from Phobos toward Mars on a pendant cable and possibly using the same tether to rendezvous with and retrieve the vehicle. In addition to these tether ideas, there are also possibilities of using tethered masses for momentum exchange between Phobos and the spacecraft. Section 3.7.4 provides more details on Phobos tether applications.

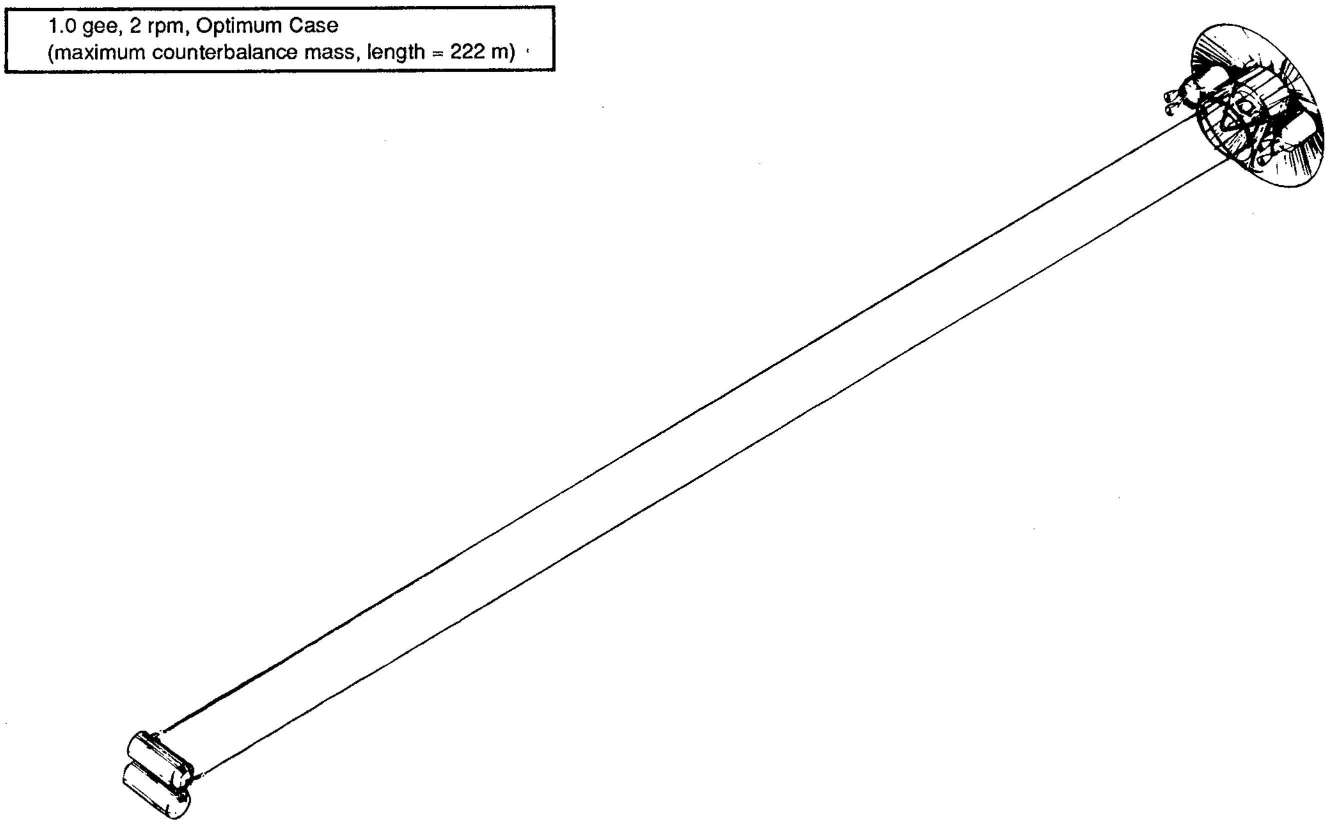


Figure 3.7-1 Tether Artificial Gravity for Phobos Gateway

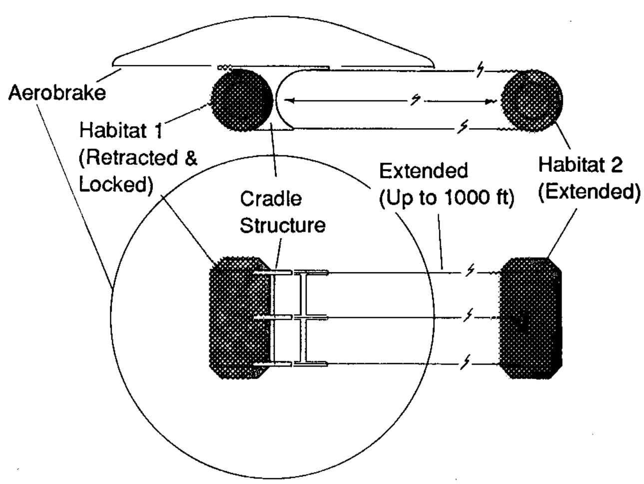


Figure 3.7-2 Tether Artificial Gravity, Rotation Around Central Hub

3.7.1 Tether History

High strength aramid fiber has been used extensively in a variety of tether applications for over 15 years. Balloon tethers produced in the seventies ranged from 10,000-ft lines of 1/8-in. diameter flown over Kwajalein in the Pacific to a pair of 85,000 ft tethers for the Air Force rated at 8,500 lbs breaking strength. The longest continuous tether of 100 km length was manufactured by Martin Marietta in the summer of 1983. The Air Force tethers represent the most Kevlar poundage ever assembled into a continuous tether for a non-marine application, weighing in at about 2,000 lbs per cable.

3.7.2 Tether Technology

To meet the challenges presented by the proposed use of tethers in support of upcoming Mars and Lunar missions, there are several thresholds of tether technology that must be crossed. Present manufacturing methods can support continuous production of any length of tether. However, there are definite processing speed limitations that may make long, splice-free, continuous tethers infeasible. Serial processing of kilometers of tether not only creates a long lead time for delivery, but statistically increases the risk. New splicing technology allows tether sections to be built in parallel, enhancing testability of midspan samples, while at the same time reducing delivery time.

The task of designing a reel to accommodate several thousand kilograms of tether mass and to successfully deploy and retrieve it is another area in which further research is required. For massive tethers (several metric tons) there is a concern about the effects of storage on a reel under a one Earth-g of tension force. Innovative approaches to tether retrieval and control need to be investigated, as do sophisticated winding techniques.

3.7.3 Tether Concerns for Artificial Gravity Systems

In the design of rotating tethered bodies for artificial gravity, it is necessary to study the nature of high strength synthetic fiber ropes and to design a system that protects their dynamic integrity. The first approach is to look at existing technology (including sewing machine bobbins, fishing tackle, and cable winches) and scale up proportionately to the range of line sizes and lengths needed for new tether applications. This analogy is not necessarily true when dealing with quantities of cable mass several orders of magnitude greater than what is normally seen on conventional winding apparatus. There are differences in power needs, materials, and environments under which they must perform.

3.7.3.1 Power Requirements—Hoist cable retrieval as used for common applications (e.g., helicopter winch, crane operations) is a power intensive system. In a space application, the design must be much more energy efficient. Spooling mechanisms that drive large reels from the rim as compared to the more conventional shaft-driven systems should be investigated to reduce the reel-turning force required. Such a system would result in a reel geometry design with a larger barrel diameter, which will be required as cable diameter increases.

3.7.3.2 Tether Material—Given the requirement to minimize tether mass on a Mars mission, the tethers must be fabricated from high strength-to-weight ratio materials (aramid fibers). Table 3.7.3.2-1 explains tether masses for different materials (assuming a 50 t mass on the end of the tether).

One major concern is the minimization of the tension/compression combination seen by the cable after it has been deployed to the required length. This is an important area of concern in an artificial gravity system, where the tether is under significant tension for several months at a time. Smaller systems have successfully operated using the reel itself (with an actuating break) to hold the load. This approach is not feasible with very high tensile loads (50 metric tons +), since there would be an unacceptable stress on the tether and the reel structure.

Another concern is the problem of how to retrieve a tether while it is under a high tensile load. This problem has been successfully addressed using a dual capstan winch that accommodates high outboard tension, while allowing for lower, controlled inboard tether tension (which is preferred for successful spooling of long tether lengths).

However, even with a generous capstan diameter, sustained high tension loading over a capstan (as would be required in an artificial gravity system) creates stress concentrations that may put the tether at risk. A possible alternative to the dual capstan is a long compression clamp that would distribute the load uniformly along the tether. Given the necessity to decouple the winding tension on the reel from the outboard tether tension, a belt driven capstan is another concept that should be considered for retrieving a high-tension tether.

3.7.3.3 Exposure—There may be a problem with long-term tether exposure to the space environment, specifically micrometeroid and space debris impacts. Modifications can be made to the tether jacket design to increase the filament density, and thereby improve resistance to particle impact. Laboratory simulations of micrometeorite bombardment can be used to compare different configurations (thicker braids, multiple layers, fine versus coarse yarn sizes) to optimize impact resistance to extend useful tether life.

The baseline used to estimate tether mass requirements as a function of length and ultimate tether strength is one kilogram per kilometer per kilonewton (1 kg/km/kN). This approximation assumes an allowance of 20% of the total mass for protective jacketing over the remaining 80% strength member. This protective jacketing should prevent tether failure from occurring due to the space environment.

Table 3.7.3.2-1 Tether Masses for a Rotating System

Material	Density, g/cm^3	Tensile Strength, MPa	Modulus, GPa	Mass, 1 rpm (kg)	Mass, 2 rpm (kg)	Mass, 3 rpm (kg)	Mass, 4 rpm (kg)
Kevlar 29	1.44	2758	62.06	229.15	57.19	25.41	14.30
Kevlar 49	1.44	2758	131.0	229.15	57.19	25.41	14.30
Graphite Fibers							
Thornel 50	1.67	2400	413.7	305.63	76.23	33.87	19.05
Thornel 75	1.82	2496	489.5	320.31	79.88	35.49	19.97
Thornel 300	1.74	2620	234.4	291.66	72.75	32.32	18.19
Thornel P	2.00	1207	393.0	730.88	181.70	80.68	45.39
Hercules A	1.91	2606	186.2	321.97	80.30	35.67	20.07
T-300/A201	2.27	1475	144.8	678.47	168.50	74.93	42.15
Graphite/Epoxy	1.53	1503	137.9	447.75	111.60	49.55	27.88
Steel Wire	7.82	4137	206.9	834.62	207.40	92.05	51.78
Titanium	4.71	1931	110.0	1079.60	267.70	118.8	66.82
Boron Fibers	2.60	3172	413.7	320.21	89.81	39.90	22.45
Boron/Aluminum	2.65	1489	213.7	785.43	195.20	86.66	48.75
Carbon Fibers	1.41	1724	—	359.42	89.61	39.81	22.40

3.7.4 Phobos Tether Application

A permanent tether facility at Phobos can reduce rocket propellant requirements by imparting a ΔV to the spacecraft through momentum exchange between Phobos and the spacecraft. In this study, the use of tethers for augmenting trans-Earth injection and Phobos-to-surface transfer was investigated (Fig. 3.7.4-1). Three methods of momentum exchange were addressed: (1) separation along the gravity gradient followed by a release, (2) libration pumping (swinging) and release, and (3) powered winching to draw the spacecraft toward Phobos followed by a release. Coriolis effects and/or a small libration prevent this last approach from resulting in an impact with Phobos.

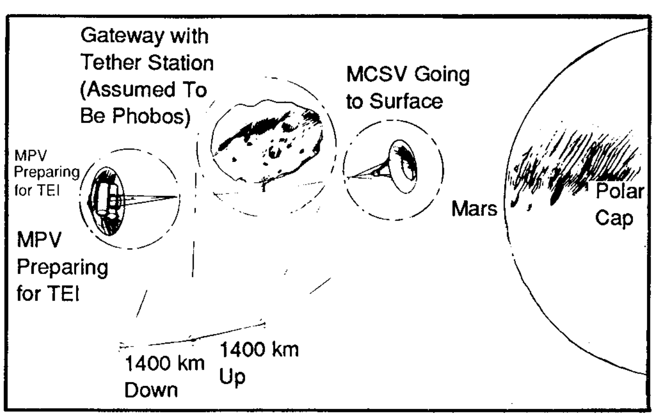
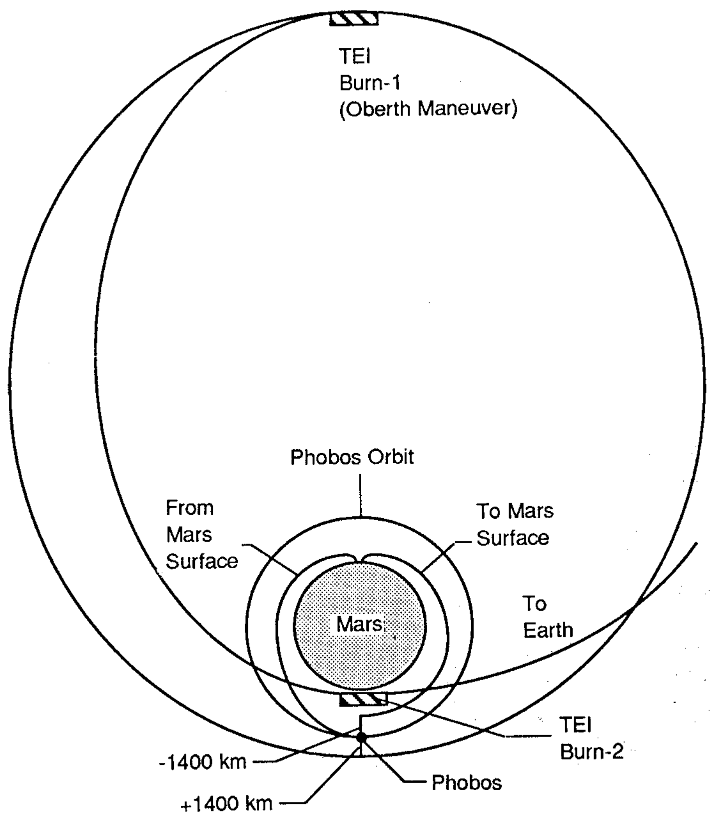


Figure 3.7.4-1 Phobos Tether Applications

Winching was eliminated from near-term consideration because of the large power requirement for reasonable tether lengths. For example, to impart a 100 m/s ΔV onto a 10 t spacecraft requires a one megawatt power source and a 2000 m tether. Greater ΔVs can be achieved with either longer tethers or more power. However, using even one megawatt of power implies a nuclear electric power system, which significantly complicates Phobos operations and increases the time to mass break-even between the tether system and the saved rocket propellant. Lengthening the tether is even less attractive than increasing power, since the required tether length goes up as the cube of the desired ΔV.

Librating tethers were eliminated because of operational complexity caused by the low frequency of libration and the low tip speeds achieved by the tether. For example, a 10 km tether, librating through a maximum 120 degree arc, has a maximum tangential speed of only 3 m/s.

This leaves only the separation and release approaches for examination. The spacecraft can be lowered toward Mars to decrease orbital energy or raised away from Mars to increase energy. For departure to or return from the surface, a 1400 km tether eliminates the apogee burn near Phobos and reduces the perigee burn near Mars (Fig. 3.7.4-2). This makes a reusable Mars Crew Sortie Vehicle (MCSV) possible. Without tethers, or a major Mars Orbital Operations System (MOOS), the ΔV is too large for a single stage MCSV.



- 1400 km Is Length Required To Put The Postrelease Periapsis in Mars' Atmosphere
- ΔV Savings is 538 m/s per MCSV Round Trip & 766 m/s per TEI Injection
- Electric Power Needed to Winch Tether & Expended MCSV to Phobos is
  1 MWe for 12.6 Hours or
  100 kWe for 5.25 Days
- Power Needed to Winch Empty Tether to Phobos is:
  100 kWe for 2.1 Days or
  10 kWe for 21 Days
- Tether Used Only to Reel-Out Piloted & Cargo Vehicle
- Tether Winched in Empty

Figure 3.7.4-2 Tether Assist at Phobos

For the down leg (toward Mars), the MCSV completes a rendezvous with the tether facility (Fig. 3.7.4-3) and initiates the downward motion with its Reaction Control System (RCS). After a short coast away from Phobos the gravity gradient begins to pull the MCSV both forward (in the direction of Phobos' velocity) and downward. After the full 1400 km is deployed, and the MCSV is stabilized, the tether is released and the MCSV enters an atmospheric intercepting orbit.

For the return trip the ascent vehicle boosts into an orbit where apoapsis is at Phobos' altitude minus the tether length. When the MCSV reaches this apoapsis, it rendezvous with the tether. Although tangential speeds are matched before hook-up, radial speeds are only matched instantaneously because the tether end is not in a free orbit. This imposes the requirement that a "smart" tether end manipulator, powered by small rockets, rendezvous with the MSCV.

If the tether is used in place of a MOOS to drop the MCSV into Mars' atmosphere and pick it up again on the return trip, the ΔV savings is derived from a 557.6 m/s perigee raise on the down leg and a 100 m/s apogee raise plus a 522.4 perigee raise on the up leg (towards Phobos). The 100 m/s is saved because the tether intercept orbit is of lower energy than Phobos rendezvous orbit (because of the 1400 km tether length savings). Given these requirements for the MOOS, and assuming a MCSV mass of 66.5 t, the propellant consumed for each round trip is 12.5 t, giving a mass payback for the tether system after only four flights of the MCSV. This analysis makes the following additional assumptions:

1) 0.9 mass fraction MOOS
2) MOOS I_{sp} = 449 sec
3) tether system mass = 53 t (26.5 t tether + 26.5 t capstan and solar arrays)
4) tether mass/length/load = 0.939 kg/km/kN
5) tether length = 1400 km
6) free hanging tension = 3.206 kN
7) loaded tension = 20.162 kN
8) factor of safety = 2.5
9) tether can be retrieved in less than five days with a 10 kW motor.

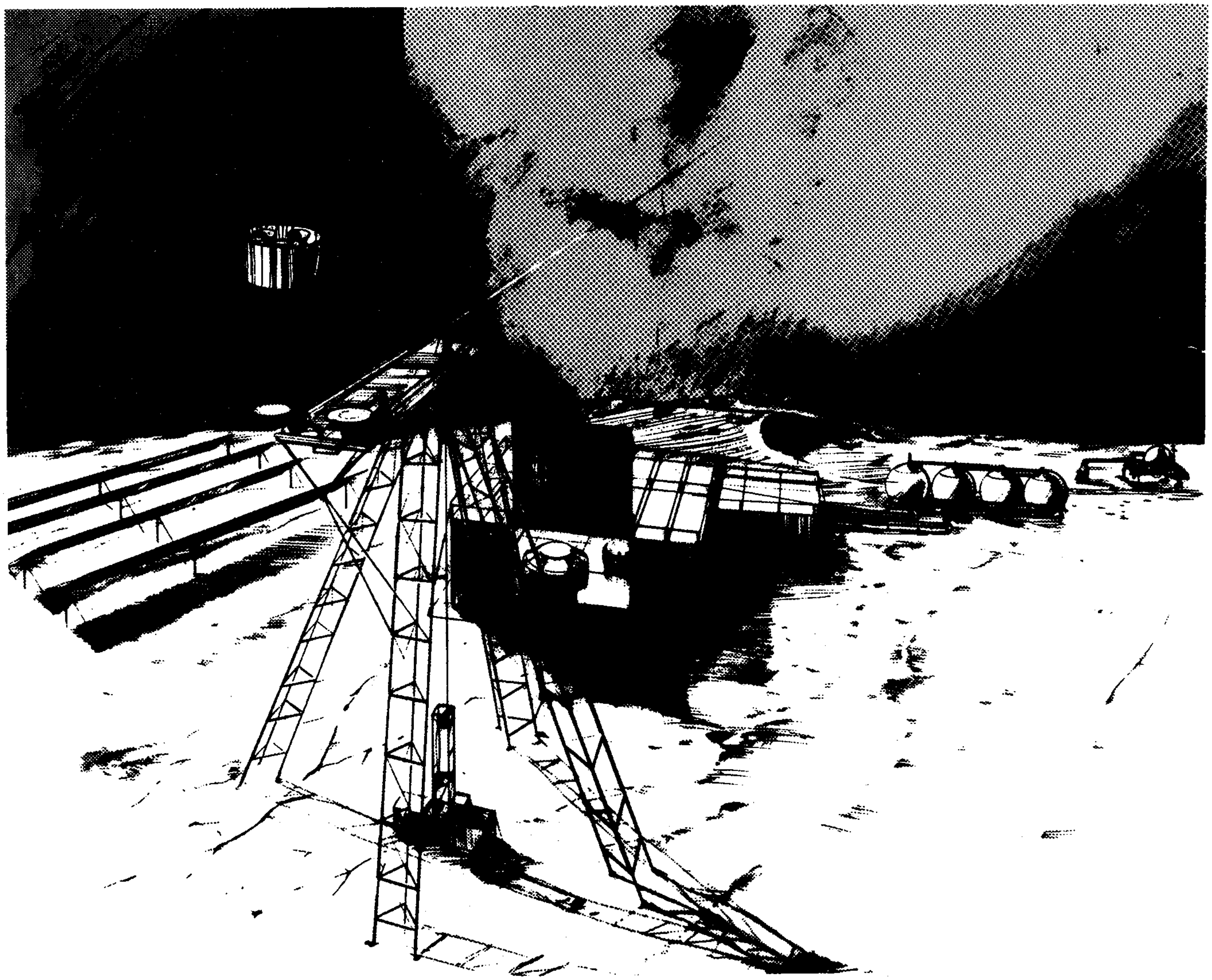


Figure 3.7.4-3 Phobos Tether Infrastructure

Trans-Earth injection (TEI) savings result by allowing the gravity gradient to raise the spacecraft away from Mars, giving it additional energy (Fig. 3.7-4). After tether release the spacecraft is still in Mars orbit and can inject toward Earth either by firing directly into a trans-Earth trajectory or by performing a retro burn, drop to within 250 km of Mars, and then burn the remaining propellant for TEI. This latter case trades the loss in orbital energy from the apogee burn against the gain in energy from performing TEI deep in the Mars' gravity well. Figure 3.7.4-4 shows the performance gains for both approaches.

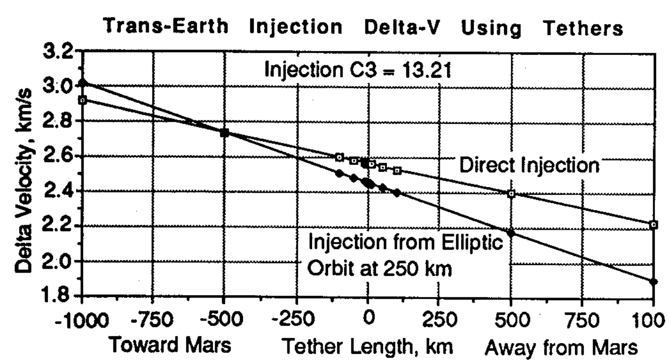


Figure 3.7.4-4 Trans-Earth Injection ΔV Using Tethers

It should be noted that the same tether may not be used for both TEI and MCSV augmentation because the necessary attach points on Phobos are on opposite sides of the moon. However, it may be possible to attach the tether station on a Phobos protrusion near its Mars terminator, allowing both operations.

3.7.5 Sample Tether System

Table 3.7.5-1 shows a sample tether system that could be used for Phobos applications.

Table 3.7.5-1 Sample Tether System

System Masses (tonnes)	
Total Tether System Mass:	75.0
Two Tether Cords (1400 km, 77 t capability)	61.80
30.914 t each: Based on Cortland date on 20,000N	
Kevlar lines including a 2.0 safety factor.	
Dual Capstan Reel Storage (x2)	1.50
Control System	1.75
Sensors, Computers, Controller	0.75
Tension Capstans and Structures	1.0
Dual Motors and Transmissions (10 kW)	2.00
Power System (10 kWe PVPA)	0.70
Micro Meteor Shield	0.75
System Structure	1.00
Landing/Departure Tower	5.00
Beams, Rocket anchors, Power lines	
Accelerator Ramps, Beacons, Landing pads	
Track to Vehicle Changeout Building	0.50

3.8 COMMUNICATIONS AND CONTROL

The communications for Mars missions will be mission drivers in some respects, whereas Lunar communications can be handled by a multitude of different approaches, ranging from use of TDRSS to a new geostationary communication satellite.

Mars missions may be divided into phases:

A.	Earth-to-orbit transportation	(ETO)
B.	Earth orbital	
C.	Earth escape	(TMI)
D.	Transfer to Mars	
E.	Mars orbital capture	(MOC)
F.	Mars orbital/landed	
	Subphase 1 - Pre-landing	
	Subphase 2 - Mars Entry and Landing	(MEL)
	Subphase 3 - Post-landing	
	Subphase 4 - Ascent, Rendezvous, and Docking	(ARD)
	Subphase 5 - Post-ascent	
G.	Mars escape	(TEI)
H.	Transfer to Earth	
I.	Earth capture/recovery	(EOC)
J.	Transport to Earth's surface; Post-landing	
Notes:		
B.	On-orbit assembly, checkout, fuel-up	
H.	Orbital capture OR direct entry	

Mission Rule (suggested): Mission Operations near-term command authority will reside with the Space-borne Command Post (SCP) whenever the round-trip communications propagation delay from Ground Control to the element in question exceeds 10 seconds and the SCP-to-element roundtrip delay is less than 10 seconds.

Definition: "Near-term authority" = command, control, deviations issuance, and emergency response for all activities that occur in time periods of less than one day.

Consequence: A strong communications link with Earth is needed only for special event coverage and emergencies at relatively short range to Earth, up to 1.5 x 10^6 km (i.e., about 0.5% of Mars-Earth maximum range), and for a period of several days.

Three separate communication users must be serviced:

Vehicle and engineering subsystems

- monitoring; command and control (C & C)

Science

- data; command and control

Human occupants

- information exchange
- human factors communications needs

The information stream can be categorized as video, sound, and data

Video is needed

- to support human needs for psychosocial interactions and maintaining the Earth-tie
- to monitor EVAs, mechanisms and actuators (e.g., antenna slewing), and IVA activities

Sound is needed

- to monitor engineering mechanisms and subsystems
- to monitor astronaut activities
- to support human needs (including interpersonal relationships, information exchange, music)

Data is needed

- to monitor engineering systems and subsystems

- to monitor crew health
- to obtain science instrument results
- to monitor Solar monitoring package

The *pacing* link-rate derives from the Video requirements. The ground communications infrastructure is driven by data *rate* requirement.

What is the minimum video support for a manned Mars mission? None? Mercury, Gemini, and early Apollo were all conducted without video. No unmanned program uses video for engineering monitoring/ housekeeping. However, for long-term missions into deep space, video will be needed just for crew support:

- as a countermeasure against psychological disconnection from Earth
- as an aid to monitor astronaut health and performance
- to maintain crew-ground working relationships at peak achievable performance

A mission design approach would be to provide two continuous downlink channels. The crew would control the primary downlink video channel. Ground mission operations has control of the secondary downlink channel, but maintains a current request for a downlink camera, but with a first and second priority. If the crew has already selected the first priority channel, the second priority is automatically invoked. Thus two downlink video channels are required, at videoconferencing quality.

The same could be used for uplink, with roles reversed. Thus two uplink video channels are required, at videoconferencing quality. During off-duty periods, selection of the crew-controlled uplink channel may be assigned to individual crewmembers, or to group vote.

During periodic high-rate comlinks (morning, noon, evening, and midnight) and during special events and emergencies, transmission of additional video channels may be opened up. At certain key periods it is assumed to be necessary to simultaneously provide each and every crewmember with a private video channel but not at highest resolution.

High and low gain links will be needed so that if pointing control is lost, a low-gain omnidirectional or wide-beam antenna can still maintain a link with Earth.

We have also suggested the idea of a free-flying nearby communications satellite with an omni link from MSS to ComSat. Advantages are that even if there is the spaceship loses orientation, some data is returned. Also the relay can get realtime live camera coverage of spaceship from far away and can be used to provide inspection capabilities. It may also be a more stable platform. Disadvantage: it will be necessary to dockup for MOC. Also, we cannot depend on this companion ComSat as the only link because of possibility of failure by various modes.

Table 3.8-1 Mars Missions Data Needs: MPV-->Earth, High Gain Link for Video

Data Source			Duty Cycle								
	kbps/30	Max	Continuous			10%			1% & Spec/Emerg*		
Purpose/Type	HzFR**	BER	# ch	FR	kbps	# ch	FR	kbps	# ch	FR	kbps
Human Factors and C & C											
Teleconferencing quality	1,500	10^{-3}	2	30	3,000	3	30	4,500	5	10	2,500
HD color, compressed, med rate	10,000	10^{-4}	--	--	--	1	3	1,000	5	3	5,000
Engineering monitoring											
B&W, good resol., low rate	3,000	10^{-3}	15	0.2	300	15	1	1,500	20	2	4,000
B&W, good resol., high rate	3,000	10^{-3}	2	1	200	2	5	1,000	5	30	15,000
Standard color TV quality	1,500	10^{-3}	5	1	250	--	--	--	--	--	--
HD color, low rate	10,000	10^{-4}	--	--	--	--	--	--	12	1	4,000
High Definition (HD) color, raw	100,000	10^{-4}	--	--	--	--	--	--	--	--	--
Solar monitor video	3,000	10^{-3}	10	0.1	100	15	0.5	750	10	0.3	1,000
Science (imaging)	3,000	10^{-4}	8	0.5	400	16	1	1,200	--	--	--

Notes:
* Spec/Emerg = Special and Emergency use. See derivation of needs. (Note: Solar monitor reduced, unless rad emergency)
Assumes pointing, power, and communications systems healthy (see low-gain backup)
** Rate in kilobits per second for a nominal frame rate (FR) of 30 Hz. Bit stream is data compressed and encoded.

Table 3.8-2 Mars Missions Data Needs: MPV-->Earth, High Gain Link for Sound

Data Source			Duty Cycle								
	kbps	Max	Continuous			10%			1% + Spec/Emerg*		
Purpose/Type	per ch	BER	# ch	ch rate	kbps	# ch	ch rate	kbps	# ch	ch rate	kbps
Voice, Conversational quality	20	10^{-2}	5	–	100	2	–	40	10	–	200
High fidelity (stereophonic CD qual)	100	10^{-3}	1	–	100	3	–	300	5	–	500

Table 3.8-3 Mars Missions Data Needs: MPV-->Earth, High Gain Link for Data

Data Source			Duty Cycle								
	Bits	Max	Continuous			10%			1% & Spec/Emerg*		
Purpose/Type	per ch	BER	# ch	ch rate	kbps	# ch	ch rate	kbps	# ch	ch rate	kbps
Engineering/housekeeping monitoring											
Nominal criticality		10^{-4}									
Low sampling rate	12		250	1/s	3	250	10/s	30	250	100/s	300
Medium sampling rate	10		100	10/s	10	100	100/s	100	70	1000/s	700
High sampling rate	8		20	100/s	16	50	1000/s	160	25	10 k	2,000
High criticality	12	10^{-6}	50	10/s	6	100	100/s	120	200	100/s	240
Science data											
Stored/buffered data	2000	10^{-4}	50	1/s	100	50	1/s	100	---	---	–
Real-time data	2000	10^{-4}	–	–	–	5	100/s	1,000	–	–	–
Solar Flare/Radiation Monitoring	2000	10^{-4}	30	1/s	60	30	10/s	600	30	100/s	600
Data base playback***	100,000	10^{-4}	1	1/s	100	10	1/s	1,000	30	1/s	3,000
Note: *** Checksum included											

Table 3.8-4 Mars Missions Data Needs: MPV-->Earth, High Gain Link Total

Totals (Video+Sound+Data)	(544 ch) **4.745 Mbps**	(595 ch) **13.4 Mbps**	(530 ch) **39.04 Mbps**
Note: 10% is 1.0 hr in the morning (at 7 a.m.), and 1.4 hr evening (6 p.m.) 1% is 7.2 minutes, twice per day (noon and nominally at midnight)			

Table 3.8-5 Mars Missions Data Needs: MPV-->Earth, Low Gain backup for Video

Data Source			Duty Cycle								
	kbps/30	Max	Omnidirectional			Broad-beam			Burst*		
Purpose/Type	HzFR**	BER	# ch	FR	bps	# ch	FR	bps	# ch	FR	bps
Crew Status											
Interior cabin views, color degraded	600	10^{-2}	–	–	–	–	–	–	2	0.1	4,000
Vehicle Status											
B&W, gross resol., low rate from external monitors)	300	10^{-2}	–	–	–	–	–	–	5	0.02	1,000

Notes:
* Omnidirectional and Broad-beam (approximately. XXX steradians) are separate and independent antenna and drive systems. Burst mode utilizes high gain system whenever attitude determination system is consistent with orientation toward DSN receivers or when receiver detects uplink communications signal.
** Rate in kilobits per second for a nominal frame rate (FR) of 30 Hz. Bit stream is data compressed and encoded.

Table 3.8-6 Mars Missions Data Needs: MPV-->Earth, Low Gain Backup for Sound

Data Source			Duty Cycle								
	kbps	Max	Omnidirectional			Broad-beam			Burst*		
Purpose/Type	per ch	BER	# ch	ch rate	bps	# ch	ch rate	bps	# ch	ch rate	bps
Crew Status											
Voice, reduced quality	3	10^{-2}	–	–	–	–	–	–	–	1	3,000
Vehicle Status											
Minimum quality sound	0.5	10^{-2}	–	–	–	–	–	–	2	–	1,000

Table 3.8-7 Mars Missions Data Needs: MPV-->Earth, Low Gain Backup for Data

Data Source			Duty Cycle								
	Bits	Max	Omnidirectional			Broad-beam			Burst*		
Purpose/Type	per ch	BER	# ch	ch rate	bps	# ch	ch rate	bps	# ch	ch rate	bps
Vehicle Status											
Vital monitors	2	10^{-4}	25	0.2/s	10	–	–	–	–	–	-
Nominal criticality		10^{-3}									
Low sampling rate	12		–	–	–	125	0.01/s	15	250	0.1/s	300
Medium sampling rate	10		–	–	–	20	0.1/s	20	70	1/s	700
High sampling rate	8		–	–	–	12	0.5/s	48	25	10/s	2,000
High criticality	12	10^{-4}	–	–	–	10	0.1/s	12	200	1/s	2,400

Table 3.8-8 Mars Missions Data Needs: MPV-->Earth, Low Gain Backup Total

Totals (Video+Sound+Data)	(25 ch) **10 bps**	(167 ch) **95 bps**	(545 ch) **4,400 bps**

Table 3.8-9 On-board Data Buffering, Mars Spaceship

39,040 kbps --> 1.4 x 10^{11} bits for 1 hr

Rationale: With 1 hr full storage, Ground Operations could have up to 5 to 37 minutes to react to anomalous results in the continuous data stream and initiate the emergency data stream to retrieve concurrent as well as 15 minutes of pre-emergency data (depending on roundtrip communication propagation times of 8 to 40 minutes).

Candidate Storage Media

80 MBy Hard Disk	6.4 x 10^{8} bits
8 mm video cassette	~ 10^{10} bits
Optical disk	~ 10^{12} bits

Table 3.8-10 Mars Spaceship <--> Earth, Synopsis

Downlink

Video for C & C, human factors: Continuous transmission of any two channels, one selected by crew and one by ground operations. Six channels of transmission for 2.4 hrs each day. Ten channels in emergency situations.

Video for hardware monitoring: 22 low-rate monitors, 15 solar images every 10 s, and 8 astrophysical images per second.

Voice: 5 channels continuous conversational quality, 1 high fidelity stereo. 15 channels during emergencies.

Data, Engineering: 420 channels (ranging from 1 to 100 Hz sampling rates). 35 kbps.

Data, Science: 100 kbps science. 60 kbps solar patrol.

Uplink

Video for Command & Control: Continuous transmission of any two channels, one selected by crew and one by ground operations.

Video for human factors: Five channels of transmission for 6 hrs each day for R & R use.

Voice: 1 channel continuous conversational quality, 5 high fidelity stereo.

Command & Control, Engineering: 100 kbps when under ground control. 30 kbps otherwise (includes provisions for data base updates).

Command & Control, Science: 1 kbps science.

4.0 HABITATS

A number of habitat studies were conducted during the course of this work. Habitats and habitability are keynotes of the mission because they drive vehicle design, especially for Mars mission scenarios. During the course of this contract, a total of 16 habitability concepts were proposed for manned Mars and lunar expeditions. One additional habitat, designed under another study, is included in this section for the sake of comparison. Standard features in both lunar and Mars mission habitats include a personal hygiene area (PH) and a Command and Control Center (CCC). Additional facilities, such as a Health Maintenance Facility (HMF) and a galley, are also included in a Mars spaceship because of the long interplanetary flight time. Nomenclature for the facilities is varied and often confusing. Table 4-1 lists all of the possible names for the facilities, highlighting in bold at the top the ones we chose as standard.

Table 4-1 Habitat Nomenclature (Including Alternatives)

Wardroom	**Stow**	**Shelter**	**Corridors**	**Galley**
Lounge	Logistics	Radiation Shelter	Hallway	Food Prep
Public Area	Storage	Rad	Passageway	Galley/Pantry
Recreation Area	Stowage	Cabin/Rad Shelter	Deck	Dining Area
Meeting Room	Equip/Consum	Pantry/Rad Shelter	Traffic Flow	
R & R	Stores	Storm Shelter		
Deck	General Storage &		**Work Areas**	**Fitness Center**
Ballroom	Logistics		Lab	Exercise Station
		Health	Maintenance	Physical Fitness
Command and	**Personal**	**Maintenance**	Work Space	Workout Room
Control Center	**Hygiene (P.H.)**	**Facility (HMF.)**	Laboratory	
(CCC)	Head	Bio-Med	Science Work Station	**Quarters**
Control Center	Laundry/Shower	Med/Lab/HMF	Engineering Station	Stateroom
Command and	Commode	Medical	Shop/Maintenance	Cabin
Control	Bathroom	Medical Lab/OR	Experiments/Investigations	Crew Quarter
Command Center	WMF		Maintenance/Housekeeping	Bedroom
Cmd Center	Toilet		Lab/Maintenance Work Stations	Private Suite
Command Console	Lavatory			Room
Piloting Station	WC			Crew Compartment
	Waste Elimination Facility			

Each facility within a habitat was assessed in the following ways;

Total Volume available: Interior volume of habitat before outfitting.

Total Floor Area Available: Floor area, before outfitting.

Walking Floor Area: Open floor area.

Walkable Volume: Walking floor area multiplied by ceiling height.

Additional Free Volume: Volume above tables and beds, under desks, and of ceiling and floor storage facilities.

Outfitted Volume: Actual volume of equipment, tables, beds, exercise facilities, etc.

A summary of the results for each habitat is shown in Table 4-2.

Table 4-2 Mars and Lunar Mission Habitats

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Free Volume, m^3	Outfitted Volume, m^3
Martian Habitats						
2-Cylinder–Martin Marietta	420.0	126.0	64.9	159.2	111.2	149.6
1-Cylinder–E. Cliffton	125.0	—	—	—	—	—
3-Cylinder–E. Cliffton	630.0	233.4	110.9	265.4	65.1	299.4
2-Cylinder (Short)–J. Danelek	265.0	88.2	72.5	177.2	49.4	38.4
2-Cylinder–Eagle Engineering	420.0	70.9	37.3	117.4	218.2	84.4
2-Disk–J. Danelek	225.0	60.3	43.0	105.6	77.1	42.3
2-Disk–Eagle Engineering	388.0	97.4	67.8	184.7	130.7	72.6
1-Disk–E. Cliffton	136.0	—	—	—	—	—
1-Disk–E. Cliffton	136.0	—	—	—	—	—
1-Disk–E. Cliffton	136.0	—	—	—	—	—
1-Disk (Mezzanine)–E. Cliffton	300.0	—	—	—	—	—
Lunar Habitats						
1-Deck LCSV	33.5	31.2	3.6	7.6	5.6	20.3
2-Deck LCSV Habitat	88.0	30.4	9.7	24.6	13.2	50.2
Alternative LPV Habitat	85.0	31.9	10.0	25.5	12.9	46.6

For those habitats with artificial gravity, the acceleration vector can be either transverse (max horizontal vista = 12.8 m) or longitudinal along the cylinder (max horizontal vista = 4.6 m), and is always transverse on disk modules (max horizontal vista = 7.6 m). The transverse cylindrical module packages most readily in the low L/D aerobrake configuration. In addition, an array of modules in this orientation allows a "running track" toroidal closure. The longitudinal module minimizes corridor volume by making the ladder a corridor also. This also benefits the health of the crew since they exercise while climbing the ladder. Lastly, there is a fall hazard, but if acceleration levels are sub-gee, this may be more acceptable. Finally, disk modules have the maximum "floor" area for the same volume, and a compromise between the two cylinder types for maximum longitudinal vista. However, disk modules do not have any potential of derivation from Space Station designs.

4.1 MARS MISSION HABITATS

4.1.1 Interplanetary Habitats

Mars mission habitats are cylinders derived from Space Station designs. These designs often employ artificial gravity to ensure the health and strength of the crew. Zero-gravity missions could adopt Space Station Freedom modules almost directly, although they were designed with the concept of 90 day (or 180 day maximum) mission for the occupants. Most also provide an exercise area.

One concept, a Martin Marietta design, is shown in Figure 4.1.1.-1. This efficient design is unique because both cylinders are sectioned into 5 levels, giving a feeling of a large structure. In fact, compared to a habitat of equal overall volume, this has 25% more floor area and 45% additional walkable volume (Table 4.1.1.-1). Finally, the crew quarters are comparatively large in these cylinders, further enhancing the relative comfort of the crew.

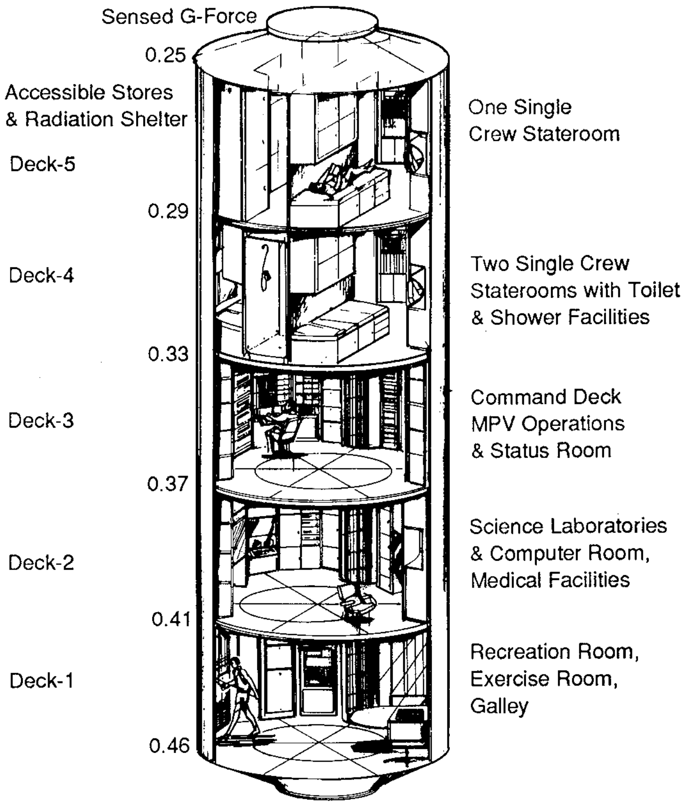


Figure 4.1.1-1 Habitat A

The second habitat, designed by E. Cliffton, is similarly tiered into 3 levels (Fig. 4.1.1.-2). As the smallest in the cylinder class (125 m^3), the crew is considerably more cramped than in the other habitats. Each of the three tiers has a different function. The bottom houses the crew quarters and P H. The center tier, the Wardroom, also has a galley and an HMF. Finally, the CCC/work station is segregated on the uppermost level, making this single facility the largest (Table 4.1.1.-2).

The next habitat is the largest in the cylinder class by 33% (Fig. 4.1.1-3, Table 4.1.1-3). 630 m^3 is divided between 3 habitat modules with stowage accounting for over 1/3 of the total volume, and corridors for 1/5th. Six crew members share 99.4 m^3 for living quarters, and six separate work areas comprise the CCC.

Table 4.1.1-1 2-Cylinder Habitat, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Quarters	138.5	42.5	12.7	31.1	34.8	72.7
Galley	33.5	5.9	2.0	5.0	19.1	9.5
Personal Hygiene	11.4	3.0	2.3	5.7	3.4	2.3
Ladder	18.2	0.5	0.5	1.1	17.0	0.0
Health Maintenance Facility	33.6	4.8	2.0	5.0	21.8	6.8
Fitness Center	16.8	4.3	1.8	4.5	6.1	6.1
Command & Control Center	84.0	32.5	21.8	53.4	4.5	26.1
Work Area	84.0	32.5	21.8	53.4	4.5	26.1
Totals	420.0	126.0	64.9	159.2	111.2	149.6

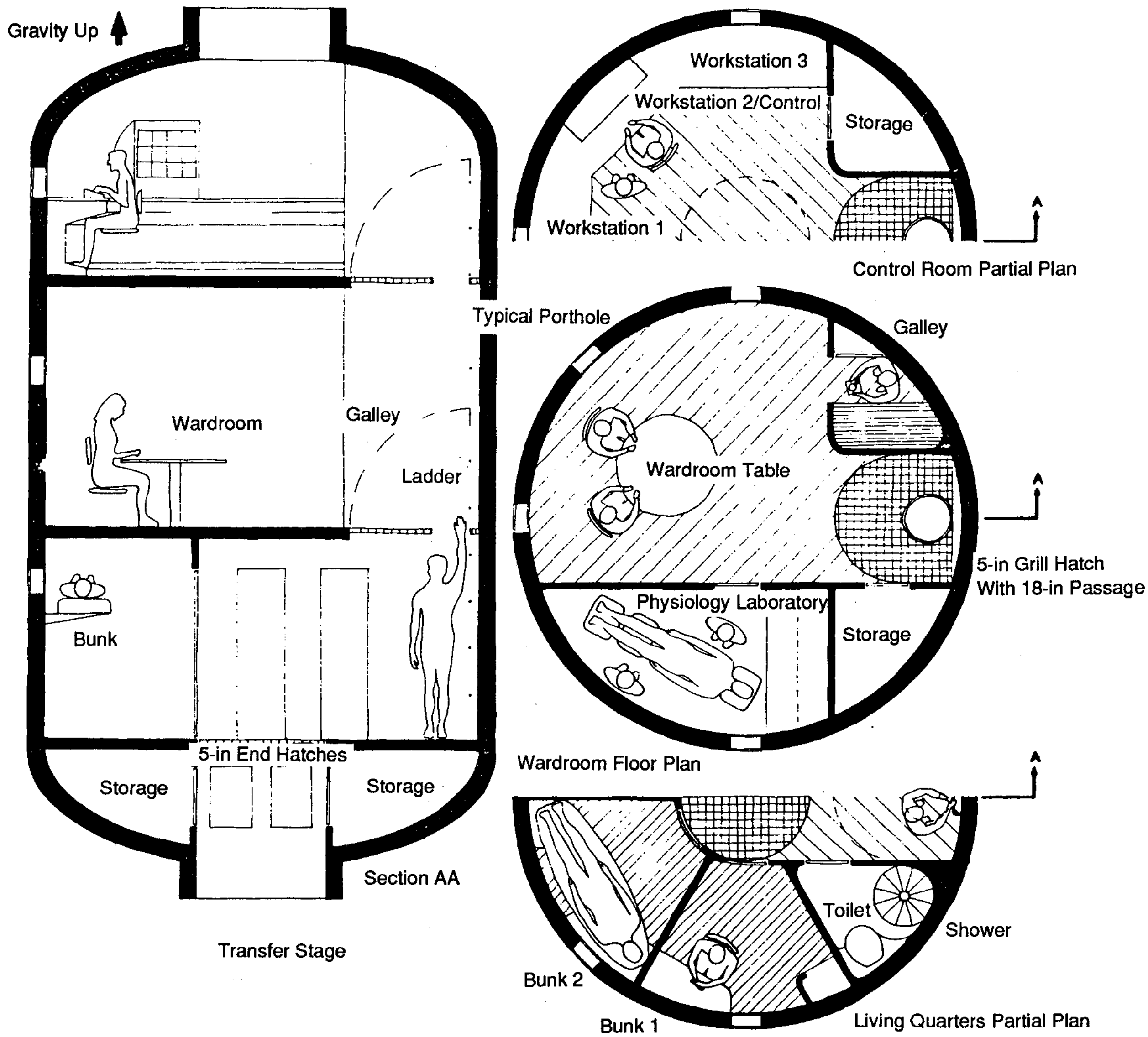


Figure 4.1.1-2 Habitat B

Table 4.1.1-2 1-Cylinder Habitat

	Total Volume Available, m^3
Quarters	19.3
Galley	3.6
Personal Hygiene	24.6
Ladder	10.2
Command & Control Center/Work Area	28.6
Stow	16.2
Wardroom	22.5
Totals	125.0

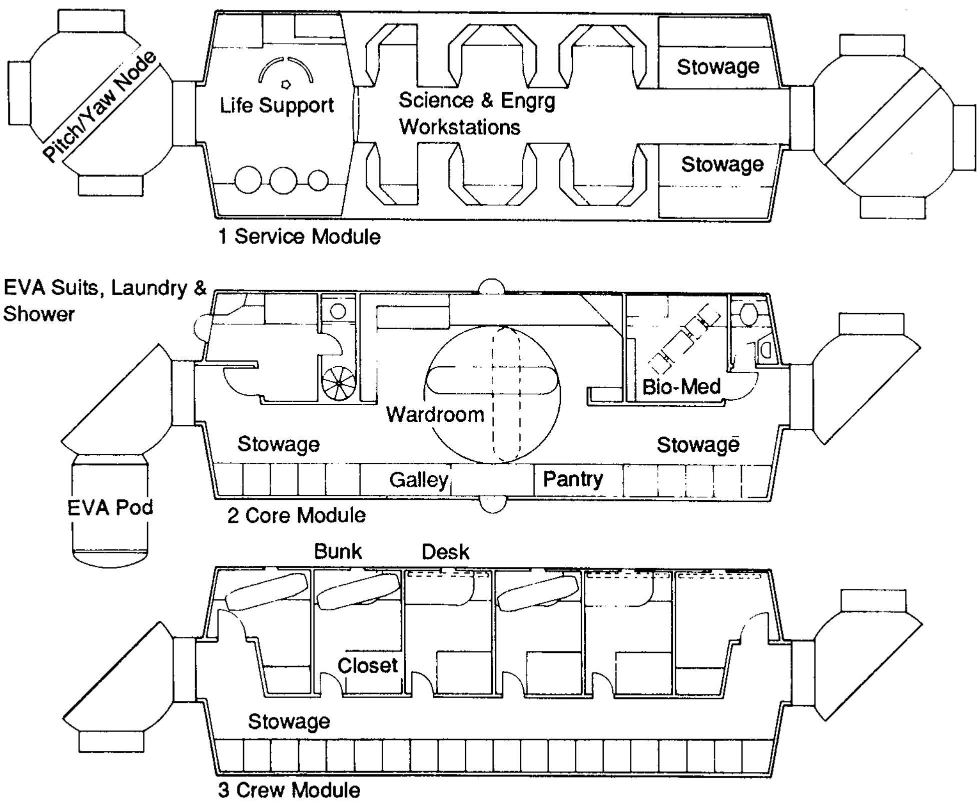


Figure 4.1.1-3 Habitat C

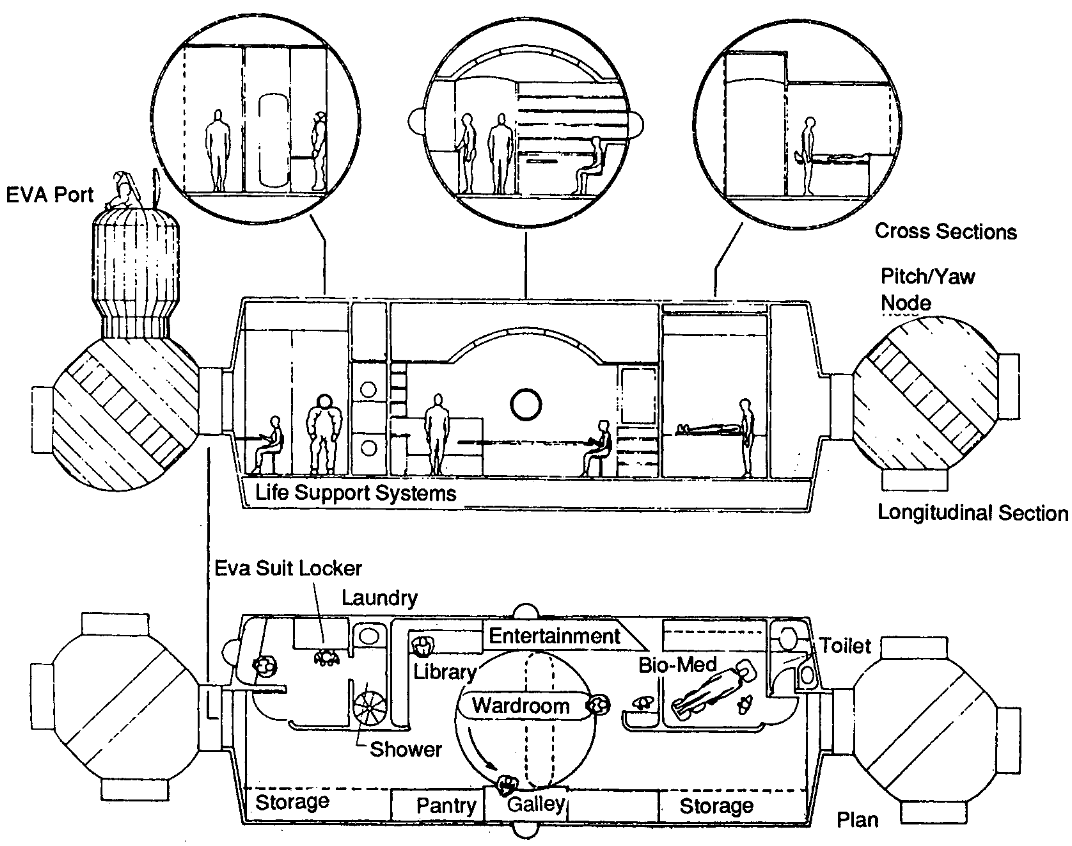


Figure 4.1.1-3 (continued)

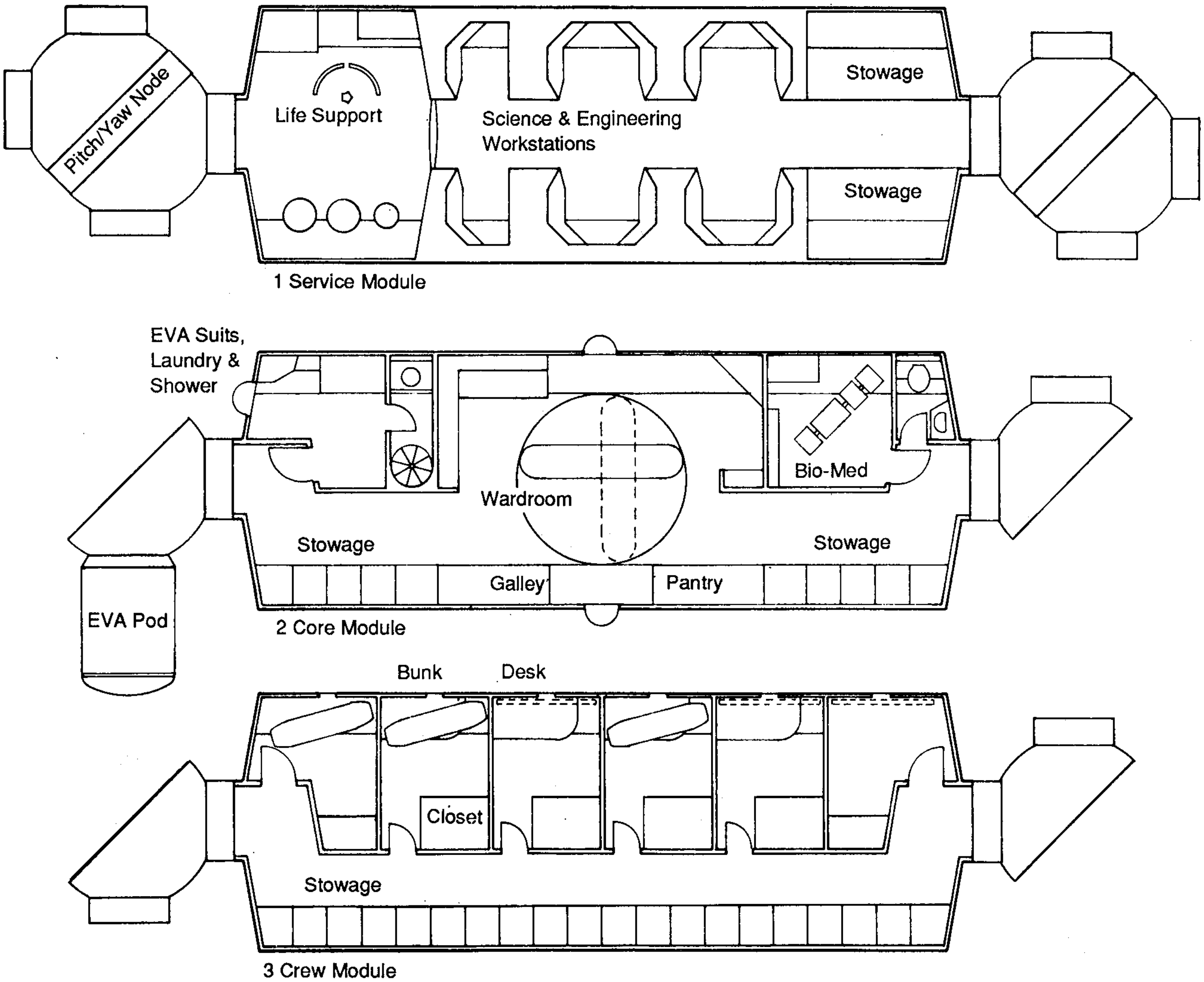


Figure 4.1.1-3 (concluded)

Table 4.1.1-3 3-Cylinder Habitat

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Corridor	110.3	45.8	45.8	110.4	0.0	0.0
Quarters	99.4	25.7	23.3	56.2	37.3	5.4
Galley	52.0	20.8	15.5	35.3	3.9	12.8
Personal Hygiene	9.8	4.1	1.8	4.4	0.0	5.5
Health Maintenance Facility	13.5	3.9	2.9	7.0	4.1	2.5
Life Support	67.4	20.9	12.1	29.2	16.7	21.5
Command & Control Center/Work Area	48.6	18.7	9.5	22.9	3.1	22.6
Stow	229.0	93.5	0.0	0.0	0.0	229.1
Total	630.0	233.4	110.9	265.4	65.1	299.4

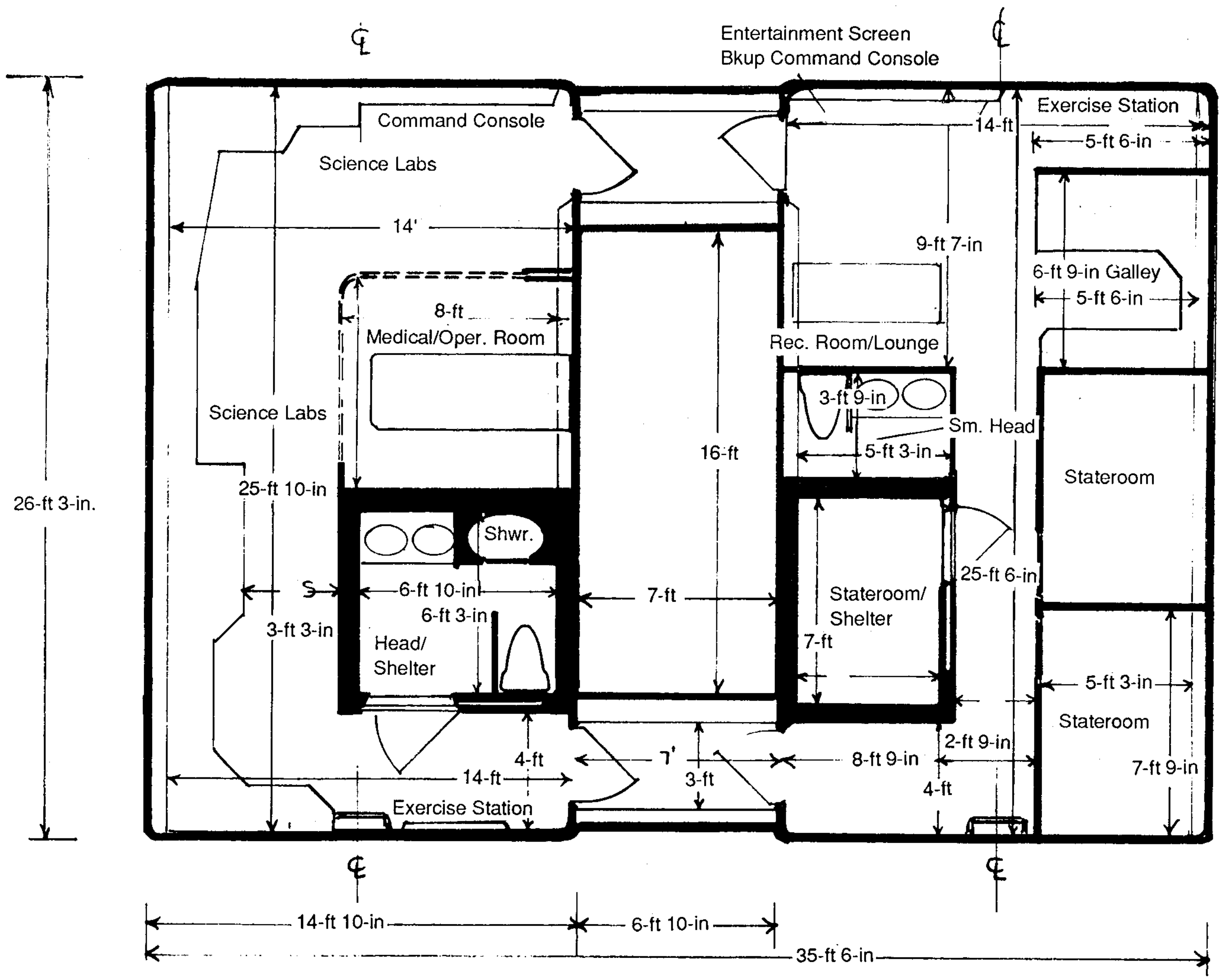


Figure 4.1.1-4 Habitat D

Table 4.1.1-4 2-Cylinder (Short) Habitat, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Corridor	73.4	30.0	30.0	73.4	0.0	0.0
Quarters	36.7	8.5	4.2	10.2	15.9	10.6
Galley	37.8	11.6	9.2	22.6	9.5	5.7
Personal Hygiene	19.8	6.2	4.3	10.6	4.6	4.6
Ladder	4.2	1.6	1.6	3.7	0.4	0.0
Health Maintenance Facility	18.5	6.1	5.0	12.2	3.5	2.9
Fitness Center	12.0	3.0	1.0	2.5	4.8	4.8
Command & Control Center	13.6	4.9	4.6	11.3	1.7	0.7
Work Area	49.0	16.3	12.6	30.8	9.1	9.1
Total	265.0	88.2	72.5	177.2	49.4	38.4

J. Danelek has designed a 2-cylinder habitat which has 265 m^3 total available volume and accommodates 3 crew (Fig. 4.1.1-4, Table 4.1.1-4). Unique to this design is the presence of two shelters and 2 exercise stations, one in each module. Corridors occupy the most volume (25%) in the habitat.

The next habitat, designed under another study, was contributed to the collection by Martin Marietta/ Danelek. Two 210 m^3 cylinders lie side-by-side, and access to either is granted through one of two connecting tunnels. Stowage occupies the most volume (Fig. 4.1.1-5, Table 4.1.1-5).

The last artificial gravity cylindrical habitat was contributed by Eagle Engineering (Fig. 4.1.1-6, Table 4.1.1-6). In this case two cylinders lie side-by-side, similar to the above arrangement. Unlike all of the previous designs, the galley is the largest facility area, followed by crew quarters. The CCC has the smallest amount of volume allocated, which is the same as the exercise facility.

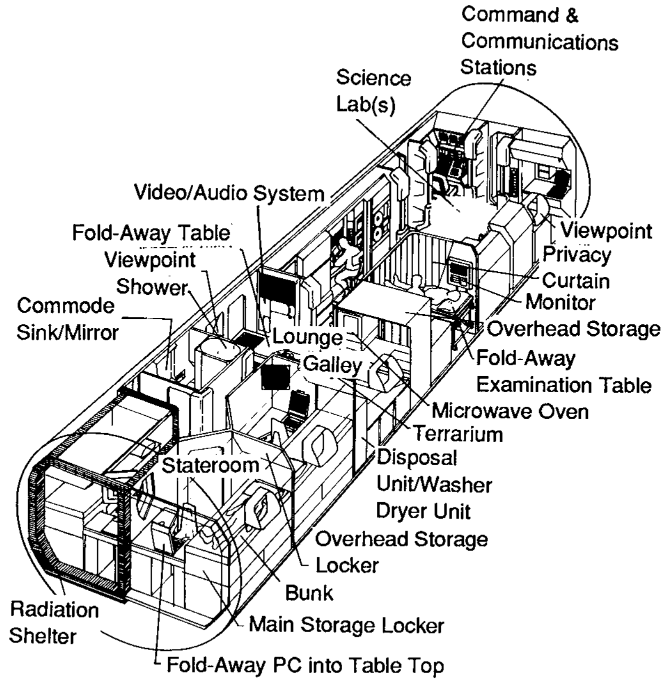


Figure 4.1.1-5 Habitat E

Table 4.1.1-5 Cylindrical Habitats, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Quarters	56.8	20.8	11.4	28.0	14.4	14.4
Galley	23.6	8.6	4.2	10.0	4.0	9.6
Lounge	19.2	7.0	6.2	15.2	3.2	0.8
Personal Hygiene	10.2	3.8	2.2	5.4	3.4	1.4
Corridor	18.2	6.8	2.8	18.2	0.0	0.0
Health Maintenance Facility	17.6	6.4	2.4	5.6	6.0	6.0
Fitness Center	6.8	2.6	1.4	3.6	1.6	1.6
Work Area	85.0	31.2	15.8	38.6	23.2	23.2
Command & Control Center	17.6	6.4	2.4	5.6	6.0	6.0
Stow	165.0	0.0	0.0	0.0	0.0	165.0
Total	420.0	93.6	48.8	130.2	61.8	228.0

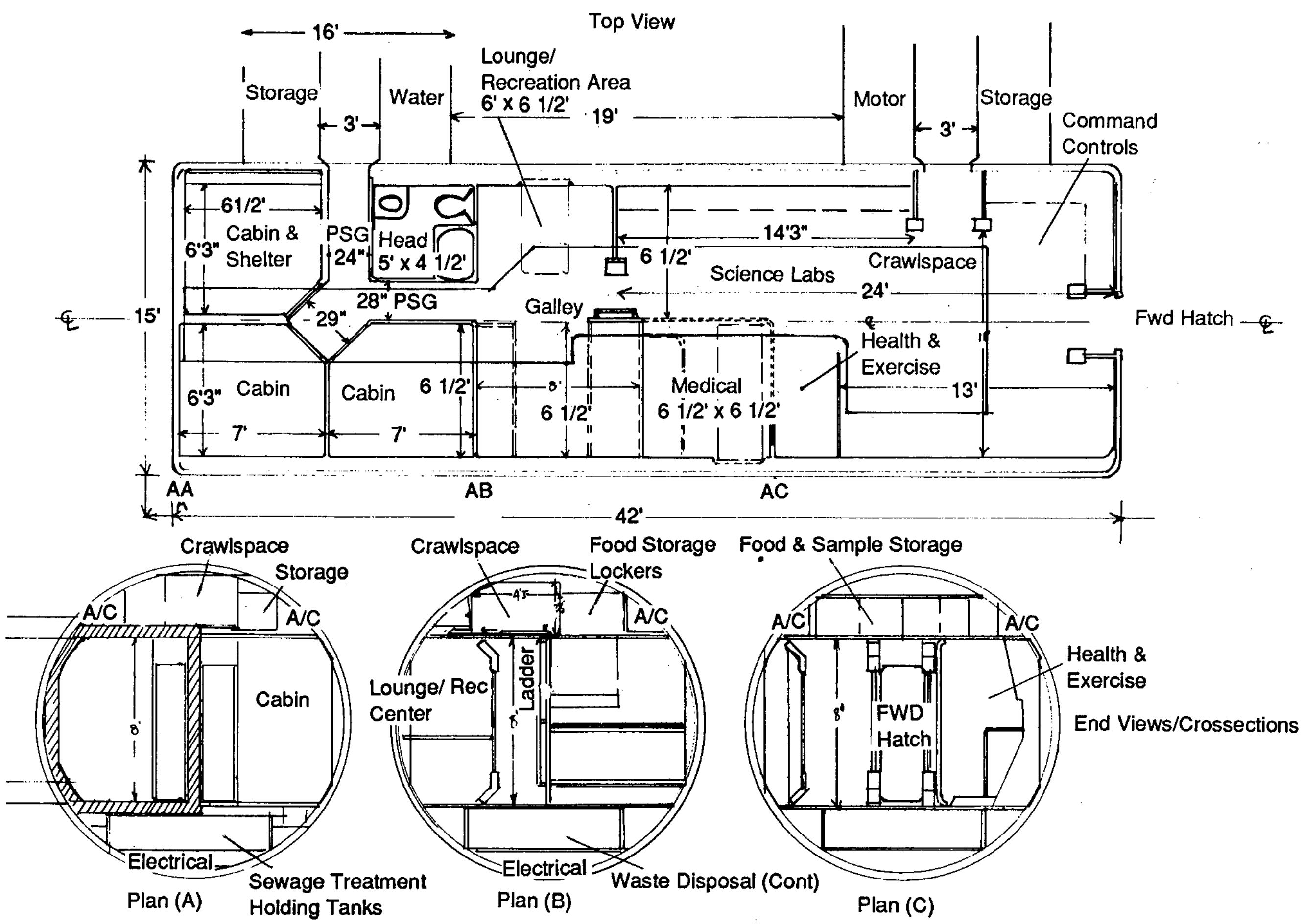


Figure 4.1.1-5 (concluded)

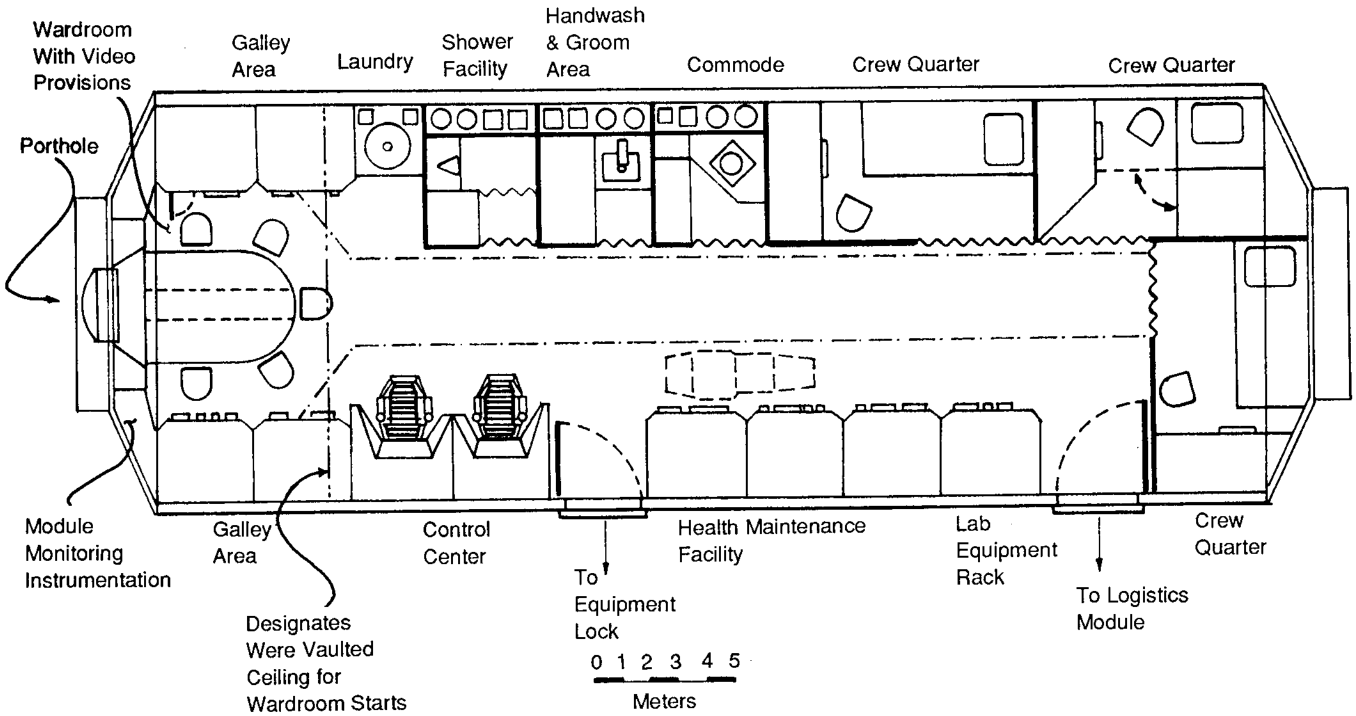


Figure 4.1.1-6 Habitat F

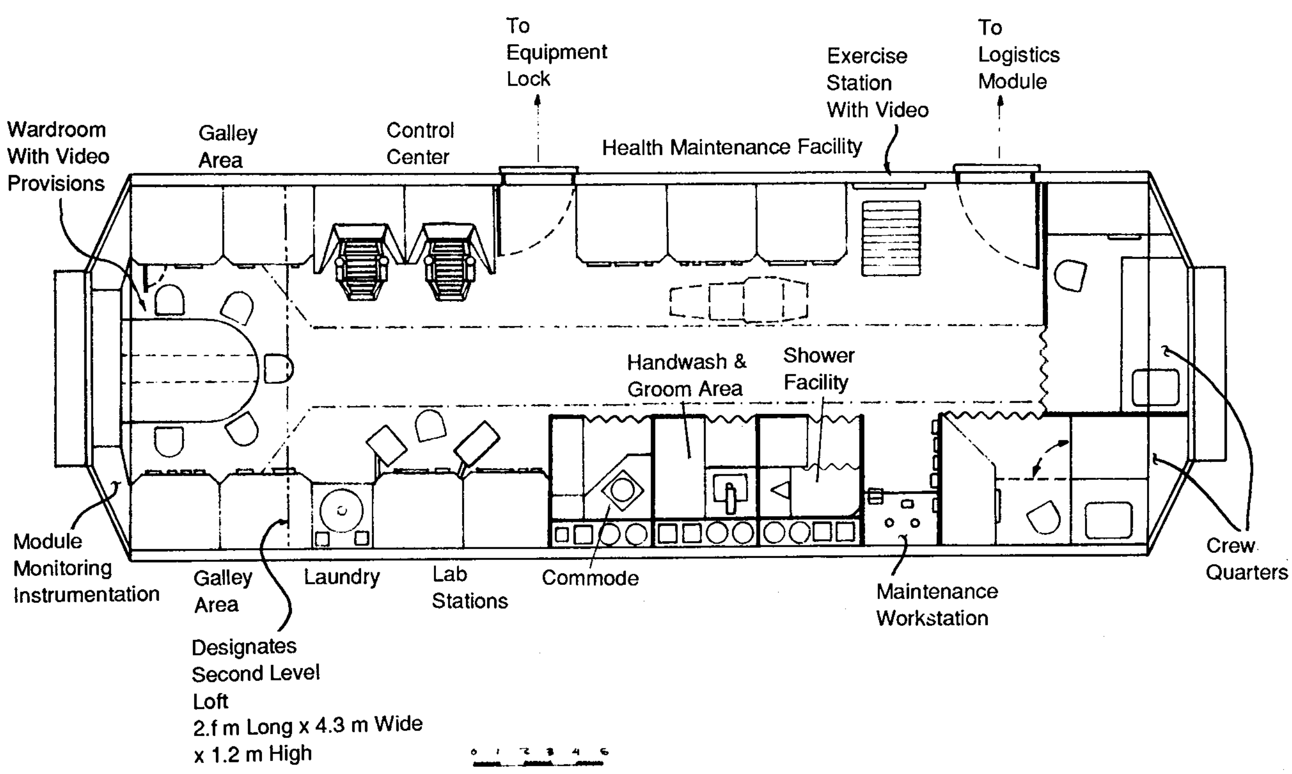


Figure 4.1.1-6 (continued)

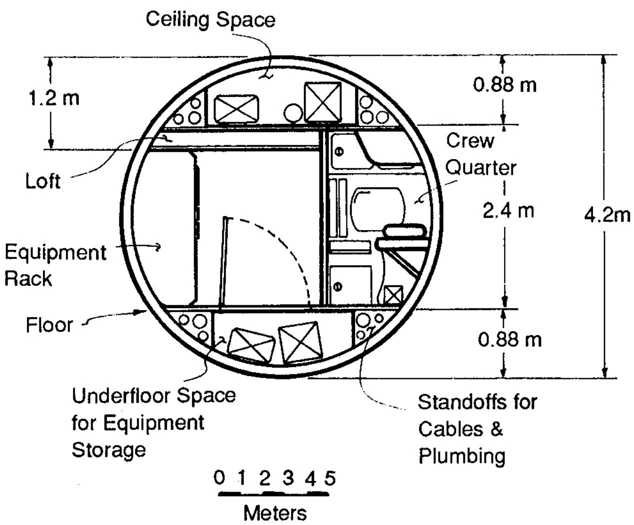


Figure 4.1.1-6 (concluded)

4.1.2 Mars Descent Vehicle Habitats

Two of the following disk habitats have artificial gravity. The 4 others have a single level with no artificial gravity. All of these habitats are designed around a centrally located hub, which is surrounded by the facilities. The maximum number of crew members per habitat is 5.

The first 2-disk habitat was created by J. Danelek. Access between the disks is provided by two opposing ladders. This disk has slightly more than half the volume of a 420 m^3 cylindrical habitat, yet it accommodates a crew of 4 (Fig. 4.1.2-1, Table 4.1.2-1).

Table 4.1.1-6 2-Cylinder Habitat, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Corridor	34.0	7.7	7.7	18.8	15.2	0.0
Quarters	96.0	12.1	10.4	25.2	66.6	4.3
Galley	118.9	17.8	6.1	41.9	48.3	28.7
Personal Hygiene	49.8	9.4	2.0	4.8	26.8	18.3
Health Maintenance Facility	48.5	11.0	3.7	9.0	21.6	17.9
Fitness Center	16.2	2.6	2.4	6.0	10.0	0.2
Command & Control Center	16.2	3.5	1.6	3.5	5.8	6.6
Work Area	40.4	6.8	3.4	8.4	23.9	8.4
Total	420	70.9	37.3	117.4	218.2	84.4

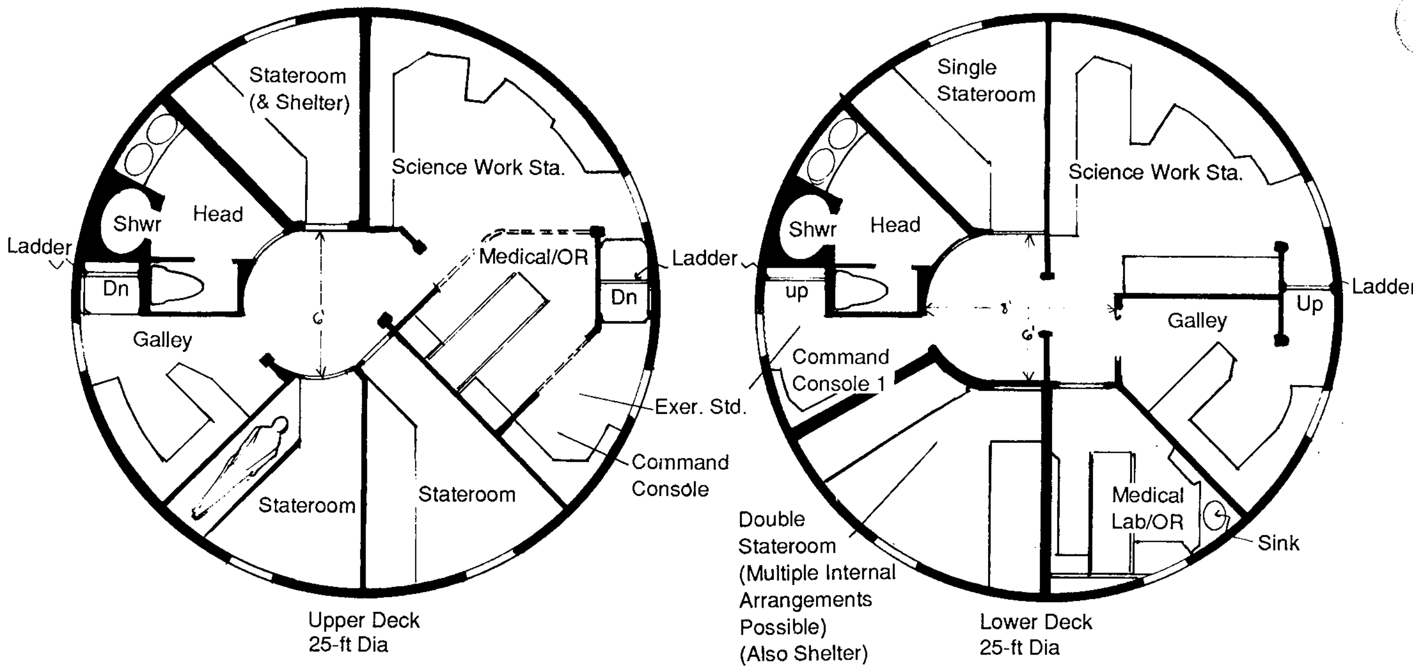


Figure 4.1.2-1 Habitat G

Table 4.1.2-1 2-Disk Habitat, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Corridor	15.6	6.4	6.4	15.6	0.0	0.0
Quarters	76.2	13.3	8.2	20.3	43.4	12.5
Galley	21.9	7.4	5.8	14.1	3.9	3.9
Personal Hygiene	20.9	4.5	2.9	4.8	7.1	7.1
Ladder	7.9	3.1	2.1	6.8	2.7	0.0
Health Maintenance Facility	24.0	8.7	6.1	15.0	2.9	6.1
Fitness Center/Command & Control Center	12.0	3.4	1.9	5.5	3.4	3.4
Work Area	46.5	13.5	9.6	23.5	13.7	9.3
Total	225.0	60.3	43.0	105.6	77.1	42.3

The other 2-disk habitat was designed by L. Guerra and B. Stump from Eagle Engineering. This is the largest of the disk habitats and it has a complement of 5 crew members (Fig. 4.1.2-2, Table 4.1.2-2). The crew quarters and galley are segregated from the CCC, HMF and storm shelter, and access between modules is achieved by a central ladder.

The following three 136 m^3 disk habitats have only one level (Fig. 4.1.2-3a, -3b, -3c, Table 4.1.2-3a, -3b, -3c) and no artificial gravity. Crew quarters are nearly identical in all three, as is the HMF. The first distinguishes itself by placing emphasis on the work area and central wardroom. The second is divided into three equal triangles except for the wardroom, which is located at the end of one passageway. The final habitat is divided into 4 sections, with a crew quarter at the end of each radiating corridor. In each habitat the central wardroom is encircled by the remaining facilities (i.e., HMF, CCC).

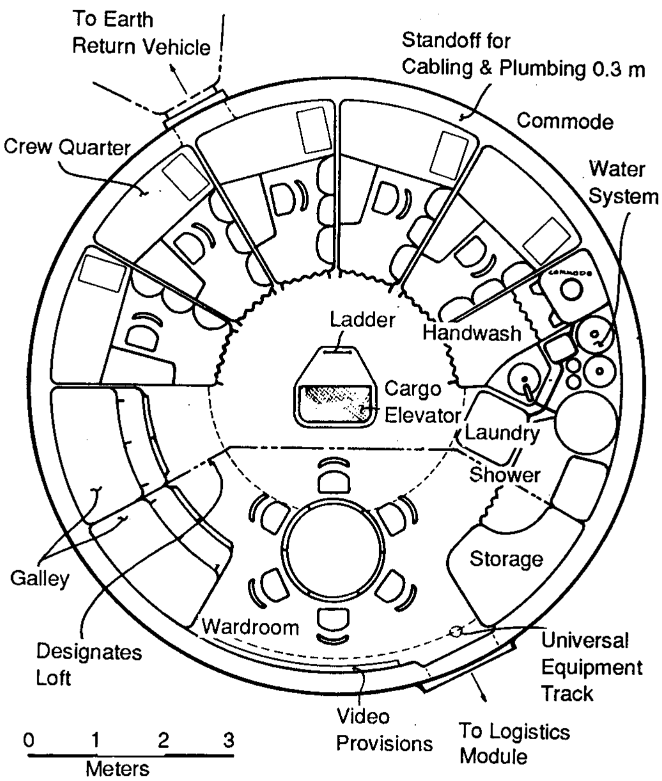


Figure 4.1.2-2 Habitat H

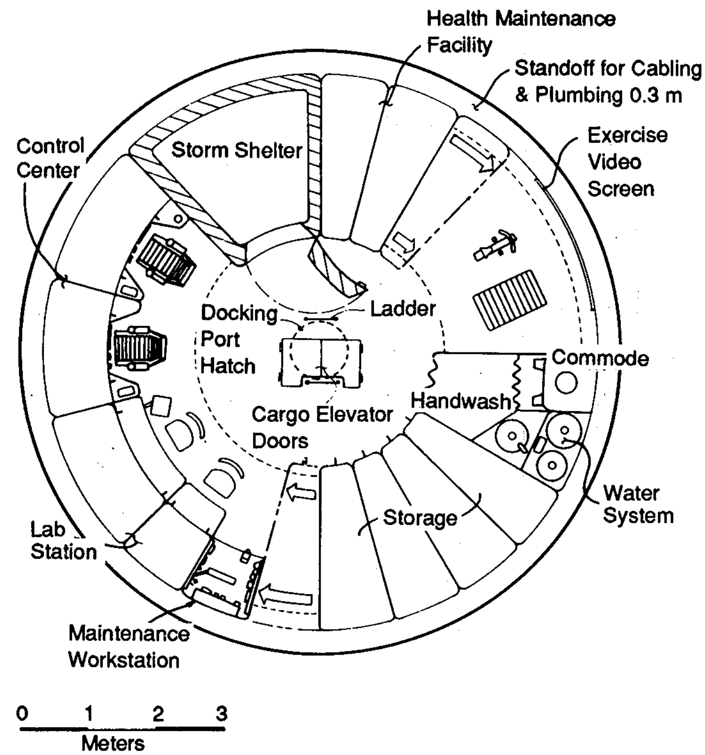


Figure 4.1.2-2 (concluded)

Table 4.1.2-2 2-Disk Habitat, Artificial Gravity

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Corridor	64.0	26.2	21.9	53.7	0.0	10.4
Quarters	65.5	16.2	9.0	22.0	25.9	17.7
Galley	64.0	16.5	10.7	26.1	23.6	14.3
Personal Hygiene	38.1	6.8	2.4	5.8	21.6	10.7
Ladder	7.6	3.0	3.0	7.3	0.3	0.0
Health Maintenance Facility	22.9	7.7	6.1	14.9	4.0	4.0
Fitness Center	22.9	8.4	7.5	18.3	2.8	2.3
Command & Control Center	25.9	6.3	3.6	8.7	10.7	6.6
Work Area	25.9	6.3	3.6	8.7	10.7	6.6
Stow/Shelter	50.3	7.8	7.8	19.2	31.1	0.0
Total	388.0	97.4	67.8	184.7	130.7	72.6

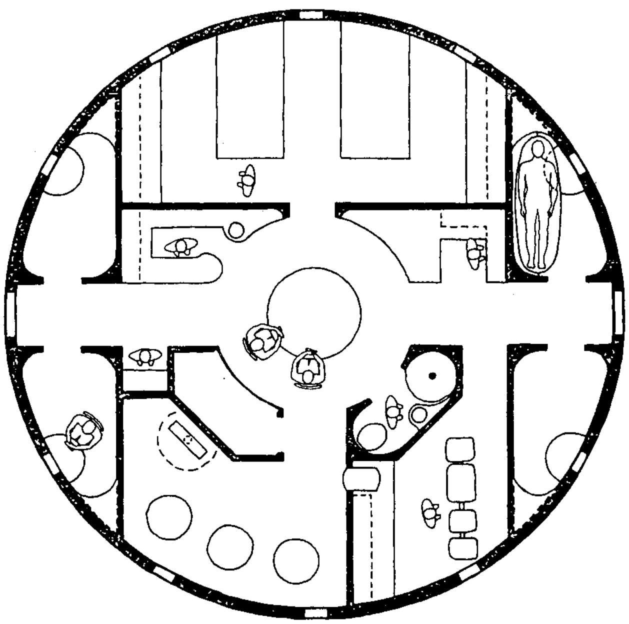


Figure 4.1.2-3a   1-Disk Habitat

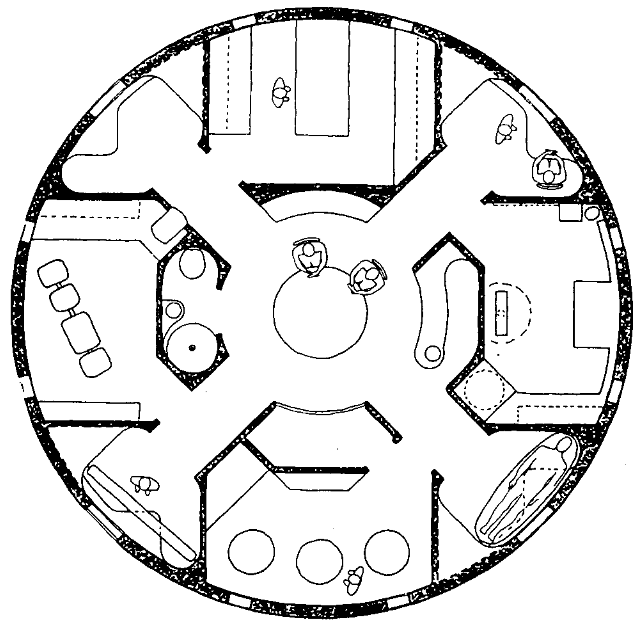


Figure 4.1.2-3c   1-Disk Habitat

Table 4.1.2-3a   1-Disk Habitat

	Total Volume Available, m^3
Corridor	13.9
Quarters	27.8
Galley	4.5
Personal Hygiene	4.7
Health Maintenance Facility	13.4
Command & Control Center	15.2
Work Area	32.0
Wardroom	15.8
Greenhouse	4.7
Entertainment & Communication	4.0
Totals	136.0

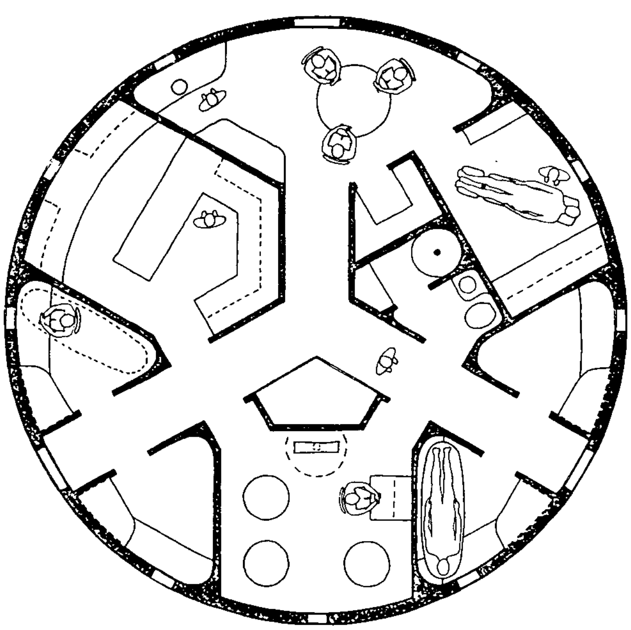


Figure 4.1.2-3b   1-Disk Habitat

Table 4.1.2-3b   1-Disk Habitat

	Total Volume Available, m^3
Corridor	20.3
Quarters	28.2
Galley	2.4
Personal Hygiene	5.8
Health Maintenance Facility	12.6
Command & Control Center	17.4
Work Area	20.6
Wardroom	25.5
Greenhouse	3.2
Totals	136.0

Table 4.1.2-3c 1-Disk Habitat

	Total Volume Available, m^3
Corridor	12.7
Quarters	28.5
Galley	4.7
Personal Hygiene	4.7
Health Maintenance Facility	16.6
Command & Control Center	16.6
Work Area	33.1
Wardroom	14.4
Greenhouse	4.7
Totals	136.0

The final disk habitat has one level and a mezzanine for crew quarters, which gives it more than twice the total volume of the three previous habitats (300 m^3, Table 4.1.2-4). Again, the facilities revolve around the wardroom hub. Corridor space consumes over 1/3 of the total volume, with the work area a distant second.

Four crew members' quarters are situated on the mezzanine, over their desks. The work area is open to its full height, up to the ceiling of the mezzanine (Fig. 4.1.2-4).

Table 4.1.2-4 1-Disk Habitat with Mezzanine

	Total Volume Available, m^3
Corridor	105.7
Quarters	35.0
Galley	6.1
Personal Hygiene	5.6
Health Maintenance Facility	16.0
Command & Control Center	14.4
Work Area	80.6
Wardroom	22.2
Stow	14.4
Totals	300.0

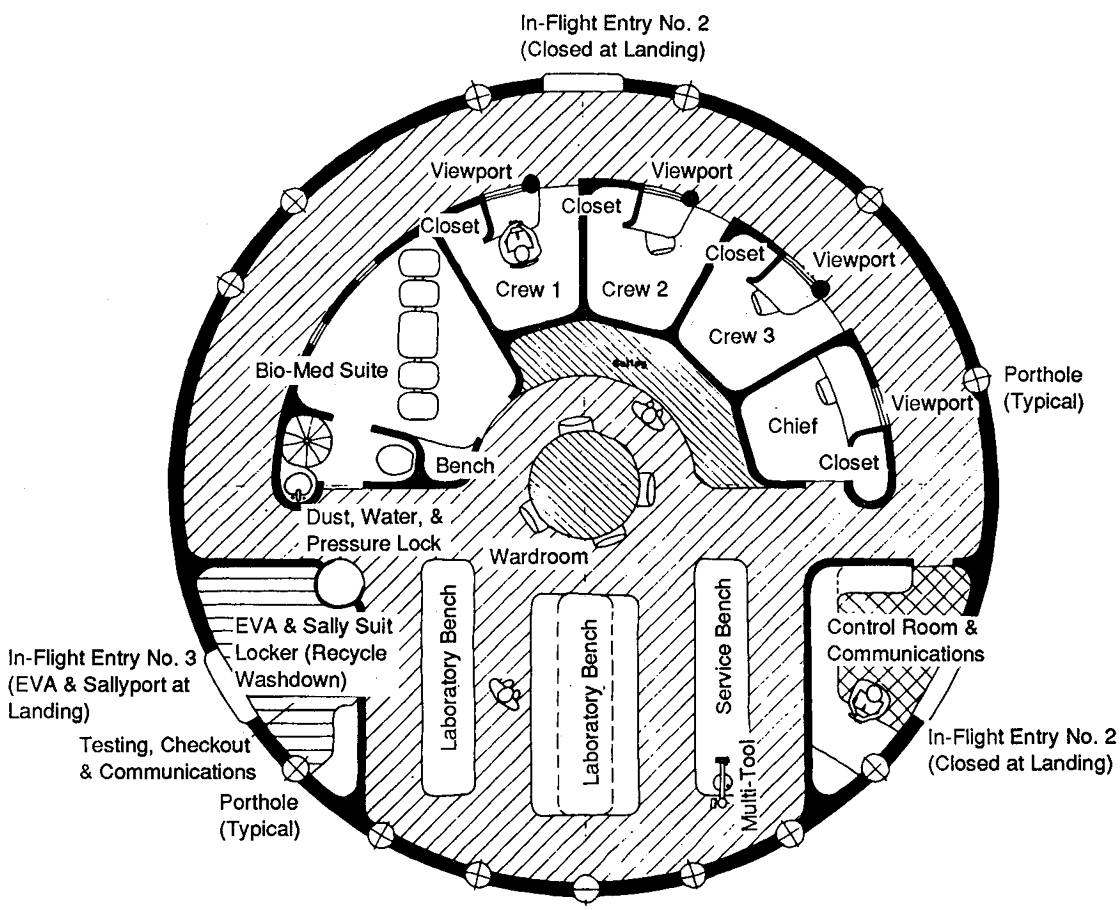


Figure 4.1.2-4 Habitat L

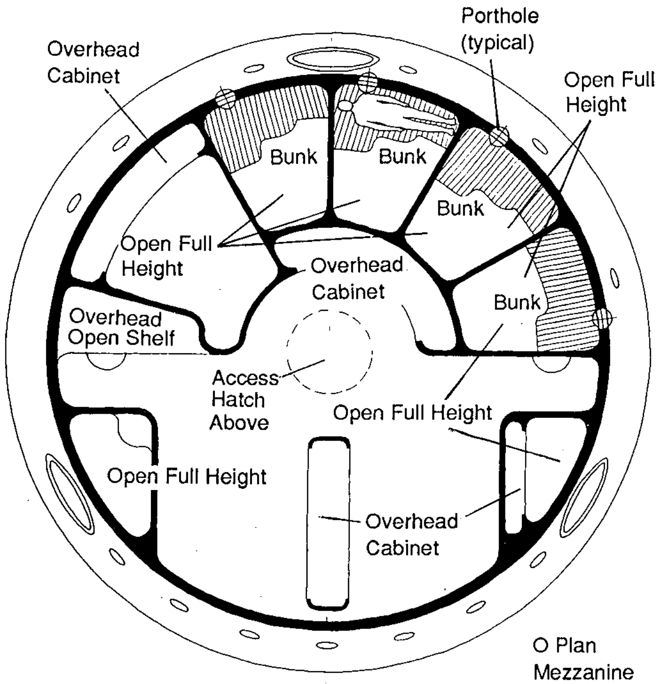


Figure 4.1.2-4 (concluded)

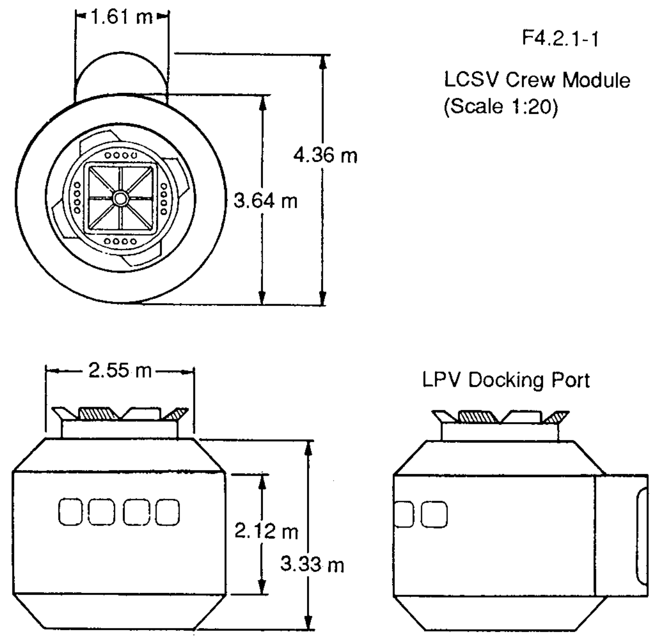
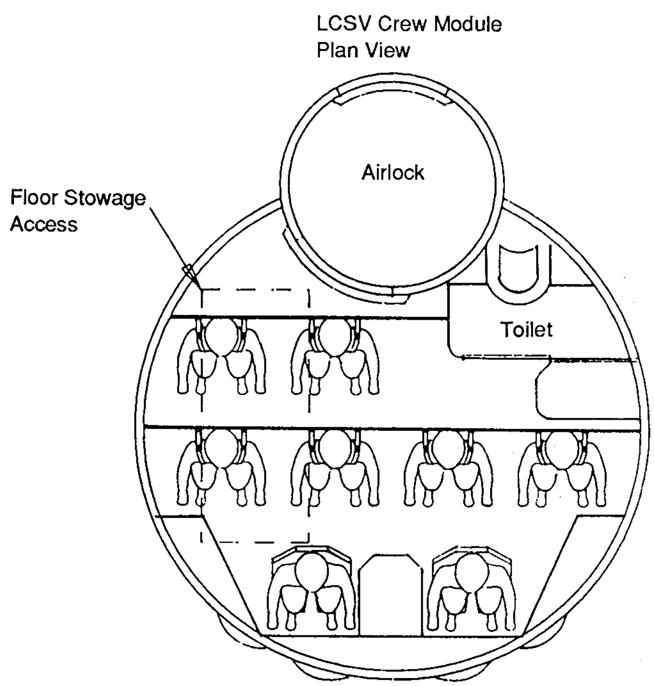


Figure 4.2.1-1 Habitat M



(Design by Eagle Engineering)

Figure 4.2.1-1 (concluded)

4.2 LUNAR MISSION HABITATS

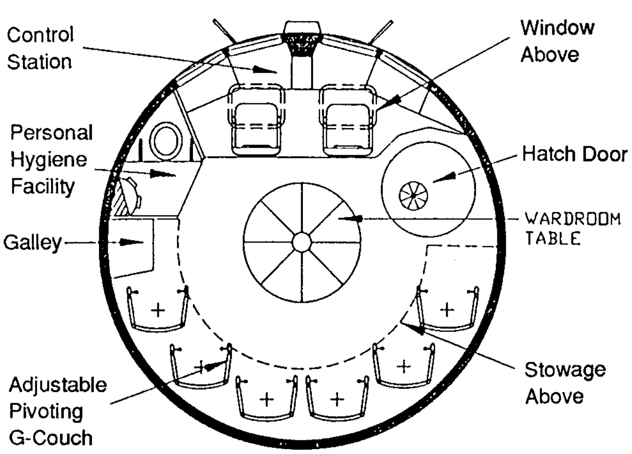
Four lunar habitats are proposed; 2 Lunar Crew Sortie Vehicles (LCSV) and two Lunar Piloted Vehicles (LPV). The first two transport 8 crew members between the lunar surface and low lunar orbit, and The LPVs make the round trip between low-earth orbit and low-lunar orbit.

4.2.1 Lunar Excursion Vehicle Crew Cab

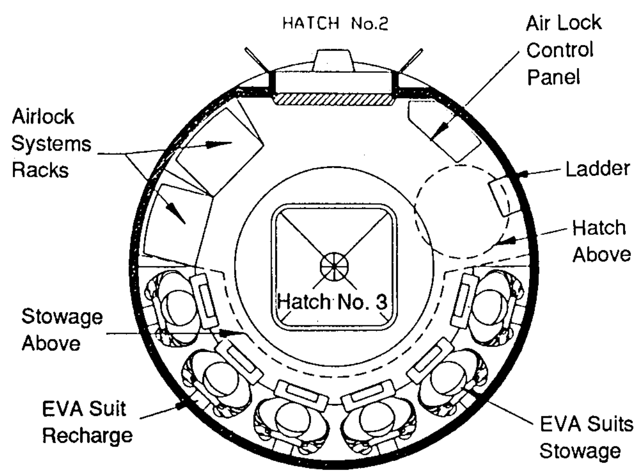
The LCSV Habitat is extremely small, allowing only for necessities such as 8 reentry/sleep couches and a PH (Fig. 4.2.1-1, Table 4.2.1-1). The alternative LCSV has nearly 33% more volume spread over 2 decks. The six crew members have access from the Flight Deck/Habitation Chamber to the Air Lock /Stowage Chamber by means of a hatch that opens onto a ladder. In addition, there is a wardroom table and galley on the Flight Deck (Fig. 4.2.1-2, Table 4.2.1-2).

Table 4.2.1-1 LCSV Habitat

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Airlock	4.3	2.0	2.0	4.3	0.0	0.0
Command & Control Center	7.4	3.5	0.0	0.0	2.5	4.9
Reentry & Sleep Area	10.0	4.7	0.6	1.3	2.9	5.8
Personal Hygiene	2.6	1.2	1.0	2.0	0.2	0.4
Propellant	4.6	0.0	0.0	0.0	0.0	4.6
Stow	4.6	0.0	0.0	0.0	0.0	4.6
Total	33.5	31.2	3.6	7.6	5.6	20.3



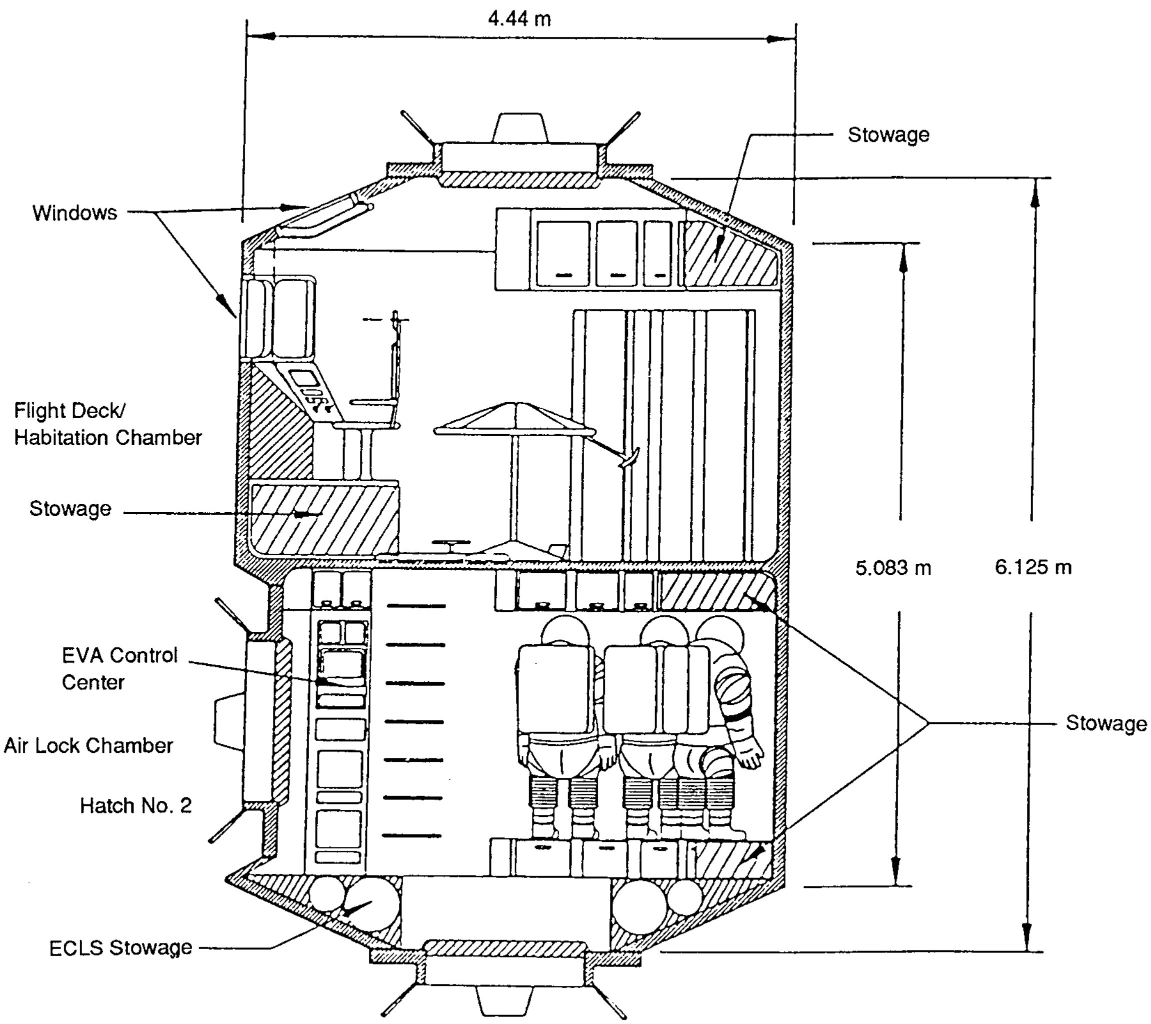
Flight Deck/Habitation Chamber



Air Lock/Stowage Chamber

Figure 4.2.1-2 Habitat N

By: Eagle Engineering (L. Guerra, B. Stump)



Crew Transfer Module Section

Figure 4.2.1-2 (concluded)

Table 4.2.1-2 2-Deck LCSV Habitat (Alternative)

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Airlock	31.4	12.6	5.2	13.0	3.7	14.7
Command & Control Center	11.6	4.5	0.0	0.0	2.3	9.3
Reentry & Sleep Area	10.0	5.0	0.0	0.0	3.3	6.7
Personal Hygiene	1.8	0.9	0.3	0.6	0.3	0.9
Ladder	1.9	1.6	0.0	0.0	1.5	0.4
Galley	0.6	0.3	0.0	0.0	0.0	0.6
ECLSS	4.6	0.0	0.0	0.0	0.0	4.6
Corridor	11.0	4.2	4.2	11.0	0.0	0.0
Stow	11.9	0.0	0.0	0.0	0.0	11.9
Wardroom	3.2	1.3	0.0	0.0	2.1	1.1
Total	88.0	30.4	9.7	24.6	13.2	50.2

4.2.2 Lunar Transfer Habitats

The LPVs have approximately the same volume as the Alternative LCSV, and accommodate many of the same facilities. The LPVs Crew Module (Fig. 4.2.2-1, Table 4.2.2-1) has a large amount of unused volume, especially above the storage compartments separating the reentry/sleep couches. The alternative LPV has slightly less volume, yet makes better use of the available volume. Because of this, the habitat has an exercise station and a larger galley. A total of eight crew members occupy two levels. (Fig. 4.2.2-2, Table 4.2.2-2)

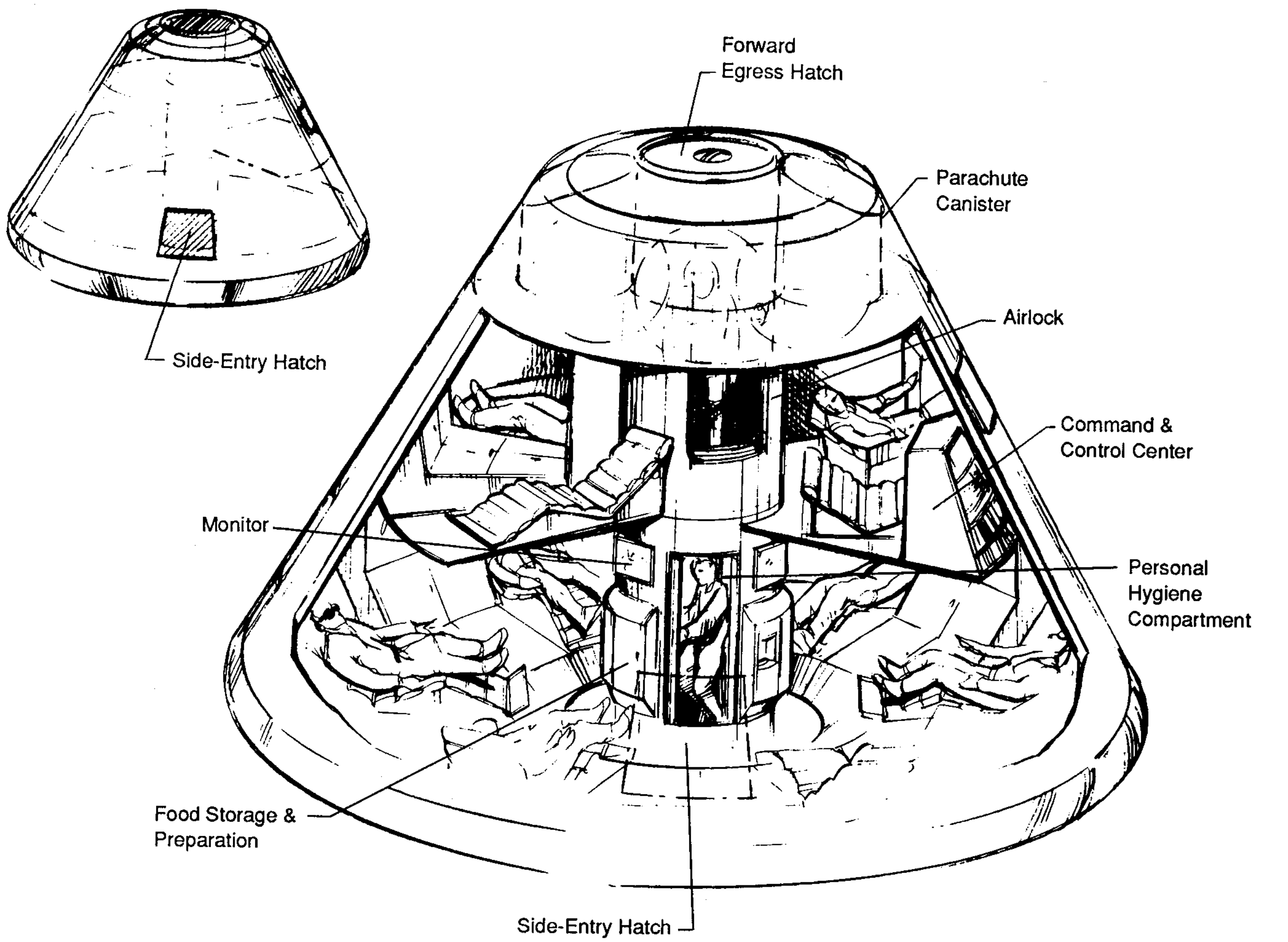


Figure 4.2.2-1 Habitat O

Table 4.2.2-1 LPV Habitat

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Airlock	2.9	1.2	1.2	2.9	0.0	0.0
Command & Control Center	5.0	7.5	0.0	0.0	2.5	5.0
Reentry & Sleep Area	20.8	14.3	0.0	0.0	1.8	18.5
Personal Hygiene	1.7	1.2	0.1	0.2	1.1	0.5
Propellant	3.1	7.8	0.0	0.0	0.0	3.1
Galley	0.3	0.4	0.0	0.0	0.0	0.3
Stow	57.2	37.1	0.0	0.0	38.1	19.1
Total	91.0	69.5	1.3	3.1	43.1	44.8

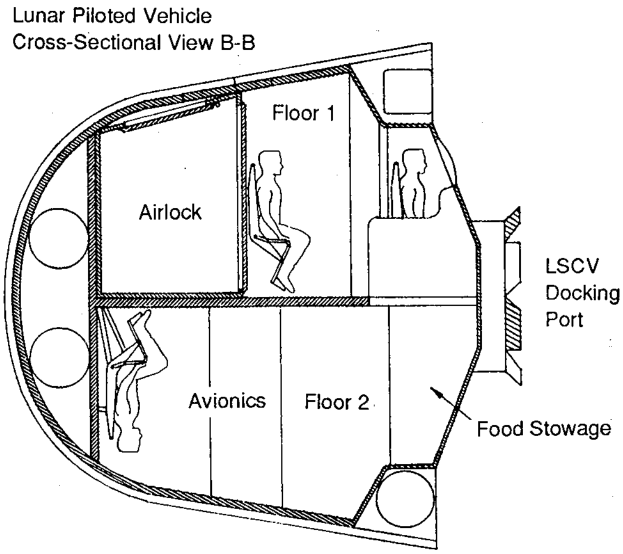


Figure 4.2.2-2 Habitat P

4.3 CREW SIZE AND COMPOSITION

Special considerations need to be made for long-term missions, such as voyages to Mars or extended tours of duty at a lunar base. Probably the most important factor from the standpoint of mission sizing is the minimum crew size.

Crew complements of from two to twenty have been suggested at various times. Somewhat surprisingly, the mass needed for habitats and consumables dominates the transportation system for long-term missions and tends to scale significantly with the number of crew members being supported. There is, therefore, ample impetus to restrict the number of astronauts to be supported on such missions. All other things being equal, it is desirable to have large crew sizes because of the division of labor, the increase in types of skills available, the number of hands that can be applied to physical and/or mental work, greater flexibility in manning and in back-up modes, and an increase in social stability and quality of life for the group. Accordingly, it is important that engineering solutions that can minimize mass and complexity growth yet accommodate larger crews should be studied with great care and attention to innovation. Fertile areas for analysis include the use of inflatable structures to increase volume, the application of high-efficiency closed life support systems, advances in minimal-volume food storage, and eventually the use of *in situ* planetary resources for habitat construction and food and propellant production.

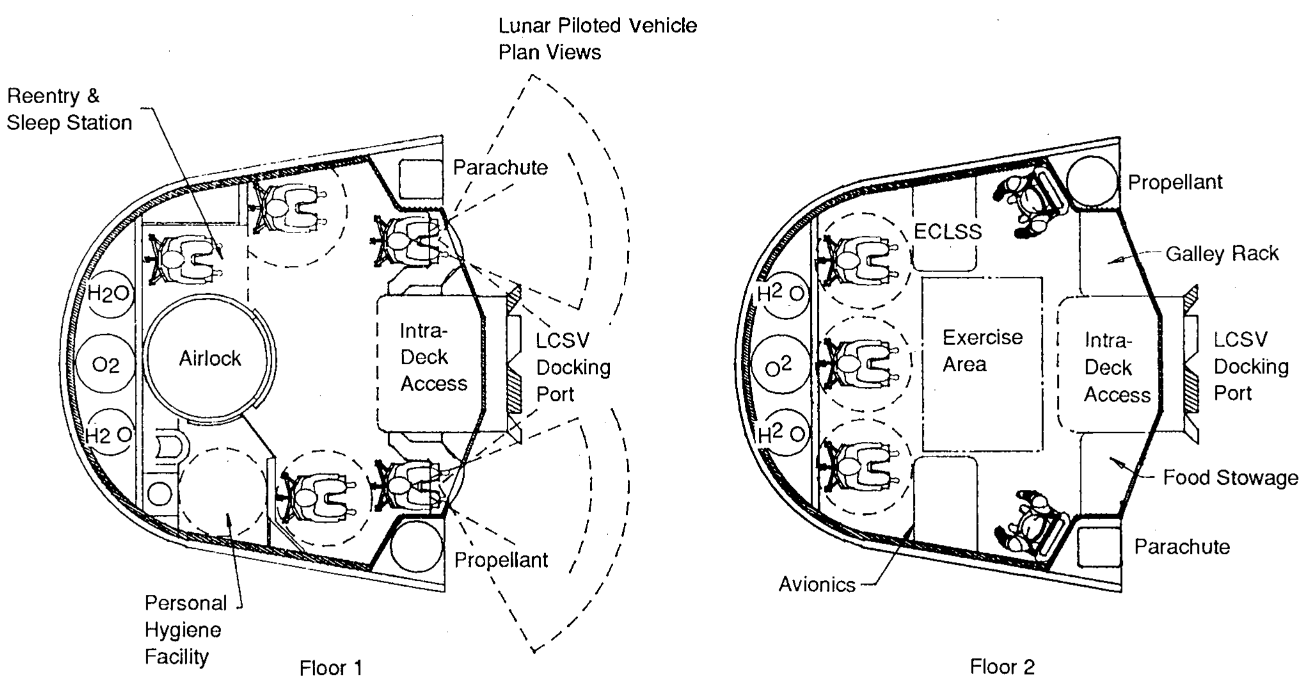


Figure 4.2.2-2 (continued)

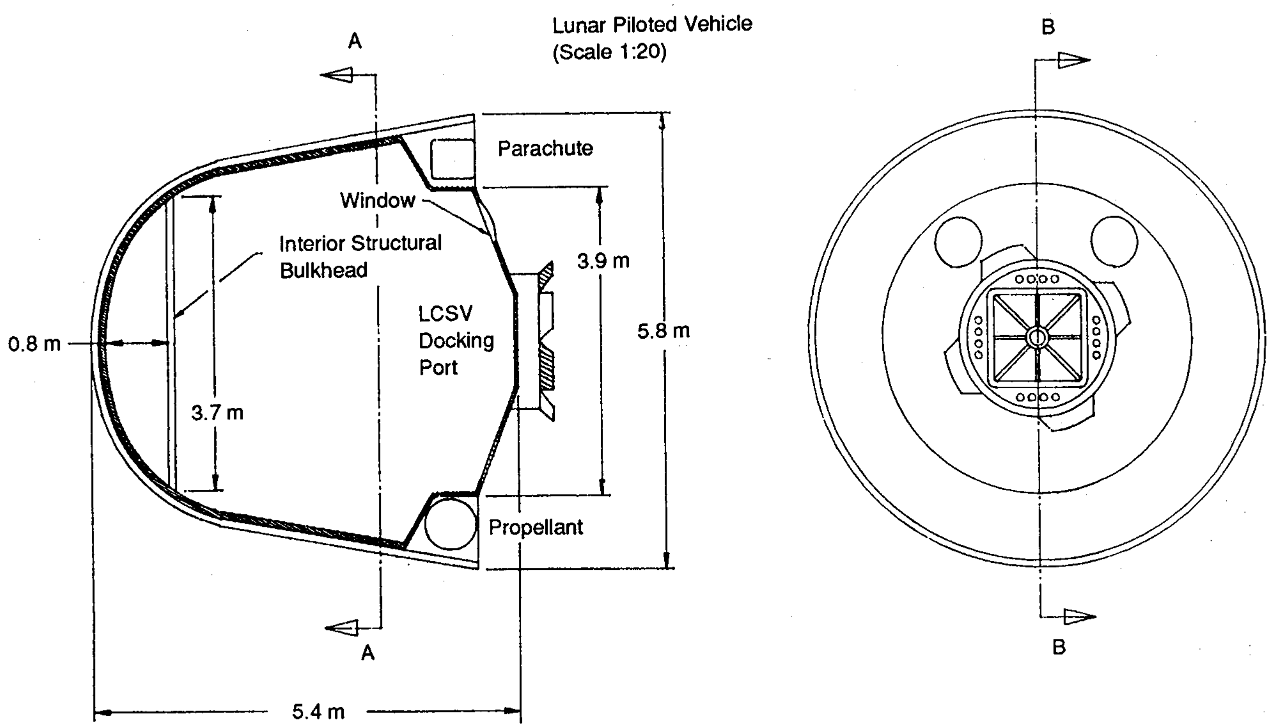


Figure 4.2.2-2 (concluded)

Table 4.2.2-2 Alternative LPV Habitat

	Total Volume Available, m^3	Total Floor Area Available, m^2	Walking Floor Area, m^2	Walkable Volume, m^3	Additional Volume, m^3	Outfitted Volume, m^3
Airlock	4.8	2.4	2.4	4.8	0.0	0.0
Fitness Center	9.8	3.3	3.3	9.8	0.0	0.0
Command & Control Center	6.0	2.7	0.0	0.0	1.0	5.0
Reentry & Sleep Area	18.0	8.0	0.0	0.0	3.0	15.0
Personal Hygiene	3.4	1.5	1.0	1.1	0.8	1.5
Ladder	5.9	1.3	0.0	0.0	4.9	1.0
Propellant	8.4	2.9	0.0	0.0	0.0	8.4
Galley	3.7	0.8	0.0	0.0	3.2	0.5
ECLSS	2.8	1.3	0.0	0.0	0.0	2.8
Avionics	2.7	1.2	0.0	0.0	0.0	2.7
Corridor	9.8	3.3	3.3	9.8	0.0	0.0
Stow	9.7	3.2	0.0	0.0	0.0	9.7
Total	85.0	31.9	10.0	25.5	12.9	46.6

The absolute minimum number of crew is still an important issue for early or caretaker missions. On the moon, with the possibility of short-term rescue (3 days to 55 days, depending on the infrastructure scenario) a small crew of two or three persons could be adequate. Deep-space missions to Mars are another matter because there is no chance of rescue, or return, for at least one year and possibly as long as three or more years. For these missions, it is concluded that a minimum crew size should be five persons.

The justification is on the basis of requiring professional skill levels for each and every critical role for the mission. One crewmember must be the commander, i.e., a group leader with ultimate authority in times of crisis, high-activity time periods, or high-risk operations. This person could have a test-pilot background where rapid, responsible actions are needed based on intense pre-mission training and sound intuitive responses. A second crewmember should be a *bona fide* medical person (i.e., a practitioner, not a surrogate). This is because of the large array of illnesses and/or accidents that could occur with non-negligible probability over the long time period of isolation. A third person is needed who is highly skilled in the engineering sciences, and needs not only diagnostic talents but also technician skills for repairs. A fourth person should be a *bona fide* scientist if exploration is a mission objective (as it has been and is currently planned for all human initiatives). This results in a crew complement of four, but with no backups of any skills. It should be pointed out that cross-training can provide some redundant capability. For example, the scientist could back up the engineer; the engineer could back up the pilot; etc. Four remains a very thin group, however, for three reasons. First, is the scientist to back up the engineer, or rather should he be given intensive medical training. Or, should the pilot be the engineer's backup. Obviously, crew selection procedures would be greatly strained by these potentially conflicting requirements. Furthermore, the group must be selected in large part for their mutual psychosocial compatibilities as well as their professional skills. Second, a key mission success scenario that must be addressed is single-fault tolerance. The "fault" in this case is loss of a crew member. The failure could be physical or mental illness, or any other form of incapacitation of one of the crew members. In this failure mode, the crew is reduced to below its minimum mission success complement. Worse, a crew member in chronic distress will place a significant additional burden on the other crewmembers who must now look after and take care of the fallen colleague. Thus, a fifth member adds a needed degree of redundancy and reserve capacity. Third, it is recognized by psychologists that for small groups that must operate at times on a consensus basis it is important to provide a tie-breaker vote. Thus, odd-numbered crew sizes are preferable to even numbers of crew. For all the reasons given above, five crew is indicated as a minimum size for very long-term exploration missions in space.

4.4 HUMAN FACTORS AND HABITABILITY

In an initial study of human factors considerations, Prof. A. A. Harrison has provided for us a set of 56 recommendations for astronaut selection and habitat module design criteria. A treatise on this subject is given in Appendix A. The recommendations may be summarized as follows:

1) Because psychological and social variables have implications for long range planning including spaceship and lander design, human factors must be taken into account early in the planning process.

2) We recommend applying a full range of interventions (selection, training, and engineering) to align the capabilities of the crewmembers and the demands of the flight.

3) Although the behavioral recommendations in this report are based on naturalistic observations and experimental research, whenever possible they should be tested further before application to the Mars mission.

4) Our overall research effort should address events before, during, and after the flight, and should include mission support personnel, crewmembers' associates, and the public as well as the crewmembers themselves.

5) Each manned spaceflight should serve as a behavioral laboratory for subsequent missions.

6) Goals of selection are: (1) to identify individuals who are technically skilled, highly motivated, and emotionally stable; and (2) to compose groups of individuals who, in the aggregate, are socially compatible and satisfy all of the technical or work requirements.

7) We recommend a five-step selection process beginning with procedures to select individuals and concluding with procedures to select teams.

8) Selection should involve sensible application of a variety of predictors including biographical data, psychological tests, behavioral tests, interviews, and assessment centers. Initial psychological screening should be on the basis of quick and inexpensive methods, and subsequent "cuts" should be based on increasingly elaborate techniques.

9) We recommend candidates who will be in their late thirties or early forties at the time of departure.

10) We recommend both women and men in the crew.

11) We recommend a multinational or ethnically mixed crew.

12) Each crewmember should have two sets of technical skills: those that apply during the flight, and those that apply during surface exploration. In each domain, each crewmember should have both primary and secondary (backup) skills.

13) Each crewmember should be interpersonally as well as technically skilled.

14) Each crewmember should have excellent mental as well as physical health.

15) Crewmembers must be compatible with one another. This is encouraged by value similarity, androgyny, high work and mastery orientation, low competitiveness, and complementary needs.

16) Highly qualified candidates who are not ultimately selected for the first Mars crews should be given assignments that allow them to put their hard-won skills to good use.

17) Although the focus of training will be the Mars crew itself, closely coordinated training should be given to mission control personnel and the crewmembers' immediate families.

18) Experienced astronauts should be well represented within the ranks of mission support personnel.

19) Flight and support personnel should be developed as a single, overall team.

20) Training should occur within environments that bear correspondence to spaceflight environments. Early training should involve special bases in Antarctica, and final training should take place in space itself.

21) Technical training should be supplemented with human relations training.

22) Astronauts need to know how to control fear itself as well as the dangerous conditions that provoke fear.

23) Biofeedback , relaxation, and meditation training will help crewmembers deal with chronic stress.

24) Physically and psychologically demanding training exercises are recommended in the interests of preparedness and morale.

25) Make the spacecraft and the Mars base as spacious as possible. We suggest an allowance of $17m^3$ of private crew quarters space per person.

26) We recommend the use of design techniques that enhance the perceived spaciousness of the spaceship and lander. These include the use of light interior colors, horizontal rather than vertical layouts, and irregular interiors that provide occupants with a range of visual distances and fixation points.

27) The spaceflight and Mars base environments should be visually coherent or legible. All facilities and equipment should be oriented within an unambiguous frame of reference including an apparent vertical.

28) As much as possible, Marsfarers should be given control over the interior configurations of the spaceship and lander.

29) Each astronaut should be assigned individual private sleeping quarters.

30) Multiple personal hygiene facilities are mandatory.

31) Include locations where individual astronauts can occasionally go to be "alone" without retiring to their cabins, and where small groups of astronauts can gather to engage in social activity.

32) At least one area should be large enough to accommodate the entire crew at any one time.

33) All pieces of equipment, restraints, and aids should be adjustable to accommodate anthropometric variations and personal preferences.

34) Lighting should do more than allow for good vision: full spectrum fluorescent lighting should be used to alleviate minor depression; area lighting should be available, and astronauts should have control over lighting intensity.

35) We recommend multiple windows.

36) A telescope should be available to enhance the visual link with Earth.

37) Pictures and other graphic designs are recommended because they offer illusions of depth and tend to diminish the negative impact of minimal interior spaces.

38) Individual personal cassette recorders are recommended.

39) We recommend against uninterrupted video surveillance.

40) Personal communications systems are desirable for communicating with other people aboard the spacecraft or at the Mars base.

41) Communication with mission control personnel and with family and friends on Earth should involve full duplex audio-video systems.

42) Advances in electronics and in liquid crystal technology should make it possible to provide each astronaut with a personal multipurpose unit that functions as a communications device, a computer, and word processor for both work and recreational purposes, and a display device for electronic microfiche and video fare.

43) Balancing cost and social psychological factors we recommend a crew of five members.

44) To achieve a good balance between decision quality and member acceptance, we recommend a mixture of autocratic and democratic decision making procedures.

45) The commander and crew should have considerable latitude to make decisions and take appropriate action.

46) Workloads must be capable of adjustment so that crewmembers are neither bored nor overworked but instead are offered an appropriate degree of challenge.

47) Desirable and undesirable tasks should be distributed fairly so that no one draws only undesirable tasks.

48) Following the work of Hackman and others, we recommend that work assignments be identifiable and whole, work assignments draw on a range of abilities and skills, work assignments have a real and demonstrable effect on mission success, individual astronauts be granted as much autonomy as possible, and astronauts be given regular feedback regarding their level of performance.

49) Planners should explore Kahn's work module forms of organization: mission requirements are first broken down into time-task units, astronauts are allowed to qualify for various tasks, and individual astronauts are allowed to piece together patterns or mosaics of assignments.

50) All crewmembers should have the opportunity to serve on Mars.

51) Group norms should be well established before the crew's departure.

52) We recommend a psychological "buddy system" in which each crewmember assumes special responsibility for the welfare and morale of another specific crewmember.

53) Individual members of the ground support team should assume special responsibility for the welfare and morale of individual crewmembers.

54) Establish a support team to monitor and boost the astronauts' morale.

55) Individual crewmembers should serve as mentors to trainees for subsequent missions.

56) Support services must be available to the astronauts' families.

5.0 PROPULSION

A number of different studies were performed in the area of propulsion to identify mission enabling and possible enhancing technology. Each of the mission propulsion systems were designed and specified, and any required advances in engine and tank design were identified. In addition, studies of advanced technology, including lunar produced propellant use and nuclear propulsion, were performed.

5.1 INTERPLANETARY TRANSFER

Propulsion systems for both the lunar and Mars interplanetary spacecraft were specified, defining propellant types, cryogenic propellant boiloff rates, propellant margins, required engine thrust levels, engine I_{sp}s, and tankage factors. I_{sp} values and boiloff rates were then varied to determine their effect on Trans-Mars Injection System (TMIS) mass, Trans-Earth Injection System (TEIS) mass, and overall vehicle mass.

5.1.1 Propellants

Both the lunar and Mars missions use a cryogenic LH_2/LOX propellant for the main propulsion systems. Storable bipropellant is used for the Reaction Control System (RCS), artificial-g spacecraft spinup/ spindown, and MAV (ascent) and MDV (descent) propulsion systems.

Assumed boiloff rates and propellant margins are shown in Table 5.1.1-1 for all stages of the five previously described missions. The boiloff rates are specified for Low Earth Orbit (LEO), interplanetary flight, and the planetary vicinity (either Mars or the moon). Because of the significantly longer storage times associated with TEIS (as compared to TMIS), better insulated tanks are assumed, resulting in lower boiloff rates.

Propellant margins are specified for ΔV, I_{sp}, and unusable propellant (bulk) in all IMLEO calculations. In addition to these margins, the lunar cases assume an additional ullage (gaseous propellant) factor of 5%.

Table 5.1.1-1 Propellant Boiloff Rates and Margins

Mission Stage	Boiloff(%/mo:LEO, IP, Planetary)	Margins (%:ΔV, I_{sp}, bulk)
Phobos Gateway:		
TMIS	3.0, N/A, N/A	1.0, 1.0, 1.0
TEIS	0.15, 0.3, 0.065	0.0, 0.0, 2.0
MAV, MDV	N/A	3.0, 0.0, 0.0
ECCV	N/A	0.0, 0.0, 2.0
Lunar Gateway:		
all	0.0, 0,0, 0.0	0.0, 2.0, 1.5 (+ 5% ullage)
Lunar Evolution:		
all	3.73, 3.73, 3.73	0.0, 2.0, 1.5 (+ 5% ullage)
Mars Expedition:		
TMIS (crew)	3.0, N/A, N/A	1.0, 1.0, 1.0
TMIS (cargo)	3.0, N/A, N/A	1.0, 1.0, 1.0
TEIS	0.55, 1.0, 0.33	0.0, 0.0, 2.0
MAV	N/A	3.0, 0.0, 0.0
Mars Evolution:		
TMIS (crew)	3.0, N/A, N/A	1.0, 1.0, 1.0
TMIS (cargo)	3.0, N/A, N/A	1.0, 1.0, 1.0
TEIS (crew)	0.33, 0.6, 0.15	0.0, 0.0, 2.0
MCSV	0.0, 0.0, 1.0	1.0, 0.0, 0.0

A study was performed to determine the effect of varying boiloff rates on the overall vehicle masses in the Mars Evolution and Mars Expedition cases. Table 5.1.1-2 shows the assumed boiloff rates on the Mars missions for conservative (high boiloff), nominal (medium boiloff), and advanced (low boiloff) technology.

Table 5.1.1-2 Assumed Boiloff Rates

Mission Stage	Low (%/mo)	Medium (%/mo)	High (%/mo)
LEO (TEIS)	0.150	0.33	0.55
Interplanetary (TEIS, crew)	0.300	0.60	1.00
Interplanetary (TEIS, cargo)	0.100	0.20	0.40
Mars (TEIS)	0.065	0.15	0.33
LEO (TMIS)	0.000	3.00	5.00

Table 5.1.1-3 Baseline Mission Masses

Mission Stage	Mass (t)
Mars Expedition:	
Total (Cargo & Human)	824.90
Total (Cargo)	208.97
Total (Human)	615.93
TMI propellant (Cargo)	118.75
TMI propellant (Human)	380.79
TEI propellant	92.21
Mars Evolution (Oppositions):	
Total	689.86
TMI propellant	418.55
TEI propellant	64.44
Mars Evolution (Conjunction):	
Total	741.87
TMI propellant	437.97
TEI propellant	34.24

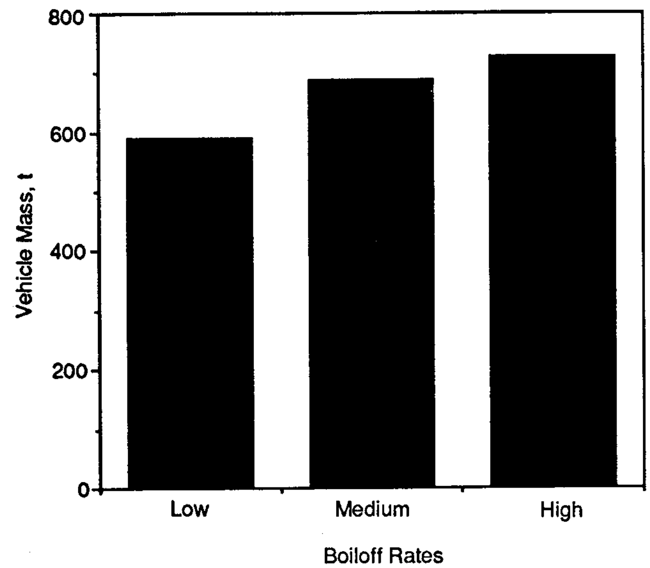


Figure 5.1.1-2 Mars Evolution (Opposition) Total Masses for Varying Boiloff Rates

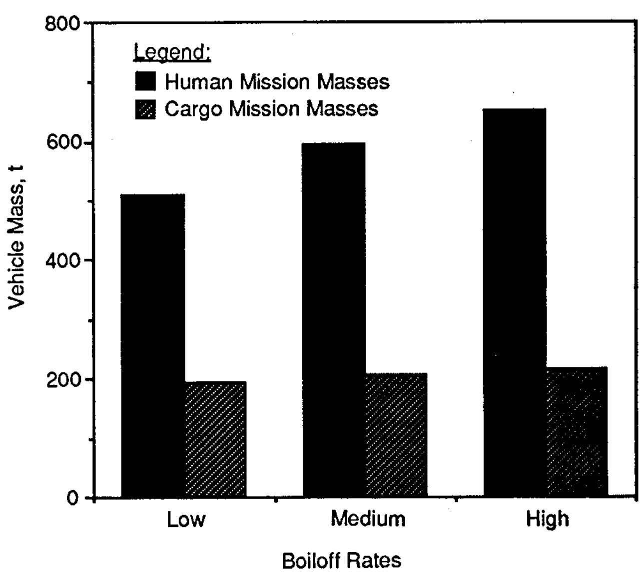


Figure 5.1.1-1 Mars Expedition Total Masses for Varying Boiloff Rates

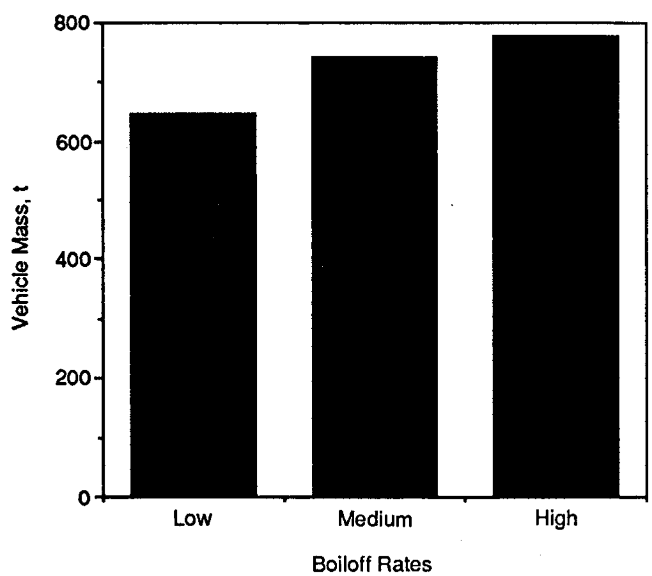


Figure 5.1.1-3 Mars Evolution (Conjunction) Total Masses for Varying Boiloff Rates

For the Mars Expedition and the Mars Evolution cases (both the conjunction and opposition class cases), the boiloff rates were varied to determine the effect of improved technology level on overall vehicle mass. The baseline mission (from Table 5.1.1-1 data) masses are displayed in Table 5.1.1-3. Figures 5.1.1-1 through 5.1.1-3 show the effect of higher boiloff rates on total spacecraft mass.

In the Mars Expedition case, an advance in technology from nominal (current state of the art) to advanced allows a 9.6% mass advantage for the human mission and a 7.2% mass advantage for the cargo mission. In the Mars Evolution opposition class mission, a similar technology advance translates to a 17.1% mass advantage. In the conjunction mission, the technology advance gives a 14.8% mass advantage. Clearly, there is potential for a substantial decrease in overall mass through research and development of low-boiloff tanks.

5.1.2 Engines

Current state-of-the-art engines are not optimized for the lunar and Mars mission thrust and I_{sp} requirements. To minimize development costs, however, old-technology engines could be modified to meet the new engine requirements. These engine baselines are summarized in Table 5.1.2-1. Figure 5.1.2-1 through 5.1.2-4 give details and illustrations of some of the tabulated engines.

Table 5.1.2-1 Interplanetary Engines

Mission Stage	Propellant	Engine	I_{sp} (sec)	Thrust (klbf)
Phobos Gateway: TMIS	LH_2/LOX	SSME der.	478	470.5
TEIS	LH_2/LOX	RL-10-X1	470	22.0
Lunar Gateway: TLIS, LAV, LTV	LH_2/LOX	RL-10-X1	470	22.0
Lunar Evolution: LCV, LPV	LH_2/LOX	Adv. cryo engine	481	15.0
Mars Expedition: TMIS	LH_2/LOX	SSME der.	471	470.5
TEIS, MOO, MOC	LH_2/LOX	RL10-X1	470	22.0
DSM, MCC	MMH/NTO	Shuttle OMS	316	6.0
RCS ETM	MMH/NTO	Marquardt R-40B	310	1.0
RCS MTE	MMH/NTO	Marquardt R-4D	311	0.1
Mars Evolution: TMIS	LH_2/LOX	SSME der.	471	470.5
TEIS, MOO, MCC	LH_2/LOX	advanced OTV	480	7.5
Art-g	MMH/NTO	Marquardt R-40B	310	1.0
RCS	MMH/NTO	Marquardt R-4D	311	0.1

Isp	4.69 kN-s/kg (478 s)
Thrust	2,093 kN (470.5 klb $_f$)
Mass	3,175 kg
Dimensions	
— Nozzle Exit Diameter	7.29 m (287 in)
— Total Length	12.1 m (478 in)

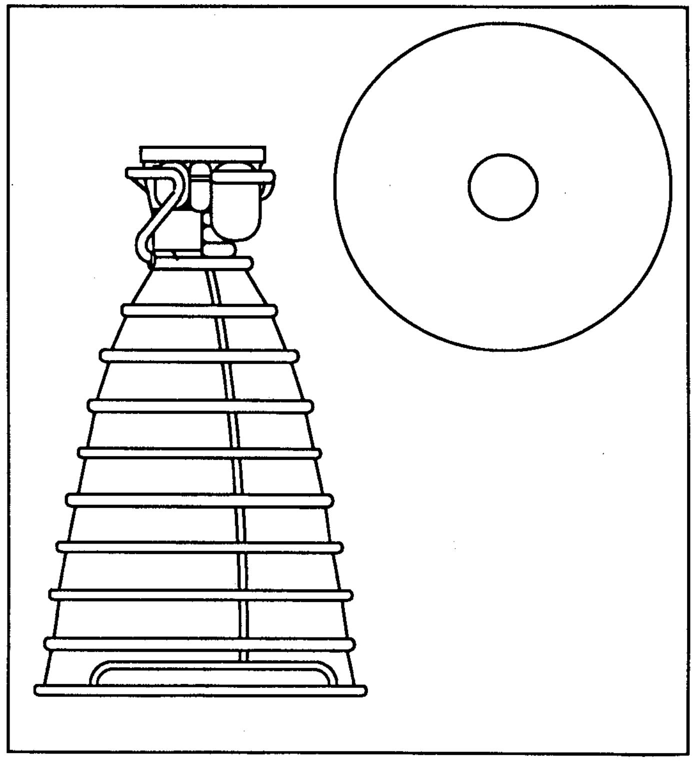


Figure 5.1.2-1 SSME Derived Engine (for TMI)

Isp	4.61 kN-s/kg (470 s)
Thrust	89.0 kN (22.0 klb $_f$)
Mass	272 kg
Dimensions	
— Nozzle Exit Diameter	2.29 m (90 in)
— Total Length	4.42 m (174 in)
— Area Ratio	400:1

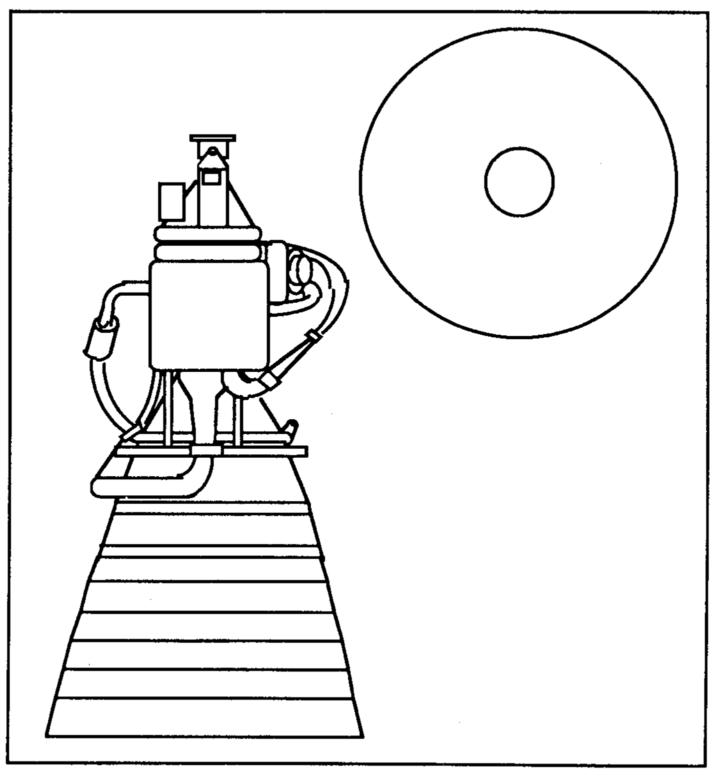
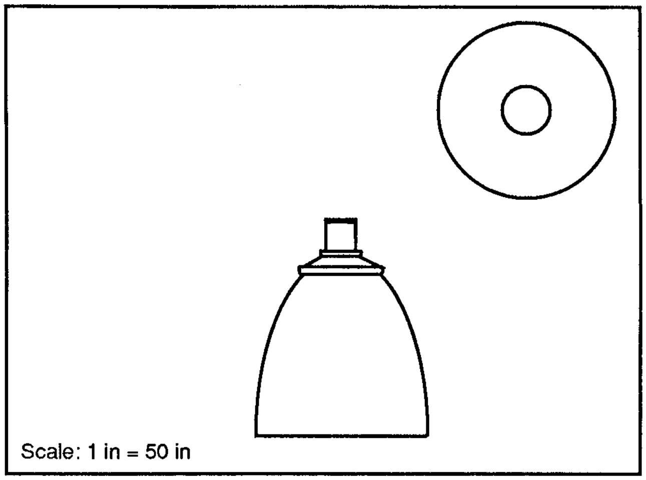
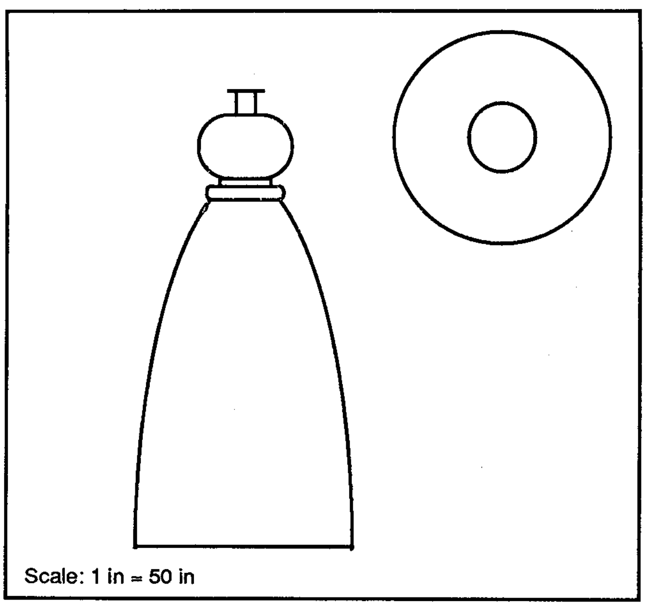


Figure 5.1.2-2 RL10-X1 Engine



Isp	3.10 kN-s/kg (316 s)
Thrust	26.7 kN (6.0 klb_f)
Mass	134 kg
Dimensions	
— Nozzle Exit Diameter	1.14 m (45 in)
— Total Length	1.45 m (57 in)

Figure 5.1.2-3 Shuttle Orbital Maneuvering Engine (OME)



Isp	4.71 kN-s/kg (480 s)
Thrust	33.4 kN (7.51 klb_f)
Mass	163 kg
Dimensions	
— Nozzle Exit Diameter	1.4 m (55 in)
— Total Length	3.05 m (120 in)

Figure 5.1.2-4 Advanced Cryogenic (OTV) Engine (RS-44 class)

The most important engine system, from a performance standpoint, is the Trans-Mars Injection System (TMIS) engine. Three possibilities are available TMIS engines:

1) One very large engine in the 300-500 klb_f thrust class.
2) An engine in the 75-100 klb_f thrust class. These would be used in one of two possible approaches:
   a) Clustered (3 to 6 engines) to achieve the thrust of the engine in case 1 above.
   b) As a single engine, combined with a multi-burn strategy where gravity losses are avoided by short burns (@ 5-15 minutes) at successive perigee passes in Earth orbit. From 3 to 6 orbits might be needed to effect escape. This strategy has the advantage that the resulting highly elliptical orbits permit plane changes to adjust between the original LEO orbital plane and the desired escape asymptote for proper trans-Mars injection.
3) Engines in the 20-30 klb_f thrust class. These would be clustered as 4 to 8 engines and used in the same strategy as 2b above.

An additional consideration is the possible use of this engine as an upper stage for a heavy lift launch vehicle. In the case of the Shuttle-Z concept (see Section 5.3), only option 1 or 2a would be viable. Options 2b or 3 would not provide the level of thrust necessary to permit the Shuttle-Z to obtain the type of performance required to satisfy the very heavy lift capabilities for Mars mission needs.

Another consideration is potential commonality with lunar mission engine needs. Option 3 is quite compatible. However, Mars missions do not need the large throttling range that will be needed for lunar landings. In addition, if lunar vehicle are re-usable, a requirement exists for very long life and a large numbers of restarts. In option 3, the total operating life of the engine need only be one week, and the number of restarts is a maximum of 9.

A single large thrust, SSME-derived engine is baselined (Ref Table 5.1.2-1) for TMIS mission examples, but from the discussion above it is seen that multi-burn strategies using clustered engines is an attractive alternative.

A performance study was done to determine the effect of varying I_{sp} for interplanetary flight. Assuming a constant I_{sp} throughout the mission, the TMIS mass, TEIS mass, and total vehicle mass were analyzed over a range of I_{sp} values for the Mars Expedition and Evolution cases. The results are shown in Figures 5.1.2-5 through 5.1.2-7.

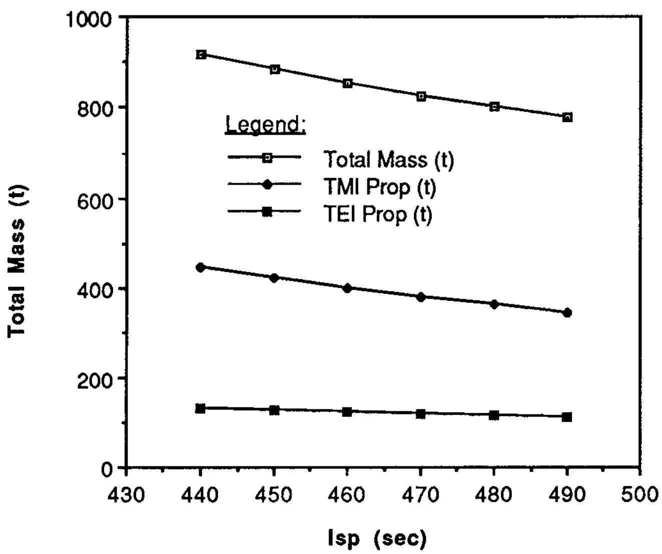


Figure 5.1.2-5 Mars Expedition I_{sp} Study

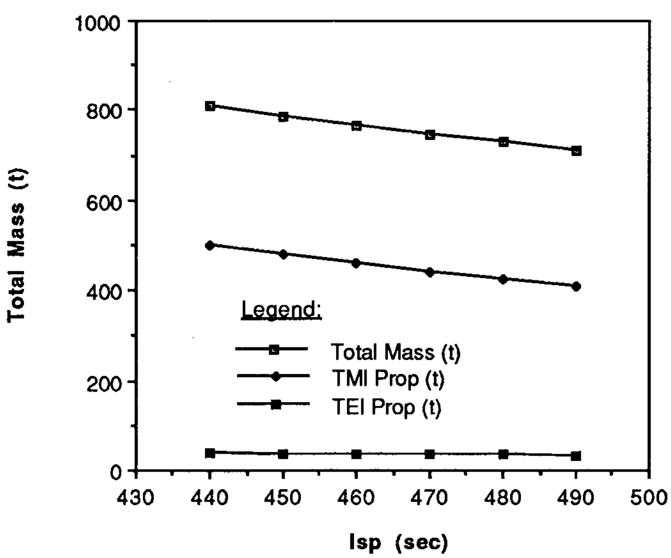


Figure 5.1.2-6 Mars Evolution (Conjunction Class) I_{sp} Study

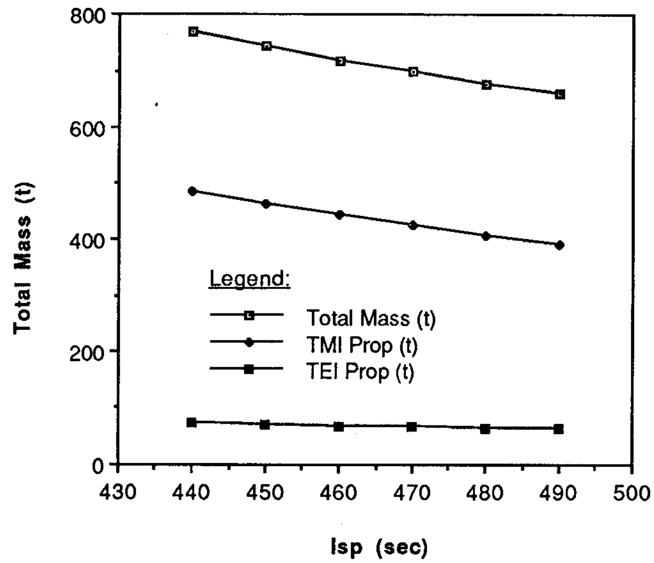


Figure 5.1.2-7 Mars Evolution (Opposition Class) I_{sp} Study

For the Mars Expedition case, a ten second increase in I_{sp} translates into a 3.5% decrease in overall vehicle mass. This total decrease consists of a 4% decrease in human mission mass and a 2% decrease in cargo mission mass. Most of the advantage is realized in the TMI propellant mass savings, where a 5.5% decrease in human mission TMIS mass and a 3.3% decrease in cargo mission mass was realized. TEIS mass dropped 3.5%.

In the Mars Evolution conjunction mission, a ten second increase in I_{sp} translates to a 2.7% decrease in overall vehicle mass. TMIS mass decreased by 4.2%, while TEIS mass dropped 1.8%. In the opposition case, the same I_{sp} change resulted in a mass savings of 3.2%, with a 4.6% change in TMIS mass and a 2.8% decrease in TEIS mass.

5.1.3 Tanks

Propellant tank masses were determined from allocated tankage factors (% of propellant mass). Table 5.1.3-1 shows the assumed tankage factors for the different mission stages.

Table 5.1.3-1 Allocated Tankage Factors

Mission Stage	Tankage Factor (%)
Phobos Gateway: TMIS	10
TEIS	15
Mars Expedition: TMIS	7
TEIS	15
RCS	5
Mars Evolution: TMIS	7
TEIS	15
RCS	5

5.1.4 Conclusions

From the data in this section, it appears to be as mass-beneficial to develop low boil-off tanks as it is to invest in improving existing engine I_{sp}. Substantial mass advantages can be obtained by reducing cryogenic boiloff.

5.2 PLANETARY DESCENT/ASCENT

The propulsion systems for both the lunar and Mars ascent/descent stages were allocated, defining propellant types, propellant margins, required engine thrust levels, engine I_{sp}s, and tankage factors.

5.2.1 Propellants

Cryogenic propellants were baselined on ascent/descent for the lunar missions and for the Mars Evolution missions. Storable bipropellants were baselined on ascent and descent for the Mars Expedition and Phobos Gateway missions.

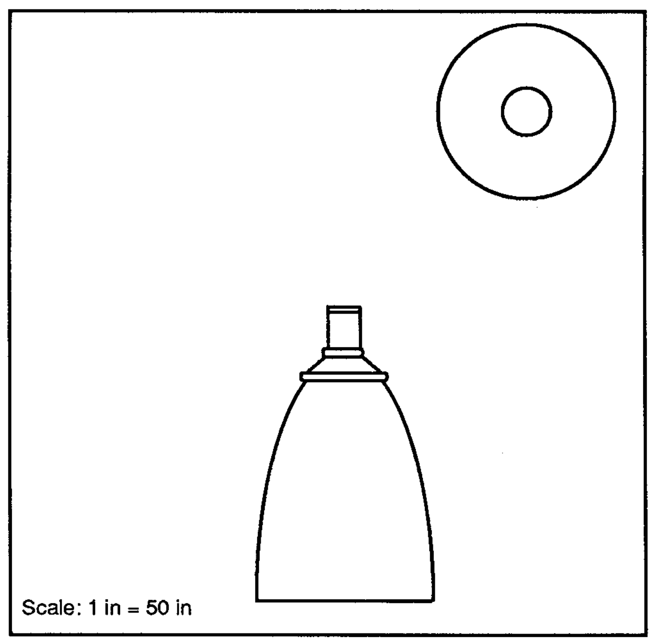
Propellant boiloff rates and margins for the ascent/ descent portion of these missions are given in Table 5.1.1-1.

5.2.2 Engines

To save on engine development, the ascent/descent engines for the Gateway missions and the Mars Expedition mission were selected as uprated versions of currently available engines. The Evolution cases both use identical advanced, near-term cryogenic engines. These engines are summarized in Table 5.2.2-1 and shown in Figures 5.1.2-2, 5.2.2-1, and 5.2.2-2.

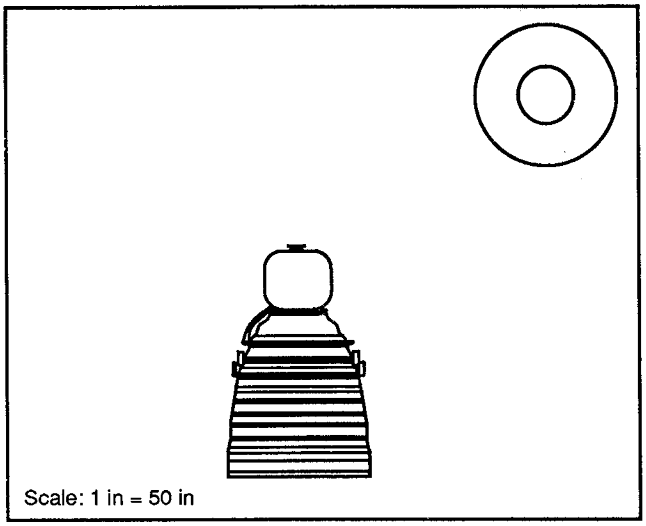
Table 5.2.2-1 Ascent/Decent Engines

Mission Stage	Propellant	Engine	I_{sp} (sec)	Thrust (klbf)
Phobos Gateway: MDV, MAV	MMH/NTO	Pump-fed OME	334	9.4
Lunar Gateway: LAV, LDV	LH_2/LOX	RL-10-X1	470	22.0
Lunar Evolution: LCSV, LCL, LPT	LH_2/LOX	Adv. cryo engine	463	15.0
Mars Expedition: MDV, MAV	MMH/NTO	Pump-fed OME	334	9.4
Mars Evolution: MCSV	LH_2/LOX	Adv. cryo engine	463	1.0 5.0



Isp	3.28 kN-s/kg(334 s)
Thrust	42.0 kN (9.44 klb_f)
Mass	141 kg
Dimensions	
— Nozzle Exit Diameter	1.12 m (44 in)
— Total Length	1.96 m (77 in)

Figure 5.2.2-1 Pump-fed Orbital Maneuvering Engine (OME)



Isp	4..54 kN-s/kg (463 s)
Thrust	66.7 kN (9.44 klb $_f$)
Mass	342 kg
Dimensions	
— Nozzle Exit Diameter	0.94 m (44 in)
— Total Length	1.52 m (77 in)
— Area Ratio	225:1

Figure 5.2.2-2 Advanced Cryogenic Engine (RS-44 Class)

5.3 EARTH-TO-ORBIT (ETO) TRANSPORTATION

For any of the described missions to occur, the required hardware must first be placed into LEO. To accomplish this, a heavy-lift launch vehicle (HLLV) is required (possibilities for HLLV include modified shuttle, or Shuttle-C and Shuttle-Z concepts.

5.3.1 ETO Manifests

Table 5.3.1-1 shows a sample ETO manifest to place the required Mars expedition hardware into LEO. This scenario assumes a HLLV capable of lifting 140 t to LEO. Figure 5.3.1-1 shows this described ETO process. Similar manifests were performed under this contract for the other mission scenarios, and can be found in earlier studies. (Reference tables in Section 1.0).

Table 5.3.1-1 Example Manifest using HLLV

Mars Cargo Vehicle (MCV)	
A.) HLLV Launch of all-up MCV (wet)	77.6 t
B.) HLLV Launch of TMIS and propellant Dock TMIS with MCV and execute TMI	131.4 t
Mars Piloted Vehicle (MPV)	
C.) HLLV Launch of all-up piloted vehicle (unmanned), with TEIS propellant off-loaded	105.1 t
D.) HLLV Launch of 92 t TEIS propellant	92.0 t
E-G.) HLLV Launch of three TMIS stages, (wet, or with prop transfer) On-orbit docking of three TMIS Stages Dock TMIS stages to vehicle	3 x 140 t
H.) STS Launch of crew Board Mars Spaceship. Checkout.	
I.) Initiate mission with TMI burn.	

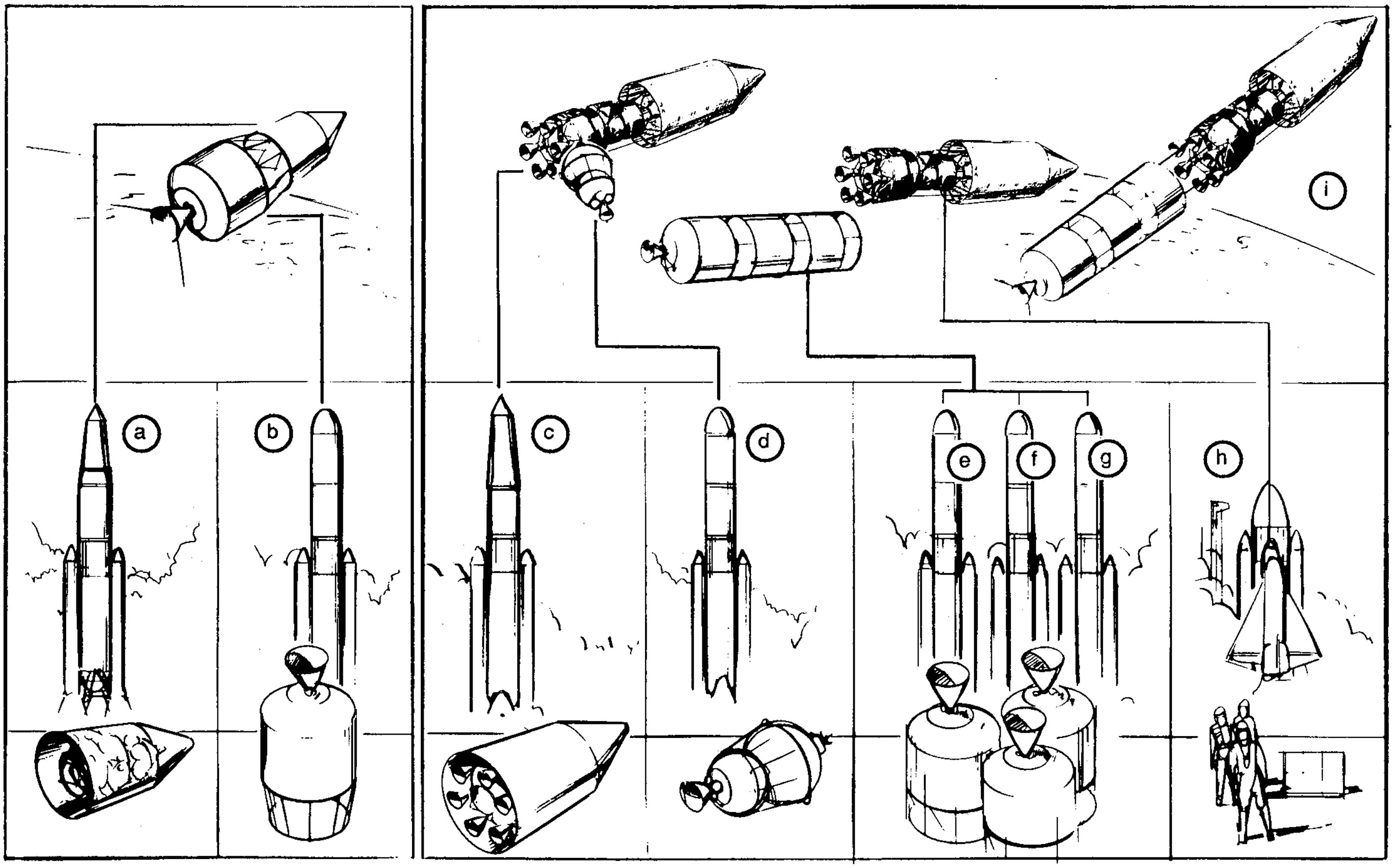


Figure 5.3.1-1 ETO Sequence for Baseline Mars Expedition Case Study

As can be observed from the above sequence, the Mars Expedition case can be performed with 7 HLLV launches and a single Shuttle mission.

5.3.2 Shuttle-Z

A Shuttle-Z can deliver 139.7 metric tonnes of useful payload to a nominal Space Station Freedom orbit of 407 km, and in roughly half-a-dozen launches can lift a complete manned Mars mission ensemble into orbit. The key to Shuttle-Z is its use of a third stage which is identical to and can be reused as the stage used to boost manned Mars missions out of Earth orbit. The reference Shuttle-Z consists of two Advanced Solid Rocket Motors (ASRM), an external tank, a side-mount cargo element that houses the three Space Shuttle Main Engines (SSME), a new third stage powered by a high expansion ratio SSME, and a 12.2 meter diameter by 15.2 meter barrel section payload fairing. This section of the study report documents how the Shuttle-Z evolved, and in the process shows several intermediate and alternative vehicle designs and their performance. The baseline Shuttle-Z's flight profile, mass statement, and cost to develop and build the first vehicle are estimated. External tank modifications and vehicle processing requirements for Kennedy Space Center are also provided.

5.3.2.1 Magnum Vehicle—Shuttle-Z's lineage goes back to late 1988 when NASA asked Martin Marietta Astronautics Group to present heavy lift booster concepts that used the National Space Transportation System's (Space Shuttle's) components. The booster had to be capable of lifting 227 to 680 metric tonnes of payload into a 407 by 407 km orbit inclined at 28.5 degrees. These ambitious lift requirements might enable NASA to launch an entire manned Mars vehicle at once.

Martin Marietta investigated two classes of ultra-large vehicles, both known as Magnum boosters. The Super Magnum family, as shown in Figures 5.3.2.1-1 through 5.3.2.1-3, has lift capabilities ranging from 575 to 666 tonnes and requires 12 SRBs for stage-1, 16 SSMEs burning in parallel for stage-2, and 4 SSMEs on stage-3. This family has a 13.7 meter core vehicle diameter and ranges in height from 113 to 142 meters. The Mini-Magnum family, shown in Figures 5.3.2.1-4 and 5.3.2.1-5, has roughly half the Super Magnum's payload capability (200 t class). This vehicle uses 6 SRBs for stage-1, seven SSMEs that start two minutes into the flight as the SRBs thrust begins to taper off for stage-2, and a single SSME for stage-3. Mini Magnums have a 10 meter core diameter and are 90 or 107 meters tall for the tanker or cargo version, respectively.

Obviously these concepts represent a large departure from the current Shuttle. In addition, the cost advantages of using already-tested STS hardware is diluted by the need for new core stages, launch platforms, and assembly facilities. This led to a more focused effort on designs of lesser capability but which had greater commonality with the Shuttle. This class of vehicle was initially named Shuttle-CZ (C for cargo, Z for NASA/Code-Z), but later renamed Shuttle-Z to avoid confusion with the Shuttle-C family of designs.

Payload	575 t
Shroud	None
GLOM	10589 t
Height	112.7 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	12 SRBs
Stage-2	13.7-m Diameter
	16 SSMEs
	2375 t Propellant
	152 t Dry
Stage-3	10-m Diameter
	4 SSMEs
	1294 t Propellant
	67 t Dry

Stage-2 SSMEs Ignited before Liftoff

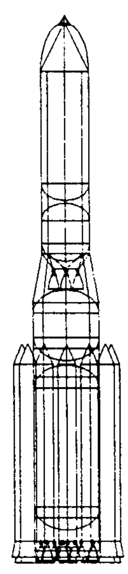


Figure 5.3.2.1-1 575 t Super Magnum Tanker

Payload	583 t
Shroud	12.2-m Dia x 24.4 m
GLOM	10538 t
Height	130.5 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	12 SRBs
Stage-2	13.7-m Diameter
	16 SSMEs
	2375 t Propellant
	152 t Dry
Stage-3	10-m Diameter
	4 SSMEs
	684 t Propellant
	42 t Dry

Stage-2 SSMEs Ignited before Liftoff

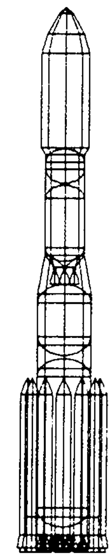


Figure 5.3.2.1-2 583 t Super Magnum

Payload	666 t
Shroud	12.2-m Dia x 24.4 m
GLOM	11215 t
Height	142 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	12 SRBs
Stage-2	20.4-m Diameter
	16 SSMEs
	2997 t Propellant
	178 t Dry
Stage-3	10-m Diameter
	4 SSMEs
	684 t Propellant
	42 t Dry
Stage-4	13.6 t Gross
	1 RL-10
	2 t Dry

Stage-2 SSMEs Ignited before Liftoff

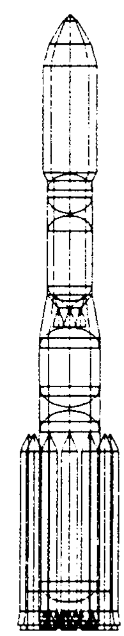


Figure 5.3.2.1-3 666 t 4-Stage Super Magnum

Payload	236 t
Shroud	None
GLOM	4625 t
Height	90 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	6 SRBs
Stage-2	8.4-m Diameter
	7 SSMEs
	723 t Propellant
	52 t Dry
Stage-3	8.4-m Diameter
	1 SSMEs (1000:1)
	475 t Propellant
	24 t Dry

Stage-2 SSMEs Ignited at SRB Separation

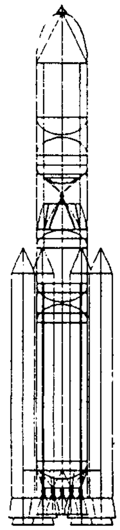


Figure 5.3.2.1-4 236 t Mini Magnum Tanker

Payload	245 t
Shroud	12.2-m Dia x 24.4 m
GLOM	4620 t
Height	106.6 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	6 SRBs
Stage-2	8.4-m Diameter
	7 SSMEs
	723 t Propellant
	52 t Dry
Stage-3	8.4-m Diameter
	1 SSMEs (1000:1)
	214 t Propellant
	13 t Dry

Stage-2 SSMEs Ignited at SRB Separation

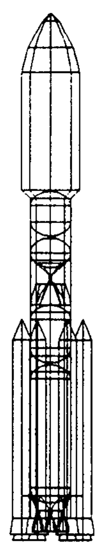


Figure 5.3.2.1-5 245 t Mini Magnum

5.3.2.2 Shuttle-Z—The Shuttle-Z concepts are based on the use of only two SRBs, 8.4 meter diameter external tanks, and side-mounted SSMEs to retain compatibility with the current launch pad flame trenches, mobile platform's carrying capability, and SRB attach points. These vehicles are all in-line, three-stage configurations with varying numbers of stage-2 SSMEs and varying degrees of propellant tank stretching. (Note: Stage-1 denotes SRMs; Stage-2 is the ET/SSME propulsion and Stage-3 indicates the new upper stage).

The smallest in this family is the 120 t payload capacity Shuttle-Z shown in Figure 5.3.2.2-1. It has three SSMEs that run at 109% full-rated thrust for stage-2 and a single SSME with a 1000:1 expansion ratio sliding nozzle for stage-3. Stage-2 has the same tank capacity as the current external tank (723 t), while stage-3's capacity is 180 t. Payload dimensions for these vehicles, as well as for the Magnum vehicles, are 12.2 meters in diameter and 24.4 meters in barrel section length.

By adding a fourth SSME to the boattail, gravity losses are reduced and 140 tonnes can be lifted into orbit. Figure 5.3.2.2-2 shows this configuration. All other elements of the design are the same as the 120 t configuration except that the four SSMEs on stage-2 are run at 100% full thrust rating instead of 109%.

The final Z vehicle that was considered uses two engine propulsion/avionics (PA) modules, each with four SSMEs running at 100%. This configuration can lift 175 t but requires stretching both stage-2 and stage-3 tanks so they can hold 1311 and 204.5 t of propellant, respectively. Figure 5.3.2.2-3 shows how this tank stretching gives the vehicle a questionable height-to-diameter ratio when considering structural efficiency and bending loads during atmospheric ascent.

Payload	120.7 t
Shroud	12.2-m Dia x 24.4 m
GLOM	2220 t
Height	106 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	2 SRBs
Stage-2	8.4-m Diameter
	3 SSMEs at 109%
	723 t Propellant
	46.4 t Dry
Stage-3	8.4-m Diameter
	1 SSME (1000:1)
	180 t Propellant
	11.7 t Dry

Stage-2 SSMEs Ignited before Liftoff

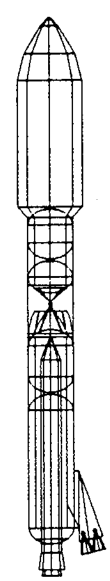


Figure 5.3.2.2-1 120 t Shuttle-Z

Payload	139.7 t
Shroud	12.2-m Dia x 24.4 m
GLOM	2243 t
Height	106 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	2 SRBs
Stage-2	8.4-m Diameter
	4 SSMEs at 100%
	723 t Propellant
	49.6 t Dry
Stage-3	8.4-m Diameter
	1 SSME (1000:1)
	180 t Propellant
	11.7 t Dry

Stage-2 SSMEs Ignited before Liftoff

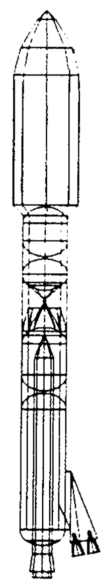


Figure 5.3.2.2-2 140 t Shuttle-Z

Payload	175 t
Shroud	12.2-m Dia x 24.4 m
GLOM	2970 t
Height	136 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	2 SRBs
Stage-2	8.4-m Diameter
	8 SSMEs at 100%
	1311 t Propellant
	95 t Dry
Stage-3	8.4-m Diameter
	1 SSME (1000:1)
	204.5 t Propellant
	45.0 t Dry

Stage-2 SSMEs Ignited before Liftoff

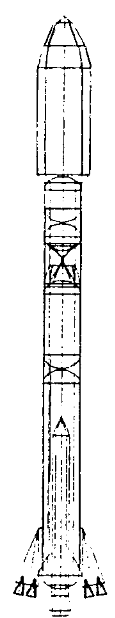


Figure 5.3.2.2-3 175 t Shuttle-Z

5.3.2.3 Reference Shuttle-Z—Interest generated by the 120 and 140 t Shuttle-Zs led directly to the reference Shuttle-Z shown in Figure 5.3.2.3-1. By placing stage-3 and the cargo on a side-mounted pod, much like the Shuttle-C, the potential exists for greater commonality with ground facilities, the current external tank, and payload servicing equipment. To compensate for the increased drag of the side mounted cargo carrier, Advanced Solid Rocket Motors (ASRM) which have 2,611 million Newton-seconds (17.6%) more impulse than the SRBs, are employed. The payload fairing barrel length is also shortened because it was found that the Mars vehicles could be packaged in less than 15 meters of length.

Payload	124.4 t
Shroud	12.2-m Dia x 24.4 m
GLOM	2266.5 t
Height	70 m
Orbit	407 x 407 km
	28.5 from KSC
Stage-1	2 ASRMs
Stage-2	8.4-m Diameter
	3 SSMEs at 109%
	723 t Propellant
	53.8 t Dry
Stage-3	7.4-m Diameter
	1 SSME (725:1)
	127 t Propellant
	13 t Dry

Stage-2 SSMEs Ignited before Liftoff

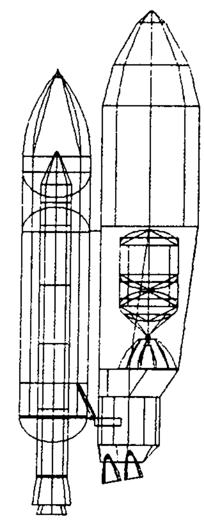


Figure 5.3.2.3-1 124 t Shuttle-Z

Table 5.3.2.3-1 is Martin's mass statement for the baseline Shuttle-Z with comparison data summarized from a study by Martin Marietta's Manned Space Systems Company (MSS). The third column is Marshall Space Flight Center's (MSFC) mass statement for their independent sizing of a Shuttle-Z configuration.

One significant difference between the Martin Marietta and MSFC configurations is the third stage. MSFC selected six Advanced Space Engines rated at 153.6 kN each, reduced the propellant load from 127 to 79.4 t, and operated the SSME core engines at 100% maximum rated thrust (compared to the 109% assumed by the Martin Marietta study). By adding a fourth SSME to the boattail, the lift capability of the Martin Marietta Shuttle-Z increases to 139 t (Fig. 5.3.2.3-2).

Table 5.3.2.3-1 Shuttle-Z Mass Breakdown and Comparison

	MMAG (t)	MSS (t)	MSFC (t)
External Tank	29.9		35.5
(propellant)	(722.9)		(723.5)
ASRMs	1209.2		1246.5
Side Pod	23.9	37.9	24.9
Strongback	13.2	29.5 (inc. boattail)	
Boattail	9.5		
Fairing	1.1	8.3	
Stage-3	13.0	12.2	14.5
SSME	3.6	5.3	
Tank	8.9	3.3	
Avionics	0.5	0.3	
Other	0.0	3.2	
(Propellant)	(127.0)		(79.4)
Payload Shroud	15.7		26.7
Payload	124.3		87.9
Total Dry Mass	1416.0		1436.0
Gross Lift-Off	2266.0		2239.0

Payload	138.7 t
Shroud	12.2-m Dia x 15.8 m
GLOM	2284.0 t
Height	70 m
Orbit	407 x 407 km 28.5 from KSC
Stage-1	2 ASRMs
Stage-2	8.4-m Diameter 4 SSMEs at 100% 723 t Propellant 57.0 t Dry
Stage-3	7.4-m Diameter 1 SSME (725:1) 127 t Propellant 13 t Dry

Stage-2 SSMEs Ignited before Liftoff

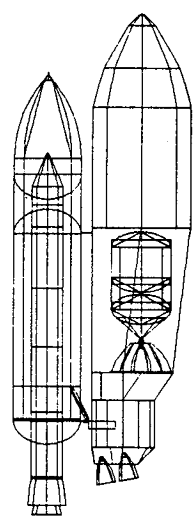


Figure 5.3.2.3-2 139 t Shuttle-Z

5.3.2.4 Mass Estimates—The external tank will require strengthening in several areas to be usable for the Shuttle-Z. Table 5.3.2.4-1 summarizes the mass adjustments associated with each modified item. The total mass is estimated to increase 1658.3 kg (20% contingencies included). This represents a 5% increase in the current mass of the external tank .

Table 5.3.2.4-1 Required External Tank Modifications

	Δ Mass (kg)
Total Increased Mass (kg)	1658.3
LOX Tank	
Aft Ogive Gores	70.8
Barrel Panels	37.2
Intertank	
Main Frame	89.1
Crossbeam	35.9
Forward SRB Fitting	88.5
Hydrogen Tank	
1129 Frame & Bipod B/U Fittings	83.7
Barrel Panels	501.7
2058 Frame & Vertical Structure B/U Fittings	100.3
Aft Dome	15.3
Forward Attach Hardware	
Bipod	150.3
Spindles	36.3
Aft Attach Hardware	
Thrust Strut End Fitting	32.9
Vertical Struts	98.3
Diagonal Strut	38.3
Upper Aft SRB Fitting	3.3
Contingency (20%)	276.4

5.3.2.5 Ground Facilities Requirements—Ground facilities at Kennedy Space Center can support the reference Shuttle-Z with some modified and new buildings. A new integration building with 50 foot crane height may be needed to integrate the elements of the side-mount module, which includes the payload, stage-3, payload shroud, and boattail with engines. This module then becomes a direct replacement for the Orbiter in a Shuttle stack and is rolled on a wheeled strongback to the Vehicle Assembly Building for integration with the external tank and solid motors. Weight limitations of the crawler-transporter and mobile launch platform are not exceeded by the Shuttle-Z's 1416 t roll-out mass or 2266 t lift-off mass. Even the crawler-transporter, which has the smallest margin, retains a 370 t weight margin.

5.3.2.6 Performance Analysis—The ascent profile for the reference Shuttle-Z is shown in Figures 5.3.2.6-1 through 5.3.2.6-4. Figure 5.3.2.6-1 shows the ASRM boost phase trajectory needed to allow sufficient fuel burn-off to obtain a reasonable SSME thrust-to-weight ratio. Figure 5.3.2.6-2 shows the axial accelerations associated with this ASRM trajectory, as well as g-levels associated with the SSME burns. Note that the acceleration is below 1 g for 110 seconds following ASRM separation (which occurs at 120 seconds). This creates significant gravity losses and suggests that adding a fourth SSME to the boattail would be a favorable variation to the reference Shuttle-Z. Despite the increased mass of the fourth SSME, a 13.4 t (approximately 10 percent) payload gain is realized with this option.

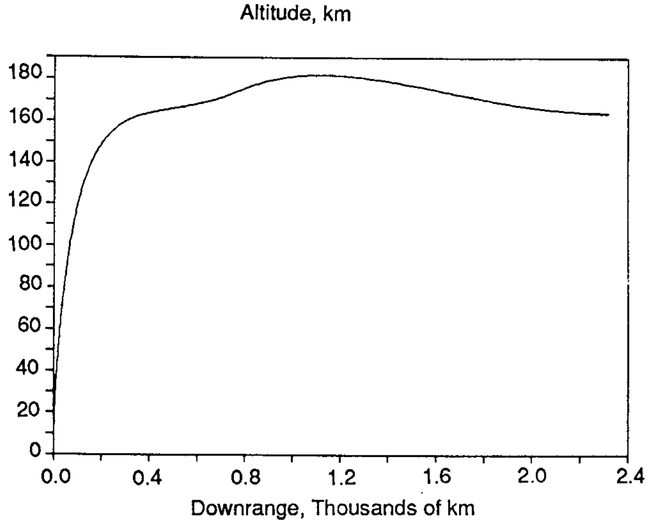


Figure 5.3.2.6-1 Shuttle-Z Altitude versus Downrange

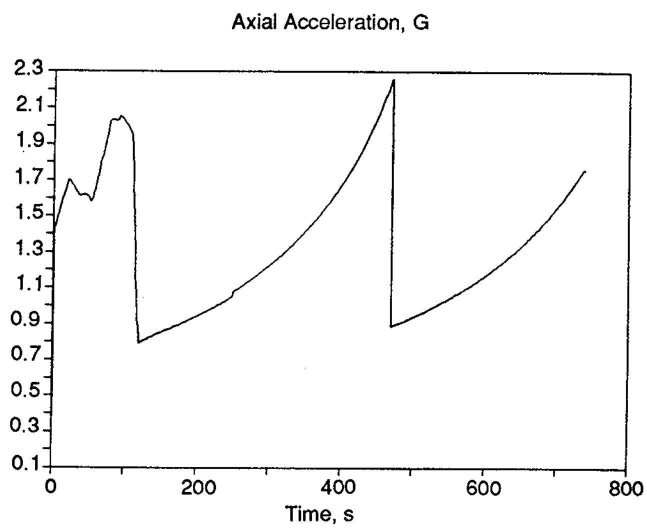


Figure 5.3.2.6-2 Shuttle-Z Axial Sensed Acceleration

Figure 5.3.2.6-3 shows the benign dynamic pressure environment experienced by the Shuttle-Z. The increased drag of the bulbous 12.2 m fairing and the heavier lift-off mass combine to reduce the maximum dynamic pressure from around 40 kPa for the Shuttle to less than 28 kPa for the Shuttle-Z. This allows the use of lighter gauge materials for the payload fairing. Figure 5.3.2.6-4 shows how the burn-out altitude (160 km) is a lower altitude limit, caused by the rapidly increased heating as vehicle speed reaches to orbital values.

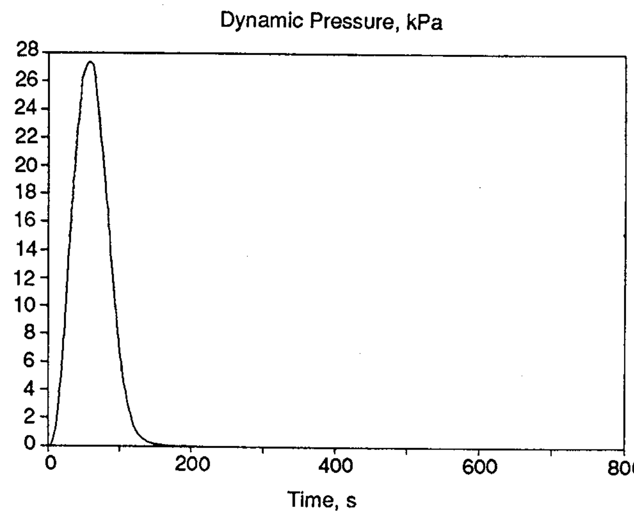


Figure 5.3.2.6-3 Shuttle-Z Dynamic Pressure

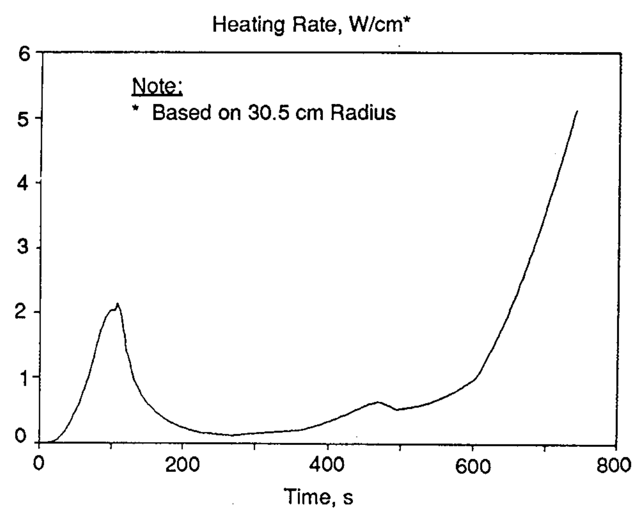


Figure 5.3.2.6-4 Shuttle-Z Heating Rate

These performance data were obtained from Fly-It, a Martin Marietta 3 degree-of-freedom ascent trajectory program, and verified on Martin Marietta's Program to Optimize Simulated Trajectories (POST). The launch site is assumed to be Kennedy Space Center with a due east launch azimuth (28.5° orbit inclination). The ultimate orbit is 407 km circular, analogous to Space Station Freedom's nominal orbit.

The vehicle ascends to a 164.3 by 722.3 km elliptic orbit, with burnout occurring at perigee. This orbit is the ΔV equivalent of performing a burn at perigee and then performing a later apogee burn to circularize at 407 km. Fly-It uses a geometrically oblate, rotating Earth model and the NASA-standard 1976 atmosphere. It has been used extensively at Martin Marietta on many programs (e.g., Advanced Launch System).

5.3.2.7 Flight Profile—The following flight trajectory was used for the Shuttle-Z ascent:

• T-8 sec	SSMEs ignite (T = 0 at ASRM ignition)
• 0.7 sec	Shuttle-Z is released (T/W >1)
• 0.7 to 10 sec	vehicle vertical rise
• 10 to 20 sec	vehicle initiates a 27.7 deg/min eastward pitch rate
• 20 to 22 sec	vehicle holds attitude until it reaches zero-angle of attach (alpha wind-aligned)
• 22 to 102.1 sec	vehicle maintains zero-alpha until altitude = 45 km
• 102.1 to 120 sec	vehicle begins an inertial pitch rate of 3.94 deg/min (continues until orbit is obtained)
• 120 sec	ASRMs released (ASRM acceleration = the core vehicles)
• 250 sec	shroud separation
• 250+ sec	external tank separation
• 469.7 sec	three SSMEs shut down, second stage released, third stage SSME start-up
• 739.7 sec	third stage completes its burn

• the apogee-raise maneuver occurs one-half orbit later, placing the third stage and payload into the final circular orbit.

5.3.2.8 Mars Mission Manifest Using Shuttle-Z— In NASA's Office of Exploration 1989 Cycle-2 studies of manned Mars missions the Shuttle-Z's capabilities (payload mass and dimensions) were specified as a constraint on the Mars vehicle's design. To demonstrate how the Shuttle-Z enables Mars missions, consider the following eight launch manifests that place all elements of an artificial gravity Mars Evolution spacecraft in LEO.

The Mars vehicle to be launched (Fig. 5.3.2.8-1) consists of a 39 m diameter aerobrake and two space station sized habitation modules that rotate out to increase artificial gravity and rotate back to place them inside the plasma wake during aerobraking. In the center of the aerobrake is the trans-Earth injection (TEI) propellant tanks and docking hub. Docked to the hub is a Mars descent vehicle (MDV) with aerobrake stowed.

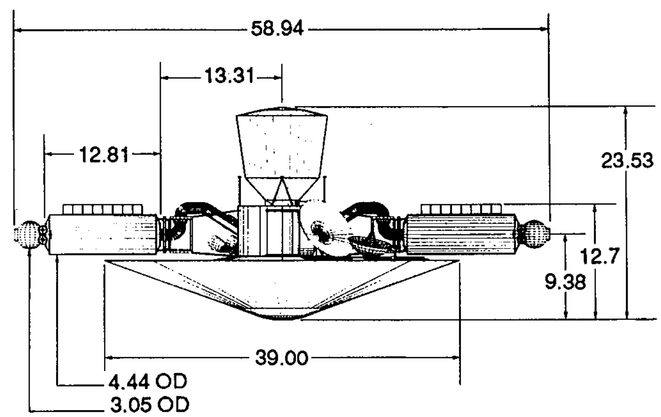


Figure 5.3.2.8-1 Artificial Gravity Mars Piloted Vehicle

The sequence begins with the launch of the 39 m aerobrake and the TEI tankage. After orbit is achieved, 50 articulated beams fold out from the aerobrake's 10 m core. The TEI tankage is launched only partially loaded to avoid exceeding the Shuttle-Z's 140 t lift capability. The next launch not only delivers the TEI engines, communications antennas, support struts, crew tunnels, docking hub, and tunnel rotation joints, but also provides the remaining TEI propellant, as this payload set would otherwise be less than the full 140 t. The third and fourth launches deliver the habitation modules and Mars descent vehicle. At this point the piloted vehicle is complete and all that remains is to attach all four of the spent Shuttle-Z third stages to the base of the aerobrake for trans-Mars injection. After four more Shuttle-Z launches (each launch delivering 124 tonnes of propellant as payload), the injection stages are filled and the mission can commence toward Mars.

If a Shuttle-C had been the only heavy lift launch vehicle available it would have taken 16 flights to lift the same mass. Furthermore, the aerobrake would have to be assembled on-orbit because the Shuttle-C's 4.6 m central core is too small. Finally, a new upper stage would need to be developed for TMI. Using a Titan-IV ELV or the Space Shuttle (with a third the lift capability of a Shuttle-C) would prove even more impractical.

5.3.2.9 Evolution from Shuttle-C—Evolution from the Space Shuttle through Shuttle-C into Shuttle-Z follows a logical and natural progression. Once Shuttle-C is operational, a large diameter version will be highly desirable because of its high lift capability to volume ratio (175 kg/m^3 versus 67 for NSTS). NASA has studied Shuttle-C payload diameters of 7.6 and 10 meters. As a Shuttle-C program develops these larger vehicles, a parallel requirement for a hydrogen/oxygen propelled upper stage will emerge to lift heavy payloads to geosynchronous orbit and to support lunar and unmanned planetary missions. If planned in advance, all these elements can flow directly into the making of Shuttle-Z.

5.3.2.10 Conclusions—This shuttle-derived HLLV study has presented several launch vehicle concepts. In retrospect, it appears that the most viable configuration worthy of further study is the four SSME boattail version of the side-mounted Shuttle-Z. It could use either 12 RL-10s or four advanced space engines with a thrust of 334 to 667 kN for the third stage. Also, we have found that a payload diameter of 10 meters is acceptable for launching the MPV's folded aerobrake.

Figure 5.3.2.10-1 shows a recommended configuration for future Shuttle-Z studies. Two areas of concern for the reference Shuttle-Z of Figure 5.3.2.3-1 exist. First, flight qualifying the SSME for an in-flight and zero start sequence and building the large expansion ratio nozzle need further study. Two, using only three SSMEs forces the heavy Shuttle-Z to loft its trajectory more, wasting propellant and payload capability.

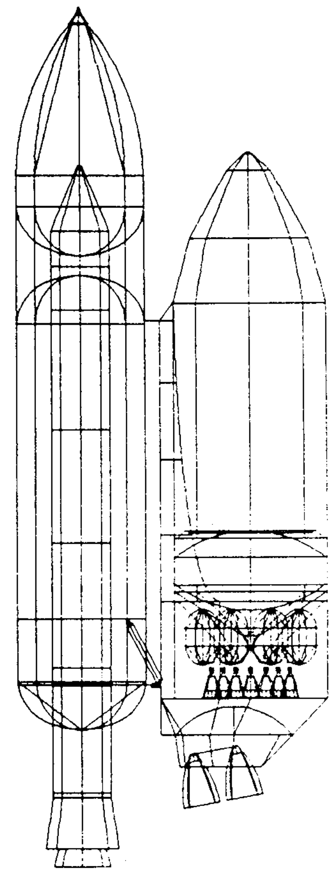


Figure 5.3.2.10-1 Alternative Configuration for Shuttle-Z

5.4 ADVANCED TECHNOLOGY

One possible method for decreasing the overall mission masses is to use lunar-produced liquid oxygen (LLOX). Advanced propulsion systems, including Nuclear Thermal Rockets (NTR), Nuclear Electric Propulsion (NEP), and Solar Electric Propulsion (SEP) are other possibilities for reducing the large spacecraft masses in these scenarios.

5.4.1 Lunar Liquid Oxygen (LLOX)

An independent study was performed to determine the feasibility of using lunar liquid oxygen (LLOX) on future Mars and lunar missions. This involved assessment of lunar produced propellant for routine cargo flights to the moon and of propellant loading node selection for trans-Mars injection (TMI). The study focused upon an oxygen and hydrogen propellant-based transportation system that delivers cargo to the moon from low Earth orbit (LEO) and also delivers propellant from either LEO or the lunar surface to the TMI node. Mission design options, vehicle mass sensitivities, and propulsion system mixture ratio characteristics were examined.

Specifically addressed for the baseline lunar cargo delivery missions were: a) the extent to which lunar produced propellants should be used in the lunar transportation system, b) the differences between employing lunar orbit as a transportation node and using a direct cargo transfer to the surface, and c) the optimum mixture ratio (oxygen to hydrogen mass ratio) for Earth-moon transfer. A sensitivity study was also performed to determine the impact of variation in Earth aerobrake mass on LLOX feasibility.

The TMI node study determined the transportation requirements of delivering propellant to each of the node options from both LEO and from the lunar surface. With this information, the total propellant requirements for a Mars mission for the various node candidates were compared, both with and without the use of lunar oxygen.

5.4.1.1 Lunar Cargo Missions and Lunar LOX— A major problem in assessing the potential benefits of lunar produced propellants is in determining an appropriate figure of merit for measuring these advantages. To avoid extensive life cycle costing analysis, a figure of merit has been employed that gives the "break even" conditions for the use of lunar propellants as compared to exclusively using propellants launched to LEO from Earth. Specifically, the cost ratio of producing propellant on the lunar surface to providing propellant in LEO has been quantified for several cases to show the conditions for which lunar produced propellant using chemical rocket transportation gives a net cost benefit over the use of Earth launched propellant.

The equations expressed in Figure 5.4.1.1-1 show how the "break even" cost ratio expression was derived. Equation 1 expresses the fact that lunar produced liquid oxygen (LLOX) will provide a cost benefit for the mission if the cost of LLOX and Earth-launched propellant in LEO (LEOPROP) for that mission are less than the costs of performing the mission with only LEOPROP. By multiplying the respective propellant amounts by the costs of providing propellant in each location, and rearranging the inequality, the ratio of cost per pound of lunar produced LOX to the cost of providing it in LEO is determined (see Equation 3). This ratio is expressed in terms of the mission propellant requirements.

When both sides of the relationship are set equal, an equation for the previously defined cost ratio "R" is determined (Fig. 5.4.1.1-2). This ratio sets the point at which the use of lunar propellants costs the same as with the case where no LLOX is used. This "break even" cost ratio has been quantified for varying degrees of use of lunar-produced oxygen, varying the amount of LLOX returned to LEO for the next trans-lunar injection (TLI). Also investigated were two separate mission profiles, various propellant mixture ratios to and from the moon, and two aerobrake mass ratios (percentages of return mass).

A Payload Delivery Mission May Benefit By Using Lunar Lox (LLOX) If:

1 $$\text{(LLOX Cost + LEOPROP Cost)} < \underset{\text{(Using No LLOX)}}{\text{LEOPROP(0) Costs}}$$

Or:

2 $$\text{LLOX * (LSURF \$/LBM) + LEOPROP * (LEO \$/LBM)} < \text{LEOPROP(0) * (LEO \$/LBM)}$$

Divide Through By Cost of Prop in LEO, Then Rearranging:

3 $$\frac{\text{LSURF \$/LBM}}{\text{LEO \$/LBM}} < \frac{\text{LEOPROP(0) - LEOPROP}}{\text{LLOX}}$$

Figure 5.4.1.1-1 Space Transportation: Lunar Oxygen Payoff

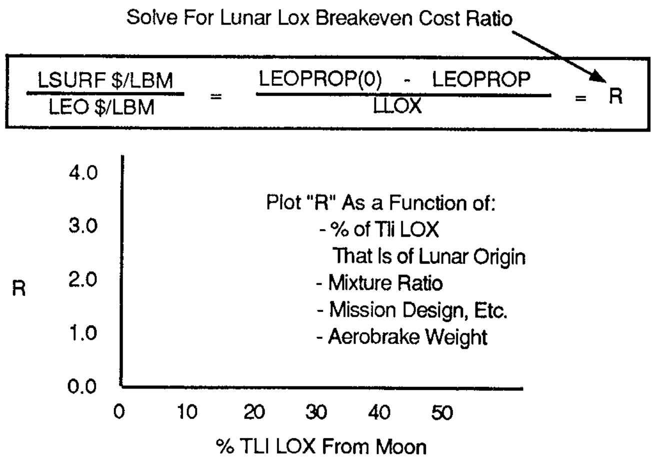


Figure 5.4.1.1-2 Cost Ratio

The ground rules used in this analysis were as follows:

1) 20 metric tons (44000 lb_m) of payload delivered to the lunar surface
2) reusable, aeroassisted space transfer vehicle based in LEO and cargo lander (if used) based on the lunar surface
3) single stage delivery of payload to lunar vicinity
4) LOX returned to LEO in main propellant tanks
5) Earth-originated hydrogen
6) LLOX for transfer vehicle return to LEO
7) LLOX for lunar lander (if a dedicated lander is used)
8) I_{sp} versus mixture ratio curves provided by Pratt & Whitney
9) vehicle subsystem scaling relationships and mission ΔVs performed by Martin Marietta

Two mission design options were considered during the study. The first uses low lunar orbit (LLO) as a transportation node and the second uses direct-to-surface delivery of the cargo. Because of the unfavorable comparison with the other two cases as a transportation node for delivering cargo to the lunar surface (OTV Phase A Study, MSFC NAS8-36108), the Earth-moon libration point L1 was not included in this analysis. L1 was considered, however, in the TMI node location study (Section 5.4.1.2).

Figure 5.4.1.1-3 shows the delivery of cargo from LEO to LLO with a lunar cargo vehicle (LCV). The payload is subsequently delivered to the surface with a lunar cargo lander (LCL). The lander then returns to LLO to transfer LLOX, of the amount desired, to the LCV. The LCV then returns to LEO to refuel with Earth hydrogen and top-off with Earth liquid oxygen.

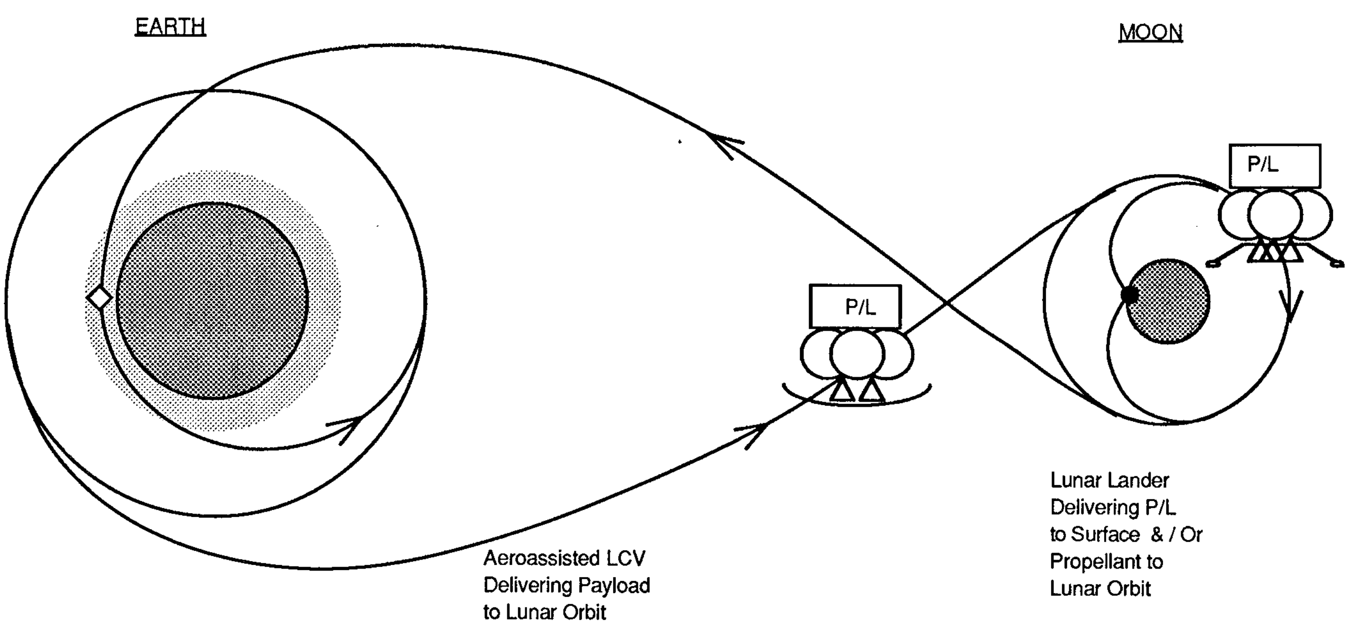


Figure 5.4.1.1-3 Lunar Orbit Mission Profile

A similar scenario is shown in Figure 5.4.1.1-4 except that the LCV delivers the cargo directly to the surface of the moon instead of rendezvousing with an LCL in lunar orbit for the payload transfer. This LCV must therefore have landing legs as well as an aerobrake. The LCV collects the LLOX on the lunar surface (as compared to the LLO transfer of the previous case) for return to LEO and for use on the next trans-lunar injection (TLI). After LLOX transfer, the LCV performs a direct ascent and returns to LEO by aerocapturing.

The extent of LLOX use was varied for each mission design to determine a possible optimum or feasibility limit for LLOX use. The mission scenarios in this study consisted of a delivery of cargo to the lunar surface, the use of LLOX for the LCV trip back to LEO, and the return of LLOX to Earth orbit. The analyzed range of LLOX use extended from 0 to 100 percent of cargo vehicle TLI lunar LOX. LLOX was also assumed to be used for all the LCL propellant needs.
The cost ratio values for LLOX use as a function of percent lunar-originated TLI oxygen is shown in Figure 5.4.1.1-5. In the direct-to-surface case, LLOX can cost more than four times as much as LOX in LEO and still "break even" with the case of using all-LEO propellants. However, this is only for the condition where no LLOX is returned to LEO for TLI (LLOX is only used for returning the cargo vehicle). By bringing LLOX back to LEO, the "break even" cost ratio of LLOX decreases for both mission scenarios (i.e. LLOX production must become less expensive to present a cost advantage over all-LEO propellant cases). It is apparent, therefore, that the use of LLOX only for LCV return to LEO (no TLI LOX returned to LEO) is the most advantageous method of LLOX employment from a cost standpoint.

It is also clear, that when considering the cost benefits of LLOX, the direct-to-surface mission design is more desirable than the mission using LLO as a transportation node. This is caused by the heavier dry mass of the cargo vehicle for this mission option (it must have landing legs plus an aerobrake) and the large, direct-ascent ΔV for lunar surface to LEO transfer. However, as shown in Figure 5.4.1.1-5, the cost-effectiveness for the LLOX direct-to-surface case crosses over the LLO node case for over 20 % of TLI LLOX returned to LEO. This is caused by aerobrake mass growth (because of returned LLOX mass growth) and the higher return ΔV for the direct-to-surface option.

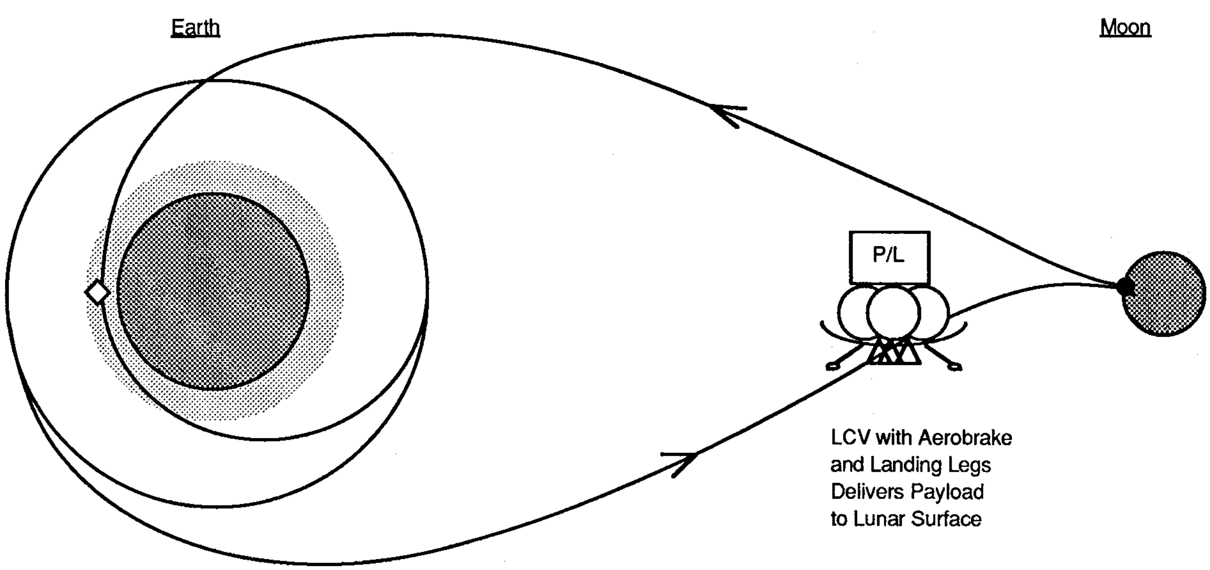


Figure 5.4.1.1-4 Direct-to-Surface Mission Profile

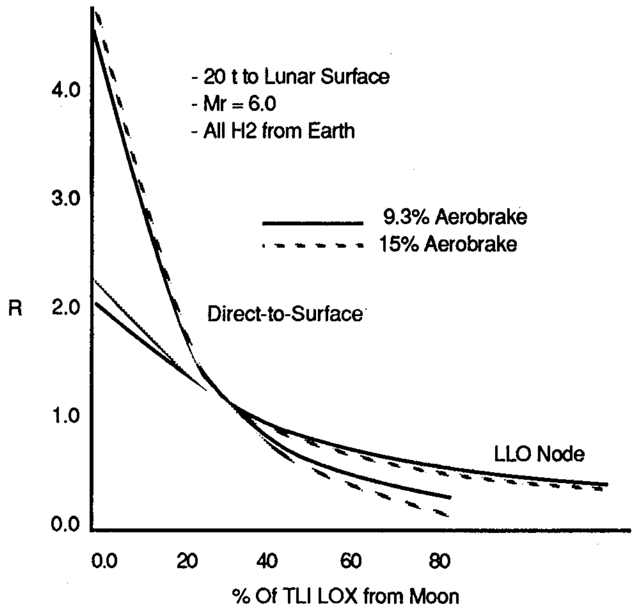


Figure 5.4.1.1-5 "Break Even" LLOX Cost Ratios

The results were initially calculated for an aerobrake mass ratio (aerobrake to entry mass) of 9.3%, corresponding to a flexible fabric aerobrake design. Because of the advances needed for this type of aerobrake, a sensitivity study was performed by varying the aerobrake mass ratio. By increasing this ratio to 0.15, a second set of "R" values were obtained and are shown in Figure 5.4.1.1-5. From this analysis, the cost-effectiveness of LLOX does not appear to be significantly affected by aerobrake mass ratio changes. The total propellant requirements increase as a result of increased aerobrake weight, but the benefits of LLOX on a per pound basis change only slightly.

It should be noted that these curves cannot be used to determine the optimum quantity of LLOX. These data simply represent the condition for which LLOX breaks even. Estimates for the per-pound costs of LEOPROP and LLOX need to be multiplied by the respective amounts of propellant used from each location to obtain a total cost comparison. With these propellant amounts (Fig. 5.4.1.1-6), one can speculate on the cost of producing lunar propellant and, from this, determine possible benefits for LLOX use.

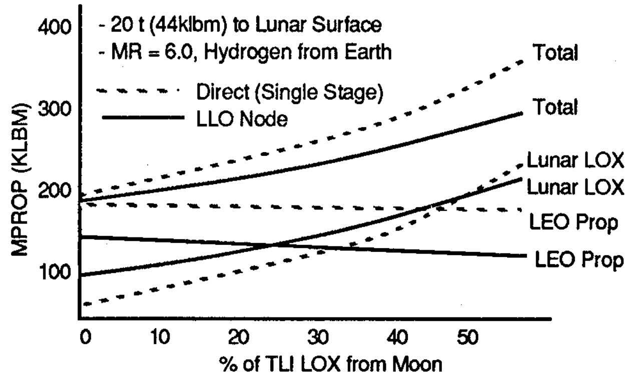


Figure 5.4.1.1-6 Propellant Usage for LOX Return

The propellant quantities shown in Figure 5.4.1.1-6 were examined to assess cost drivers related to propellant quantity, including: (1) operations for increased amounts of transferred propellant, and (2) greater or lesser vehicle unit costs for larger vehicles, or greater flight operations costs for more missions with the same size vehicles. The propellant requirements grow significantly with increasing amounts of LLOX usage (Fig. 5.4.1.1-6), quantifying the costs mentioned above would provide further rationale for not returning large amounts (e.g., larger than 20%) of TLI LLOX

The total propellant requirement for the LLO node case is slightly less than for the single stage direct-to-surface case. However, in both cases, two vehicles will likely be involved in a cargo delivery mission, either by having a two-stage LCV for the direct case, or by having an LCV and LCL in the LLO node case. In using two stages for the direct-to-surface option, an approximate six percent reduction can be made in overall propellant requirements over the single stage case. Therefore, with both mission designs involving two "stages" to complete the delivery, the mission operations costs would be approximately equal, or perhaps even favor, the direct-to-surface case. However, the performance comparison between these mission options is highly dependant upon the ΔVs and vehicle mass estimates corresponding to each mission profile and therefore requires further investigation. The intent of this study was not to choose between the two mission profiles, but only to present their impacts on the LLOX benefits.

A further study was performed to determine the effect of varying propellant mixture ratio (MR) on the cost-effectiveness of LLOX. Although I_{sp} decreases for mixture ratios greater than about 6.0, operating at a higher MR could provide performance benefits by minimizing Earth-launched hydrogen, assuming abundant and relatively inexpensive LLOX. The study results shown earlier (which assumed a MR of 6.0) were recalculated with varying MRs for: (1) the trip to the moon (upleg), (2) the trip back from the moon (downleg), and (3) both legs. By increasing the downleg MR, which influences the operation of the LCL, very little cost benefit was realized. No significant effect could be found for greatly increasing the MR over 6.0, except in cases where a large percentage of LLOX is used for TLI (Fig. 5.4.1.1-7). In addition, there was no benefit to increasing the upleg MR over 6.0.

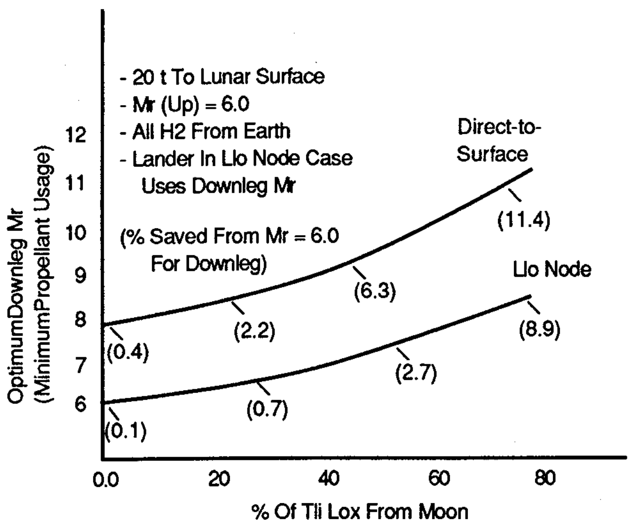


Figure 5.4.1.1-7 Mixture Ratio Optimization

Hauling large amounts of LOX back to LEO may not provide a payoff. It appears, that providing LLOX for the downleg (trip back from the moon) provides a non-negligible cost benefit. To determine the quantity of return propellant that should be lunar-generated, a sensitivity study was performed. Figure 5.4.1.1-8 shows the "R" values for the direct-to-surface and LLO node cases as a function of LLOX used for cargo vehicle return. Because of the low ΔV required for the cargo vehicle to inject out of LLO, the LLO node case is relatively insensitive to return propellant usage. Yet, because of the large ΔV of ascent and the larger dry mass of the cargo vehicle, the direct-to-surface case is strongly dependant on the availability of LLOX at the surface of the moon. Figure 5.4.1.1-9 shows the total propellant requirements for the two cases.

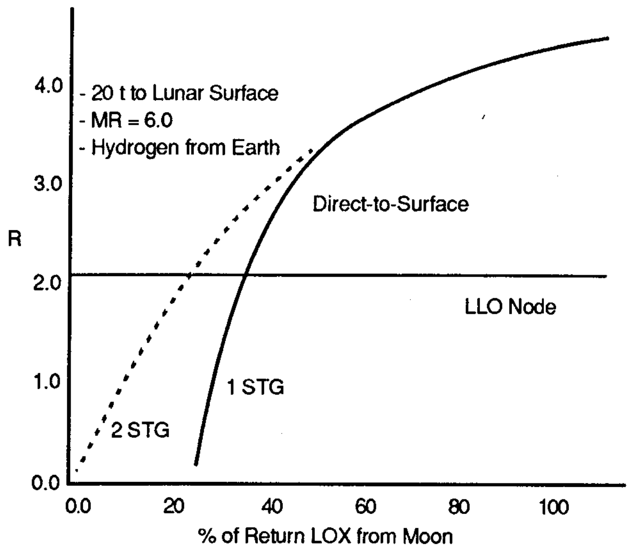


Figure 5.4.1.1-8 "Break Even" Cost Ratios for LCV Return

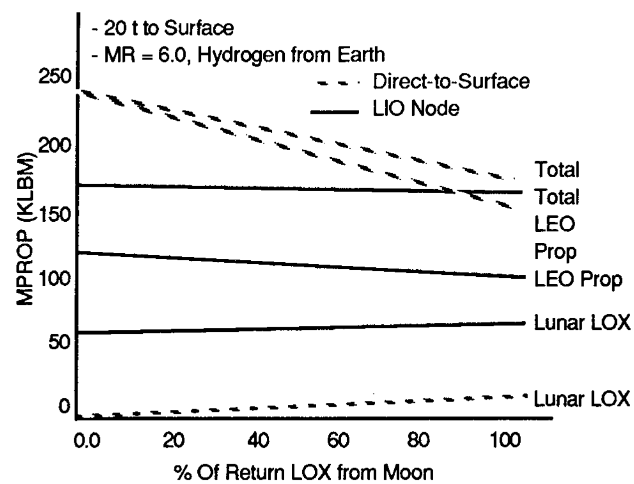


Figure 5.4.1.1-9 Propellant Usage for LCV Return

By combining Figures 5.4.1.1-5 and 5.4.1.1-8, the complete range over which LLOX usage is shown (Fig. 5.4.1.1-10). The results of this study for lunar cargo delivery show that LLOX offers the greatest cost benefits when it is used only for LCL operations between the surface of the moon and LLO (for the LLO node case) or if used for the return to LEO leg of the direct-to-surface mission option. In addition, LLOX use is most beneficial in the direct-to-surface case.

5.4.1.2 Trans-Mars Injection Node Location & Lunar LOX—To allow the extensive vehicle buildup required for a manned Mars mission, an as-sembly platform or propellant loading depot in the Earth-moon system may be needed. Because of large propellant requirements for a Mars mission, it is useful to reduce ΔV of trans-Mars Injection (TMI) and acquire propellant from an optimum location. Figure 5.4.1.2-1 shows node candidate locations within the Earth-moon system that are possible locations for an assembly location and/or propellant loading depot.

There are several issues surrounding the choice of a TMI node location. One, of course, is accessibility to the node by Earth surface-originated elements, including crew, payload(s), and the TMI vehicle. Another issue is the risk involved in doing the buildup and propellant loading operations at the node compared to the same risks at LEO (or launching it in one piece from the ground).

Possible optimum locations for TMI propellant loading, differing from LEO, include: (1) a highly elliptical Earth orbit (HEEO), (2) lunar orbit, and (3) the Earth-moon libration point (L1). The crew/payload and TMI vehicle loaded with hydrogen for the mission (altogether a relatively small fraction of the total TMI stage mass) could first be injected into one of these node locations. The LOX would then be loaded onto the stack prior to TMI. Because TMI LOX may be available to some of these node locations from the lunar surface at a lower cost and higher transportation efficiency than obtaining it from LEO, the overall cost of the mission could drop.

Figure 5.4.1.2-2 shows the transfer vehicle propellant requirements for delivering LOX from either LEO or the lunar surface to various node locations. These data represent the number of kilograms of transportation propellant required to deliver one kg of LOX from either of the propellant sources to each of the candidate nodes. From this figure, it appears to be very inefficient to deliver lunar LOX to LEO because of the large transportation penalties. This is caused by the ground rule that all hydrogen must come from Earth. This number is reduced from 25 to about 6.0 if the hydrogen is available on the moon. Lunar orbit (LLO), on the other hand, may be a desirable node for using lunar LOX because of the high transportation efficiency of LLOX to LLO transfer. Depending upon the relative costs of the transportation propellant used, HEEO or L1 may also be good locations for the used of LLOX for TMI. The data of Figure 5.4.1.2-2 combined with TMI performance sensitivities for injecting the TMI vehicle out of each node provide the rationale for selecting a TMI node location.

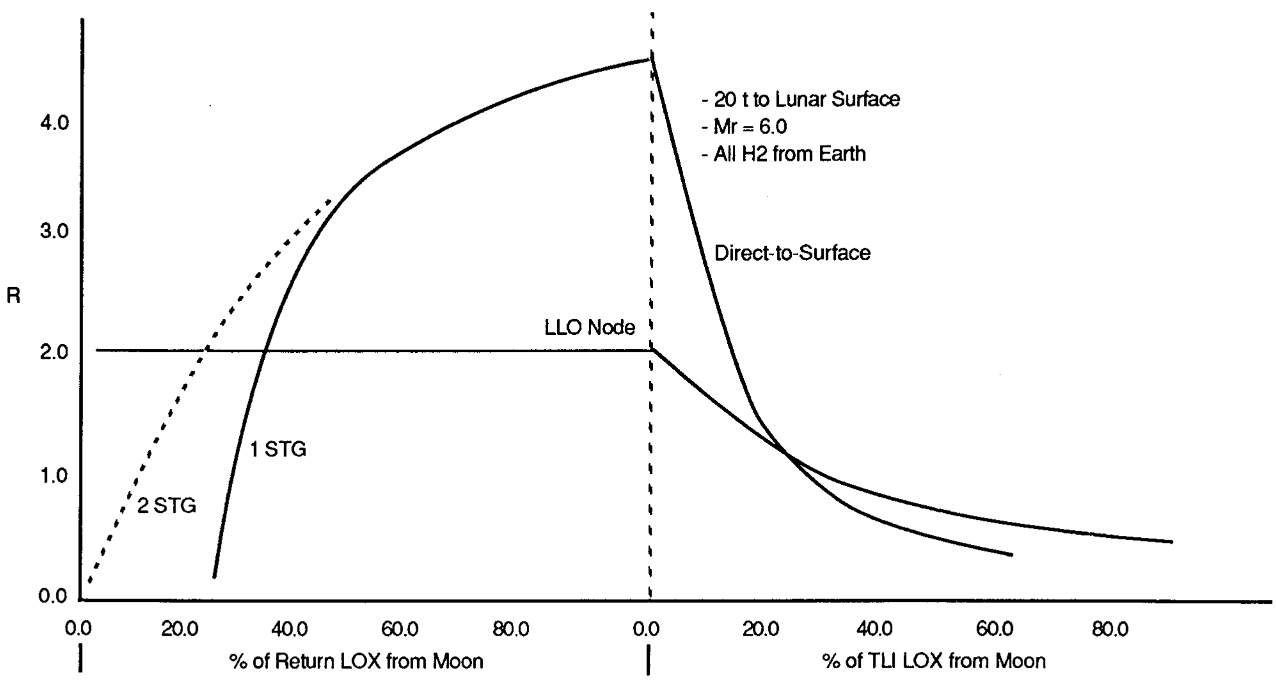
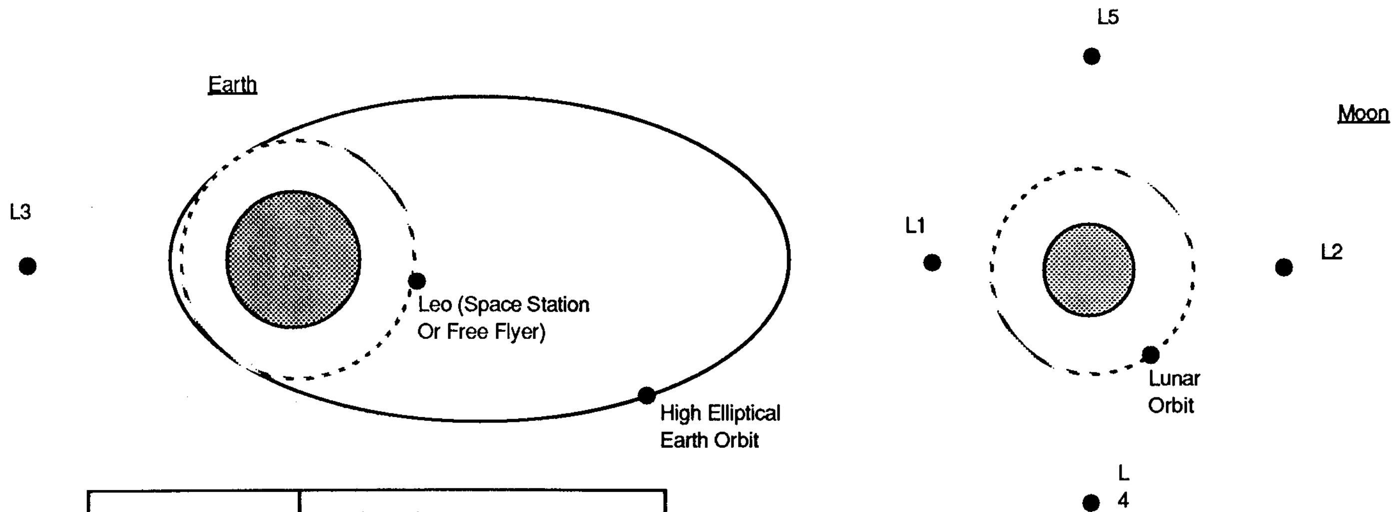


Figure 5.4.1.1-10 "R" Values Over Range of LLOX Usage



Node Location	Why Attractive
Lunar Orbit	Near Lunar Surface For Lunar Propellant Delivery
Libration Points	Gravitationally Stable, Not A Deep Gravity Well, L1 Perhaps The Best
High Elliptical Earth Orbit	Near Earth Escape, About "Halfway" Between Earth And Moon
Low Earth Orbit	Use Of Leo Space Station (Or Co-orbiting Facility), Nearest Earth

Figure 5.4.1.2-1 Trans-Mars Injection Node Candidates

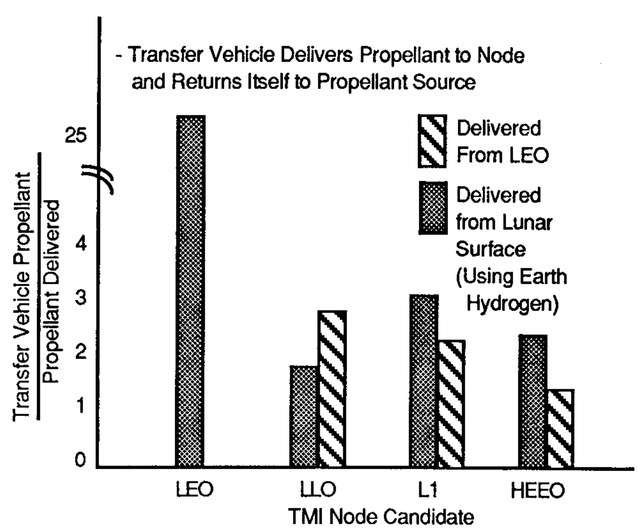


Figure 5.4.1.2-2 Propellant Delivery Sensitivities

To analyze the node efficiencies, the TMI ΔV requirements as a function of departure C3 are needed. These velocity requirements are shown in Figure 5.4.1.2-3. In the case of a LLO-located TMI stage, the spacecraft must first kick out of that orbit and fall toward the Earth before performing the final TMI propulsive burn at periapsis. This case takes advantage of the Earth swingby, but is penalized by the initial ΔV required to inject out of lunar orbit. The HEEO case, on the other hand, still takes advantage of its velocity at periapsis, but is not penalized by initial ΔV required to enter an Earth swingby trajectory. These ΔV data were used to compute the quantity of TMI propellant required per unit mass of TMI payload for the various departure node locations.

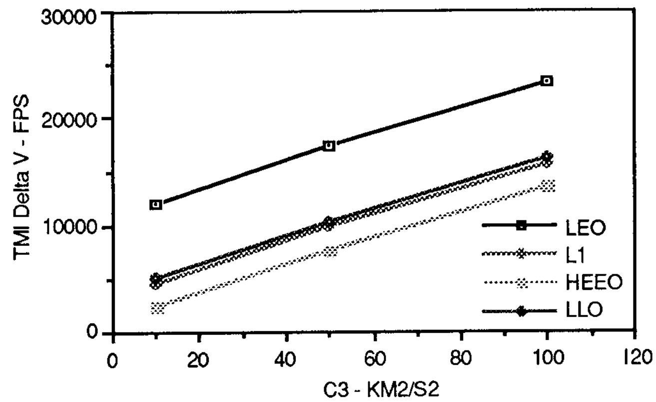


Figure 5.4.1.2-3 ΔV versus C3 for Each Node

The equation in Figure 5.4.1.2-4 shows how the transportation sensitivities from Figure 5.4.1.2-2 and those derived from Figure 5.4.1.2-3 were used to compute the total propellant requirements to transport a payload from LEO through the possible non-LEO TMI departure node towards Mars. The first term in this equation calculates the amount of propellant involved to transport the TMI LOX to the departure node location. "N", a payload mass fraction, is derived from the ΔV information in Figure 5.4.1.2-3 and "M" is from Figure 5.4.1.2-2. The second term computes the propellant required to transport the TMI dry vehicle and hydrogen from LEO to the TMI node. The last two terms consist of the propellant required to transfer the TMI payload from LEO to the node and the actual TMI propellant required per TMI payload mass.

The results of this equation are shown in Figure 5.4.1.2-5 for various departure C3s. The LEO TMI node case that uses lunar LOX was not considered because of excessive propellant requirements. Except for in the lunar node case, less total propellant was required when Earth-only propellants were used. However, this does not mean that the total propellant cost is also less. If lunar propellants are abundant and relatively inexpensive, compared to providing propellant in LEO, these cases may actually pay off for a Mars mission. Therefore, it is necessary to understand the constituents of these total propellant amounts (whether from the Earth or the moon) so that corresponding cost factors can be applied to propellant supplied from various locations.

Figures 5.4.1.2-6 to 5.4.1.2-8, which correspond to cases with LLOX as TMI propellant, show the propellant required quantities from each of the propellant sources for the different TMI node locations. These propellant amounts, when combined with cost factors associated with each source of propellant, provide rationale for selecting propellant source locations and TMI LOX loading and departure node locations. For example, in the highly elliptical Earth orbit (HEEO) case (Fig. 5.4.1.2-6), the total propellant requirements for LLOX use are greater than for the Earth-only propellant comparison case. Therefore, for the HEEO/ lunar LOX case to "break even" with the total propellant costs of the comparison case, lunar surface LOX must be less expensive than propellant in LEO. How much less expensive it needs to be depends upon the departure C3 of interest.

$$N \times \left(\frac{6}{7}\right) \frac{\text{LBM O2}}{\text{LBM TMI Prop}} \times M + N \times \left(\frac{1}{7} + 0.09\right) \frac{\text{LBM Trans Prop}}{\text{LBM H2 \& Dry}} \times K$$

$$+ K + N = \boxed{\frac{\text{Total Prop}}{\text{LBM P/L}}}$$

$$N = \frac{\text{LBM Tmi Prop}}{\text{LBM P/L}}$$

$$M = \frac{\text{LBM Trans Prop}}{\text{LBM O2}}$$

$$K = \frac{\text{LBM H2 \& Dry}}{\text{LBM TMI Prop}}$$

Figure 5.4.1.2-4 Total Propellant Calculation for TMI

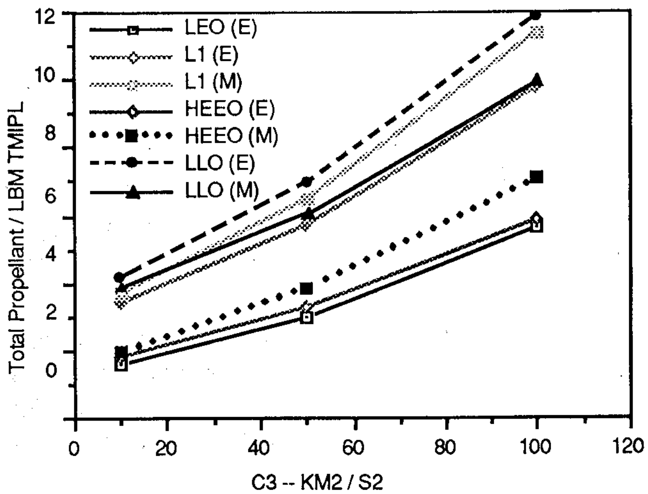


Figure 5.4.1.2-5 TMI Node Location Parametrics

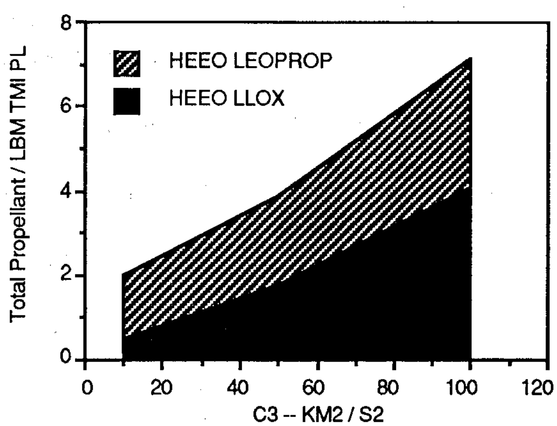


Figure 5.4.1.2-6 Propellant Split-HEEO Node w/LLOX

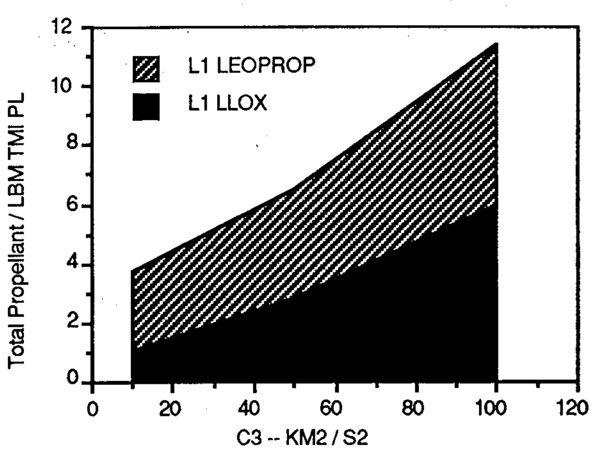


Figure 5.4.1.2-7 Propellant Split-L1 Node w/LLOX

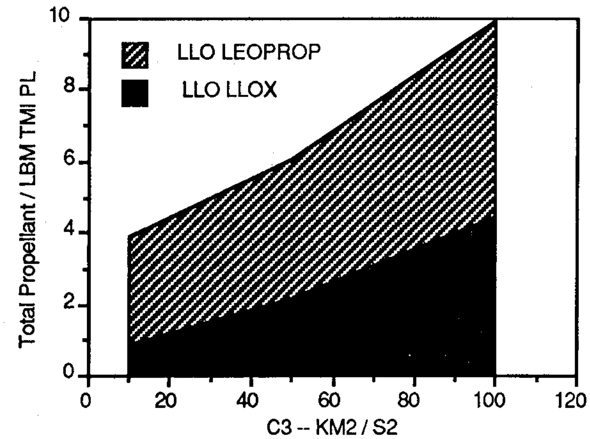


Figure 5.4.1.2-8 Propellant Split-LLO Node w/LLOX

5.4.1.3 Lunar LOX Conclusions—The analysis in this study indicates that if LLOX is used in the LLO node case, it should provide all of the lunar lander LOX needs. For the direct-to-surface scenario, lunar LOX usage for the moon to LEO return trip is recommended.

For both mission profiles, it is not recommended to return a large percentage (> 20%) of LLOX needed for subsequent lunar missions. To provide precise numbers, cost estimates for propellant in LEO and on the surface of the moon should be multiplied by this studies calculated propellant quantities to provide an absolute total propellant cost comparison. Mixture ratio increase above 7.0 or 8.0 is not recommended because of the minor gains in performance.

Trans-Mars injection node location optimization is a function of: (1) the efficiency of transferring propellant from its source to the candidate node locations, (2) the costs of providing the propellant at the source, and (3) the costs of transporting the dry TMI vehicle and payload to the node from LEO. The data presented here indicates that lunar produced LOX would need to be less expensive to produce on the lunar surface than the cost of propellant in LEO to provide a cost benefit for use on TMI.

The node location that appears to have the best potential for taking advantage of lunar LOX is a highly elliptical Earth orbit (HEEO). This is caused by the fact that less total propellant is required for the HEEO case, as well as the fact that a higher percentage of the overall propellant amounts are of lunar origin. However, unless liquid oxygen is significantly cheaper to produce on the lunar surface than providing it in LEO, Mars vehicle buildup and trans-Mars injection should occur in low Earth orbit with all-Earth propellants.

5.4.2 Nuclear Thermal Rockets (NTR)

Nuclear Thermal Rocket (NTR) technology offers the potential of sharply reducing the described missions' IMLEO masses.

5.4.2.1 NTR Use on Mars Missions—A study was performed to determine the mass benefit obtained by using NTR over a purely cryogenic system. The analyzed mission assumed a roundtrip voyage from LEO to a 250 km by 1 sol martian orbit. Table 5.4.2.1-1 shows the ΔVs used for this analysis.

Table 5.4.2.1-1 Assumed ΔVs

ΔVs (km/s)			
	Conjunction	Medium Energy	High Energy
Earth Departure	3.8	4.5	5.5
Mars Arrival	1.5	2.5	5.1
Mars Departure	1.5	2.5	4.4
Earth Arrival	3.8	4.5	6.2
Flight Time (days each way)	220-330	120-170	80-120

The following additional assumptions were made in this analysis:

1) 60 tonne roundtrip payload
2) aerobraking at Earth and Mars.
3) aerobrake mass fractions of 15% for the conjunction mission, 20% for the medium energy mission, and 30% for the high energy mission
4) NTR masses of 15 t for the conjunction mission, 20 t for the medium energy mission, and 30 t for the high energy mission
5) staged engine burns
6) cryogenic rocket stages with a mass fraction of 0.9
7) cryogenic I_{sp} = 470 sec; NTR I_{sp} = 900 sec

The results of this study can be seen in Table 5.4.2.1-2.

Table 5.4.2.1-2 Cryogenic versus NTR Mars Missions

Initial Mass in LEO (t)			
	Conjunction	Medium Energy	High Energy
Cryo/no aerobrake	956	2479	23211
Cryo/aerobrake	317	555	1567
NTR/no aerobrake	289	480	1408
NTR/aerobrake	195	282	547

It can be seen that as the missions increase in energy, the benefit of combining using NTR becomes more and more pronounced. For very quick trips (less than 120 days), the combination of NTR and aerobrake becomes a necessity.

5.4.2.2 NTR Use on Lunar Missions—Nuclear thermal rockets (NTR) were evaluated against cryogenic LOX/H_2 propulsion for the Lunar Gateway case, with a 6 t crew cab making a round trip from LEO to the lunar surface, and a 20 t cargo vehicle making a one-way trip from LEO to the lunar surface.
The following assumptions were made in this analysis:

1) human mission with 6 t crew cab and 4 t spacecraft (roundtrip LEO-Luna)
2) cargo mission with 20 t LEO-Luna cargo and 5 t spacecraft
3) NTR I_{sp} = 900 sec; cryogenic I_{sp} = 465 sec
4) 600 MWt NTR with a mass of 3 t
5) cryogenic engines with a mass of 0 t
6) aerobrake mass fraction of 15%,
7) cryogenic propellant tank fraction of 7.5%
8) NTR propellant tank fraction of 10%.
9) ΔV to leave or enter LEO = 3.357 km/s,
10) ΔV to leave or enter LLO or TEI = 1.17 km/s.
11) ΔV to land on the moon from LLO = 2 km/s,
12) ΔV to return to LLO from Luna = 1.9 km/s
13) ΔV for direct flight from Luna to TEI = 2.508 km/s.
14) ΔV for periapsis raise after aerobraking = 0.275 km/s.

The resulting total initial masses in LEO (IMLEO) for each mission in lunar gateway are shown in Table 5.4.2.2-1 for a direct to surface trajectory profile and for a LLO node stopover trajectory profile.

Table 5.4.2.2-1 NTR versus Cryogens (IMLEO in tonnes)

	OTV to LLO node, cryo lander (t)		LEO to Luna Direct Descent (t)	
	All Propulsive	Aerobrake	All Propulsive	Aerobrake
Human:				
Cryogenic	147.00	104.00	211.00	114.00
NTR	74.00	64.00	64.00	50.00
Cryo/NTR	1.99	1.63	3.30	2.28
Cargo:				
Cryogenic	147.00	127.00	202.00	154.00
NTR	86.00	80.00	85.00	76.00
Cryo/NTR	1.71	1.59	2.38	2.03

It can been seen that using cryogenic propulsion instead of NTR increases the total IMLEO of lunar missions by a factor ranging from 1.6 to 3.3.

5.4.2.3 Example Mars Mission NTR Spacecraft—An NTR spacecraft design point was specified for possible use on a Mars mission. Table 5.4.2.3-1 shows the approximate mass of this NTR-based spacecraft. The manned vehicle uses a high energy transfer orbit to reduce the Earth-to-Mars transfer time to 100 to 170 days. The cargo vehicle transfers to Mars on a minimum energy orbit which takes about 220 to 300 days.

Table 5.4.2.3-1 NTR Vehicle Design Point

Component	Cargo (t)	Human (t)	Comments
Engine	10.0	10.0	5000 MW_{th}, 900 sec I_{sp}
Shield	10.0	10.0	
Propellant	225.0	225.0	hydrogen
Tankage	25.0	25.0	
Aeroshell	25.8	25.8	
Payload	-	60.0	high energy orbit, round trip
Payload	185.0	-	minimum energy orbit, round trip
TOTAL	480.8	355.8	

This vehicle is shown in Figures 5.4.2.3-1 and 5.4.2.3-2. It uses a NERVA engine, shown in Figure 5.4.2.3-3.

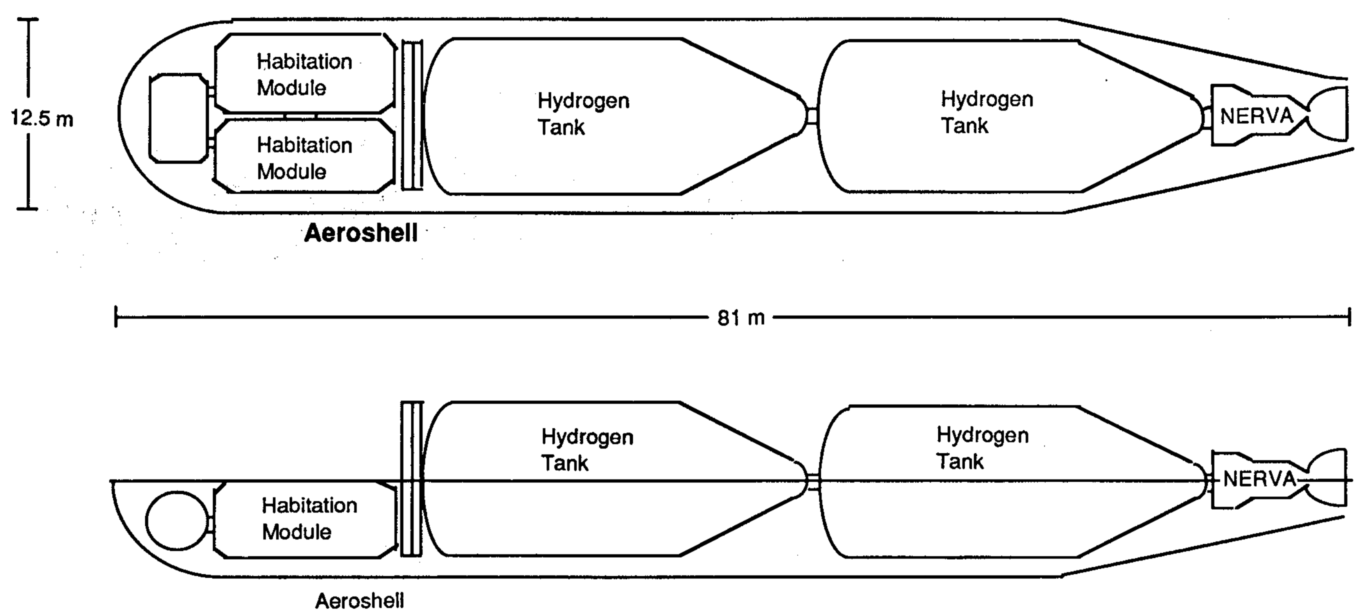


Figure 5.4.2.3-1 NTR Interplanetary Transfer Vehicle

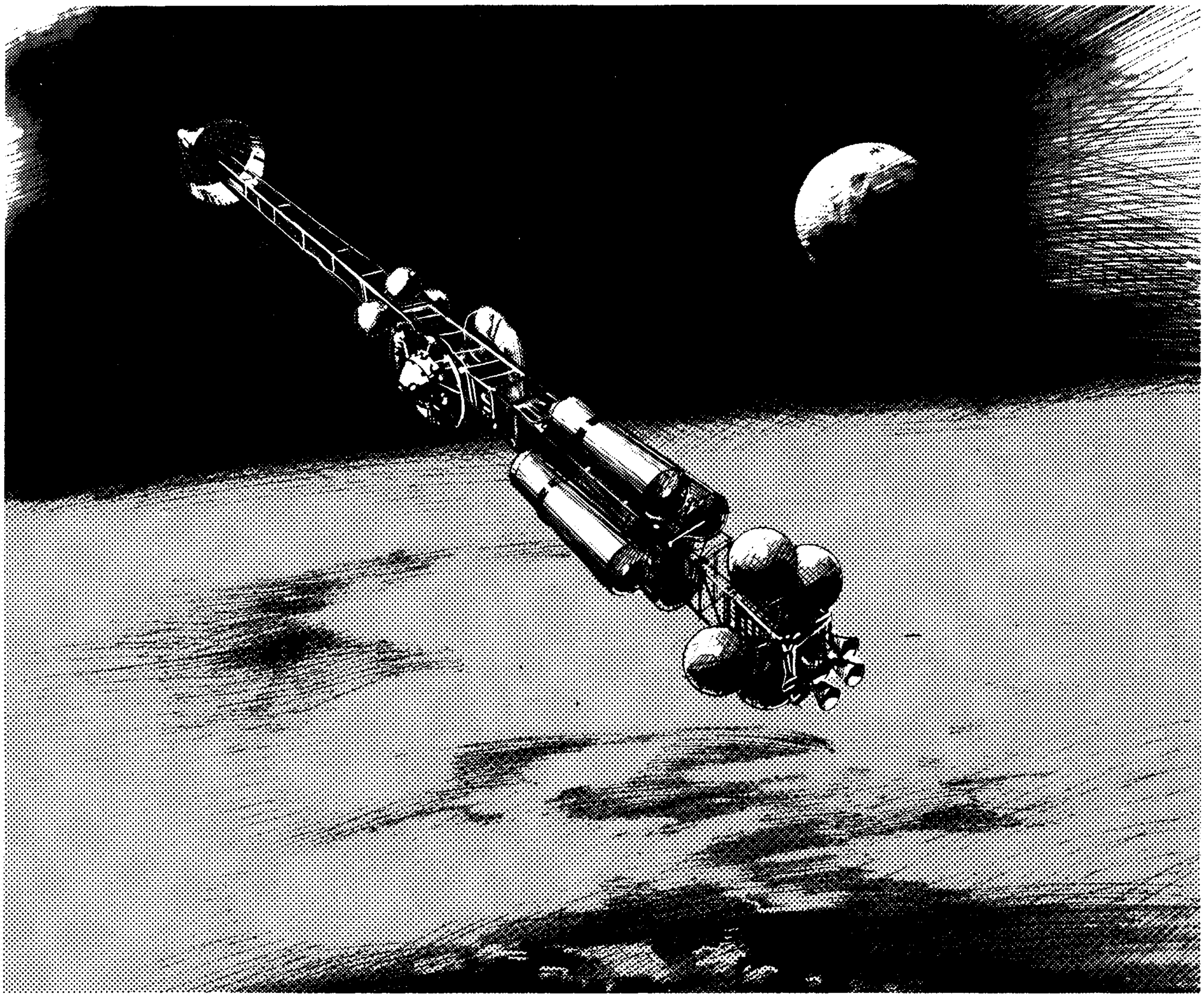


Figure 5.4.2.3-2 NTR Interplanetary Transfer Vehicle (Artist's Conception)

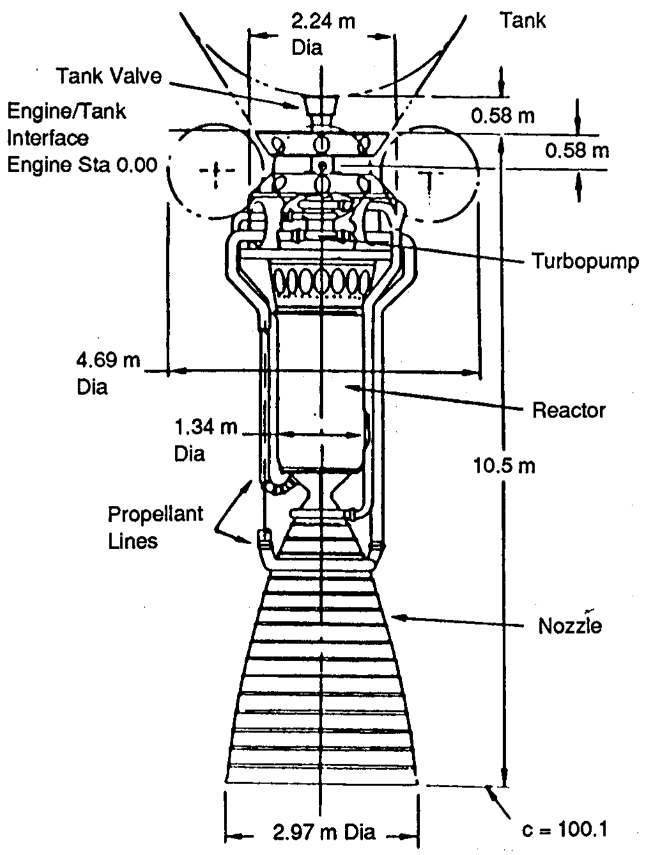


Figure 5.4.2.3-3 NERVA Flight Engine

5.4.3 Nuclear Electric Propulsion (NEP)

Another possibility for reducing the mission masses is to use Nuclear Electric Propulsion (NEP). This section shows two different concepts for possible NEP vehicles, the first using ion thrusters, and the second employing magnetoplasmadynamic (MPD) thrusters.

5.4.3.1 Case Study 4 Nuclear Electric Cargo Vehicle—A study was performed to generate a nuclear electric cargo vehicle for use in low-thrust transfer of cargo to Mars and the moon. The following assumptions were made in generating a baseline vehicle:

1) alpha = 15 kg/kWe
2) reactor output = 5 MWe
3) system efficiency = 63%
4) power control and distribution assembly efficiency = 90%
5) ion thruster efficiency = 70%
6) I_{sp} = 6000 sec
7) maximum payload = 910 t
8) tankage and propellant reserves = 10% of nominal propellant mass
9) payload adaptor structure = 5% of maximum payload

Table 5.4.3.1-1 tabulates the masses for this ion-thrusted, NEP vehicle, and Figure 5.4.3.1-1 and 5.4.3.1-2 show and summarize this electric cargo vehicle.

Table 5.4.3.1-1 NEP Cargo Vehicle Summary

Subsystem	Mass (t)
Nuclear Subsystem	48.6
Reactor	11.9
4-Pi Radiation Shield	36.7
Power Generation	5.0
Turbines	2.8
Alternators	2.2
Thermal Subsystem	3.6
Bubble Membrane Radiator	1.7
Auxiliary Cooling	1.0
Coolant (potassium)	0.9
Pwr Condit & Dist	5.2
Ion Engines	3.9
Structure	2.0
Payload	910.0
Payload Adaptor	45.5
Propellant (Argon)	212.0
Tankage	21.2
Contingency (20%)	27.0
TOTAL (dry)	1072.0
TOTAL (wet)	1284.0

Dry Mass	1,072,000 kg
(Includes Max. Payload)	910,000 kg
Payload Mass	
(Max.)	804m3
Payload Volume	
(As Shown)	Argon
Propulsion System	
Propellant Type	
Engines	
Number	7 (2 Spare)
Type	Ion
Thrust (Total)	21 N
Isp (6000 s)	58.9 kN-s/kg
Propellant Mass	
(Max.)	212,000 kg
Tank Mass	21,200 kg
Total Mass	1,284,000 kg

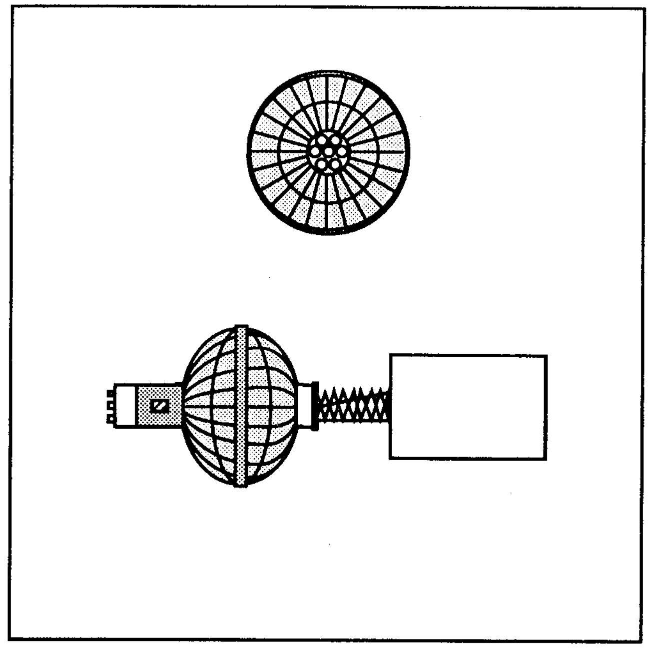


Figure 5.4.3.1-1 Nuclear Electric Cargo Vehicle Summary

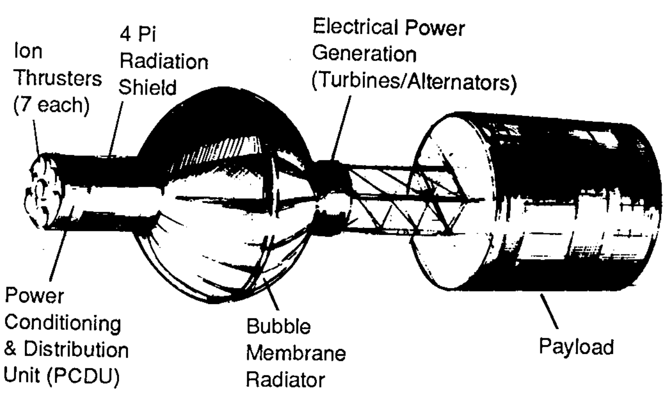


Figure 5.4.3.1-2 Nuclear Electric Cargo Vehicle (Artist's Conception)

5.4.3.2 400 t Capacity NEP Cargo Vehicle—A 400 t capacity, MPD-thrusted cargo vehicle was also generated for this study. This example NEP vehicle is summarized in Table 5.4.3.2-1 and shown in Figure 5.4.3.2-1. The propellant quantity specified is designed to transfer the spacecraft between Earth and Mars.

More details on NEP can be found in Appendix D.

Table 5.4.3.2-1 NEP Vehicle Design Point

Component	Cargo (t)	Comments
Reactor	13.7	26.7 MWth
Shield	3.3	65 cm thick LiH
Turbines	20.8	produces 5 MWe
Radiator	20.3	3694 square meters
Propulsion	23.5	MPD thrusters, 3.65 MW, 6000 sec I_{sp}
Structure	10.0	single truss
Propellant	167.6	argon
Tankage	16.8	
Payload	400.0	one-way, returns to Earth empty
TOTAL	676.0	

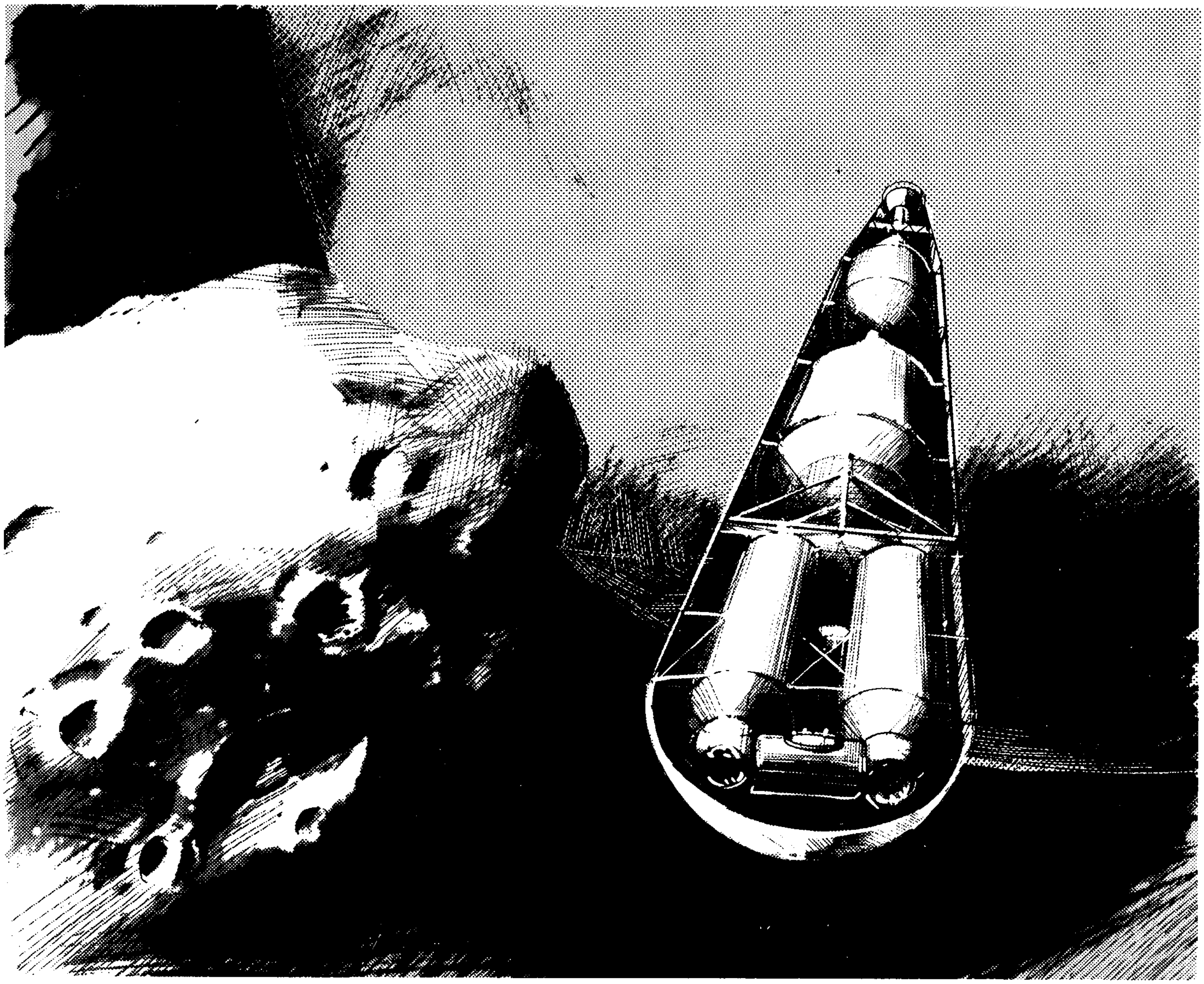


Figure 5.4.3.2-1 MPD NEP Cargo Vehicle (Artist's Conception)

5.4.4 Solar Electric Propulsion (SEP)

Four Mars Cargo Vehicle (MCV) options were sized for the Phobos gateway to compare a SEP vehicle to a LH_2/LOX propelled vehicle. The mission is from a 300 km LEO to Phobos. The assumptions made for the comparison are summarized below:

Mission assumptions:

1) payload delivered to Mars = 100 t
2) low thrust ΔVs: 15.3 km/s for LEO to Phobos, 7.78 km/s Earth escape to Phobos
3) coplanar injections and insertions

SEP assumptions:

1) alpha = 31 kg/kWe total
2) I_{sp} = 2800 seconds
3) power = 500 kWe
4) efficiency = 0.58
5) tank fraction = 0.05 of propellant mass
6) structural fraction =0.015 of gross mass
7) SEP assumed to be in continual sunlight with average solar radiation of 1.2 AU

Chemical assumptions:

1) I_{sp} = 480 seconds
2) stage sized for C3 = 10 km^2/s^2
3) aerobraking at Mars (with chemical apogee kick for Phobos circularization)
4) tank fraction = 0.075 of propellant mass
5) structural fraction = 0.015 of gross mass
6) initial stage T/W = 0.2
7) bare engine T/W = 75
8) aerobrake mass = 0.15 of braked mass (AKM + 100 t payload)

The four options considered were:

A) all solar electric stage from LEO to Phobos.
B) combination of chemical and SEP stages. The chemical stage boosts the SEP stage and payload from LEO to 3000 mi circular orbit to bypass the inner Van Allen belt, decrease solar occultation time, and decrease the flight time. The SEP system then takes the payload from 3000 mi orbit to Phobos with continuous thrust.
C) all chemical stages for both the TMI and the circularization at Phobos. This option uses an aerobrake for the initial capture at Mars leaving the payload in an 80 by 5976.5 km orbit. Both the aerobrake and circularization stage masses are included for the TMIS calculation at Earth.
D) combination of chemical and SEP stages. The chemical stage boosts the SEP stage from LEO to a C3 of zero (Earth escape). The SEP stage then takes over and delivers the payload to Phobos by continuous thrust.

Figure 5.4.4-1 shows the results of the four options with LEO masses (excluding the payload mass) and total flight times. It is clear that option D is not a viable choice, because of the fact that it results in more LEO mass than the all-chemical option. The remaining options (A, B, and C) provide a trade-off between flight time and initial mass. Option A yields the lowest initial mass (about half that of the chemical option) but has a flight time of 3.6 years. The intermediate case (option B) results in an intermediate flight time and initial mass. The all-chemical stage (option C) provides by far the shortest flight times but has a high a mass fraction (2.5 times the payload mass).

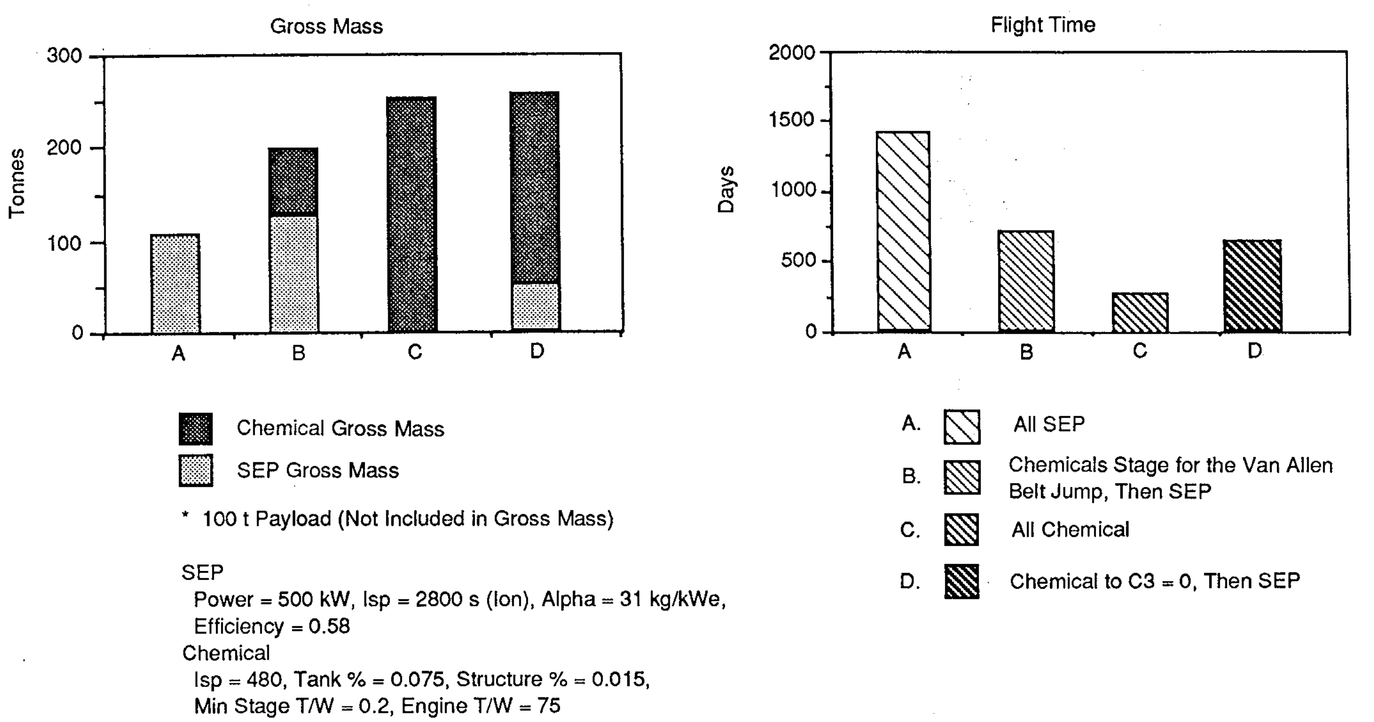


Figure 5.4.4-1 Total LEO Masses for Phobos Rendezvous using SEP

Several changes can be made to the SEP system to improve its performance. If the I_{sp} is increased, the propellant mass is reduced, but at the cost of linearly increasing the flight time. If power is increased, the flight time decreases, increasing the inert mass and the overall mass in LEO. The only factors that decrease both flight time and LEO mass are overall efficiency and specific mass. Both values used in this study are conservative and represent near-term technology (1991).

The overall conclusion of this study was that the use of SEP can reduce LEO mass, but at a penalty of longer delivery times. The optimal stage configuration depends upon relative priorities between flight times, acceptable radiation doses, and launch masses.

6.0 NODE SUPPORT AND ON-ORBIT VEHICLE ASSEMBLY

The types of node support provided in LEO will depend on the infrastructure assumptions attendant with the case study. In some cases, little infrastructure is assumed, such as for Case Study 1 (which had no node), whereas in most other case studies a full-up major LEO node was assumed to be available or to be provided. Space Station Freedom and independent nodes have been considered. A variety of infrastructure elements are applicable to this functional need (Table 6-1).

6.1 FREE-FLYER NODES

In certain case studies, a fully independent node was considered. Although the Transportation Integration Agent did not have a major responsibility here, an example has been provided of a method of accommodating the build-up of a major vehicle on a gravity-gradient stabilized truss. In the scenario outlined in Figures 6.1-1 through 6.1-3, the "T"-shaped framework allows the assembly of a Mars transportation vehicle alongside. The aerobrake is assembled from petals by an attached robot arm. Each of several launches brings up elements as well as increments of LOX propellant. The final launch brings up the liquid hydrogen, since it is the most susceptible to boil-off losses. The assembly backbone contains all the necessary support functions, such as communications and power generation.

Table 6-1 Candidate LEO Infrastructure

Space Station Freedom	**Orbital Maneuvering Vehicle (OMV)**
Technology: ECLSS demo; Hab/Lab/Log modules	Assembly and transport of large components
Research: Micro and artificial gravity effects	Personnel trans between facility and Mars spaceship (MSS)
Facility Support Services: as a node for support serv, including	**Robotic Equipment**
- Astronauts (EVA and Mars mission habitats IVA)	Flight Telerobotic Servicer (FTS); Mobile Servicing Platform (MSP)
- Robots and manipulators: FTS and MSP	**NSTS (Shuttle)**
- Vehicles: OMV	Personnel transportation to/from Earth
- Safe haven, rescue, emergency return-to-Earth (CERV)	On-orbit assembly of modules (assist via RMS)
- Checkout or repair of equipment prior to final installation	Cargo delivery of equipment, consumables, or small elements
- Stores and spares (consumables and equipment)	Return of hardware to Earth
Fluids Depot	**HLLV**
Cryopropellants; Storable propellants (for topping off after use)	Delivery of major elements
Inert gases (including pressurants)	Delivery of propellant; aerobrakes
	Delivery of large (e.g., 25-foot diameter) modules (or, by ACC)

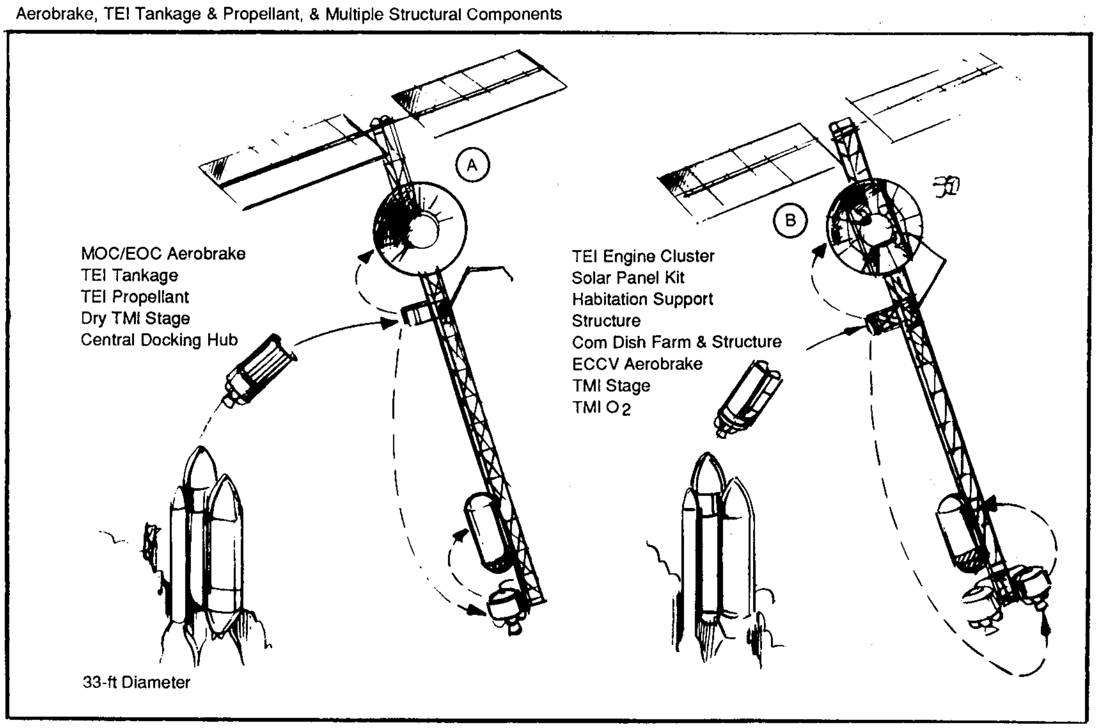


Figure 6.1-1 Assembly Sequence on a Free-flyer Node (Part 1 of 3)

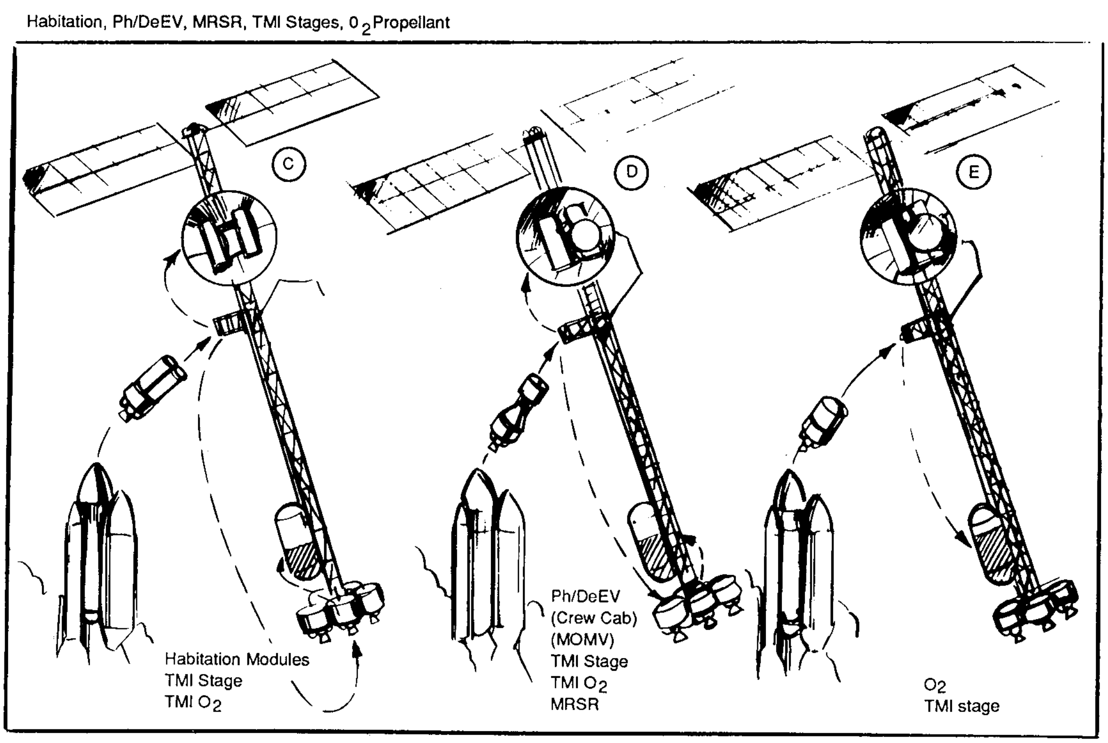


Figure 6.1-2 Assembly Sequence on a Free-flyer Node (Part 2 of 3)

Final O_2 Tanker Flight & H_2Tanker Flight

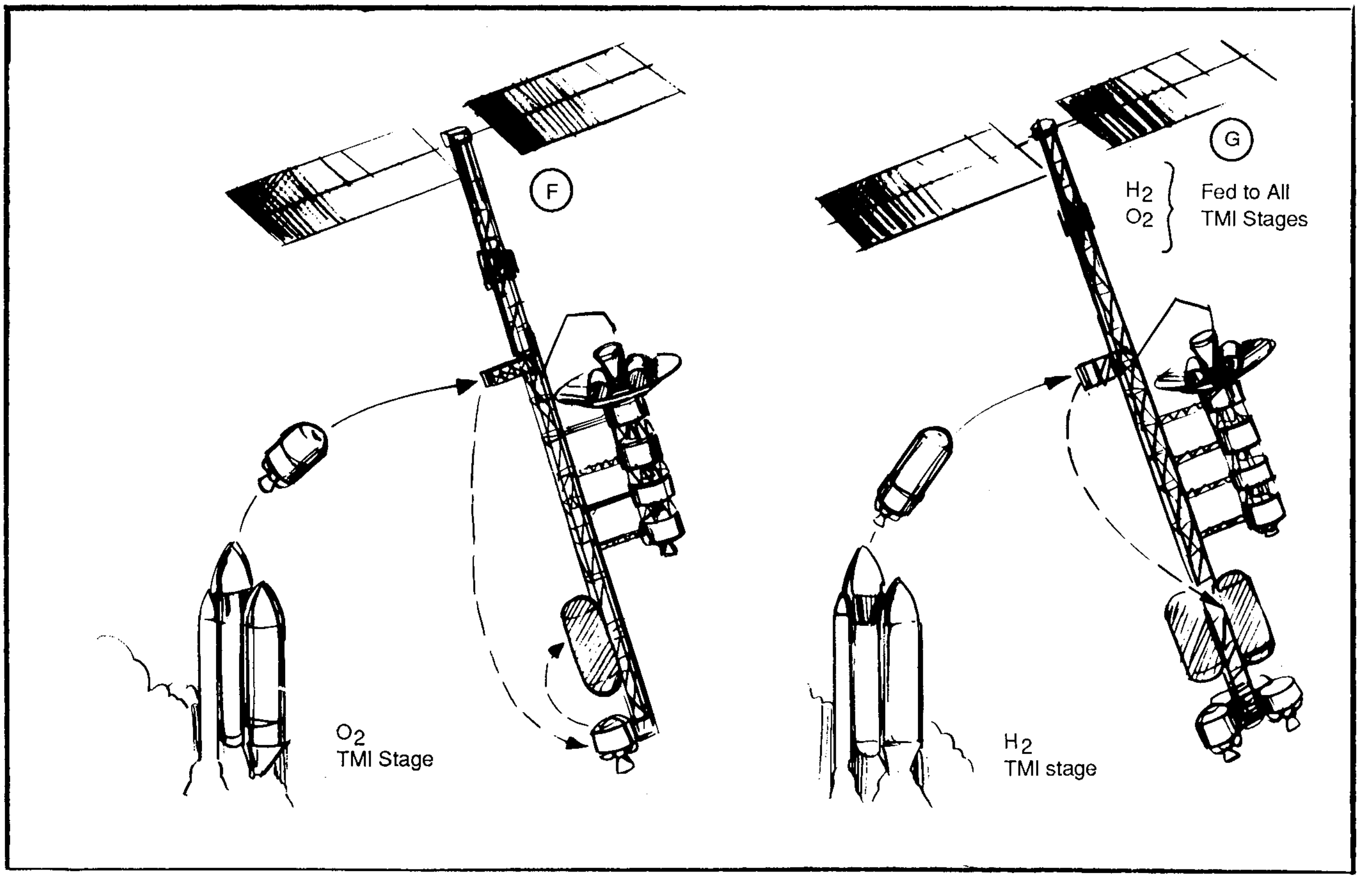


Figure 6.1-3 Assembly Sequence on a Free-flyer Node (Part 3 of 3)

6.2 UTILIZATION OF SPACE STATION FREEDOM

A set of options for use of Freedom Station in support of these human exploration missions has been developed. These are based on usages that vary from minimum involvement to the extreme of applying Freedom as a transportation node with full support and servicing of vehicles. In order to bound the problem in a manageable way and to allow a derivation of needs and ultimately to set requirements, the applications have been analyzed in terms of three levels: minimum usage, nominal usage, and maximum usage. These "min/ nom/ max" levels are then analyzed across the functional support areas derived for an orbiting node. In Table 6.2-1, we have provided the guidelines for the selected definitions of each of these levels. In the subsequent tables, these guidelines are applied and the resultant options described accordingly. It should be noted that for any given future mission or case study scenario, a combination of these capabilities may be necessarily applied to different areas to meet case study levied requirements.

Table 6.2-1 Min/Nom/Max Usage Guidelines

Minimum ("min"): No use of Freedom Station as a Node. Only use of Freedom is for R&TD to enable manned Mars missions and certain aspects of Lunar Base.
Nominal ("nom"): Use of Freedom Station as a Node, but with relatively little impact.
Lunar vehicles are stored at Freedom Station, but re-fueled at separate Depot.
Mars vehicles are free-flyers in vicinity of node.
Conveyance between Freedom and Mars vehicles is via use of the ECCV, STS, PhEV,
MAV, a manned OMV or other manned spacecraft.
The Mars vehicle functions as Command Central as soon as it becomes habitable.
Maximum ("max"): Full Freedom Station support at the maximum envisioned level.
Lunar vehicles are constructed/assembled and are fueled and fully maintained, including major refurbishment.
Mars vehicles are assembled at Freedom and remain attached until completed.
Freedom is Command Central. Vehicles are recovered and refurbished.

Functional applications are delineated in Table 6.2-2. This provides the list of applications which must be addressed in detail in order to scope the use of Space Station Freedom for the levels of use under consideration.

The approach that is taken to analysis of these options is given in Table 6.2-3. Tables 6.2-4 through 6.2-28, which follow provide the results of this approach. Lunar missions are summarized in Tables 6.2-29 through 6.2-31, and Mars missions are summarized in Tables 6.2-32 through 6.2-34.

Table 6.2-2 Functional Applications of Freedom Station to Exploration Missions

Technology	**Repository Utilization**
- Flight demo and verification of Advanced ECLSS	- Consumables Stores
- Hab/Lab/Log Modules and Nodes (designs)	- Equipment and Spares
Research	**Astronaut Safety**
- Microgravity Effects and Countermeasures	- Safe Haven Support
On-orbit Assembly	- Rescue Capacity (incl. return-to-Earth)
- Direct use of Freedom Station	**Propellant and Fluids Handling/Storage**
- as support to STS Orbiter and RMS	- Cryopropellant Depot
On-orbit Construction	- Fluid Transfers
- Aerobrake construction	**Earth Departure Launch Node**
- Other: large scale solar cell arrays; habitat permanent seals; etc.	- TMI and/or TLI Launch Support
On-orbit Storage	- Command and Control
- Hangar Protection	**As the Earth Planetary Return Node**
Support Personnel	- Retrieval (spaceship, astronauts)
- EVA Support	- Refurbishment (spaceship)
- IVA Support	**Support Services**
	- Telerobotic Support
	- Flight Crew Transfers

Table 6.2-3 Approach to Analysis

Lunar Missions	
Min	Apollo-like (no requirement for Freedom Station)
Nom	Aspects of Case Study 2.0 of FY88 > Protective LEO storage; propellant refueling of vehicle
Max	Case Study 2.1 of FY 89 > Full maintenance, including module/major component change-outs
Mars Missions	
Min	"No nodes" Mars mission
Nom	Phobos Gateway Study (for NASA/Headquarters)> one opposition and three conjunction missions > IMLEO = 575 t to 700 t per mission
Max	Case Study 2.0 of FY88, Human Expeditions to Mars> three split/sprint 8-crew missions> IMLEO total = 6600 t over a period of six years

Table 6.2-4 Flight demo and verification of Advanced ECLSS

Lunar Missions (note: these comments apply to Lunar Base, not transportation)	
Min:	On-orbit operation, with specialized diagnostics equipment to monitor performance. Not used to support human life or substitute for mainline Freedom ECLSS.
Nominal:	On-orbit *use* of individual, selected LSS components (e.g., CO_2 removal, oxygen regeneration, water recycling, etc.), with diagnostics of performance and reliability, for one month.
Max:	On-orbit *use*, as a total ECLSS system, for a minimum of one <u>month</u>.
Mars Missions	
Min:	On-orbit operation, with specialized diagnostics equipment to monitor performance. Not used to support human life or substitute for mainline Freedom ECLSS.
Nominal:	On-orbit *use* of individual, selected LSS components (e.g., CO_2 removal, oxygen regeneration, water recycling, etc.), with diagnostics of performance and reliability, for 6 months.
Max:	On-orbit *use*, as a total ECLSS system, for a minimum of one <u>year</u>.

Table 6.2-5 Hab/Lab/Log Module Designs

Lunar Missions (note: these comments apply to Lunar Base, not transportation)	
Min:	Use Freedom Hab module design, with no modifications. Use Hab module design for fabrication of Lunar surface habitat modules.
Nominal:	Remove racks. Customize interiors. No checkout of new module designs at Freedom.
Max:	Specialized designs. Early hab module sent to Space Station Freedom for integration and checkout to verify operability and habitability.
Mars Missions	
Min:	No modifications to modules. Use Hab module design for fabrication of Mars spaceship habitat modules.
Nominal:	Remove racks. Customize interiors. No checkout of new module designs at Freedom.
Max:	Specialized lightweight designs. Early hab module sent to Space Station Freedom for integration and checkout to verify operability and habitability. Removal of orbital debris shields prior to launch, *or* provide a hangar for shelter against debris (this requires a trade-off study by Space Station).

Table 6.2-6 Microgravity Effects on Humans and Countermeasures

Lunar Missions	
Min:	Not required for Lunar missions. STS flights yield adequate data.
Nominal:	N/A
Max:	N/A
Mars Missions	
Min:	Evaluate effect of exercise, diet, drugs, and any other countermeasure against the deleterious physiological effects of microgravity (bone demineralization, muscle atrophy, cardiovascular deconditioning, vestibular dysfunction, immune system shifts, etc.)
Nominal:	Same as "min", but add small-animal research project to evaluate countermeasures, including on-board periodic centrifugation (which could be upscaled to humans)
Max:	Provide human-rated centrifuge *or* artificial gravity environment (spinning modules with minimum of 56-ft swing radius). Evaluate artificial gravity capability to protect human health and well-being over extended duration (6 months). *Note*: A method of accomplishing artificial gravity testing *without* utilization of Freedom is via early habitation of the Mars spaceship module system.

Table 6.2-7 Use of Space Station Freedom

Lunar Missions	
Min:	none
Nominal:	see requirements for EVA and telerobotic support (e.g., FTS and OMV)
Max:	Attachment of crew cabs or other modular systems to propulsion modules. Requires manipulation of assemblies in the 5 to 15 t class.
Mars Missions	
Min:	none
Nominal:	see requirements for EVA and telerobotic support (e.g., FTS and OMV)
Max:	Multiple support for emplacement of major systems which were transported separately to Freedom Station (e.g., TEIS propulsion modules, photovoltaic arrays, ECCV, Science and ComSat packages). Assumed to require special systems to manipulate subassemblies in the 10 to 50 t class.

Table 6.2-8 Freedom Station in conjunction with STS

Lunar Missions	
Min:	none
Nominal:	none
Max:	Multiple docking/berthing of STS in vicinity of attached spaceship
Mars Missions	
Min:	none
Nominal:	Support STS for use of RMS in on-orbit assembly of free-flyer spaceship
Max:	Multiple docking/berthing of STS in vicinity of attached spaceship

Table 6.2-9 Aerobrake Construction

Lunar Missions	
Min:	none
Nominal:	none (aerobrake deployed)
Max:	Major joining of pre-fabricated aerobrake sections. Use slice or pie-section models of aerobrake construction. Assume need for large-mass precision positioning, fastening using bolts, pins, welding or bonding, and quality assurance inspections. Aerobrake sizes in the 50 to 80-ft class.
Mars Missions	
Min:	none
Nominal:	none (all-up or self-deployable aerobrake)
Max:	Major joining of pre-fabricated aerobrake. Use slice or pie-section models of aerobrake build-up. Assume need for large-mass precision positioning, fastening using bolts and pins (no welding or bonding), and quality assurance inspections. Aerobrake sizes in the 70 to 135-ft class.

Table 6.2-10 Other Construction

Lunar Missions	
Min:	none
Nominal:	none
Max:	none identified for Transportation.
Mars Missions	
Min:	none
Nominal:	none
Max:	Provide vacuum-tight sealing of joined members, using welding or adhesive fillers. Assume both EVA and IVA operations required, but with labor covered in "Personnel Support" requirements.

Table 6.2-11 Use of Space Station Freedom

Lunar Missions	
Min:	none
Nominal:	see requirements for EVA and telerobotic support (e.g., FTS and OMV)
Max:	Attachment of crew cabs or other modular systems to propulsion modules. Requires manipulation of assemblies in the 5 to 15 t class.
Mars Missions	
Min:	none
Nominal:	see requirements for EVA and telerobotic support (e.g., FTS and OMV)
Max:	Multiple support for emplacement of major systems which were transported separately to Freedom Station (e.g., TEIS propulsion modules, photovoltaic arrays, ECCV, Science and ComSat packages). Assumed to require special systems to manipulate subassemblies in the 10 to 50 t class.

Table 6.2-12 Freedom Station in Conjunction with STS

Lunar Missions	
Min:	none
Nominal:	none
Max:	Multiple docking/berthing of STS in vicinity of attached spaceship
Mars Missions	
Min:	none
Nominal:	Support STS for use of RMS in on-orbit assembly of free-flyer spaceship
Max:	Multiple docking/berthing of STS in vicinity of attached spaceship

Table 6.2-13 Aerobrake Construction

Lunar Missions	
Min:	none
Nominal:	none (aerobrake deployed)
Max:	Major joining of pre-fabricated aerobrake sections. Use slice or pie-section models of aerobrake construction. Assume need for large-mass precision positioning, fastening using bolts, pins, welding or bonding, and quality assurance inspections. Aerobrake sizes in the 50 to 80-ft class.
Mars Missions	
Min:	none
Nominal:	none (all-up or self-deployable aerobrake)
Max:	Major joining of pre-fabricated aerobrake. Use slice or pie-section models of aerobrake build-up. Assume need for large-mass precision positioning, fastening using bolts and pins (no welding or bonding), and quality assurance inspections. Aerobrake sizes in the 70 to 135-ft class.

Table 6.2-14 Other Construction

Lunar Missions	
Min:	none
Nominal:	provide hangar for orbital debris/micrometeoroid protection
Max:	Provide hangar for orbital debris/micrometeoroid protection; Protected environment for changeout of major assemblies
Mars Missions	
Min:	none
Nominal:	none
Max:	Provide hangar for orbital debris/micrometeoroid protection; Protected environment for changeout of major assemblies

Table 6.2-15 EVA Support

Lunar Missions	
Min:	none required
Nominal:	Vehicles launched all-up. Major need for Freedom crew member(s) to perform inspections of Lunar Vehicle upon each return to Earth. Size for total of 10 EVA's (4 hours productive work per EVA) per recertification. Provide 10 additional EVAs (at 6 hours productive work each), on the average, to allow for repairs or maintenance.
Max:	Major Freedom crew member involvement in construction, assembly, checkout, and test. Provide a total of 25 EVA's, at 6 hours each, per vehicle. Assume same vehicle maintenance as for Nominal case.
Mars Missions	
Min:	none required
Nominal:	Freedom crew member(s) perform inspections or assist Mars spaceship assembly/checkout crew in EVA work tasks. Size for a total of 15 EVA's, at 6 hours productive work per EVA.
Max:	Major Freedom crew member involvement in construction, assembly, checkout, and test. Provide a total of 75 EVA's, at 6 hours each.

Table 6.2-16 IVA Support

Lunar Missions	
Min:	none required
Nominal:	Freedom personnel board Lunar Vehicle upon arrival (whether from Earth or from Moon) to inspect, test, checkout equipment ("receiving inspection"). Provide two crew half-time for three weeks.
Max:	Freedom personnel provide installation and fabrication work for vehicle assembly and refurbishments. Assume two full-time crew members, with up to 10% utilization of four additional crew over a six month period per vehicle.
Mars Missions	
Min:	none required
Nominal:	Need for one dedicated crew member involvement in Mars spaceship assembly for support, monitoring, some teleoperation, and as a liaison with Freedom operations. Occasional support of up to 3 crew members, but at not more than a 10% impact on total Freedom crew time (excluding dedicated crew person)
Max:	Major Freedom crew member involvement in construction, assembly, checkout, and test. Assume 4 full-time crew members for 12 months, with part time utilization of up to 10% of remaining crew time.

Table 6.2-17 Telerobotic Support

Lunar Missions	
Min:	none required
Nominal:	Required only for on-orbit propellant transfer. · Size for one week of dedicated FTS application per vehicle refueling.
Max:	STS RMS, FTS and/or MSP. Assume 60 days of dedicated MSP and 20 days of FTS for construction, per vehicle.
Mars Missions	
Min:	none required
Nominal:	none required
Max:	Mobile Servicing Platform (MSP) and FTS Assume three STS visits with use of RMS, per Mars vehicle. Allow 30% of MSP per vehicle for FTS over a 6 month period. Dedicated use of MSP by Mars spaceship for 6 months, per vehicle.

Table 6.2-18 Flight Crew Transfers (Freedom <->Crew Vehicles)

Lunar Missions	
Min:	none required
Nominal:	Use tethered EVA *or* MMU/EVA *or* access tunnel IVA.
Max:	same as nominal
Mars Missions	
Min:	none required
Nominal:	Use STS *or* manned OMV *or* CERV *or* other
Max:	Use tethered EVA *or* MMU/EVA *or* access tunnel IVA.

Table 6.2-19 Consumables Stores

Lunar Missions (note: Surface IA may have major needs)	
Min:	none
Nominal:	Top-off consumables
Max:	All consumables transferred from on-board storage (Logistics Node or other appropriate storage). Assume 1 metric tonne of consumables: food, water, air revitalization chemicals (incl. LOX).
Mars Missions	
Min:	none
Nominal:	Top-off consumables
Max:	All consumables transferred from on-board storage (Logistics Node or other appropriate storage). Assume 10 metric tonnes of food (2 tonnes of which is frozen).

Table 6.2-20 Equipment and Spares

Lunar Missions	
Min:	none
Nominal:	Storage for set of critical spares only. Assume two double racks of volume. Provide minimal repair capabilities (one-rack lab space; crew time covered above; normal electronics lab tools)
Max:	Full set of spares, including tanks, engines, avionics packages, and components. Assume eight double-rack equivalents and 2 t of attached payload equipment.
Mars Missions	
Min:	none
Nominal:	Storage for set of critical spares to support launch only. Assume two double racks of volume. Provide minimal repair capabilities (one-rack lab space; crew time covered above; normal electronics lab tools)
Max:	Full set of spares, including double sets of ECLSS spare components. Extensive special-purpose equipment servicing facility. Assume 12 double-rack equivalents.

Table 6.2-21 Safe Haven Support

Lunar Missions	
Min:	none
Nominal:	Baseline Freedom availability, as required, but utilizing only normal support and safety services.
Max:	Enhanced medical facilities to handle major credible incidents resulting from construction accidents, propellant spills, etc. Safe haven for 8 additional crew.
Mars Missions	
Min:	none
Nominal:	Nominal Freedom Station available, as required
Max:	Enhanced medical facilities to handle major credible incidents resulting from construction accidents, propellant spills, etc. Safe haven for 5 additional crew.

Table 6.2-22 Rescue Capacity

Lunar Missions	
Min:	none
Nominal:	Responsibility of Lunar spaceship support crew
Max:	Standby vehicles and crew members, fully suited when necessary to backup critical and hazardous operations by Lunar crew. Rescue vehicle could be CERV or specialized vehicle, and could be autonomous, teleoperated, or piloted.
Mars Missions	
Min:	none
Nominal:	Responsibility of Mars spaceship support crew
Max:	Standby vehicles and crew members, fully suited when necessary to backup critical and hazardous operations by Mars crew. Rescue vehicle could be CERV or specialized vehicle, and could be autonomous, teleoperated, or piloted.

Table 6.2-23 Cryopropellant Depot

Lunar Missions	
Min:	none (transfers occur only between ETO propellant tankers and lunar vehicles in LEO)
Nominal:	No attached facility, but separate depot free-flying outside CCZ of Space Station Freedom.
Max:	Attached. Assume 75 metric tonne capacity of cryopropellant
Mars Missions	
Min:	none (transfers occur only between ETO propellant tankers and Mars vehicles in LEO)
Nominal:	No attached facility, but separate depot free-flying outside CCZ of Space Station Freedom.
Max:	Attached. Assume 500 metric tonne capacity of cryopropellant

Table 6.2-24 Fluid Transfers

Lunar Missions	
Min:	none
Nominal:	Gravity-assisted transfer (e.g. tether, spin-up, or propulsive acceleration) *or* zero-g propellant acquisition and transfer capability. Note: No requirement on Freedom except to support the transfers at Depot with EVA or teleoperated robot(s).
Max:	Active pumping, zero-g propellant acquisition. Mass gauging in microgravity.
Mars Missions	
Min:	none
Nominal:	Gravity-assisted transfer (e.g. tether, spin-up, or propulsive acceleration) *or* zero-g propellant acquisition and transfer capability. Note: No requirement on Freedom except to support the transfers at Depot with EVA or teleoperated robot(s).
Max:	Active pumping, zero-g propellant acquisition. Mass gauging in microgravity.

Table 6.2-25 TMI or TLI Launch Support

Lunar Missions	
Min:	none
Nominal:	none
Max:	Reboost or more direct propulsion to position spaceship for TLI. (note: reboost may require long lead times and thereby constrain Freedom Station altitude strategies). Freedom Station provides storage for rescue STV (spare piloted vehicle or cargo vehicle that can be outfitted within 7 days notice with a crew cab).
Mars Missions	
Min:	none
Nominal:	No phasing of orbit. However, Freedom Station includes a rescue vehicle for an aborted or incomplete TMI propulsive burn(s) (e.g., manned STV)
Max:	Reboost or more direct propulsion to position spaceship for TMI. (note: reboost may require long lead times and thereby constrain Freedom Station altitude strategies). Freedom Station includes a rescue vehicle for an aborted or incomplete TMI propulsive burn(s) (e.g., manned STV)

Table 6.2-26 Command and Control

Lunar Missions	
Min:	none (C&C from Earth and on-board computer control)
Nominal:	C&C central is directed by ground control, except when within the CCZ of Freedom.
Max:	Freedom Station directs mission.
Mars Missions	
Min:	none (C&C from Earth and on-board computer control)
Nominal:	C&C central is on Mars spaceship, directed by ground control, but only backed-up by Freedom monitoring
Max:	Same as nominal (no advantage to Freedom Station providing countdown and launch control instead of ground control)

Table 6.2-27 Retrieval

Lunar Missions	
Min:	none (astronauts recovered by direct entry to Earth)
Nominal:	Lunar Vehicle is brought to CCZ. Astronauts recovered via shirt-sleeve docking tunnel or via EVA to Freedom airlock.
Max:	same options as for Nominal case.
Mars Missions	
Min:	none (astronauts recovered by direct entry to Earth)
Nominal:	Provide retrieval vehicle (STV or OMV) for astronauts. Minimal special facilities for astronauts because of near-term transfer to STS for flight to Earth. Mars vehicle not recovered at Space Station Freedom.
Max:	Mars Spaceship recovered by dedicated vehicle (STV) and docked at Space Station Freedom for subsequent refurbishment and refueling. Astronauts recovered and given enhanced medical attention. Gradual readjustment to gravity possible by use of on-board centrifuge or artificial gravity vehicle. Provide temporary quarantine facilities. Provide sample storage and toxicity testing facilities. Assume two STS dedicated visits to bring up medical and de-briefing specialists.

Table 6.2-28 Refurbishment

Lunar Missions	
Min:	none (all return vehicles expendable)
Nominal:	Support equipment for minimal refurbishment—simple replacements and resupply of consumables (no major storage on Freedom Station required). Storage and access to 2 t of materiel and re-supply consumables.
Max:	Vehicle captured and re-attached to Freedom Station. Requirement for docking to a Freedom Station module or special port to allow continuous shirt-sleeve access. Freedom Station has facilities for repair, maintenance, and sufficient storage for replacement consumables (food, ECLSS soft goods, space suits, etc.) See requirements under "Repository Utility" and "Support Personnel" above. Use twice the original amount of IVA support, but assume EVA support needs are the same.
Mars Missions	
Min:	none (all return vehicles expendable)
Nominal:	Provide minimal services as a backup staging point for supplies and work crews.
Max:	Vehicle captured and re-attached to Freedom Station. Freedom Station has major facilities for repair, maintenance, and sufficient storage for replacement consumables (food, ECLSS soft goods, space suits, etc.) See requirements under "Repository Utility" and "Support Personnel" above. Use twice the original amount of IVA support, but assume EVA support needs are the same.

Table 6.2-29 "Min" Freedom/OEXP Interfaces, Lunar Missions

Technology	
Flight demo and verification of Advanced ECLSS	Demo only in Freedom. No LSS substitution
Hab/Lab/Log Modules and Nodes (designs)	Copy of module designs, no re-design
Research	
Microgravity Effects and Countermeasures	N/A to Lunar missions
On-orbit Assembly	
Direct use of Freedom Station, support with STS	none required
On-orbit Construction	
Aerobrake, Other (solar cell arrays; habitat permanent seals)	none required
On-orbit Storage	
Hangar Protection	none required
Support Personnel	
EVA Support, IVA Support	none required
Support Services	
Telerobotic Support, Flight Crew Transfers	none required
Repository Utilization	
Consumables Stores, Equipment and Spares	none required
Astronaut Safety	
Safe Haven Support	none required
Rescue Capacity (incl. return-to-Earth)	none required
Propellant and Fluids Handling/Storage	
Cryopropellant Depot	none required (transfers are HLLV-->LV)
Fluid Transfers	none required
Earth Departure Launch Node	
TLI Launch Support, C & C	none required (Earth-based C & C)
As the Earth Planetary Return Node	
Retrieval (spaceship, astronauts)	none required (astronauts direct entry)
Refurbishment (spaceship)	none required

Table 6.2-30 "Nom" Freedom/OEXP Interfaces, Lunar Missions

Technology Flight demo and verification of Advanced ECLSS Hab/Lab/Log Modules and Nodes (designs)	*Use* of selected ECLSS components customize interiors for Lunar Base applicability.
Research Micrогravity Effects and Countermeasures	Not applicable to Lunar missions
On-orbit Assembly Direct use of Freedom Station, support with STS	See requirements for EVA/telerobotic
On-orbit Construction Aerobrake, Other (solar cell arrays; habitat permanent seals)	None (aerobrake deployed). Minor support to STS Orbiter used at free-flyer
On-orbit Storage Hangar Protection	Provide hangar against orbital debris
Support Personnel EVA Support, IVA Support **Support Services** Telerobotic Support, Flight Crew Transfers	20 EVA's (4-6 hr) IVA: 2 crew, half-time, 3 wks One wk FTS for propellant transfer, per vehicle. Use EVA or access tunnel IVA for crew
Repository Utilization Consumables Stores, Equipment and Spares	Top-off consumables Critical spares, maintain 2 double-racks
Astronaut Safety Safe Haven Support Rescue Capacity (including. return-to-Earth)	No additional facilities
Propellant and Fluids Handling/Storage Cryopropellant Depot Fluid Transfers	Depot remains outside CCZ Minor support of transfers (see teleop)
Earth Departure Launch Node TLI Launch Support, C & C	no requirement, except when vehicle in CCZ
As the Earth Planetary Return Node Retrieval (spaceship, astronauts) Refurbishment (spaceship)	Astronauts via shirt-sleeve tunnel or EVA Minimal equipment: 2 t of supplies

Table 6.2-31 "Max" Freedom/OEXP Interfaces, Lunar Missions

Technology Flight demo and verification of Advanced ECLSS Hab/Lab/Log Modules and Nodes (designs)	Substitution of total ECLSS for 1 mo(LBase) customize interiors for Lunar Base applications.
Research Microgravity Effects and Countermeasures	Not applicable to Lunar missions
On-orbit Assembly	Manipulation of assemblies, 5-15 t. Direct use of Freedom Station, support with STS Multiple docking of STS near OEXP vehicles
On-orbit Construction Aerobrake, Other (solar cell arrays; habitat permanent seals)	Joint pre-fabricated aerobrake (50-80 ft dia)
On-orbit Storage Hangar Protection	Protected environment for changeouts Provide hangar against orbital debris
Support Personnel EVA Support, IVA Support	25 EVA's (6 hr) per vehicle IVA: 2 crew, full-time; 4 crew, 10%-time
Support Services Telerobotic Support, Flight Crew Transfers	60 d MSP, 20 d FTS per vehicle. Use EVA or access tunnel IVA for crew
Repository Utilization 2 t critical spares, maint.; 8 double-racks	1 t consumables Consumables Stores, Equipment and Spares
Astronaut Safety Safe Haven Support Rescue Capacity (including return-to-Earth)	Enhanced medical facilities (major accident) Standby rescue vehicle, EVA astronauts
Propellant and Fluids Handling/Storage Cryopropellant Depot Fluid Transfers	Attached to Freedom, 75 t cryopropellant Pumped, zero-g acquisition. Gauging.
Earth Departure Launch Node TLI Launch Support, C & C	Reboost strategy, TLI staging. Rescue STV Freedom directs mission
As the Earth Planetary Return Node Retrieval (spaceship, astronauts) Refurbishment (spaceship)	Astronauts via shirt-sleeve tunnel or EVA Vehicle returns. Double IVA, same EVA

Table 6.2-32 "Min" Freedom/OEXP Interfaces, Mars Missions

Technology Flight demo and verification of Advanced ECLSS Hab/Lab/Log Modules and Nodes (designs)	 Demo only in Freedom. No LSS substitution Copy of hab module design (no re-design)
Research Microgravity Effects and Countermeasures	Evaluate exercise, diet, drugs, other countermeasures
On-orbit Assembly Direct use of Freedom Station, support with STS	 none required
On-orbit Construction Aerobrake, Other (solar cell arrays; habitat permanent seals)	 none required
On-orbit Storage Hangar Protection	 none required
Support Personnel EVA Support, IVA Support	 none required
Support Services Telerobotic Support, Flight Crew Transfers	 none required
Repository Utilization Consumables Stores, Equipment and Spares	 none required
Astronaut Safety Safe Haven Support Rescue Capacity (incl. return-to-Earth)	 none required none required
Propellant and Fluids Handling/Storage Cryopropellant Depot Fluid Transfers	 none required (transfers are HLLV-->MSS) none required
Earth Departure Launch Node TMI Launch Support, C & C	 none required (Earth-based C & C)
As the Earth Planetary Return Node Retrieval (spaceship, astronauts) Refurbishment (spaceship)	 none required (astronauts direct entry) none required

Table 6.2-33 "Nom" Freedom/OEXP Interfaces, Mars Missions

Technology Flight demo and verification of Advanced ECLSS Hab/Lab/Log Modules and Nodes (designs)	 *Use* of selected ECLSS components, 6 mo. customize interiors for Hab module
Research Microgravity Effects and Countermeasures	Same as "min", but add small-animal lab and centrifuge
On-orbit Assembly Direct use of Freedom Station, support with STS	 See requirements for EVA/telerobotic
On-orbit Construction Aerobrake, Other (solar cell arrays; habitat permanent seals)	None (aerobrake deployed). Minor support to STS Orbiter used at free-flyer
On-orbit Storage Hangar Protection	 none required
Support Personnel EVA Support, IVA Support	15 EVA's (6 hr productive work) IVA: 1 crew, full-time; 3 crew, 10%-time
Support Services Telerobotic Support, Flight Crew Transfers	No telerobotic support Use CERV, manned OMV, or MSS vehicle.
Repository Utilization Consumables Stores, Equipment and Spares	Top-off consumables Critical spares, maintain 2 double-racks
Astronaut Safety Safe Haven Support Rescue Capacity (incl. return-to-Earth)	 No additional facilities
Propellant and Fluids Handling/Storage Cryopropellant Depot Fluid Transfers	 Depot remains outside CCZ Minor support of transfers (see teleop)
Earth Departure Launch Node TMI Launch Support, C & C	No requirement, except rescue aborted TMI C & C is on Mars Spaceship (MSS)
As the Earth Planetary Return Node Retrieval (spaceship, astronauts) Refurbishment (spaceship)	 none required Minimal support as a staging point

Table 6.2-34 "Max" Freedom/OEXP Interfaces, Mars Missions

Technology Flight demo and verification of Advanced ECLSS Hab/Lab/Log Modules and Nodes (designs)	 Substitution of total ECLSS for 1 year Specialized light-weight design, custom.
Research Microgravity Effects and Countermeasures	 Provide ≥56-ft radius external art-g habitat
On-orbit Assembly Direct use of Freedom Station, support with STS	Manipulation of assemblies, 10-50 t Multiple docking of STS near OEXP vehicles
On-orbit Construction Aerobrake, Other (solar cell arrays; habitat permanent seals)	Vacuum-tight sealing; PVPA arrays Join pre-fabricated aerobrake (70-135ft dia)
On-orbit Storage Hangar Protection	Protected environment for changeouts Provide hangar against orbital debris
Support Personnel EVA Support, IVA Support	75 EVA's (6 hr) per vehicle IVA: 4 crew, full-time; 4 crew, 10%-time
Support Services Telerobotic Support, Flight Crew Transfers	100% MSP, 30% FTS, 6 mo.; per vehicle. Use EVA or access tunnel IVA for crew
Repository Utilization Consumables Stores, Equipment and Spares	10 t consumables 15 t critical spares, maint.; 12 double-racks
Astronaut Safety Safe Haven Support Rescue Capacity (incl. return-to-Earth)	5 additional crew (safe haven) Enhanced medical facilities (major accident) Standby rescue vehicle, EVA astronauts
Propellant and Fluids Handling/Storage Cryopropellant Depot Fluid Transfers	 Attached to Freedom, 500 t cryopropellant Pumped, zero-g acquisition. Gauging.
Earth Departure Launch Node TMI Launch Support, C & C	Reboost strategy, TMI staging. Rescue STV Freedom directs first and last wk of mission
As the Earth Planetary Return Node Retrieval (spaceship, astronauts) Refurbishment (spaceship)	 Astronauts quarantine, rehab facilities Vehicle returns. Double IVA, same EVA

7.0 AEROASSIST

Aeroassist for human exploration missions is the use of aerodynamic braking in a planetary atmosphere to efficiently reduce the orbital energy of a spacecraft. In the case of a hyperbolic encounter with a planet, such an atmospheric maneuver can be used to capture (aerocapture) a spacecraft into a closed park orbit. The same technique is applied to landing on the surface of a planet from an initial closed park orbit where velocity reduction in the atmosphere slows the vehicle for terminal descent. The MMSS investigated aeroassist as a means of reducing the overall IMLEO. A fairly wide range of encounter velocities with Mars and Earth were considered, as well as their implications. A variety of aerobrake shapes were considered, with lift to drag ratios (L/D) of 0.2 to 1.0. The use of artificial-g conditioning in the cruise phases of the mission was important for it could result in the crew being able to withstand the higher g levels that resulted from some of the high-energy encounter missions. Multipass capture was important at Earth to reduce the peak loads and heating that were encountered. At Mars, the use of a one sol park orbit was important to reduce the trans-Earth injection burn requirements. This also has the effect of reducing deceleration loads for the aerocapture maneuver.

The lunar mission studies used aeroassist in the Earth's atmosphere to enable the efficient capture of the reusable propulsion stages. Because the aeroassist maneuver is less strenuous than a direct entry to the surface of the Earth, reusable flexible insulator TPS technology was used rather than the ablators that were used on Apollo. The aerobrake sizes were primarily set by wake impingement constraints on the propulsive stage because of the generally long dimensions of those vehicles. Deceleration loads in the aeroassist maneuver were kept below 4 g's through the use of a slightly higher L/D than would be required for pure error management as well as the use of load relief trajectory control techniques.

The Mars Expedition Case study considered a manned Mars mission that minimized the use of new technology. By expending hardware as it went this mission was able to reduce its IMLEO mass requirements. A result of this was that aerobrakes did not have to be reused and that each aero device was fresh when employed. Because the Earth capture only involved the recovery of crew, a fairly simple Apollo command module approach was used, which returned only a small crew cabin. This mission study used sprint class transfer trajectories that minimize the time spent in cruise but also result in fast encounters with Mars and the Earth. The encounter C3s for this study were 60 at Mars and 116 km^2/sec^2 at Earth.

The wide range of missions envisioned for the Mars Evolutionary Case Study represented a more complex mission requirement. In this study class, reusable spacecraft that perform round trip Mars missions were a central theme. The multiple flight opportunities required adaptable packaging for the cruise configuration vehicle. The use of artificial gravity in transit allowed the crew to maintain better conditioning for the aerocapture deceleration loads. The use of more advanced technology to achieve these goals was indicated. This included the use of low L/D symmetric brakes that afford good packaging capability. The groundruled Mars encounter C3 for this study phase was 60 km^2/ sec^2, the same as for the Mars Expedition study. This value is significantly higher than the minimum C3s for conjunction class missions of about 10 km^2/sec^2. As will be seen later, this represents a driver for entry deceleration g-loads. Earth encounter C3 was 25 km^2/sec^2.

Aerocapture error analysis is crucial to establishing L/D requirements, which in turn is a major driver of vehicle configuration and constraints. It is of fundamental importance to establish the level of control required to control the entry trajectory. This analysis was performed for a range of entry conditions and concluded that a minimum L/D of 0.2 was required for Mars and Earth aerocapture and an L/D of 0.14 was needed for lunar return aeroassist. The use of excess lift for inclination changes is not an optimum solution since it is more mass efficient to perform the plane changes propulsively at the apoapsis of the park orbit. The use of symmetric blunt cone configurations at these levels of L/D minimizes the construction difficulties and gives good packaging characteristics. The highest encounter energies at these L/D levels result in a Mars capture peak load of 8.6 g's. Whether the crew can be sufficiently conditioned before entry to accept these loads is an open issue that requires better definition. Solutions include reducing the encounter energy below C3 values of 38 km^2/sec^2 or the use of high L/D biconic aeroshells that invoke much more stringent packaging constraints.

7.1 AEROCAPTURE L/D

Control of the entry process of an aerocapturing vehicle is critical to establishing the proper exit conditions. Previous work on flight vehicles (Gemini, Apollo, Shuttle, Viking, etc) as well as prior aerocapture studies (AFE, AOTV, MRSR) has shown that the most efficient control method is the use of a stable trim angle of attack that produces lift. This is in contrast to drag control techniques such as variable surface area flaps, aerospike, and the variable-volume ballute all of which have inadequate control margins. The lift vector orientation is controlled to produce trajectory changes through the use of a closed loop guidance process. The L/D of a lifting entry vehicle has critical effects on the configuration of the entire system. High L/D vehicles (greater than about 0.5) require the use of biconic shapes, with attendant packaging constraints. Between about 0.3 and 0.5 a raked ellipse may be used, such as the AFE flight experiment. Below an L/D of 0.3, symmetric cones may be used. There is much to be gained by reducing the analysis and manufacturing complexity through use of symmetric configurations. The shape of the aerobrake also has a major effect on the configuration of the other vehicle elements and thus the L/D requirements must be understood at more than a shallow level. A complete assessment of the entry errors is thus required to establish acceptable L/D levels.

The highly elliptical one sol Mars park orbit (250-km periapsis x 33800-km apoapsis) used throughout the study is an efficient location for the relatively heavy Earth return spacecraft because it is higher in the gravity well (thus minimizing the Mars departure propellant required) while still maintaining a low periapsis that is close to the atmosphere for aerocapture. In addition, this type of loose orbit reduces the loads and heating of aerocapture. The one sol orbit is also quite efficient for performing propulsive plane change, requiring only about 8 meters per second for every degree of plane change at apoapsis. Because of this efficiency, plane changes are better performed with engines rather than with aerodynamic lift in the entry phase. It is not optimum to perform plane change in the atmosphere, where higher L/D and TPS requirements increase the aerobrake weight faster than propellant savings are realized. Thus, the fundamental L/D requirements are only those necessary to control the exit apoapsis and correct for atmospheric flight errors.

Feasibility of aerocapture maneuvers is established by a process of assessing basic control levels through parametric and associated error analysis, followed by closed loop simulations to verify control adequacy. The following discussion traces an example of this process using the Mars aerocapture at an encounter C3 of 60 km^2/sec^2.

Precision encounter navigation is required to effectively control aerocapture into the desired Mars orbit. Although Earth-based radionavigation is available, onboard optical navigation using cameras and/or other celestial trackers will almost certainly be required for man-rating requirements because as with the Apollo lunar missions, the possibility of loss of ground communications requires an independent on-board navigation capability. Figure 7.1-1 shows the accuracies achievable for optical tracking of the martian moon Deimos during the terminal encounter phase for a variety of encounter C3s. For navigation measurements terminated at entry minus 6 hours, the uncertainty in predicted periapsis altitude is ±0.52 km. For continuing observations, the uncertainty is even less.

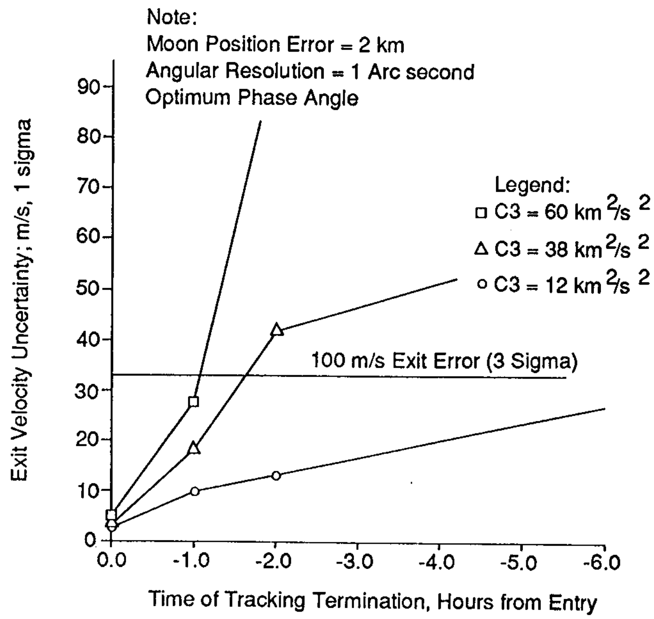


Figure 7.1-1 Mars Encounter NAV, Optical Measurements of Deimos

Figure 7.1-2 shows aerocapture control parametrics for the one sol park orbit capture case with an encounter C3 of 60 km^2/sec^2. The data plots regions of correctable vacuum periapsis altitude versus L/D as defined by the limiting conditions of continuous lift up and lift down boundary trajectories. Vacuum periapsis is used as a normalizer because it is directly relatable to the encounter targeting. Notice that many of the vacuum periapsis values are negative for the higher L/Ds. In these cases a lift vector up orientation rapidly pulls the vehicle away from the ground, the actual periapsis altitude as flown always being positive (no ground impact). The gap between these control limits is called the entry control corridor, which increases from zero at a zero L/D to around 260 km for a 1.0 L/D. A given vehicle L/D is capable of operating within the vacuum periapsis limits shown while still maintaining the desired exit orbit. The other information on the parametric charts shows the increase in peak deceleration loads for the lift up boundary. It may be seen that if the entire control capability of a 1.0 L/D vehicle is used it results in very high loads (43 g's). Thus the use of high L/D must generally be tempered with the fact that most of its control capability is unusable.

These parametrics are then compared with the results of error analysis to derive minimum L/D requirements.

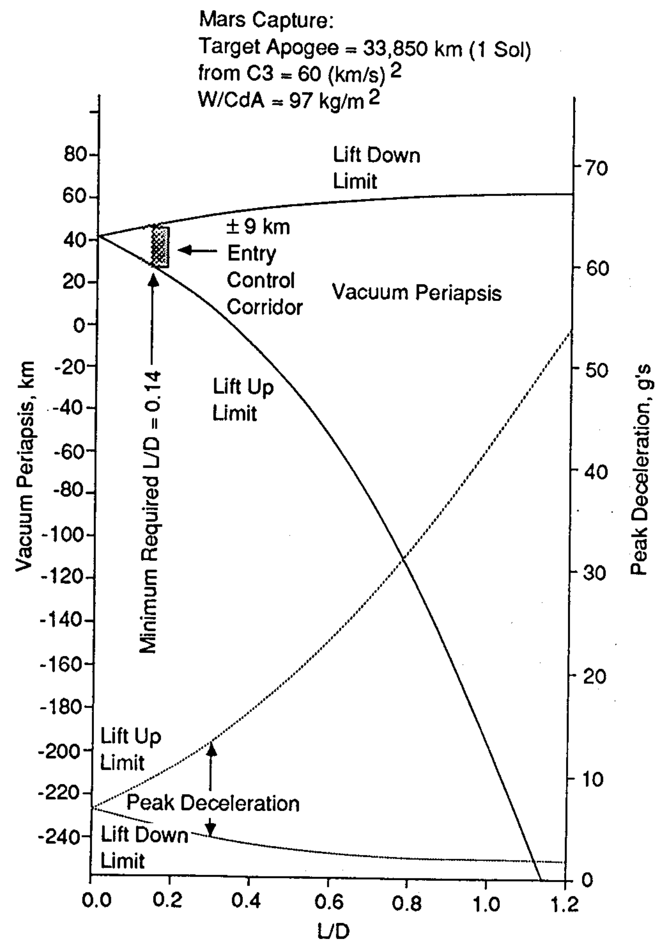


Figure 7.1-2 Mars Aerocapture Parametrics, C3 = 60 $km^2/seconds^2$

Table 7.1-1 summarizes the results of error assessments conducted for the C3=60 aerocapture phase. The principle errors affecting the encounter/entry phase are grouped into entry targeting and entry aerodynamics. These errors estimated for the 3 sigma uncertainty level and are normalized to equivalent vacuum periapsis altitude as well as entry flight path angle. This latter measure is the variation in inertial flight path angle evaluated at a reference altitude of 125 km. The navigation uncertainty (±0.52 km) corresponds with an onboard autonomous system able to make planet/moon observations up to 6 hours before entry as was described above. The terminal correction burn that performs final targeting is assumed to occur 5 hours from entry to keep execution errors low. Using representative inertial measurement unit errors and an inertial alignment of 0.1°, the errors associated with this burn are ±1.10 km at periapsis.

Table 7.1-1 Mars Aerocapture Erro Analysis, C3 = 60 km²/seconds²

	Equivalent Periapsis Variation		Comments
	Kilometers	Flt Path angle at 125 km	
Targeting Errors			
Final Midcourse (E-5 hr)			Midcourse Executed at Entry -5 hr
Execuation Errors	± 1.10 km	± 0.08°	1% Error on 6 Mps Maneuver
Navigation Errors	± 0,52 km	± 0.04°	NavSat Tracking, 3 Sigma
Aerodynamic Variation			
Atmospheric Variation	± 4.91 km	± 0.33°	± 50% Density Variation
L/D Uncertainity	± 4.82 km	± 0.33°	± 2.0° at 12° Angle of Attack (±17% L/D)
RSS of Variations	± 6.98 km	± 0.48°	
RSS + 30%	± 9.08 km	± 0.62°	

Notes:
For C3 = 60 km2/sec2, L/D is Minimum
Use 0.2 L/D to Also Span Lower Energy Encounters

Of the aerodynamic variables, the dominant one is that resulting from Mars atmospheric density variations. The above assessment assumed that the density could be known to within ±50% as represented by the COSPAR warm and cool atmospheric models for the northern hemisphere of Mars. An encounter during a dust storm, although boosting the atmosphere's effective density up 100% over its clear state, would still be predictable to within ±50% using far-encounter and/or Earth-based measurements. The overall uncertainty in vacuum periapsis resulting from this density variation is ±4.91 km.

The next most significant aerodynamic variable is the variation in lift to drag ratio (L/D). This arises from variations in the vehicle center of gravity (CG) location as well as uncertainties in the aerodynamic coefficients, both of which affect the static trim attitude and thus the L/D. Previous assessments of this variation for a variety of aeroassisted vehicles (AFE, AOTV, etc) indicates a net trim attitude variation of ±2° can be expected. When this variation is mapped into a low L/D vehicle configuration (a control requirement consistent with this level of errors), a net variation in periapsis of 4.82 km results. These variations were assumed to be all independent of each other and so were RSS'ed into a net variation of ±9.08 km which included a 30% margin for control lags.

When this net variation is mapped into the parametric control chart (Fig. 7.1-2), a minimum required L/D of 0.14 is derived. Lower energy encounters considered in the mission study space drive this L/D up because of the loss of lift component with reduced aero delta-v. To accommodate these lower energies a universal minimum L/D of 0.2 was adopted for aerocapture at Mars.

The above analysis is parametric in nature and requires the use of a closed-loop simulation with induced dispersions to verify the robustness of the derived L/D. This testing was performed using the closed loop aeroassist simulation tool (CLAAS). This guidance package uses a predictor-corrector algorithm to steer a lifting vehicle to a desired target exit apoapsis and orbital plane. Results of testing for the 0.2 L/D configuration are shown in Table 7.1-2 for the encounter C3 of 60 km^2/sec^2 and target exit orbit of one sol elliptical The high and low pressure COSPAR as well as Viking 1 and 2 atmospheres, perigee targeting variations, and angle of attack uncertainties are used as environmental variations. Only the entry flight path angle uncertainty is "known" to the guidance software; the other variations are unknown and must be reacted to. For each profile the exit orbit characteristics are tabulated along with the correction burn magnitudes required to inject into the final target orbit . In addition, the peak loads are tabulated. It is readily seen by inspection of Table 7.1-2 that this simulation demonstrates stable behavior of the 0.2 L/D vehicle with minimal variation in the correction velocity.

Table 7.1-2 Mars Aerocapture, Closed Loop Testing

Dispersion	Exit Orbit		ΔV to Reach Park Orbit, Mps**	Peak Loads. g's
	Periapsis, km	Apoapsis, km		
Nominal	37.6	33845.5	12.3	6.91
Low Pres Atmos	41.0	36129.8	23.4	6.89
High Pres Atmos	34.9	31323.2	28.1	6.39
Viking 1 Atmos	45.4	35533.1	20.3	6.78
Viking 2 Atmos	32.7	31460.8	27.3	6.54
ΔPer = + 2.78 km*	37.7	33844.7	12.3	6.65
ΔPer = - 2.78 km*	37.8	33854.2	12.2	6.76
Δ Alpha = + 2.0°	37.9	33853.7	12.2	7.01
Δ Alpha = - 2.0°	37.4	33846.9	12.3	6.82
RMS of Deltas from Nominal	3.6	1585.7	9.1	0.26

Notes:
* Delta Flight Path Angle = ± 0.10° (at 125 km)
** Final Park Orbit of 250 x 33851 km is Reached through ΔV1 at Apoapsis Followed by ΔV2 at Periapsis (ΔV = ΔV1 + ΔV2)

The 0.2 L/D is a low enough value that symmetric blunt shield aerobrakes may be used. Blunt shields are attractive because of packaging flexibility for the wide variety of different resources that are necessary for long duration flight including habitation modules, in-flight science experiments, and communications gear. These shapes are also attractive because they are easier to analyze and manufacture than asymmetric concepts. The 70° Viking symmetric blunt aerobrake configuration is consistent with a 0.2 L/D and exhibits good stability characteristics. This proven configuration consists of a 70° half angle cone centered on a spherical nose cap (nose radius is one fourth of the brake diameter). The region protected from wake impingement is described by a 30° half-angle cone centered on the aerobrake base. Although this cone is actually a skewed cone caused by the angle of attack effect, the 30° symmetric cone fits within the skewed cone and simplifies the packaging constraints.

Parametric analysis of the aerocapture for the highest Mars encounter velocity of 60 km^2/sec^2 indicates that the low L/D results in a peak load of 8.6 g's. If the crew is conditioned through the use of artificial gravity these g-loads are probably tolerable because they exist for only a short duration, exceeding 5 g's for only 40 seconds. During such an aeromaneuver the crew are strapped into g couches, similar to those used in the Mercury program, with their backs aligned to the vehicle-fixed aerodynamic acceleration vector. If, however, a lower g requirement is enforced and these high encounter velocities are maintained, the use of higher L/Ds will be required for load relief. In the load relief technique, excess lift is used to bias the flight altitude of the vehicle higher into the atmosphere. In this way the deceleration is spread over a longer time interval, reducing the peak loads. With the higher L/D strategy the peak heating is also reduced but the integrated heat load is increased, which drives up the thickness of the TPS. The same entry corridor requirements are maintained but since the maneuver capability envelope increases with L/D (Fig. 7.1-2), the operational control corridor can be moved to the right along the skipout boundary, cutting out a large unusable region below. To investigate the effect of increasing the vehicle L/D for load relief, the MMSS study also considered the design and implications of a 1.0 L/D biconic aeroshell for the Mars Expedition case study.

7.2 ATMOSPHERIC DENSITY SENSITIVITY IN AEROCAPTURE

A critical aspect for an aerocapture vehicle is the amount of uncertainty associated with the expected density and wind structure of the atmosphere. The Viking entry vehicles encountered significant shear features in the atmosphere of Mars. The significance of such structures is in their effect on the control rates and guidance correction capability. Because there have been limited numbers of entry vehicles into the Mars atmosphere there is great uncertainty as to the magnitude of these variations. Opinions vary as to the magnitude of effects expected to be encountered.

One way to approach the problem is to assess parametrically the effect of a variety of artificial density shears on a reference vehicle. Low L/D entry vehicles obviously do not have as much control authority as do higher L/D configurations and so should be inherently more sensitive to atmospheric variations. By testing the 0.2 L/D entry concept, already designed to withstand reasonable steady state atmospheric variations, a worst case assessment of time varying density sensitivity should result. Previous dispersion testing of these L/Ds has yielded good stability for time varying density variations as represented by the two Viking entry profiles. These configurations were then analytically tested against various synthetic atmospheric structures to thresholds of degraded performance.

A series of tests were conducted for atmospheric discontinuities modeled by simple step and square functions to assess the scale of critical disturbances. The first step was to determine where along the entry profile the vehicle was most sensitive to atmospheric discontinuities. The vehicle model used was the manned Mars 0.2 L/D configuration with a peak roll rate of 20 deg/sec (about 2 rpm) aerocapturing into a 1 sol orbit from an encounter hyperbola with a C3 of 60 km^2/sec^2. By first using a step function that increases the atmospheric density 50% over the nominal level, sensitivity to shear location was assessed as a function of time into the aeromaneuver. It was found that the most sensitive location was just past periapsis where the vehicle is still near peak loading but the control capability is decaying because of decreasing dynamic pressure.

Holding this peak response location fixed, a series of density pulse functions approximating finite shear structures were introduced to determine sensitivities. The effect to the exit orbit was measured by the park orbit trim ΔV requirements. The aero maneuver was somewhat arbitrarily considered failed if its trim burn requirements exceeded 100 meters per second. As can be seen in Figure 7.2-1, density pulses with lateral dimensions of 60 km exceeded this threshold if they raised the local density 60% or more above the expected background level. If, however, the scale of the density pulse was reduced to 30 km (Fig. 7.2-2), the magnitude of density increase had to be 180% above background to fail the maneuver. Finally, density pulses with 12 km dimensions had negligible ΔV effects for density increases up to 200%. These results should be considered preliminary but do indicate some interesting trends for data resolution requirements of the Martian atmosphere. Such missions as Mars Observer are hoped to greatly improve our knowledge base relevant to future aerocapture missions. Resolution of the atmospheric structure to a certainty of 30-40% on a horizontal scale of 60 km is indicated. Early unmanned missions such as MRSR may incorporate higher L/D ratios and may be able to cope with a lower atmospheric resolution, consistent with a poorer knowledge base for the planet.

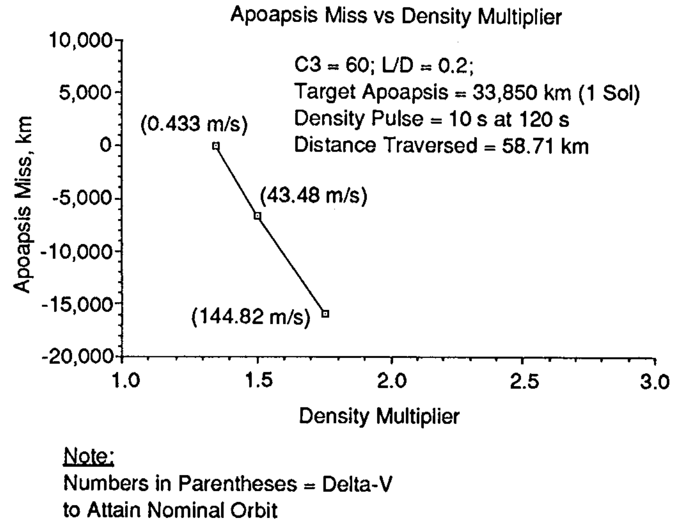


Figure 7.2-1 Mars Aerocapture Density Sensitivity, 60 km Feature

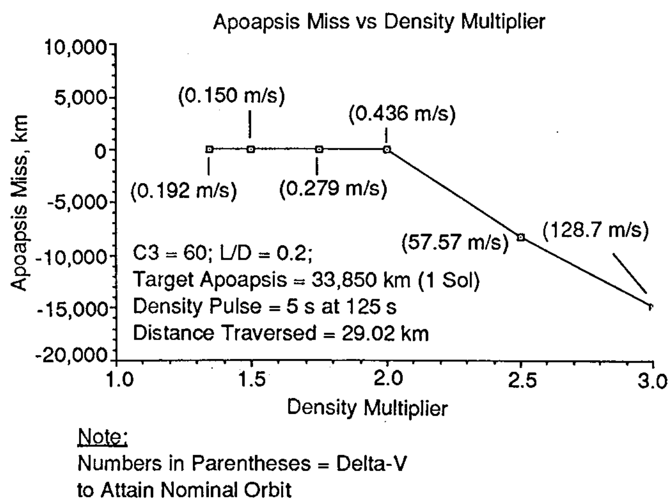


Figure 7.2-2 Mars Aerocapture Density Sensitivity, 30 km Feature

7.3 MARS LANDING

The Mars landing phase begins with deorbit of the Mars lander from the one sol park orbit where the Earth return vehicle resides. The majority of the deceleration occurs while the vehicle is travelling at hypersonic velocities. Error assessments and control parametric studies indicate that an L/D of 0.17 is adequate to maintain control of the entry trajectory. The primary problem is with the knowledge or navigation uncertainty required to perform a precision landing. A variety of navigation techniques are possible, but they all require acquisition of external reference data during the entry process. Continuous ranging to the orbiting mother ship or ComSat can yield landing accuracies of 5 km. Radar terrain correlation and landmark tracking processes depend on the recognition of features below and can reduce landing in accuracies well below 1 km. Acquisition of a ground beacon near the landing site can drive the landing accuracies down to less than 100 meters if the beacon is acquired in time to make the necessary corrections. Terminal descent can further eliminate targeting errors to achieve a pinpoint landing, if required. Because the landing phase of the mission is conducted from a one sol park orbit, the loads and heating levels are 1/3 to 1/2 of these experienced in the aerocapture phase. Because of this, the deceleration aerobrakes for entry could readily be constructed of the flexible TPS materials to keep their mass down (Ref Sect. 7.6.2). In many cases, this also gave packaging advantages since these aerobrakes could be folded for the trans-Mars cruise and thus not block antenna and solar panel view angles as well as being protected from the flow effects of Mars aerocapture.

Once the vehicle reaches an altitude of 6 km the parachute descent phase begins. The primary constraint to be maintained is that the parachute deploy velocity be maintained below Mach 2.2 (which translates to 500 mps) to allow a stable deployment. This is accomplished by choosing the ballistic coefficient of the aerobrake (Fig. 7.3-1). For a low L/D vehicle the ballistic coefficient must be around 100 kg/m^2 to maintain deploy velocities in the acceptable range. Deployment of the parachute slows the vehicle and allows differential drag to effect the jettisoning of the aerobrake. A three parachute cluster was used to raise reliability. At an altitude of 1.5 km the terminal descent engines are ignited to perform the final deceleration to a soft landing. A total of 350 mps of delta-v is allocated to slow the lander, with a throttle ratio of 3:1. This level of throttleability gives optimum overall performance by minimizing descent delta-v loss while keeping down the growth of the vehicle engine mass.

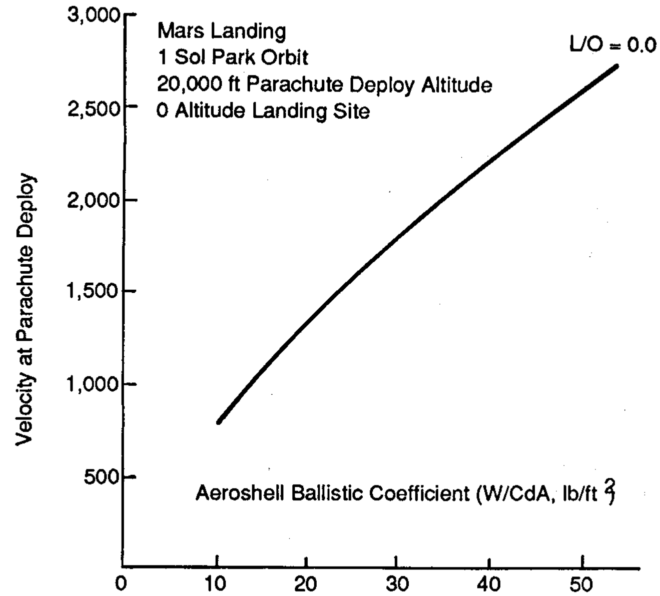


Figure 7.3-1 Mars Landing—Parachute Deploy Velocity versus W/CdA

7.4 EARTH CAPTURE

Because of a better characterized atmosphere and extremely accurate navigation infrastructure the control problem at Earth is simpler than that for Mars. A similar error analysis to that conducted above indicates a required 9 km control corridor. For Earth encounter C3s ranging from 25 to 116 km^2/sec^2, this results in minimum L/D requirements that are less than the 0.2 required for Mars capture. In the case of an Apollo-style Earth crew capture vehicle (ECCV) return the Apollo L/D of 0.3 satisfies this requirement. For a roundtrip vehicle returning the entire Mars spaceship back to Earth, the basic Mars capture L/D of 0.2 also satisfies this L/D requirement. Thus, the issue with Earth capture is not control requirements but the reduction of heating and loads for the high velocity encounters. In this instance, the 116 km^2/sec^2 C3 is the most stringent driver.

The use of an intermediate capture ellipse with a 4 day orbital period allowed the significant reduction of loads and heating by more evenly distributing the velocity reduction between two aeromaneuvers (Fig. 7.4-1). In this scenario the incoming spacecraft first captures into this extremely elliptical park orbit, the vehicle is tracked while it coasts in this orbit with a new entry solution being derived and a small perigee adjustment burn being executed during the downward leg, the second entry then places the vehicle in the low park orbit suitable for retrieval. In the case of direct entry missions to the surface of the planet this second entry would look much like an Apollo lunar profile since the entry velocities are nearly the same. Figure 7.4-2 shows the reduction in entry heating achieved by using this multipass capture technique for a C3=25 km^2/sec^2 Earth encounter. By using the 4-day intermediate orbit the heating rates are reduced to levels consistent with the use of flexible TPS for a round-trip Mars mission with a lightweight aerobrake.

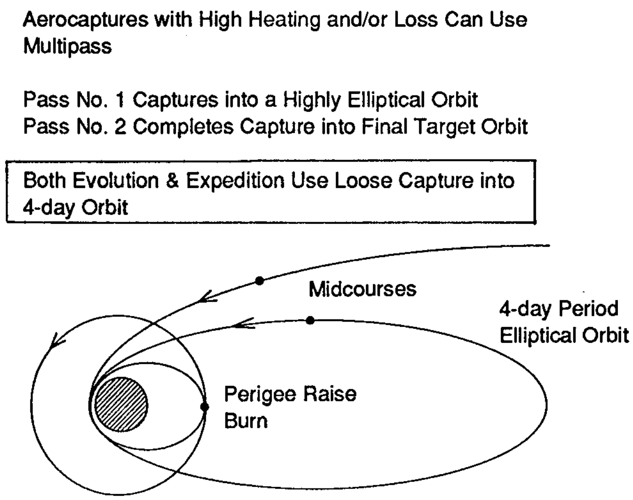


Figure 7.4-1 Multipass Aerocapture Overview

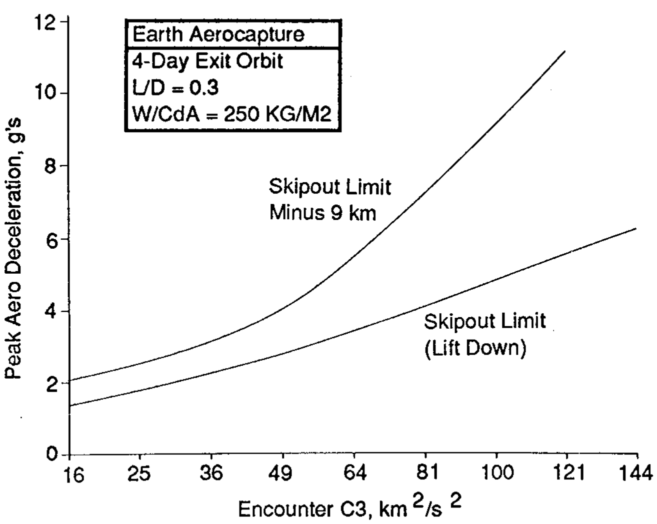


Figure 7.4-2 Earth Multipass Aerocapture Heating, C3 = 25 $km^2/seconds^2$

Very fast Earth return conditions presented significant deceleration loading and heating problems. The use of the intermediate capture ellipse helps this problem but doesn't eliminate it. In Figure 7.4-3 the peak loads for various encounter C3s are shown. Two curves are presented. The first shows the loads at the skipout boundary for a 0.3 L/D ECCV as used in the Mars Expedition Case Study. The highest C3 considered in the study was 116 km^2/sec^2, which results in a peak deceleration load of 5.2 g's. Unfortunately the skipout boundary does not represent a workable entry corridor because it has zero thickness. A bottom end operational flight boundary is produced using the 9 km control corridor requirement derived by error analysis. When the vehicle is flown to this entry boundary the second load curve in Figure 7.4-3 is produced. It may be seen that the g-load at this condition is much higher, being somewhat in excess of 10 g's. The 5 g peak load is exceeded at about an encounter C3 of 64 km^2/sec^2. If lower g's are required for the very high C3 of 116, greatly higher L/Ds in the range of 1.0 to 1.5 will be required of the ECCV. Heating for these fast encounters rules out the use of reusable insulator TPS materials, relying on single-use ablators instead.

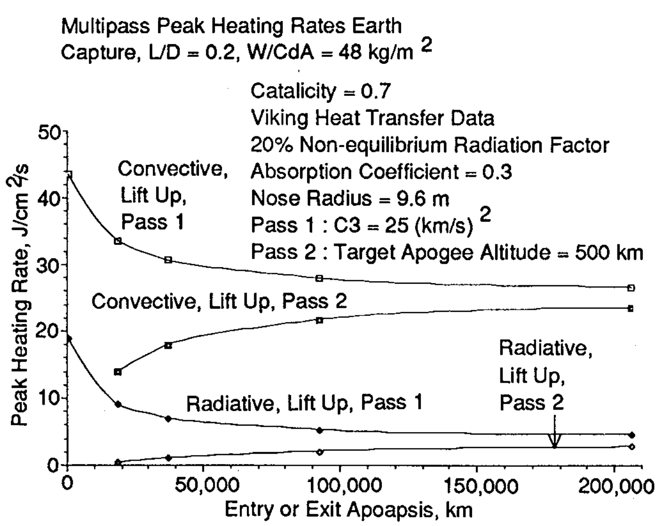


Figure 7.4-3 Earth Multipass Aerocapture Loads, Pass Number 1

7.5 AEROASSIST FOR LUNAR MISSIONS

The lunar missions were performed using all-reusable elements as per the study groundrules. These studies used an aeroassist maneuver in the Earth's atmosphere to enable the efficient capture of the reusable propulsion stages back to a Space Station orbit. Because the aeroassist maneuver is less strenuous than a direct entry to the surface of the Earth, reusable flexible insulator TPS technology was used rather than the ablators that were used on Apollo. It is important to use lightweight technology to keep the mass of the aerobrake down since the break-even point for lunar missions is for an aerobrake weight 110% of the return payload. The aerobrakes were of the flexible concept as is described below (Sect. 7.6.2). This enabled very lightweight aerobrakes (8% to 9% of returned payload for optimized designs). It also allowed the vehicles to be launched to low Earth orbit in a "ready to fly" condition because the nature of the flexible concept allows the aerobrake to be deployed on reaching orbit.

The lunar aerobrake sizes were primarily set by impingement constraints on the propulsive stage because of the generally long dimensions of those vehicles. Basic error analysis derived minimum required L/D values for lunar return of 0.11, which results in peak loads of 4.8 g's. This value was increased to an L/D of 0.14 with minimal brake design impact to allow load reduction control, bringing the peak load down to 4.0 g's. The lunar aerobrakes are summarized in Table 7.5-1. Three different classes of lunar vehicles were considered: the lunar cargo vehicle (LCV), the lunar piloted vehicle (LPV), and the lunar cargo vehicle returning a 13-ton payload of lunar LOX (LCV+LLOX). To minimize the aerobrake development cycle a common brake was designed for the LCV and LPV. Since the LCV is essentially the same vehicle as the LPV, less its payload, it has about 1/3 the dry mass. Thus this common brake is oversized for the LCV that results in its being 27% mass fraction of the empty LCV mass. The other vehicles, having aerobrakes tailored for their application, have mass fractions in the range of 8% to 9%.

7.6 AEROBRAKE DESIGN STUDIES

Aerobrake design studies were conducted for the 0.2 L/D blunt aerobrake operating at a peak g-load of 10 g's and the 1.0 L/D biconic for a load of 5 g's. These designs were thus sized for worst case capture condition of C3=60 km^2/sec^2 at Mars. In addition, two fundamental aerobrake design philosophies were used for the blunt configurations, reflecting the use of an allrigid as well as a partially flexible concept. The all-rigid concept represents a relatively conventional design approach with a rigid aerodynamic structural surface covered with a rigid TPS. The primary problem with this concept is the method of assembly, because it is too large to fit into a launch vehicle shroud. To minimize the problems of on-orbit assembly, the partially flexible aerobrake configuration was also studied. Here the technological problem is how to validate a non-rigid aerobrake for reuse for hypersonic entry.

Table 7.5-1 Lunar Evolution Aerobrake Summary

	LCV Return Brake	LPV Return Brake	LCV + LLOX Return Brake
Payload wt, MT	5.6	16.1	29.3
Diameter, m	15.9	15.9	19.5
W/cDa, kg/m2	20	57	69
Peak g-Load	4.0 g	4.0 g	4.0 g
TPS Weight, kg			
RSI	83	83	83
FSI	585	585	905
Brake Strucuture, kg	656	656	982
Total Brake Weight, kg*	1523	1523	2266
Aerobrake Fraction **	27%	9%	8%

Notes:
* Includes 15% Margin
** % of Payload Only

7.6.1 Rigid Aerobrake (L/D 0.2)

A detailed structural design of an all-rigid aerobrake was performed to characterize the mass of such a system for a large trans-Mars spacecraft. All-rigid concepts have the advantage of commonality with contemporary entry vehicle designs and thus minimize new technology associated with the entry phase itself. Because of the large size of these aerobrakes (too large for any launch vehicle fairing) they require on-orbit assembly. Thus the design concept must incorporate the capability for delivery to orbit in pieces followed by integration at an assembly fixture. The reference design used base data from the Mars Evolution mission with a spacecraft mass of 195 tons arriving at Mars with a C3 of 60 km^2/sec^2 and then returning a payload of 75 tons to Earth with a C3 of 25 km^2/sec^2. This results in a peak deceleration load of 10 g's, which is then scaled up by a 1.5 factor of safety.

A hot structure concept using advanced carbon-carbon composites allowed the structural and high temperature capacity functions to be combined into one material. The carbon-carbon is thus a load carrying element as well as the primary thermal barrier. The high backface temperature of the carbon-carbon requires that there be a thermal blanket on the backside of the aerobrake to prevent heat re-radiation to the payload. A 31-meter diameter aerobrake was used, which is the minimum size available to fit the trans-Mars spacecraft within the required impingement cone. A launch vehicle payload shroud limit of 12.5 m was specified by MASE that sets the maximum width of the disassembled aerobrake segments. The aerobrake was designed as a central circular segment surrounded by 8 petal segments (Fig. 7.6.1-1). This design concept is self-reinforcing under airloads and thus minimizes the number of on-orbit fasteners required. The joints of the segments use slotted shear pins for alignment and high temperature fasteners to maintain compression of Inconel flex seals. An overview of a representative section of joint is shown in Figure 7.6.1-2. A BOSOR4 linear buckling analysis was used to size the front face thickness at an average of 0.5 cm. To resist buckling, stiffeners were incorporated in the carbon-carbon layup as described in Figure 7.6.1-3. The aerodynamic loads are transmitted to the payload through a 12.5-m diameter payload interface ring consisting of two end rings held together by tubular trusswork. The overall aerobrake weight is 27.9 metric tons.

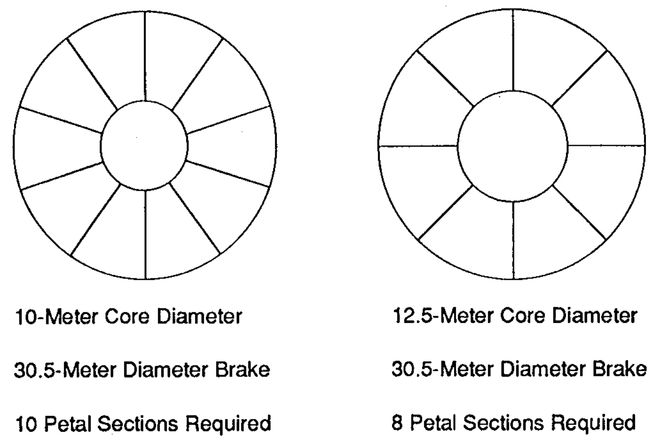


Figure 7.6.1-1 Rigid Aerobrake Petal Layout

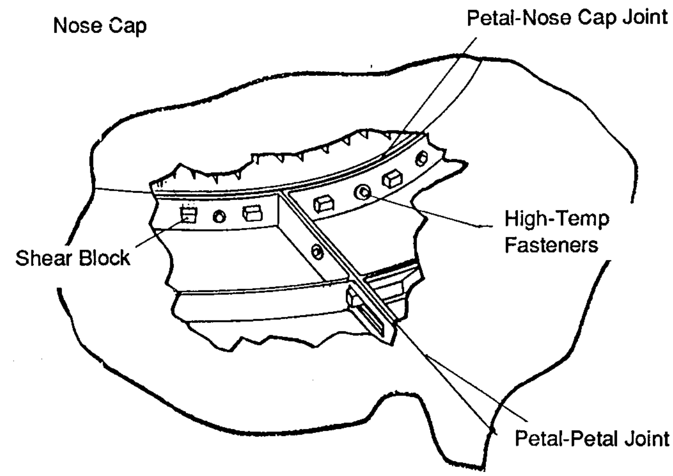


Figure 7.6.1-2 Rigid Aerobrake Joint Concept

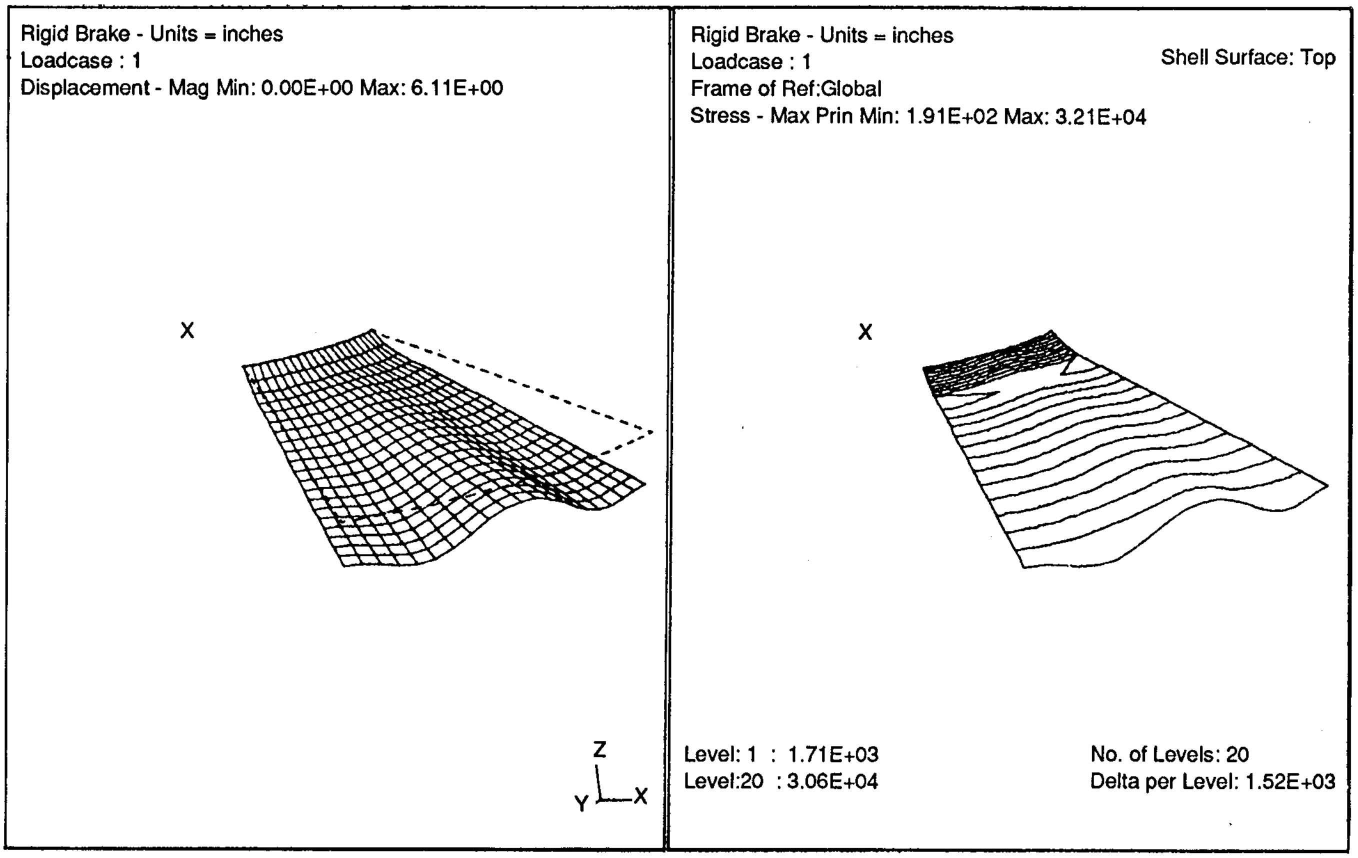


Figure 7.6.1-3 Rigid Aerobrake Stiffener Layout

The on-orbit assembly sequence is shown in Figures 7.6.1-4 through 7.6.1-6. The tapered segments are stacked side-by-side in the launch vehicle shroud in a fixture mounted to the payload interface ring and central aerobrake core section. Their attachment capability is used to lock them down into this transport facility. Once the vehicle has reached the on-orbit assembly facility, a remote manipulator arm places the core section onto a rotating assembly fixture. This rotating fixture acts like a lazy susan to present the section to be assembled to the manipulator arm. Individual petals are unlocked and removed from the launch transport rack where they are slid into place on the central circular core segment. The slotted shear pins self-align the pieces as they are emplaced whereupon their locking elements are activated. Once the aerobrake assembly is complete, the launch transport rack is removed and the vehicle is ready for outfitting with the rest of the elements of the Mars vehicle.

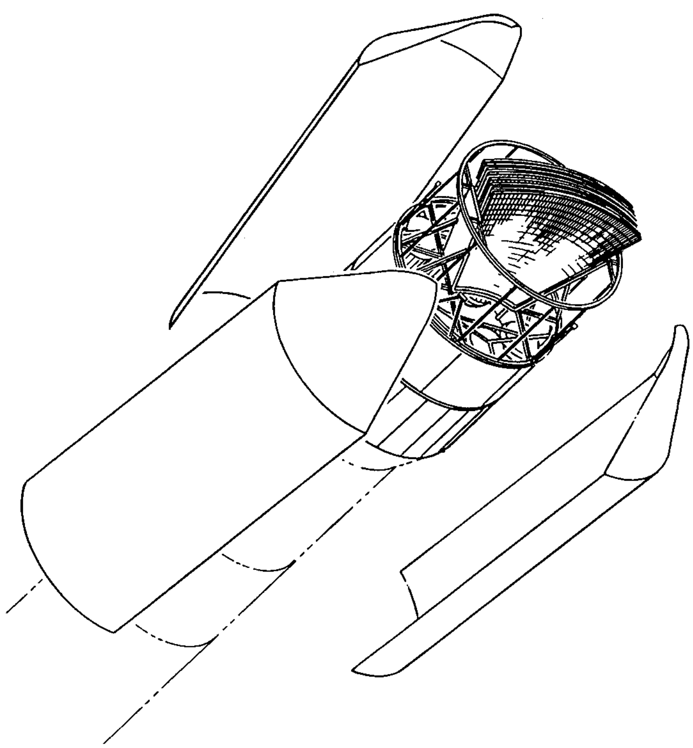


Figure 7.6.1-4 Rigid Aerobrake Assembly Number 1

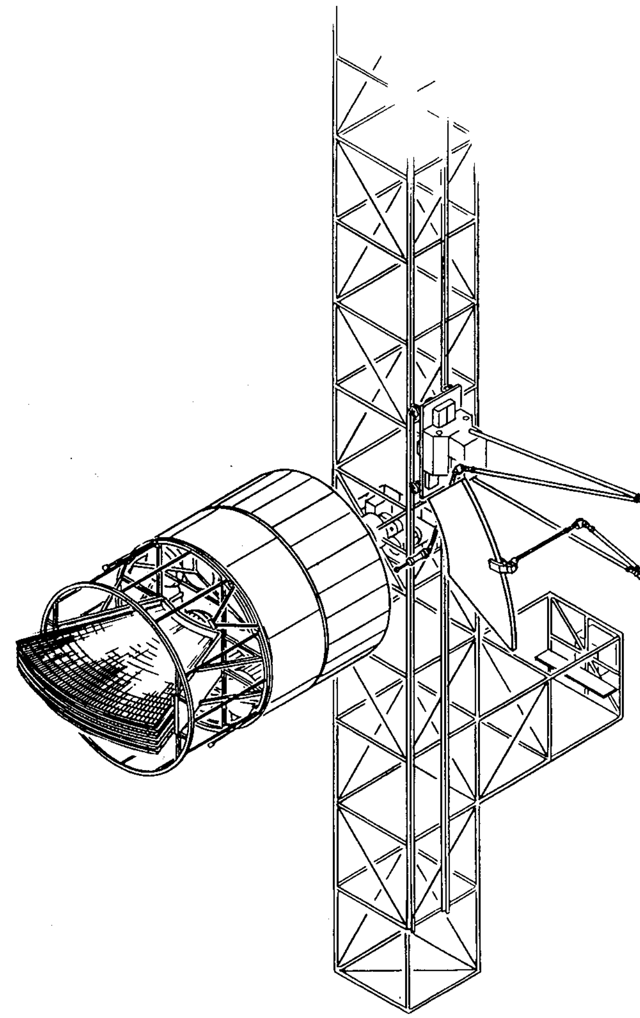


Figure 7.6.1-5 Rigid Aerobrake Assembly Number 2

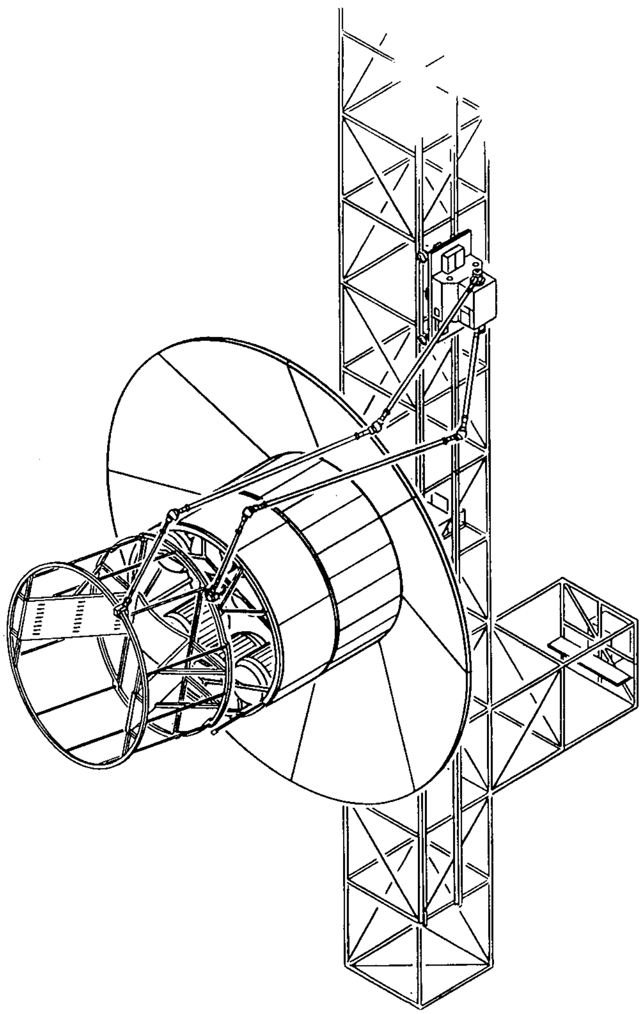


Figure 7.6.1-6 Rigid Aerobrake Assembly Number 3

A significant issue could be the validation of the flightworthiness of this aerodynamic structure. Although much can be done with good design practices that assure fail-safe locking, the requirement for validation measurements after assembly may be unavoidable. Position verification switches can indicate the seating of joints to a coarse level. The compression force along the joint face can be determined with embedded piezoelectric devices. Gas leaks can be identified with laser interferometry techniques. Ultimately the confidence in a load-carrying device such as this must be tested in an integrated fashion, however. A flight test of the aerobrake in an aeromaneuver through the Earth's atmosphere will most likely be required before the vehicle can be committed to a Mars mission. Such a flight shakedown could be accomplished unmanned and could test out other spacecraft systems without the vehicle leaving the Earth orbit environment. This profile can be established by first raising the vehicle's orbit propulsively and then targeting to a simulating entry where structural and gas integrity can be measured and recorded for validation analysis.

7.6.2 Flexible Aerobrake (0.2 L/D)

An alternate form of blunt shield aerobrake is represented by the flexible concept (Fig. 7.6.2-1). This concept uses a flexible TPS material such as the Tailorable Advanced Blanket Insulation (TABI) being investigated at Ames Research Center to provide an umbrella-like brake configuration. The flexible TPS insulation thickness is sized to maintain a 600° F backface temperature, consistent with the gas sealant material as well as the supporting graphite polyamide structure. This flexible annulus surrounds a more conventional rigid center core section. Radial beams, reinforced by compression struts, react the pressure loading of the entry into a circular payload interface ring. Previous aerobrake experience has indicated that the flexible concepts yield the lightest overall system weight. Because they can be stowed for launch and then deployed in orbit they eliminate many of the problems of on-orbit assembly of an aerodynamic/structural element. The fundamental problem with flex concepts is the lack of a suitable design base to allow confidence in their entry characteristics. Initial tensile testing has been performed on samples of candidate TPS to establish thermal response and structural characteristics however much more detailed analysis is required to establish confidence. Poorly characterized areas include hypersonic flutter, thermal embrittlement, and the number of acceptable re-uses. Currently, the TABI material is capable of 34 W/cm^2 heating rate and is anticipated to be capable of 40 W/cm^2 in the future.

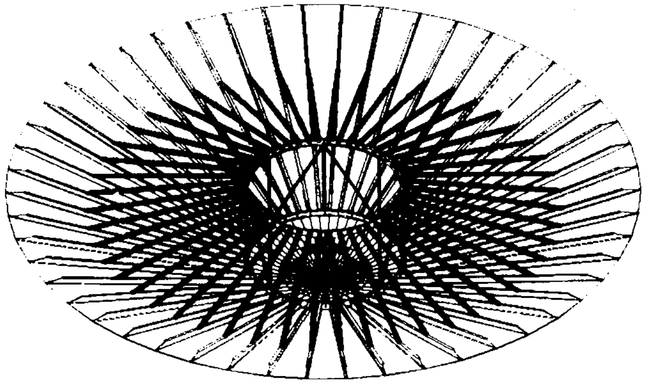


Figure 7.6.2-1 Flexible Aerobrake Overview

Design analysis of this concept was performed with the same groundrules as the all-rigid concept. The reference design used data from the Cycle 2 Evolution mission with a spacecraft mass of 195 tonnes arriving at Mars with a C3 of 60 km^2/sec^2 and then returning a payload of 75 tonnes to Earth with a C3 of 25 km^2/sec^2. This results in a peak deceleration load of 10 g's at Mars which is then scaled up by a 1.5 factor of safety. A large brake diameter of 38.4 meters is required to reduce the local heating that results from this energetic Mars encounter, maintaining the flexible material within thermal limits. The Earth return heating and loads are kept manageable by the use of a dual aeropass capture as described in the Earth return section. To maintain the flex material within pull-test tension limits, 50 radial beams were required to support the fabric. Each beam had an I-beam cross section with a height of 38.1 cm and 14.2-cm width supported by a 32.5-cm wide compressive strut (Fig. 7.6.2-2). At the central core, the thermal load is within limits of conventional ceramic tiles. For the purposes of this concept definition shuttle, FRCI-12 tiles were used as the TPS for this central section, although it is recognized that more durable options will be available at the time of Mars missions. The mass of this aerobrake was derived to be 3.4 t of TPS and 19.0 t of structure for a total overall mass of 25.8 t, which included a 15% design margin. This represents 13.2% of the payload weight. A reduced energy encounter aerobrake was also analyzed where the Mars encounter C3 was limited to a maximum of 38 km^2/s^2 representative of an opposition class mission. The reduction in heating allowed a smaller diameter aerobrake of 32.9 meters. This coupled with the reduced deceleration loads of 6.0 g's resulted in an aerobrake weight of 14.2 t. This represents 7.3% of the payload weight and indicates the large penalty paid for high energy encounters. The stowed configuration of the flexible aerobrake is shown in Figure 7.6.2-3. The radial beams fold upward into a cylindrical package. Not shown are a series of folded struts at the tips of the radial beams that lock into the circumference of the deployed brake to provide torsional stability as well as the edge curvature required to reduce edge heating. This aerobrake is deployed on reaching orbit by releasing locks in the radial beams. The action of springs and/or motor-driven actuators in the compression struts pushes the beams outward to their flight positions where they are locked into place. The rotation of deploy cams at the base of the beams brings the flexible TABI material at their base up against the core structure, sealing it against gas leakage. Once the deployment sequence is complete and structural integrity verified, the rest of the elements of the Mars vehicle are integrated into the aerobrake.

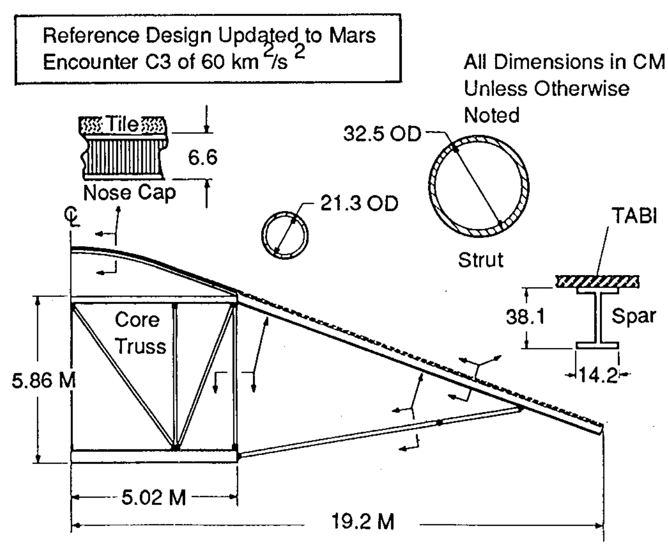


Figure 7.6.2-2 Flexible Aerobrake Design

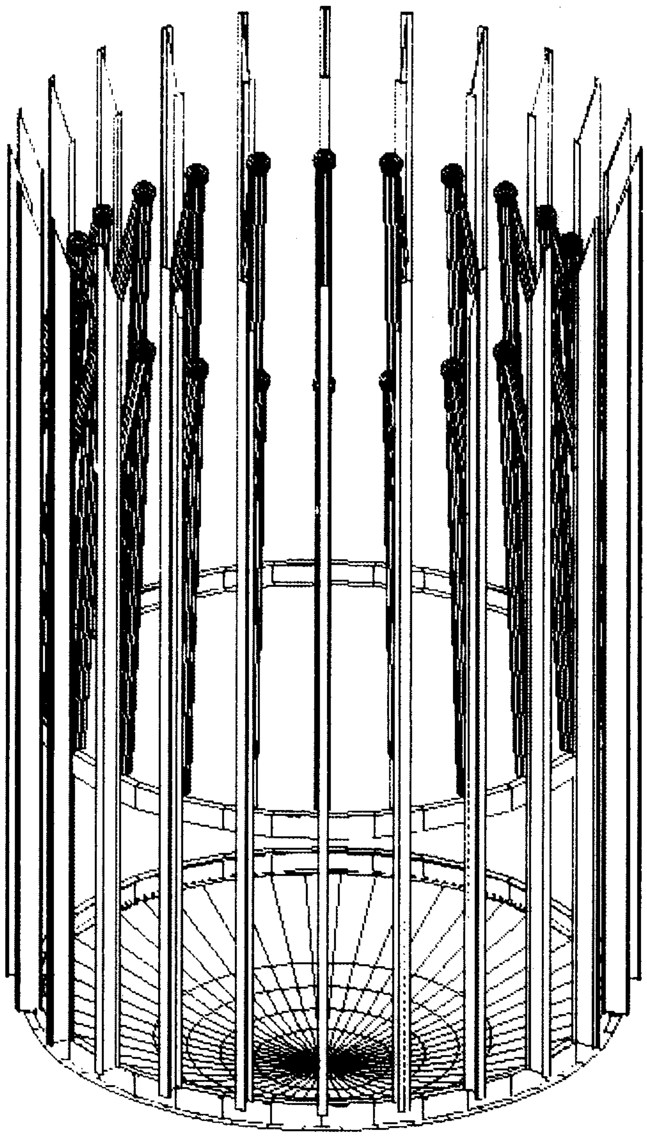


Figure 7.6.2-3 Flexible Aerobrake, Launch Configuration

7.6.3 Biconic Aerobrake (1.0 L/D)

High L/D aerobrake concepts are useful in reducing deceleration loads in energetic aeromaneuvers through the use of atmosphere-skimming load relief techniques. In these methods the large lift vector is oriented generally downward, allowing the vehicle to fly higher in the atmosphere. This reduces the peak aerodynamic load at the expense of a longer flight time and higher integrated heating loads. In addition it reduces the peak heating rate, at the expense of a higher integrated heat load. In this study a 1.0 L/D concept was specified for the Mars Expedition concept.

To achieve an L/D of 1.0 a biconic configuration is required (Fig. 7.6.3-1). The concept used is based on a configuration used for the Mars Rover Sample Return Mission (MRSR). When flown at an angle of attack of 29°, the desired L/D 1.0 is achieved with good longitudinal stability characteristics. The reference design used a Mars encounter C3 of 60 km^2/sec^2, the same as for the previous blunt shield designs. When flown at the bottom of the operational entry corridor (18.5 km from skipout) this configuration experiences a peak deceleration load of 5.0 g's as contrasted with the blunt shield load of 8.6 g's. Thus, the use of lower L/D can reduce peak loads by about 40%.

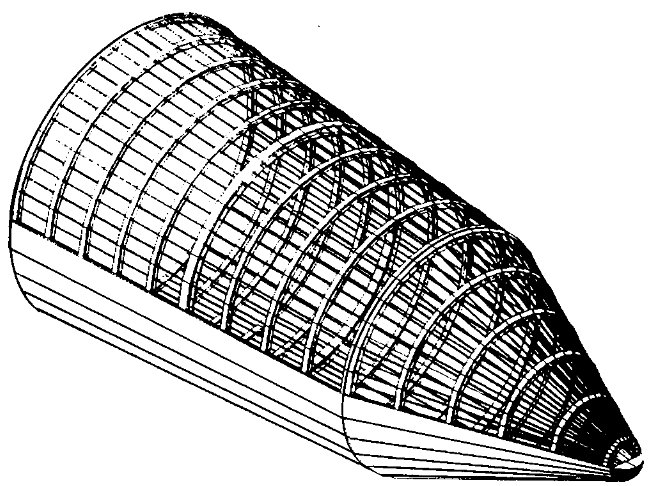


Figure 7.6.3-1 Biconic Aerobrake Overview

A design study produced mass estimates for this vehicle used a 200 metric ton payload delivered to a one sol Mars orbit. Because of the demanding heating environment, the aerobrake was assumed to be a single-use device for Mars capture only. The aerobrake was 12.5-m in diameter and 27-m long. A conventional skin-stringer structural design was used with the payload secured inside by a four-point suspension system similar to the shuttle sill fittings. For the identified 5-g loads, the mass of the brake was 20.3 metric tons for a graphite polyamide construction. The biconic has a relatively high ballistic coefficient of 2200 kg/m^2, which results in very high heating rates relative to the blunt configurations. This requires the use of ablator TPS with an estimated total mass of 10 metric tons. Although this represents a 15.0% fraction of the payload, it must be born in mind that this aerobrake is only capable of a single use (no Earth return re-use), whereas the blunt configurations have multi-use capability. The weight of TPS would increase considerably for multiple use, although it is not clear whether an ablator could be successfully used twice. Other difficulties with the biconic aerobrake is its limited packaging volume and severe thermal balance constraints. In the latter case, this drove the payload elements to be deployed on rails out the back end for the trans-Mars coast.

7.7 AEROASSIST STUDY CONCLUSIONS

Aeroassist is a fundamental technology for future space initiatives. As with any advanced technology, its use is subject to constraints, some peculiar to the nature of a hypersonic glider, but most just now beginning to be understood fully. The application of blunt cubic aeroshells gives excellent packaging advantages that can be well used in a multidimensionally complex manned Mars vehicle. The use of flexible aerobrake configurations have certain benefits—primarily the elimination of on-orbit assembly but require much more technological maturity before their use can be adopted. The on-orbit assembly of rigid aerobrakes requires a great deal of work in the area of verification/ validation of the completed aerodynamic structure. High encounter energies drive up aerocapture heating and g-loads, which must be considered carefully before selecting mission options that produce them. Mission design must take into account the special constraints imposed by the use of aeroassist to fully realize their benefits.

8.0 CONCLUSIONS AND RECOMMENDATIONS

During the course of this work, a number of alternative approaches and options were studied for implementing human missions to Mars and the moon. In no case should these studies be considered as fully comprehensive, or the last word for any particular trade because after the initial start of this contractual study, the approaches became quite constrained by systems-level requirements levied from outside the purview of this effort. Furthermore, it has become apparent that few, if any, trades can be made totally independent of architecture concepts and technology-readiness assumptions. Conducting free-standing trades is a procedure that can produce unacceptable results—an example was the decision that both sprint and nonaerocapture should be chosen for the first Mars mission, each on the basis of a safety (or conservatism) concern. Thus, the first conclusion/recommendation (CR) of this study is:

CR-1. Options and alternatives should be propagated entirely through the set of missions, scenarios, or architectures under consideration at any given time to fully assess a trade-off.

Corollary A: An integrated system of analyses and computational tools is necessary to provide a disciplined and accurate way of accomplishing these trades.

Corollary B: A final answer on the most satisfactory overall approach, under any given set of groundrules, will not be reached without considerable iteration, adjustment of assumptions, and reassessments.

Corollary C: As new approaches to the mission objectives are conceived, or new technologies become considered, many previous trade conclusions must be revisited and revalidated.

Space Station Freedom (SSF), or some similarly long-lived manned space capability, is an absolute prerequisite to manned Mars missions. The research and development of countermeasures against zero-gravity physiological effects and the gaining of experience in how to maintain the well-being of small crews for very long time periods in an isolated, confined, and hazardous environment (ICHE) are elements of any success-oriented program for long-term missions. In addition, SSF can be used as an in-space demonstration platform for new technologies. It could also serve as a transportation node for storage and servicing of vehicles.

CR-2. The Space Station Freedom (SSF) project will satisfy the need for essential infrastructure in preparation for interplanetary travel and establishment of long-lived planetary bases.

Corollary A: Determining the hooks and scars in the SSF design for future studies and capabilities needed to support human exploration missions should be a paramount priority as the Freedom Station design matures.

Corollary B: Because SSF does not provide artificial gravity, except on the very small scale of an internal centrifuge, a second manned platform, an Artificial Gravity Research Facility, in LEO may ultimately become necessary.

The development of advanced Environmental Control and Life Support Systems (ECLSS) has progressed significantly during the two decades since Skylab. Instrumentation designs are at hand for providing much of the needed long-lived and advanced physical/ chemical recycling. Unfortunately, the Technical Demonstration program initiated by the SSF program has been severely de-emphasized and the LSS closure on-board Freedom Station has been postponed. It has been identified by NASA that long-lived ECLSS is a technology that is not yet mature. This contract work has also identified the criticality of low-power ECLSS, because it is a major driver of the power systems for the interplanetary Mars flight and the early landing missions (until or unless *in situ* resource use is transitioned in as the major power consumer).

CR-3. A vigorous ECLSS technology development program should be revitalized. This effort could be increased significantly in the very near-term with good productivity because of the groundwork, planning, and hardware development that has already been accomplished for the Technology Demonstrations effort.

Corollary A: A highest priority objective should be low-power, long-lived, in-space maintainable ECLSS systems.

Additional conclusions regarding the ECLSS system include:

CR-4. The studies show that if hygiene water usage is minimized through careful design and cleanliness protocols, then no make-up water is required as long as ample non-dehydrated food is provided and the water recovery from all wastes (other than solid wastes) achieves between 90% and 95% recyclability.

CR-5. The use of cryogenic hydrogen and oxygen (H/O) propellant for propulsion requirements after leaving Earth has the advantage that a free-return or minimum-power abort could free-up utilization of these propellant resources in an emergency mode to augment or almost totally supply life support requirements. These resources, in themselves, could supply breathing oxygen, drinking water, electrical power, heat on demand, and cooling power.

Corollary A: Other propulsion approaches, including both electric and thermal nuclear propulsion, or non H/O chemical pairs, provide none or at best a minimal amount of these resources.

CR-6. For human exploration missions, a Heavy Lift Launch Vehicle (HLLV) is needed to reduce the number of launches and the amount of on-orbit assembly required. The recommended payload-to-LEO capability should be:

Mars missions: 100 to 200 t for propellant; 50 to 100 t for dry payload
Lunar missions: 50 to 100 t

Item A: The Shuttle-C provides for the high end of lunar missions and an extremely high-reliability approach for launching expensive dry hardware for the Mars missions.

Item B: The Shuttle-Z concept provides for all needs of Mars missions, including the automatic placement into LEO of stages that can be used for TMIS.

Item C: The Advanced Launch System (ALS) holds the promise of reducing Earth-to-orbit (ETO) launch costs to a degree that is extremely significant in overall cost projections for a manned Mars mission.

Item D: The Soviet Energiya HLLV provides, at 100 to 150 t, many of the capabilities for lifting of payload mass into LEO, including the large propellant loads needed by Mars missions.

Payload shroud diameter is important to allow for large-sized habitats, to allow greatest possible flexibility in packaging, and to minimize on-orbit assembly.

CR-7. A large-diameter payload bay is more important than long payload bays for an HLLV. A diameter of at least 8 to 10 m is highly desirable.

CR-8. At Mars, from the standpoint of minimizing IMLEO, a highly elliptical orbit is preferred over a low circular Mars orbit. The 1-sol orbit used by the Viking missions has many advantages.

CR-9. Arrival and departure declinations at Mars can cause propulsion penalties in achieving rendezvous with Phobos or Deimos. Detailed studies indicate that to achieve IMLEO savings with the use of Phobos propellants, it will be necessary to provide shuttle tanker capabilities to transport propellants from Phobos to the user, rather than have the large and heavy user spacecraft be transported to Phobos using Earth-supplied propellant.

Corollary A: The previously proposed Phobos base may be untenable.

CR-10. Lightweight and low boiloff cryopropellant tanks are very high-leveraging developments for reducing IMLEO for all missions, and should be intensively studied.

CR-11. An advanced space engine for H/O cryogens is desirable.

Item A: Improving specific impulse is of importance, but not as highly leveraging as some other technologies (e.g., boiloff management) relative to the large investments that are needed to make modest percentage reductions in IMLEO.

Item B: A long-lived, restartable engine is desired, and could be enabling for certain vehicle designs.

Item C: Wide-ranging throttling is required to support the lunar-landing propulsion profile. Alternatively, multiple classes of engines will have to be used.

Item D: A compact engine configuration, made possible at high performance by high chamber pressures, is beneficial for certain vehicle designs of lunar landers.

Item E: The thrust-to-weight (T/W) value for a given engine is not a critical parameter in most cases because of the long firing times and the large amounts of propellant that are used. The major exception is for the nuclear thermal rocket, where the engine includes the massive reactor and the propellant loads are considerably reduced because of the high specific impulse of the system.

CR-12. The thrust levels needed for trans-Mars injection can be minimized by a multi-burn escape strategy. A total thrust of 450 to 666 kN (100 to 150 klb_f) is acceptable for this strategy with most manned Mars mission scenarios.

Because the human complement is not only the primary payload but also a key component of the system, a number of human factors considerations must come into play so that not only the safety of the crew is preserved but their performance is maintained near peak at all times.

CR-13. For long-term missions such as flights to Mars, a minimum of five crew members is recommended:

one pilot/commander

one engineer/technician

one medical doctor/dentist

one scientist,

one floater/back-up person/tie breaker.

Corollary A: Under certain Lunar Base conditions, the same criterion would apply. These conditions would be long-term tours of duty and the absence of (or limited reliability of) a rapid return rescue capability.

CR-14. The design of habitats for long-term habitation must include minimum crew facilities as well as sufficient comforts to guarantee a quality of life that optimizes performance. When protracted stay is required in ICHE, special design approaches are required.

Corollary A: Both physical and psychosocial factors are of great importance in solving this problem.

Items 1-56. Numerous human factor considerations, ranging from crew selection procedures to background noise control and lighting strategies, are presented in Section 4.4 and Appendix A.

CR-15. The inert weight needed for providing a radiation storm shelter to protect against solar particle events can be reduced to a very small value by use of on-board and external resources.

Strategy A: During the interplanetary Mars flight, crew consumables (including food) can be stored in the walls of the shelter. As consumables are used, the solid wastes produced can be substituted for the removed supplies.

Corollary: dumping of wastes to minimize mission mass cannot be permitted beyond a certain minimum level to provide shielding.

Strategy B: On the moon, local regolith can be bagged or piled to provide any necessary shielding, including amounts adequate to eliminate unnecessary exposures to energetic galactic cosmic rays.

Strategy C: On Mars, the atmospheric mass is sufficient to provide shielding against all solar particle events previously detected. If additional protection is desired or required, the martian soil is just as suitable for bulk shielding as lunar regolith.

CR-16. Artificial gravity vehicles should be given strong consideration for missions to Mars because of the fact that implementation studies have not demonstrated any major vehicle design impacts, whether through use of rigid rotating spacecraft or with tethers separating major components.

CR-17. For Earth return from Mars, a limitation of encounters C3s below 64 km^2/s^2 will permit elliptical orbits with peak deceleration loads of less than 5 g.

CR-18. For chemical rocket transportation, it does not appear to transport Lunar LOX to LEO from the moon, unless the production operations and transportation costs are lower than simply launching LOX to LLO from the Earth's surface.

CR-19. LLOX has maximum benefit from LLO $\leftrightarrow$ Lunar Surface transportation and LLO $\rightarrow$ LEO.

CR-20. O/F mixture ratios greater than 7 or 8 do not significantly effect Lunar all-chemical transportation.

CR-21. The orbital node appearing to have the best potential for taking advantage of Lunar LOX for Mars Missions is a high elliptical Earth orbit (HEEO).

Appendix A

Marsflight Human Factors

MARSFLIGHT HUMAN FACTORS

The Martin Marietta Manned Mars Mission Study

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> In earth orbit, human frailties can be relatively easily tolerated. If things get too bad, the Earth is close at hand. In 1985, the London *Daily Telegraph* reported that one cosmonaut commander had to return to earth 2 months into a much longer mission. He was, according to his flight engineer, a "bundle of nerves." A trip to the moon, an 8-day round trip, doesn't present much more of a problem than orbiting the earth. But how about far out, where planets of our solar system beckon? There things will get rough indeed, and groups of travelers will have to be selected with the greatest care. "Group dynamics" will no longer be psycho-babble, but a matter of life and death (Collins, 1988, p. 251).

INTRODUCTION

Spacefarers are confronted with conditions which are rarely encountered in everyday life including isolation, confinement, deprivation, and danger (Table 1). These conditions are typically construed as stressful. Consequently, such conditions impact performance and subjective well being. Whereas the relationships among stress, performance, and well being are complex, continued high stress can lead to negative attitudes, inefficiencies, errors, and deteriorations in physical and mental health. As Harrison and Connors (1985) note:

> A manned space mission may be viewed as a complex biotechnical and sociotechnical system consisting of manufactured and human parts. Malfunction or failure within any part of this system, be it structural, mechanical, electronic, or human, forces readjustments within the system as a whole. If successful, these readjustments involve wear-and-tear on the system, and the loss of back-up capabilities that might be needed in the future. If unsuccessful, the mission fails to achieve its goals, or possibly ends in disaster.

The Importance of Psychological and Social Considerations

Among the risks of ignoring the behavioral dimensions of Marsflight are the psychiatric incapacitation of one or more crewmembers and the potentially dangerous results of seemingly small errors and inefficiencies, either alone or in combination. Additionally, psychological and social variables are of importance because they will affect the quality of the astronauts' lives.

The Threat of Major Psychiatric Episodes

Continued exposure to hardships in space could lead to incapacitating forms of mental illness resulting in the loss of the services of formerly functioning crewmembers (Bluth, 1987; Santy, 1987; Space Science Board, 1987). A distinguishing feature of the Mars mission is that it will involve unprecedented combinations of conditions that are likely to intensify the already formidable pressures associated with spaceflight and increase the risk of psychiatric casualties. These conditions include crewmembers drawn from widely differing backgrounds, unprecedented distances from Earth, and unprecedented travel times (Brady, 1983; Clearwater, 1985; Connors & Harrison, 1988; Connors, Harrison & Akins, 1985, 1986; Harrison & Connors, 1984, 1985; Helmreich, 1983; Kanas, 1985; Nicholas, 1987; Santy, 1987; Space Science Board, 1987).

Table 1

MARSFLIGHT CONDITIONS

Isolation from Home Community isolation from family, friends, acquaintances, and society-at-large; reduced variety in interpersonal relations and in the number of social roles that can be performed; no opportunities to make new friends and acquaintances during period of absence from Earth

Confinement with Limited Number of Other People lack of privacy; little or no opportunity to avoid another crewmmember; necessity of getting along with peers

Deprivation very limited interior space; few or no fresh foods; limited work/recreational opportunities and materials; limited hygiene facilities; few or no luxuries; little or no opportunity to go outdoors; little or no opportunity for a "vacation"

Danger lethal external environment; limited medical/health facilities; total dependence on life support systems; little or no chance of rescue

Small Errors With Large Consequences

Even if prolonged spaceflight does not produce psychiatric casualties it can still lead to consequential human malfunctions. Given the demanding and unforgiving nature of spaceflight, even minor performance decrements can be dangerous. Also, the cumulative adverse effects of many small errors, omissions, and inefficiencies could prove fatal.

Optimum Management of Human Resources

A third reason for a careful consideration of the behavioral dimensions of spaceflight is that even satisfactory missions can be improved by taking these dimensions into account (Helmreich,1983). That is, missions which proceed without accommodating the attitudes, behaviors, and social needs of the participants are likely to proceed with a low level of efficiency and high human cost. Why settle for conditions that are merely adequate --- a half glass, so to speak --- when more thoughtful preparations yield peak efficiency and morale?

Table 2

A MODEL FOR CREW BEHAVIOR AND MISSION OUTCOME

Interventions ->	Criterion Behaviors ->	Outcomes
	Task Competence	Safety
Selection	Sustained Motivation	High Performance
Training	Emotional Stability	High Quality of Life
Engineering	Positive Social Relations	Positive Public Relations

A Model for Crew Behavior and Mission Outcome

The forms and potential results of psychological interventions are illustrated in a three part model based on interventions, criterion behaviors, and mission outcomes (Connors & Harrison, 1988). This model is depicted in Table 2. *Interventions* refer to the methods or procedures that encourage certain kinds of behaviors. *Criterion behaviors* are the desirable attitudes and behaviors that are the immediate goals of the intervention. *Outcomes* are the favorable results of the criterion behaviors. Thus:

interventions -> criterion behaviors -> favorable outcomes.

Interventions

The primary application of psychology is to improve the fit between the person and his or her environment. There are three ways of achieving this:

o *Selection* involves choosing people whose intellectual and emotional resources are such that they are likely to do well within the situation of interest, in this case, spaceflight and Mars exploration environments.

o *Training* involves modifying the person to fit the situation.

o *Engineering* involves designing environments to be compatible with their human users. In this model, engineering encompasses not only the design and provisioning of the spaceship and lander but also the structuring of social relationships and the design of tasks.

Criterion Behaviors

The direct goals of selection, training, and environmental engineering are criterion behaviors. Based on the work of E. K. E. Gunderson and others, these are task competence, sustained motivation, emotional stability, and positive social relations (Gunderson, 1973).

o *Task Competence* Although an incompetent worker is unwelcome in any environment, he or she is totally unacceptable in a potentially lethal environment where each person must maintain high performance if the group is to survive.

o *Sustained Motivation* Prolonged isolation and confinement are associated with a loss of motivation. Those who participate in the Mars mission must remain willing as well as able to perform.

o *Emotional Stability* Emotionally unstable people cannot be trusted to perform their duties in a satisfactory manner. Such people prove disruptive because other people have to direct attention away from their own work to deal with them, and because emotionally unstable people are likely to create turmoil.

o *Positive Social Relationships* Isolation and confinement tend to heighten social tensions and hostilities and direct energy away from constructive activities. Another goal of psychological intervention is to promote harmonious relationships within the crew and between the crew and mission control.

Mission Outcomes

Intrinsically desirable in and of themselves, criterion behaviors give rise to other desirable results. These include safety, high performance, a high quality of life, and positive public relations (Connors & Harrison, 1988).

o *Safety* Competent, motivated, emotionally stable individuals who are receptive to other people's ideas and views are unlikely to make decisions or act in ways which jeopardize the mission's welfare.

o *High Performance* High output and low error rates also follow from the criterion behaviors.

o *High Quality of Life* To the extent that the criterion behaviors prevail, mission participants should enjoy a high quality of life during transit and on the surface of Mars. Studies of quiality of life in industrial settings show that a high quality of life is associated with favorable attitudes and with personal high commitment to an organization's mission (Steers, 1984).

o *Positive Public Relations* Astronaut behavior affects the public's confidence in and support for manned space programs. The image projected by spacefarers has been of concern in the American and Soviet space programs since their inception, and this is unlikely to change in the future. The criterion behaviors described above are likely to sustain public support for subsequent manned space ventures (Harrison, 1986).

Longitudinal and Contextual Factors

Whereas our attention is drawn to the voyage itself, a complete analysis must be *longitudinal* in the sense that the mission involves a sequence of events. As Harris (1986, 1987) has pointed out, these are:

o the attraction and selection of qualifed crew candidates
o training
o supporting the crew during the mission proper
o easing the transition back into the home community

Behavioral analyses must also be *systems oriented* and encompass not only the crew itself but also the relationship of the crew to mission control, crewmembers' families and friends, and society-at-large.

Sources of Data

Analyses of the psychological and social dimensions of spaceflight should follow the ways of science - careful, naturalistic observation and controlled experimentation. Thusfar, apart from case histories, we have relatively little data from space itself. However, over the years we have gained substantial information from spaceflight-analagous environments (Table 3) which share some of the environmental characteristics of spaceflight. It is studies undertaken in these environments, as well as studies conducted in more mundane settings, that provide the basis for the present recommendations. Prior to the departure of the Mars crew, it is crucial to obtain additional data (Brady, 1983; Christensen & Talbot, 1985; Connors et al., 1985, 1986; Nicholas, 1987; Santy, 1987; Space Sciences Board, 1987; Stuster, 1986).

Table 3

SOURCES OF BEHAVIORAL DATA FOR SPACEFLIGHT

Spaceflight Environments past space missions, especially extended missions or missions involving relatively large crews; current and future missions, especially Space Station and Lunar Base (Boeing, 1983a)

Polar Environments Palmer, Pole, Siple and other relatively small Antarctic bases (Bluth, 1987; Harrison, 1988)

Submarine and Marine Environments nuclear submarines and research submersibles; historical voyages of exploration and discovery; merchant marine environments, especially supertankers and superfreighters; small Coast Guard and Navy ships on extended patrol (Boeing, 1983b; Finney, 1987; Weybrew, 1987)

Remote Worksites offshore oil rigs, and remote military bases including SAC Control Centers, missile silos, certain remote national parks, Coast Guard LORAN sites, and possibly airships

Laboratory Simulations spaceship simulators and laboratory settings which include substantial degrees of isolation and confinement (Altman, 1973; Haythorn, 1973)

Evaluation Requirements

Although the recommendations in this paper are based on cumulative research, whenever possible recommendations should be tested prior to implementation on the Mars mission. As noted by Harrison, Caldwell, Struthers & Clearwater (1988), these tests should occur in high fidelity mock-ups and simulators, spaceflight-analagous environments, and aboard the Space Station. Additionally, there should be careful evaluation of the behavioral aspects of the mission itself so that adjustments can be made for subsequent missions.

Preliminary Recommendations

1. Because psychological and social variables have implications for long range planning including spaceship and lander design, human factors must be taken into account early in the planning process.

2. We recommend applying a full range of interventions (selection, training, and engineering) to align the capabilities of the crewmembers and the demands of the flight. The immediate aims of these interventions are high task performance, high emotional stability, and positive social relations. The ultimate goals are survivability and safety, high performance, high quality of life, and good public relations.

3. Although the behavioral recommendations in this report are based on naturalistic observations and experimental research, whenever possible they should be further tested prior to application to the Mars mission. These tests should involve high-quality simulators, spaceflight-analagous environments, and the Space Station.

4. We require additional behavioral research to plan the Mars mission. The overall research effort should address events before, during, and after the flight, and should include mission support personnel, crewmembers' associates, and the public as well as the crewmembers themselves.

5. Each manned spaceflight should serve as a behavioral laboratory for subsequent missions.

CREW SELECTION

Selection refers to choosing from among a large pool those candidates who, as a result of extensive training, will be the best able to accomplish the mission. Selection involves not only choosing outstanding individuals but choosing an aggregate or group of people whom, taken together, will perform synergistically and satisfy all mission requirements.

Methods

Among the common predictors of human performance are (1) biographical data, (2) psychological tests, (3) behavioral tests, (4) interviews, and (5) assessment centers. The current report draws upon all of these and recommends a multistage selection process which requires fast and inexpensive means for initially screening large groups of candidates followed by increasingly painstaking methods for selecting first the most promising individuals and then the most promising team.

Biographical (Life History) Data

Biographical data -- age, gender, education, work history, and the like -- is highly objective. Of many tens of thousands of volunteers, most will be eliminated promptly on such objective bases as age (too young or too old), weak academic credentials, lack of appropriate technical specialization, and so forth. Those who survive this initial screening should undergo a relentless background check for signs of incompetence, emotional instability, or an inability to get along with others.

Psychological Tests

Psychological tests may be divided into three general categories: ability, interests, and personality.

Tests of Ability IQ tests or tests of general ability tend to be most effective when the pool includes candidates who have very different levels of intelligence. This will not be true of the pool of candidates for the Mars mission. Since serious candidates will have advanced academic degrees it is unlikely that IQ tests would make meaningful discriminations among them.

Interest Inventories Interest inventories compare candidates' attitudes and values with those of people who have made satisfactory career adjustments. To the extent that the examinee's interests correspond with those of people within an occupation, the examinee is likely to find that line of work personally involving and rewarding. Whereas interest inventories have not yet been applied to groups that have successfully adapted to isolation and confinement, this application is a simple research task.

Personality Tests Personality tests are applicable in three ways. The first is to use diagnostic tests (for example, the Minnesota Multiphasic Personality Inventory and the Rorschach Inkblot Test) to find people who are likely to experience adjustment problems. The second is to use tests specifically designed to identify people who are likely to show positive forms of behavior that are of advantage to the mission. The third is to use tests such as the Thematic Apperception Test (TAT) to identify people who are likely to "mesh together" and form compatible groups.

Behavioral Tests

Past behavior is the best predictor of future behavior. Having candidates undertake realistic Marsflight operations aboard the Space Station or in high-fidelity simulators will provide good estimates of that candidate's level of skill. Research with submariners has shown that candidates who do poorly during simulated missions almost never do well during actual missions (Weybrew, 1987).

Interviews

Carefully planned and well conducted interviews by seasoned interviewers can aid the selection process. Interviewers should include:

o *technical specialists* who can evaluate the candidate's level of expertise

o *psychiatrists and psychologists* who can assess the candidate's motivation and emotional stability

o *the crew commander and experienced astronauts* to ensure that nobody is selected whom the commander will find objectionable and remind the candidates that their primary obligation is to the crew

Assessment Centers

Assessment centers involve prolonged, in-depth evaluation of a limited number of promising candidates. The defining characteristics of this method are (1) multiple measures, including varied psychological and behavioral tests, special exercises, and interviews, and (2) the use of multiple raters, so that each candidate is evaluated from many different perspectives. These procedures reduce the biases or errors that are associated with any individual assessment device or evaluator and hence increase predictive validity.

Selection Variables

Potentially useful selection variables include demographics (age, gender, and ethnicity), technical skills, and personality.

Age

Providing that a candidate enjoys excellent health, maturity is more likely than youth to confer an advantage for space exploration. Young individuals have had relatively little time to build technical expertise and are likely to be drawn to activities that are difficult to sustain in the close confines of a spacecraft. Current plans for the Space Station involve a crew with an average age of 40 (NASA, 1987), and middle age seems appropriate for a Mars crew.

Gender

Both men and women are capable of performing the tasks required for Mars exploration. A "mixed sex" crew more closely approximates the conditions found on Earth than does an all male crew, and the inclusion of members of both sexes will increase social variety. Drawbacks could include prejudicial or sexist attitudes such that members of one sex view members of the other as incapable of performing all required duties, and the formation of close affectional bonds that provoke jealousies or otherwise cause social frictions.

Ethnicity

There are at least three good reasons for choosing an international crew: (1) including representatives of many different nations should lead to international financial backing; (2) compared to a national pool of candidates, an international pool will contain greater talent; (3) an international crew may help reduce world tensions. However, an international crew could complicate the planning process.

Anthropometric Constraints Physical differences among people of different ethnic backgrounds which will have to be taken into account when planning interior spaces and person-machine interfaces (Clark & Corlett, 1984; Sanders & McCormick, 1987). On the one hand, there must be enough room to accommodate large people; on the other hand, workspaces and living spaces must be arranged so that tasks can be performed by small people. It makes sense to follow Space Station guidelines. Facilities and equipment should be suitable for all but the smallest (5% of oriental females) and largest (5% of American males) potential astronauts (NASA, 1987) .

Cultural Variability International crews mean that cultural differences will have to be taken into account. Second language proficiency levels that are adequate under normal conditions may prove inadequate given the need for highly technical communications under spaceflight conditions (Connors, 1987). Each person will have to be truly fluent in a common language to ensure accurate communication under highly trying conditions. Culturally based preferences and aversions in such areas as diet, recreational activities, and privacy levels will have to be taken into account. Possible problems include the formation of coalitions or "blocs" along ethnic lines, and prejudicial attitudes towards people from other cultures.

Skills

Crewmembers must possess two broad sets of qualifications or skills. First, each crewmember will require *technical skills* which can be applied to tasks during the transit and surface exploration phases. A person's skills for each phase may or may not be closely related to each other. In the aggregate, at least four types of tasks must be performed by the crew (Connors et al., 1985; Kanas & Fedderson, 1971):

o *flight operations tasks* involve command, navigation, flight engineering, systems monitoring, and telecommunications

o *scientific investigative tasks* involve research: the generation of new data that have relevance beyond the immediate flight

o *environmental support tasks* involve the maintenance of facilities and the management of supplies

o *personnel support tasks* include promoting physical and mental health and attending to the morale of the crew as a whole.

"Doubling up" is likely in the sense that one person may have to perform more than one work role. Also, to the extent that each person can gain expertise in more than one role, he or she can provide backup in the case that another crewmember is incapacitated. For each astronaut, it should be possible to identify four role or task sets as noted in Table 4.

Table 4

TASK CATEGORIES

	Primary	Secondary
In Flight	Set 1	Set 2
On Surface	Set 3	Set 4

Each crewmember will also require human relations skills. Negative interpersonal attitudes and social tensions are commonly-reported results of isolation and confinement. Social tensions within an isolated and confined group are more likely to result in social withdrawal than in open aggression, since there appears to be tacit recognition that open conflict is unacceptable. Irritability and anger are often redirected towards outsiders. A requirement for Marsfarers is to create a minisociety which is not necessarily free of tensions but one in which neither internal conflicts nor disputes with outsiders take on destructive properties.

Mental Health

Unfortunately, people with poor histories of adjustment are often drawn to dramatic and unusual undertakings. A high level of technical expertise should not be allowed to override a candidate's poor personal and social adjustment. As Connors and Harrison (1988) have noted:

> Marsfarers should be stable in the sense that they think clearly and rationally; display emotional reactions that are appropriate and useful in the situation; have generally positive views about themselves, their colleagues, and their mission; and refrain from acting in ways that their colleagues find troublesome. Upon return to Earth, the Marsfarers should be able to reassimilate into their families and home community with a minimum of disruption and bother. Finally, the overall impact on their future lives should be positive.

Social Compatibility

Members of space crews resemble the pieces of a jigsaw puzzle in that the crucial consideration is the way the different pieces combine with one another to form a coherent and attractive pattern (Altman, 1973; Haythorn, 1973). Research on crew compatibility is in its infancy, but there are certain helpful leads.

Attitude and Value Similarity People who have similar social, moral, and ethical values get along better than do people who have differing values, a finding which may raise some concerns about the welfare of multinational crews. The Mars mission itself will be an important shared value or point of mutual interest for all participants. Commonalities might be further increased by selecting individual crewmembers in such a way that each pair of all possible pairs of crewmembers has shared interests, even though the specific interests that are shared vary from pair to pair.

Androgyny In many societies, men are expected to be autonomous, independent, somewhat dominating and aggressive, and emotionally inhibited. These are *instrumental* or task-oriented activities. Women are expected to be warm and nurturant and to openly display their feelings. These are *expressive* activities. According to Helmreich, we might seek *androgynous* people who are both instrumental and expressive (Helmreich, 1983; Helmreich, Wilhelm & Runge, 1980). Crewmembers who are exclusively expressive may be perceived as falling short of the high performance standards required for a successful space mission. Candidates who combine the best of the "masculine" with the best of the "feminine" may be in a particularly good position to help the crew reach its technical objectives while at the same time minimizing social tensions.

Achievement Orientation without Competition Helmreich and his associates (1980) also recommend individuals who work hard because they find their work intrinsically satisfying or else because they have an interest in improving their own level of efficiency and performance. They add that we should avoid those individuals who are "competitive" in the sense that their hard work is motivated by a desire to "look better" than their colleagues since such individuals tend to promote unhealthy rivalries.

Need Compatibility People's needs can mesh in ways that affect interpersonal relationships. People have compatible needs when the satisfaction of one person's needs results in the satisfaction of the other person's needs. For example, if each of two people has strong affiliative needs, each person can find satisfaction through associating with the other. Laboratory studies have shown that in comparison to people with incompatible needs, people with compatible needs adapt better to one another and to the conditions of isolation and confinement (Altman, 1973; Haythorn, 1970; 1973).

Recommended Selection Sequence

We recommend a multistage selection process in which the candidate pool is successively narrowed in a series of "cuts" until a final crew is selected. Whereas the early "cuts" are based on crude and inexpensive methods, the final decisions are based on intensive and extensive combinations of procedures. Since the crew will have to function as a team, we recommend that assessments be made of entire crews of individuals as well as of specific individuals (Table 5).

Nonchosen Candidates

There will necessarily be many candidates who are highly qualified and who have undergone extensive training and yet who are not selected for the initial Mars mission. It will be important to find good uses for these valuable human resources. Some may be promising candidates for Space Station, Moon Base and subsequent Mars missions. Others may perform outstanding services at remote work sites on Earth, or serve as mission control personnel. The point is that these candidates' personal investments and the investments society makes in them should not go to waste (Clark, 1987)

Table 5

A MULTISTAGE SELECTION PROCESS

Decision Point	Decision Basis
5-final group	select best-performing crew on the basis of performance in spaceflight or spaceflight-analagous environments
4-initial group	select five complete crews on the basis of the intermeshing of technical skills and predicted interpersonal compatibility
3-final individual	select the most promising individuals on the basis of assessment center methodology
2-intermediate individual	select the most promising individuals on the basis of interviews, background checks, and initial psychological/ behavioral tests
1-initial individual	identify viable candidates on the basis of biographical facts

Crew Selection Recommendations

6. Goals of selection are (1) to identify individuals who are technically skilled, highly motivated, and emotionally stable and (2) to compose groups of individuals who, in the aggregate, are socially compatible and satisfy all of the technical requirements of the mission.

7. We recommend a five step selection process beginning with procedures to select individuals and concluding with procedures to select teams.

8. Selection should involve sensible application of a variety of predictors including biographical data, psychological tests, behavioral tests, interviews, and assessment centers. Initial psychological screening should be on the

basis of quick and inexpensive methods, and subsequent cuts should be based on increasingly elaborate techniques.

9. We recommend candidates who will be in their late thirties or early forties at the time of their departure.

10. We recommend including both women and men in the crew. However, it will be necessary to:

- o prevent gender prejudices
- o insure that male-female parings do not prove disruptive.

11. We recommend a multinational or ethnically mixed crew since this will allow planners to draw crewmembers from an immense pool of talent and also provide political and economic advantages. However, it will be necessary to make sure that:

- o facilities and equipment accommodate people of varying sizes
- o crewmembers share a common language
- o culturally based preferences and aversions are taken into account.

12. Each crewmember should have two sets of technical skills: those that apply during the flight, and those that apply during surface exploration. In each domain, each crewmember should have both primary and secondary (backup) skills.

13. Each crewmember should be interpersonally as well as technically skilled.

14. Each crewmember should have excellent mental as well as physical health.

15. Crewmembers must be compatible with one another. To these ends we recommend crewmembers who have:

- o similar values and attitudes
- o androgynous characteristics
- o high work and mastery orientation but low competitiveness
- o complementary needs

16. Highly qualified candidates who are not ultimately selected for the first Mars crews should be given assignments which allow them to put their hard-won skills to good use.

TRAINING

Training transforms astronauts raw potentials into the precise skills required for a successful mission. Training goals include developing individual talents and forging loose aggregates of individuals into smoothly functioning flight and surface crews.

A Broad Conception of the Marsflight Team

Although most training efforts will be directed towards the flight crew it will also be necessary also to prepare mission personnel and crewmembers' families.

Mission Control

Flight and mission control personnel are parts of a larger team. It is particularly important that crewmembers and support personnel be able to clearly understand each other and to relate to each others' circumstances. Mission control personnel must be highly sensitive to the difficulties of working and living in space. Thus, experienced astronauts should be included on the ground support team, and at least some crewmembers should gain experience in mission control prior to their departure.

Crewmembers' Families

The selection and training processes will be long and arduous. Training may take longer than the mission itself; the entire project could demand a five to ten year commitment. Immediate family members need to understand the selection and training procedures, and to develop realistic and clear expectations regarding the mission itself.

Authentic Training Settings

As much as possible, training should involve authentic situations and tasks and take place in spaceflight-analagous and spaceflight environments.

Antarctica Antarctica is an appropriate site to study human behavior under spaceflight-analagous conditions and an excellent potential training ground for space (McKay, 1985; Bluth, 1987; Harrison, 1988). Many of the conditions that will unfold as we establish camps, bases, and colonies on the moon, on Mars, and beyond, have prototypes on the southernmost continent on Earth. It would be relatively easy and inexpensive to construct an Antarctic base which mimics spaceflight and lander architecture and design.

An Antarctic training base could serve several important purposes.

- o *A Testing Ground for Habitats, Equipment, and Supplies* Antarctica is a useful site for testing spaceship and lander architectural configurations, equipment, and supplies. Antarctic tests can help us assess material under rugged conditions.

- o *A Testing Ground for People* Antarctica is an appropriate site for conducting mission-relevant medical and behavioral research and for developing appropriate training procedures.

- o *A Testing Ground for Scientific Procedures* Antarctica provides a useful location for developing and testing many of the scientific research procedures that are to be employed on Mars.

Space Station and Lunar Base Since the Space Station is likely to be in operation well prior to the Mars Mission, it will provide an ideal setting for Mars bound astronauts to learn how to perform difficult or complex tasks under conditions of microgravity. Similarly, if the Moon Base is established, it will provide a useful training grounds for surface activities.

Methods

Elementary training procedures are well known. These include *direct tuition* (learning from lectures and texts), *modeling* (learning through observing accomplished performers) and *practice* (learning by doing). In general, training should proceed (1) from direct tuition through modeling to practice, (2) involve increasingly difficult tasks, and (3) involve increasingly realistic operations and settings. At any point, tasks should be graduated in terms of difficulty but within the reach of the trainees. Feedback should be prompt and accurate, and increasingly proficient performances should be a requirement for praise or other forms of reward.

Training Considerations

Training needs include (1) technical training; (2) human relations training; (3) stress regulation; and (4) team building.

Technical Training

Although crewmembers selected for the Marsflight training will already be experts in their fields there remain three basic challenges in the area of technical training.

o Crewmembers must learn how to apply their general skills (for example, astronomy) to the specific requirements of the mission (for example, making astronomical observations from space or from the Martian surface).

o Crewmembers must learn how to do their work under spaceflight conditions such as weightlessness, cramped quarters, and limited equipment and supplies.

o Crewmembers must develop their secondary or "backup" skills which can prove crucial in the case that another crewmember is incapacitated.

It will be necessary for critical skills to be "overlearned," that is, learned well enough that they can be put to use in an almost automatic way under highly stressful conditions.

Human Relations Training

Given the rigours of living under conditions of unprecedented isolation and extremely close confinement for an extended period of time, mission participants should be trained in human relations. Both crewmembers and mission controllers must be sensitive to signs of withdrawal and hostility and prepared to react appropriately.

Coping With Threat

At some point, Marsfarers are likely to encounter frightening conditions. This can lead to poor performance through the following sequence:

threat -> fear -> high stress -> poor performance

Two types of training can help Marsfarers cope with dangerous conditions. These are threat control and fear control (Leventhal, 1970; Leventhal & Lindsley, 1972).

Threat Control Threat control exists when a person knows how to counteract dangerous conditions. For example, if an astronaut knows exactly how to deal with an electrical fire, the fire becomes less dangerous, that is, less of a threat. To the extent that astronauts are able to counteract threat they can also minimize fear and maintain high levels of performance.

Fear Control It is not always possible to anticipate dangerous conditions. If a problem arises for which there is no well-rehearsed solution, a high level of fear may make it difficult to respond appropriately. A second line of defense is *fear control*, that is, being able to control the bodily and psychological states that define fear. This can be accomplished through conditioning astronauts to remain calm in dangerous situations and through training them to make responses which are incompatible with fear.

Alleviating Chronic Stress

The cumulative effect of prolonged isolation, confinement, deprivation and risk can cause chronic stress. There are several procedures for combatting stress (Levine, 1987).

o *biofeedback training* involves learning how to control psychophysiological stress responses (heart rate, sweating, breathing rate, etc.)

o *relaxation training* involves relaxing muscles, achieving comfortable postures, and acquiring other responses which are incompatible with stress

o *meditation training* helps people focus on calm inner events rather than stressful external events

Team Building

Additional training will be required to weld individual astronauts into a closely-knit, high-performing team. *Cohesiveness* is commonly used to refer to the "groupiness" of a group; that is, the extent to which group members consider the group important, are highly involved in group activities, and are committed to the group and the group's values (Cartwright, 1968). Generally speaking, cohesiveness is high when:

o the crew is friendly and supportive and satisfies each crewmember's social and emotional needs

o the crew engages in activities that each crewmember finds intrinsically satisfying

o the crew allows members to achieve goals that they could not achieve as individuals

o crewmbers have a high investment in the crew in that they have undergone considerable trouble and expense to gain membership

Tough physical and mental conditioning exercises promote threat control, fear control, and team building. Centuries of experience preparing warriors for combat shows that subjecting recruits to strenuous, dangerous, and even life-threatening conditions builds self-confidence, reduces fear, and increases the ability to respond constructively. Such training also forges strong interpersonal bonds within groups of trainees. Ranger, Special Forces, and Marine Reconnaissance courses which involve such challenges as parachute and survival schools produces team members who are highly committed to each other's well being.

Preparatory and In-flight Training

Although all critical skills must be learned prior to departure, in-flight training can be a constructive way of passing time on the long Earth-Mars journey.

Training Recommendations

17. Although the focus of training will be the Mars crew itself, closely co-ordinated training should be given to mission control personnel and the crewmembers' immediate families.

18. Experienced astronauts should be well represented within the ranks of mission support personnel.

19. Flight and support personnel should be developed as a single, overall team.

20. Training should occur within environments which bear correspondence to spaceflight environments. Early training should involve special bases in Antarctica, and final training should take place in space itself.

21. Technical training should be supplemented with human relations training.

22. Astronauts need to know how to control fear itself as well as the dangerous conditions that provoke fear.

23. Biofeedback training, relaxation training and meditation training will help crewmembers deal with chronic stress.

24. Physically and psychologically demanding training exercises are recommended in the interests of preparedness and morale.

ENVIRONMENTAL ENGINEERING[1]

Habitats, facilities, supplies and procedures must be designed in such a way as to accommodate the behavioral requirements of the crew. Early spaceflight environments provided a relatively low level of comfort, but a level that was acceptable given the brief mission durations. Marsflight environments, which will be inhabited by the same crew for extended periods of time will have to be liveable. The spacecraft and the Mars base will require high-quality working, living, and recreational facilities, carefully designed equipment, and well chosen supplies.

Architecture and Design

Architectural and design elements have profound effects on human behavior. Major goals include:

- o adequate interior space
- o supporting varied work, living, and recreational activities
- o ensuring privacy
- o high environmental coherence or legibility
- o adequate availability of windows or viewing ports
- o excellent illumination
- o good noise control
- o good odor control

Interior Volume

Truly comfortable working and living modules will be required despite cost, mass, and volumetric constraints. Required habitable volume per person increases as a function of mission duration but reaches asymptote at approximately six months (NASA, 1987). Table 6 provides estimates of tolerable (5 m^3 per person) and optimal (17 m^3 per person) interior volumes given crews of size two to twelve and assuming a mission of over six months.

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1. Parts of this section are drawn from the final reports of NASA Grants NAG 2-357 (Harrison, Sommer, Struthers & Hoyt, 1986) and NAG 2-431 (Harrison, Caldwell, Struthers & Clearwater, 1988). I am particularly indebted to Barrett Caldwell and Nancy Struthers for their contributions to the development of these earlier reports.

Table 6

INTERIOR VOLUME REQUIREMENTS FOR MISSIONS OF SIX MONTHS OR MORE

Crew Size	Tolerable	Optimal
2	10	34
3	15	51
4	20	68
5	25	85
6	30	102
7	35	119
8	40	136
9	45	153
10	50	170
11	55	187
12	60	204

To ensure that each area will be large enough to support necessary activities it will be necessary to identify *activity envelopes* which will accommodate people of differing sizes and provide ample storage areas for supplies and equipment. Activity envelopes for microgravity-adjusted individuals ranging from the 5th percentile oriental female through the 95th percentile American male have been presented in NASA STD-3000 (NASA, 1987). In addition to physical requirements, social requirements will have to be taken into account. For example, within each culture, different distances are considered comfortable for different types of social interaction (see Sommer, 1969). Most non-intimate task-related interactions between two or more persons will require that each person be separated by approximately three feet of space.

There are several techniques for increasing the apparent size of an area without increasing actual volume. These techniques, which tend to reduce perceived crowding and the associated stress, include the use of horizontal rather than vertical layouts (Nixon, 1986), the use of light or desaturated interior colors (Mandel, Baron & Fisher, 1980; NASA, 1987; Raybeck, 1987; Schiffenbauer, Brown, Perry, Shulak, & Zanola, 1977), the availability of windows and artworks with distant vanishing points (Al-Sahhaf, 1987; Haines, 1987; Nixon, 1986; Weybrew, 1987), and irregular rather than symmetrical interior design configurations (NASA, 1987; Wise, 1987).

Functional Requirements

The Mars crew will be an entirely self-sustained microsociety for a period of years. The spaceship, and then the lander, will have to support all living, recreational, and work activities, most likely in the absence of resupply. A major part of future planning will be a comprehensive *activity audit* to identify all of the living, working, and recreational activities that will occur throughout the course of the mission. Some of these activities will involve individual astronauts, others will involve the entire Mars crew. Table 7 only hints at the variety of activities that the spaceship and lander will have to support (NASA, 1987).

Table 7

SAMPLE MISSION ACTIVITIES

o sleep	o private recreation
o body cleansing	o waste elimination
o dressing and undressing	o clothing maintenance
o medical care	o life sciences experimentation
o systems monitoring	o physical sciences research
o eating	o exercise
o recreation	o meetings and teleconferences

User Definition

In the absence of strict volumetric limitations, it would be possible to have a large number of task-dedicated areas. In the case of the spaceship and lander, there are strict volumetric limitations, and some interior areas will have to be multipurpose. Such areas should be made definable and redefinable by their occupants. Flexibility can be incorporated through the careful planning of "hard" architectural features (interior dimensions, walls, hatches, etc.), the use of lightweight or "soft" features (screens, moveable partitions, fold-down and pop-out furniture and the like); the availability of small personal items that can be used to stake out temporary territories, and the creative use of color, light, and decor.

Privacy

Privacy exists to the extent that an area's occupants can limit unwanted forms of social intrusion. Restricted access serves a number of important functions (Altman, 1973, 1975; Bossley, 1976; Foddy & Finighan, 1980; Marshall, 1974; NASA, 1987; Nixon, 1986; Raybeck, 1987):

o privacy encourages the high degree of concentration required for complex scientific and technical tasks

- o privacy provides "down time" for rest and recuperation
- o privacy helps people manage their relationships with one another

The opposite of privacy --- crowding --- is associated with psychophysiological and other indicators of stress, performance deterioration, and poor health (Evans, 1979; Epstein, 1981; Karlin & Epstein, 1979).

Privacy is increased by separating people in physical space, establishing visual and auditory cut-offs which reduce the extent to which they see and hear one another, and keeping public areas neat, clean, and odor free.

Environmental Legibility

Microgravity makes possible design configurations which are not possible under normal gravitational conditions. However, configurations which do not parallel those found on Earth can be confusing. Spacecraft environments should provide coherent frames of reference so that occupants can immediately orient themselves to a location and the equipment and supplies within it. This is particularly important under emergency conditions.

An important contributant to environmental legibility is a true vertical, a cue which is not available under conditions of microgravity. To some extent one can compensate for the lack of a true vertical by arranging the environment "as if" it were in a gravity field, and, according to recent unpublished research by Coss, Barbour and Clearwater (1987), by such simple expedients as having the "floor" and the lower half of the wall a darker color than the "ceiling" and the upper half of the wall.

Windows

Windows serve a multitude of important purposes (Haines, 1987). They give astronauts something to look at when they want a change of pace, or when they want to avoid looking at another person. By providing distal visual fixation points, windows can help reduce feelings of crowding. They provide an important visual link with Earth. This link can be enhanced by the availability of a telescope (Clark, 1987).

Illumination

Proper illumination and minimal glare are essential. Lamp color can have important psychological effects. A subjectively assessed dimmension of light, lamp color depends on the illumination level, the distortion or exaggeration of various segments of the emitted visual spectrum, and other factors (Boud, 1973; Harrison et al., 1988). People associate lower levels of illumination and redder sources of light with night and social activities, and higher levels of illumination and bluer colors with daytime and increased activity. Variations in illumination and color can reproduce changing conditions on Earth and set the stage for work, recreation, socializing, sleep, and other activities.

Fluorescent lights offer a number of technical and behavioral advantages (Boud, 1973; Harrison et al., 1988). Compared to incandescent lights, fluorescent lights are efficient, resistant to vibration, generate less radiant heat, and are more durable. Fluorescents increase the accuracy of color perception, and replicate illumination on Earth. People's moods are very much affected by the amount of sunlight that they receive each day; holding temperature constant, decreasing amounts of sunlight are associated with increased depression. Daily exposure to full spectrum fluorescent light alleviates depression stemming from insufficient sunlight.

On Earth, diurnal cycles are determined in large part by *Zeitgebers* (time givers) such as sunrises and sunsets. During Marsflight, interior lights are the *Zeitgebers*, and, within limits, bodily cycles will synchronize with their onset and offset. As the spacecraft proceeds towards Mars, interior illumination should initially duplicate day-night cycles found on Earth but then shift towards the cycles found on Mars. During the return journey, the reverse shift would help crewmembers re-adjust to normal conditions.

Sound Control

Even apart from launch phases spacecraft tend to be noisey environments, with an adverse impact on speech communication, relaxation, and sleep. For long duration missions, it will be particularly important to limit noise. The recent CHABA report (NRC, 1987) addressing issues of sound and vibration control on the Space Station recommended that:

o work stations should have sound levels not exceeding 55 dBA

o for sleeping areas, background sound levels below 45 dBA are preferred, while levels up to 60 dBA are tolerable

o the risk of significant hearing loss is negligible given continuous (24-hr) exposures to noises at the 80 dBA range

o hearing conservation programs are important when occupants are exposed to 8-hours a day or more of noise at or above 85 dBA

Noise reduction will be difficult and expensive given that the simplest solutions (for example, increasing the mass of sound barriers) are likely to be too expensive. The most promising construction techniques for reducing vibration and noise are the use of highly viscoelastic materials (VEMS) in walls and the use of sound absorbing facings. Additional sound reduction can be obtained through constructing well-sealed enclosures, including rubberized and magnetic door seals (Beranek, 1981; Bullis Crema, Barboni & Castellani, 1982; Doelle, 1972; Harrison et al., 1988).

Two other techniques for noise control are masking and negative sound. *Masking* involves using white sound generators or other sound sources to overpower the unwanted sound. Unfortunately, whereas masking tends to make intermittent sounds less annoying, it contributes to the overall din. Masking may be useful when the masking sound is pleasant (for example, a preferred form of music), under the control of individual listeners, and presented by means of close-fitting headphones. *Negative sound* consists of electronically-generated soundwaves which are the mirror-image of the sound waves associated with loud, continuous, or highly repetitive environmental noises. In effect, negative sound neutralizes or cancels the unwanted environmental sound.

Odor Control

Unpleasant odors can be distracting and annoying. Sensing other people's body odors is a component of crowding, and lingering body odors serve as "contaminants" which render a space less habitable (Altman, 1975; Engen, 1982). Odor control has two aspects, prevention and correction. In the area of prevention, there should be at least one personal waste management facility for every four crewmembers (Stuster, 1986). There should be generous allowances for partial and total personal cleansing, and ample opportunity for clean changes of clothes. This will require laundry facilities aboard the spacecraft.

Masking and decontamination are the two major methods for eliminating unpleasant odors. *Masking*, or the use of pleasant scents to overpower unpleasant ones, increases the concentration of undesirable substances in the atmosphere and is also a poor choice because the masking odors themselves may be unpleasant. A high level of *decontamination* or odor elimination might be achieved through a three-part procedure consisting of (1) particulate removal through *electrostatic processes;* (2) chemical filtration or *dry scrubbing*, and finally (3) passing air through an *ultraviolet irradiation* chamber.

Communications Systems

It is essential to insure excellent communication (1) among members of the space crew, (2) between crewmembers orbiting Mars and those on the surface of the planet, and (3) between the Mars crews and people on Earth.

Internal Communication

Spaceflight conditions make it difficult to achieve accurate verbal and nonverbal communication (Connors, 1987). Verbal communication is likely to be impaired by a high ambient noise level. If the enviroment has sound levels above 50 dBA, occupants will require assistance for adequate speech communication (NRC, 1987). Microgravity, and, on occasion, cumbersome protective gear make it difficult to transmit and "read"nonverbal cues (Connors, 1987). Restraints and positioning devices that locate conversants on the same horizontal plane and in mutual "heads up" positions can alleviate this problem.

External Communication

External audio and video monitoring enables mission control to keep close tabs on the crew's progress and provides the opportunity to intervene when they see fit to do so. Additionally, information gained through external surveillance tends to be newsworthy, and, presented through the mass media, can generate public support. However, unremitting surveillance invades privacy, undercuts the crew's sense of dignity and autonomy, and hurts performance. Crewmembers should be able to shut off remote surveillance systems or have the freedom to move to a location that is "off camera" (Berry, 1973; Bossley, 1976).

Communication with family and friends is an important opportunity that is highly valued by isolated people (Connors et al., 1985; Connors, 1987). High quality, full duplex audio/video external communication systems are desirable, although problems will be posed by "round trip" communication delays of up to 40 minutes. Privacy is an important consideration. Other astronauts should not intrude when one astronaut is using the system. Additionally, the system should include safeguards to discourage electronic "eavesdropping" by unauthorized parties.

Supplies

High quality supplies are imperative. All supplies must be extensively pre-tested and consistent with astronaut preferences. Astronauts should have an active hand in selecting supplies for the mission, and there should be some personal as well as group selections. Astronauts should have the opportunity to include personal possessions in their kits.

Food

Research reviewed by Connors et al. (1985) suggests that under conditions of isolation and confinement food has special significance. The preparation and consumption of food is an important social activity. Food that is perceived as unappetizing or untasty leads to grumbling, dissatisfaction, and hostility towards mission planners. There are likely to be "runs" on certain types of foodstuffs with the result that some foods are consumed early in the mission while other foods are eaten only as a matter of necessity. Food preferences on Earth do not correspond in perfect fashion with food preferences in space, so foodstuffs will have to undergo sensory evaluation in space.

Clothing

Pitted against the sheer practicality of jumpsuits will be astronauts' needs for variety and personal expression in clothing. Some combination of regulation and personally chosen leisure attire may provide an optimal balance. Although some disposable garments can be provided, it is unlikely that there will be a sufficient supply for the entire trip. For this reason there should be some means for laundering.

Recreational Supplies

Provision will have to be made for both individual and group recreational activities. The people chosen for the Mars Mission are likely to have a strong work ethic, and additional work may turn out to be one of the most popular forms of recreation. For this reason, it has been proposed that on truly prolonged missions it might be useful to provide secondary work (that is, work not directly related to the mission itself, for example writing research papers, computer programming, planning future missions, and the like).

Studies of people in isolation and confinement suggest that despite expectations of self-enhancemen("I'll learn German") escapist activities often win out ("I played solitaire"). Also, people in isolated settings tend to avoid competitive games which increase rivalries under already tense conditions. Provision must be made for passive, escapist activities.

Environmental Engineering Recommendations

25. Make the spacecraft and the Mars base as spacious as possible. We recommend for each astronaut an allowance of $17m^3$ of utilizable interior space.

26. We recommend the use of design techniques which enhance the perceived spaciousness of the spaceship and lander. These include:

- o the use of light interior colors
- o horizontal rather than vertical layouts
- o irregular interiors which provide occupants with a range of visual distances and fixation points

27. The spaceflight and Mars base environments should be visually coherent or legible. All facilities and equipment should be oriented within an unambiguous frame of reference including an apparent vertical.

28. As much as possible, Marsfarers should be given control over the spaceship and lander environments.

- o moveable or rearrangeable furnishings will encourage the redefinition of areas to include greater or lesser numbers of people
- o moveable panels and screens may be used to expand or contract work, living, and recreational areas
- o restraining devices should allow astronauts to vary their orientations to one another

29. Following Helmreich et al. (1980), Stuster (1986), and others, we recommend that each astronaut be assigned individual private sleeping quarters. Each cabin should provide visual privacy and freedom from noise.

- o occupants should be able to adjust ventilation and temperature
- o cabins should be equipped with storage space for personal belongings, and pull-down or pop up tables for solitary work or recreational activities
- o inhabitants should be allowed to choose cabin decor including colors and pictures or posters
- o within private rooms, occupants should have the option of monitoring internal and external communications.

30. Multiple personal hygiene facilities are mandatory. They should be located as far away from work areas as possible, be well ventilated, and provide complete privacy.

31. Include locations where individual astronauts can occasionally go to be "alone" without retiring to their cabins, and where small groups of astronauts can gather to engage in social activity.

32. At least one area should be large enough to accommodate the entire crew at any one time.

33. All pieces of equipment, restraints, and aids should be adjustable to accommodate anthropometric variations and personal prefences. Positioning devices should be relocatable.

34. Lighting should do more than allow for good vision.

- o full spectrum flurorescent lighting should be used to promote positive moods

- o area lighting should be made available to help astronauts redefine areas

- o variable intensity illumination will help astronauts regulate the social atmosphere

35. Following Haines (1987), we recommend a generous availability of windows.

36. A telescope should be available to enhance the visual link with Earth.

37. Pictures and other graphic designs are recommended because they offer illusions of depth and tend to diminish the negative impact of minimal interior spaces.

38. Individual personal cassette recorders are recommended. They will:

- o allow astronauts to enjoy personally preferred music without having to take the musical preferences of other astronauts into account

- o provide masking sounds which reduce the annoyance caused by intermittent environmental sounds

- o make possible to use *audio check lists* which leave hands free for work activities

39. We recommend against uninterrupted remote video surveillance. There should be places where crewmembers can go to get "off camera," and these places should include private cabins as well as personal hygiene areas.

40. Personal communications systems will be desirable for communicating with other people aboard the spacecraft or at the Mars base. Inflight paging or intercom systems should make it possible to attract the attention of specific individuals without disturbing others.

41. Communication with mission control personnel and with family and friends on Earth should involve full duplex audio-video systems. Such systems should allow for privacy in the sense that users are free from surveillance by other astronauts and secure in the sense that they are not subject to electronic eavesdropping.

42. Advances in electronics and in liquid crystal technology should make it possible to provide each astronaut with a personal unit. Approximately the size of a lap-top computer, this would include a standard keyboard, a built-in microphone, a speaker or headphone jacks, and a miniature video camera on a gooseneck. This device could function as:

- o a communications device
- o a computer and word processor for both work and recreational purposes
- o a display device for electronic microfiche and video fare

SOCIAL ENGINEERING

Social engineering involves structuring or patterning the relationships among individuals and is reflected in the distribution of social power and assigned tasks. Social engineering considerations include crew size, command and decision making structures, task design, crew rotation, social norms, and social support.

Crew Size

Sheer practicality dictates that the crew size will fall somewhere within the small group range (approximately 2 to 12 members). Building a mission around a single crewmember is discouraged because of the likely adverse effects of complete social isolation and because in the event of incapacitation there is absolutely no back-up. Within the small group range, increasing crew size has a number of effects, some advantageous and some disadvantageous for the conduct of the mission (Table 8).

Table 8

LIKELY EFFECTS OF INCREASING CREW SIZE

Advantages

- o increased range of abilities, talents, skills
- o increased task specialization
- o increased back-up capabilities
- o increased social variety
- o increased group stability
- o decreased likelihood of work overload

Disadvantages

- o increased expense
- o increased crowding
- o increased problems of social coordination and leadership
- o increased likelihood of subgroup formation
- o decreased motivation
- o decreased sense of personal responsibility

As Kanas and Fedderson (1971) note, two-person groups are likely to be riddled with tensions, and three person groups are likely to lack cohesion because they typically split into a two-to-one coalition. Also, even-numbered groups are likely to form even numbered coalitions. Group stability appears to increase up to seven or so members; although Kanas and Fedderson expect increasing stability beyond seven members, this is unproven and such gains may be offset by declining motivation and increased difficulties gaining the necessary level of interpersonal coordination. The evidence is far from complete, but on the basis of social psychological considerations alone, we suspect that the optimum crew size may be five. Beyond that, crews of seven, nine, and eleven are appropriate, but as we proceed upwards the disadvantages of increasing crew size gain salience.

Of course, at Mars, an odd-numbered crew may have to split into two subgroups, one of which will be even-numbered and both of which will be less than optimal size. An odd-numbered crew is not imperative, since decision rules can be formulated to thwart deadlocks.

Command Structure

Command structure determines who is entitled to influence whom. Issues include the allocation of power and authority to the crew, and the distribution of decision making power within the crew.

Internal Authority Traditionally, manned spaceflight has involved military or quasi-military command structures. Such structures take the form of a clear-cut hierarchy in which individuals at one level have the authority to direct the activities of individuals at lower levels. Alternatives to hierarchical or autocratic decision making procedures (under which the commander reaches a decision on his or her own) are a range of consultative procedures (under which the commander solicits advice and opinions but still makes the final decision) and democratic procedures (under which crewmembers make decisions through consensus or ballot). Essentially all contemporary thought suggests that no single approach is useful for all situations (Steers, 1984; Wexley & Yukl, 1977). Some of the variables relating to the optimality of autocratic and democratic procedures are presented in Table 9.

Table 9

AUTOCRATIC AND DEMOCRATIC DECISION MAKING PROCEDURES

Autocratic When:	Democratic When:
o leader is especially capable	o all members capable
o leader has complete knowledge	o members have key info
o time is crucial	o time is unimportant
o stress levels are low	o stress levels are high
o member acceptance unimportant	o member acceptance important

There are definite advantages to encouraging crewmembers to contribute to the decisions that will affect them. Under consultative and democractic procedures, crewmembers would be more likely to understand the rationale underlying the decision and the decision itself is less likely to be seen as impractical or ambiguous. The decision is more likely to be consistent with crewmember abilities and interests. Although decision quality is a very complex issue, given the high capabilities of all crewmembers, decisions based on democratic procedures are likely to have an edge. Democratic procedures do take time, and for this reason alone are not appropriate in all situations. The Soviets report success using autocratic procedures for decisions regarding the mission itself, and democratic procedures for decisions regarding life aboard the spacecraft.

Table 10

CENTRALIZED VERSUS DECENTRALIZED DECISION MAKING

Centralized Recommended When:

- o there are good communications between mission control and the crew
- o Earth based managers have access to staff specialists and other information sources which are not readily available to the crew
- o Earth based managers have the same knowledge of "local" (spaceflight and Mars base) conditions as does the crew
- o decision speed is only a minor consideration

Decentralized Recommended When

- o there are poor or delayed communications between mission control and the crew
- o staff specialists and other information sources are readily available to the crew
- o Earth based managers do not have a clear understanding of "local" (spaceflight and Mars base) conditions
- o decision speed is a major consideration

Mission Control and the Crew Mission control represents a higher authority which sets overall mission goals and procedures. The question is to what level and what extent should mission control determine crew activities. For present purposes, authority is *centralized* to the extent that mission control retains decision making power; authority is *decentralized* to the extent that it is delegated to the crewmembers. Variables relating to optimal centralization (Wexley & Yukl, 1977) are presented in Table 10.

Under Marsflight conditions, decentralization may have the advantage. Communication with Earth will be delayed if not erratic; there will be specialists aboard, and in comparison to controllers stationed hundreds of thousands of miles away, the crew is likely to have an appreciably better understanding of the realities of the situation. Additionally, decentralization permits quick responses to emergency conditions.

Task Design

Role loading, or the sheer amount of work that each person is assigned, is a critical consideration. It will be very tempting to assign each crewmember prodigious amounts of work in order to keep the number of crew members to a minimum. This would be a false economy. Prolonged heavy work assignments lead to stress and to cumulative fatigue which set the stage for inefficiency, error, accident, and dissatisfaction. Studies throughout the past century have repeatedly shown that shortened work weeks, decreased workdays, and increased rest periods lead to increased productivity not only on a per hour basis, but, within very broad limits, in terms of the absolute amount of work. In preparing assignments, remember that work easy to complete under normal conditions can be very difficult to accomplish under conditions of microgravity. Also, the Mars mission will be a very lengthy mission making it impossible to sustain the work schedules which were acceptable on flights measured in days and weeks. Dangers associated with excessive or insufficient workloads can be reduced by having experienced astronauts take part in task design and work scheduling (Weick, 1977).

As much as possible, each astronaut's assignments should be involving and engaging. Tasks should not be allocated in such a way that some crewmembers get all of the "choice" assignments while other crewmembers get the "dirty work." Research by Hackman and Oldham and others (Hackman & Morris, 1975) suggests that motivation is high when:

- o workers find their work meaningful
- o workers feel responsible for the results of their work
- o workers have knowledge of the actual results of their work
- o work draws upon a range of skills
- o work has a visible result
- o work is significant in the sense that it impacts the mission ss
- o work is whole and identifiable (rather than fractionated)
- o work has a visible result

Connors, Harrison and Akins (1985, p. 284) have suggested applying Kahn's (1973) system of work modules to extended duration spaceflight. As these authors explain:

> This is implemented by first determining the shortest length of time that is economically feasible and psychologically meaningful

to work at a given task, such as navigating, analyzing data, or cooking. For purposes of illustration, let us assume this unit of time to be two hours. Time task units define work modules. From the overall perspective, a large-scale mission might consist of thousands of modules involving scores or hundreds of crewmembers performing hundreds of tasks. Under conventional forms of organization, missions might consist of a certain number of shifts or watches, each of which requires repetitive activities on the individual worker's part.

Under the work module system, a crewmember would be allowed to qualify for several different kinds of tasks (such as navigating, analyzing data, and working in the galley) and then construct his or her own schedule using the requisite number of modules. For example, one crewmember might choose two modules of navigating, one of analyzing data, and one of working in the galley to satisfy the requirements of an 8-hr shift. Still another might change job content by day of the week. Moreover, Kahn's system could provide a crewmember with the opportunity to vary the way he or she distributes work in the course of an overall mission. Thus, within the limits established by the individual's qualifications and the organization's needs, crewmembers could, in effect, construct their own jobs.

Crew Rotation

The issue of who is the first to step foot on Mars will be of immense psychological as well as symbolic and perhaps political significance. Order of Extra-Vehicular Activity (EVA) will have to be well thought out and agreed to by all parties in advance. It is essential that the crewmembers consider the method for determining the order of EVA order to be fair.

Crew rotation --- ensuring that ultimately each crewmember has the opportunity to set foot on the surface of the planet --- should help sustain individual motivation and minimize conflicts. For example, given a crew of five, three could land while two remain in orbit. Half-way through the mission, two of the three who had landed would return to the orbiter while the remaining two would would join the experienced surface explorer. Under this scenario, all astronauts would visit Mars, and continuity would be provided by the astronaut who remained at the Mars camp throughout (Clark, 1987).

If there are overlapping successive missions, at any one time one half of the crew at the Mars base could be newcomers while the other half could be experienced "Martian hands." Specific crewmembers at the surface base could be paired with their replacements in Martian orbit and remain in close communication. The idea is that through electronic conferencing systems the first crewmembers on Mars serve as mentors or trainers of their individual successors.

Social Norms

Social norms are unwritten rules that regulate the attitudes and behaviors of group members. Norms specify mandatory, optional, and inadmissible behaviors. Under conditions of isolation and confinement group norms are likely to be strictly enforced. The penalty for violating group norms is rejection. In closed social systems such as spacecrews in flight the penality of rejection can be devastating since there is nowhere else to turn for social acceptance.

Groups which evolve norms over time are likely to withstand isolation and confinement better than are groups which ignore the issues until tensions reach a high level and then try to reach agreement on how things should be done (Altman, 1973). This suggests that crew norms should be well in place prior to the mission. Also, officially prescribed rules and regulations are oftren pre-empted by *emergent* or informal norms that arise in the course of group interaction. This means that a mission's sponsors will have only partial control over group behavior during the mission.

Social Support

Also of value will be mechanisms which provide emotional and social support. Social support can come from both within and outside the crew.

Within-Crew Support Training in interpersonal relations and group dynamics will promote mutual tolerance and support within the spacecraft. Support might also be promoted by means of a psychological "buddy system" where crewmembers are paired and each member of the pair assumes special responsibility for the welfare and morale of the other (Connors & Harrison, 1988).

External Support Mission control personnel should be well trained in interpersonal relations and aware of the needs and idiosyncrasies of each crewmember. The "buddy system" might be extended so that individual flight controllers have special responsibilty for maintaining the morale of specific astronauts. To perform these functions reliably, mission control personnel will themselves have to be carefully screened for competence, motivation, emotional stability, and social sensitivity. The Soviets have reported success with ground-based "Psychological Support" units which monitor the mental health of the astronauts, offer advice and encouragement, arrange for teleconferences with celebrities and find other pleasant surprises (Boeing, 1983a).

Frequent communication with family members reinforces ties with home and reassures each party about the other party's welfare. Communication with family members is not a panacea; people in isolation and confinement can get upset if they are unable to communicate with home at a designated time, and frequently report that following communication with home there is a mild psychological "let down." Also, methods will have to be developed to overcome the distracting if not disorienting result of communication delays of up to forty minutes.

If successive Mars crews are in training at the time of the mission, frequent communication between the astronauts in flight and the trainees may be of use. The astronauts in flight can serve as mentors who help prepare the trainees while the trainees provide astronauts in flight with encouragement and support. The performance of the astronauts in flight is likely to benefit as they strive to be excellent role models for astronauts in training.

Family Astronauts' families will themselves require social support. They will need a full understanding of both training and mission procedures. They will need to deal with the fact that a spouse, parent, child or sibling will be setting forth on an unprecedented training program and voyage that may involve years of separation. They are likely to need protection from the media and from curiosity seekers. Past experience suggests that, over time, the family adjusts to one member's absence. However, that person's return causes new dislocations in the family system and, even as the family had to adjust to the person's departure, it has to re-adjust to that person's return. Support services will have to be extended to help the astronauts and their families readapt to one another following the astronauts' return to Earth.

Social Engineering Recommendations

43. Balancing cost and social psychological factors we recommend a crew of five members.

44. To achieve a good balance between decision quality and member acceptance, we recommend a mixture of autocratic and democratic decision making procedures. Subject to constraints imposed by the crewmembers' knowledge and ability and by time, crewmembers should contribute to the decisions that affect them.

45. Because communication with Earth is likely to be delayed and because the crew on the scene is likely to have the best appreciation of "local" conditions, the commander and crew should have considerable latitude to make decisions and take appropriate action.

46. Workloads must be capable of adjustment so that crewmembers are neither bored nor overworked but instead are offered an appropriate degree of challenge. Achieving optimal workloads will require involving experienced astronauts and mission participants in the work planning process.

47. Desirable and undesirable tasks should be distributed across the different crewmembers so that no one individual is overloaded with undesirable tasks.

48. Following the work of Hackman and others, we recommend that

- o tasks be identifiable and whole
- o tasks draw upon a range of abilities and skills
- o tasks have a real and demonstrable impact on mission success
- o individual astronauts have considerable autonomy
- o astronauts be given regular feedback regarding their level of performance

49. Planners should explore Kahn's work module forms of organization.

- o mission requirements are first broken down into time-task units
- o astronauts are allowed to qualify for various tasks
- o astronauts are allowed to piece together patterns or mosaics of assignments

50. All crewmembers should have the opportunity to serve on Mars.

- o the order in which different crewmembers set foot on Mars should be established prior to the mission
- o individual crewmembers in orbit should be in close contact with the individuals whom they will replace on the surface
- o the crewmembers on the surface should serve as individual tutors or mentors for those in orbit
- o successive surface parties should consist of mixtures of novices and experienced explorers

51. Group norms must evolve prior to the crew's departure.

52. We recommend a psychological "buddy system" in which each crewmember assumes special responsibility for the welfare and morale of another specific crewmember.

53. Individual support personnel assume special responsibility for the welfare and morale of individual crewmembers.

54. Following Soviet practices, a support team can be established to monitor and boost the astronauts' morale.

55. Individual crewmembers should be assigned specific trainees for subsequent missions. The crewmembers serve as the trainees' tutors and mentors, and the trainees provide their individual sponsors with support and encouragement.

56. Support services must be available to the astronauts' families.

CONCLUSIONS

Over the past decade or so there has been increasing clamor for paying more attention to the psychological and social aspects of spaceflight. This call has come not only from psychologists and popular authors, but also from presitigious scientific panels (Space Science Board, 1987) and even from astronauts themselves (Collins, 1988). Behavioral considerations are important in and of themselves, and because they will be one of the many determinants of mission failure and success. Human factors will become more, rather than less salient, as we move towards extended-duration interplanetary missions.

Good will on the part of the astronauts will not preclude problems in such areas as motivation, emotion, and social relations any more than it will prevent bone decalcification, cardiac deconditioning, and sleep disturbances. At present, we do not have all the answers. We do not even have all of the questions. We do have sketchy information from space itself, and slowly increasing amounts of data from environments that bear some resemblance to the Mars bound spaceship and lander. However, behavioral problems are identifiable and solvable, and they can be identified and solved through naturalistic observation and controlled experimentation. The time to increase our understanding of the human side of Marsflight is now, for the answers that we find will have important implications for establishing mission parameters and perfecting spaceship and lander design.

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Appendix B

Mars Interplanetary Mission Modules Conceptual Design Study

MARS INTERPLANETARY MISSION MODULES

A CONCEPTUAL DESIGN STUDY

Prepared for
Martin Marietta Astronautics Group
Denver, Colorado

By
Eagle Engineering, Inc.
Houston, Texas

EEI Contract TO 88-39
EEI Report No. 89-246

Foreword

This task was performed during the period from November, 1988 to June, 1989 by Eagle Engineering, Inc. under subcontract to Martin Marietta Astronautics Group, Denver for the NASA George C. Marshall Space Flight Center. The major objective of this study was to define two conceptual designs for the manned portion of an artificial-gravity, Mars interplanetary vehicle. The report provides itemized mass, volume, and power statements of the mission modules and supporting systems. Exterior sketches and interior planviews of each conceptual design are included.

Dr. Ben Clark was the Martin Marietta Project Manager.

Bill Stump was the Eagle Project Manager for the contract governing this task. Lisa Guerra was the Eagle Task Manager for this particular task. In addition the following Eagle employees contributed to this effort:

Technical Support

Eric Christiansen
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Table of Contents

List of Abbreviations

ACS	Attitude Control System
ATCS	Active Thermal Control System
CCC	Command & Control Center
CM	Center of Mass
Cyl	Cylindrical Modules Configuration (2 modules)
Disk	Disk Module Configuration
DMS	Date Management System
DSN	Deep Space Network
ECCV	Earth Crew Capture Vehicle
EDC	Electrochemical Depolarized Cell
EVA	Extravehicular Activity
HMF	Health Maintenance Facility
IMM	Interplanetary Mission Modules
LEO	Low Earth Orbit
MCSV	Mars Crew Sortie Vehicle
MPV	Mars Piloted Vehicle
# units "on"	number of equipment actively using power
PV	Photovoltaic solar array
SFE	Static Feed Electrolysis
S.S. Freedom	Space Station Freedom
VCD	Vapor Compression Distillation

Executive Summary

This study was initiated to develop preliminary conceptual designs of pressurized volumes for a Mars piloted vehicle. The mission profile is defined as a conjunction class mission, with a duration of approximately three years, for a crew of five persons. To accommodate the crew, two independent module configurations were analyzed. One configuration consists of two cylindrical modules (sized in accordance with the Space Station Freedom habitation and laboratory modules) connected by an airlock and a logistics module with pressurized access to the Mars landing vehicle. The other configuration is a single disk module with a diameter of 8.4 meters, three levels, an attached airlock and logistics module, and pressurized access to the Mars lander. Both configurations adhere to a conservative safe-haven requirement in which mission and life critical systems are duplicated to ensure mission success. Figure A illustrates the two module concepts outfitted with supporting mission systems.

The mission modules are part of a Mars Piloted Vehicle (MPV) consisting of two segments - the crew-occupied modules with supporting systems and the propulsion system mated to an aerobrake. The MPV segments are attached to each other by two structured tethers, as part of an artificial gravity system. The vehicle is intended to spin at a rate of 2 rpm to produce a gravity environment ranging from 3/8 g to 1 g. Periods of vehicle despin, particularly while in Mars orbit, necessitate module adaptability to 0-g conditions.

This conceptual design study focused on the definition of the module configurations and the analysis of associated systems. The basic system parameters of interest include mass, power required, volume, and area. In addition to a systems analysis, a volumetric study of the disk module interior and the cylindrical module interiors was performed and interior planviews were outlined.

The systems addressed in this analysis pertain to the pressurized facilities (i.e. life support system) or provide external support to the modules (i.e. power generation system). The pressurized facilities are defined by the following system areas:

- Life support system: a physical/chemical system in which the air and water needs of the crew are regenerated at a 98% closure level;
- Active thermal control system: a single-phase water loop internal to the modules for the acquisition and transport of cooling loads;
- Crew consumables: water, oxygen, nitrogen, and dry goods such as food and clothing used throughout the interplanetary mission to Mars;
- Module interior: seven functional areas defined as quarters, galley/wardroom, command and control center, lab and maintenance work areas, personal hygiene, health maintenance facility, and electrical systems;
- Structures: primary structure, the pressure shell and additional supports, and secondary structure, the equipment racks and internal supports;
- Logistics module: a separate, pressurized module for stowage of consumables, system spares and replaceable units;
- Radiation shelter: a shielded environment for crew protection during solar flare events - the cylindrical module configuration uses the logistics module and the disk module configuration uses a specially shielded volume internal to the module;

- Airlock: an equipment lock and crew lock for depressurization, egress, ingress, and repressurization for two EVA crew members.

To provide external support to the pressurized modules the following systems are included in this study:

- Power system: two retractable photovoltaic solar arrays located on the sun-viewing side of the vehicle;
- Energy storage system: two nickel-hydrogen, rechargeable batteries;
- External thermal control system: two radiators located perpendicular to the solar arrays;
- Attitude control system: four thruster assemblies using storable monomethylhydrazine for fuel and nitrogen tetroxide for the oxidizer;
- Artificial gravity equipment: a two tether system consisting of the tethers, spools, and motors (note that the artificial gravity propulsion system is not included in this design portion of the vehicle);
- Communications: two, 5 meter diameter antennas permitting TV transmission and data transference to Earth;
- Support structure: a lattice of trusswork to connect external systems and distribute the loads during gravity spin-ups, and a 1 meter walkway encircling the pressurized modules for EVA capability;
- Solar telescope: an externally mounted telescope for solar flare observations and astronomical experimentation.

The study results are presented in Tables A and B. Table A provides a summary comparison between the cylindrical modules and the disk module configurations in terms of module sizing, mass, and operations. Table B compares the mission module configurations with respect to the module systems and support systems.

Figure A Conceptual Design Configurations

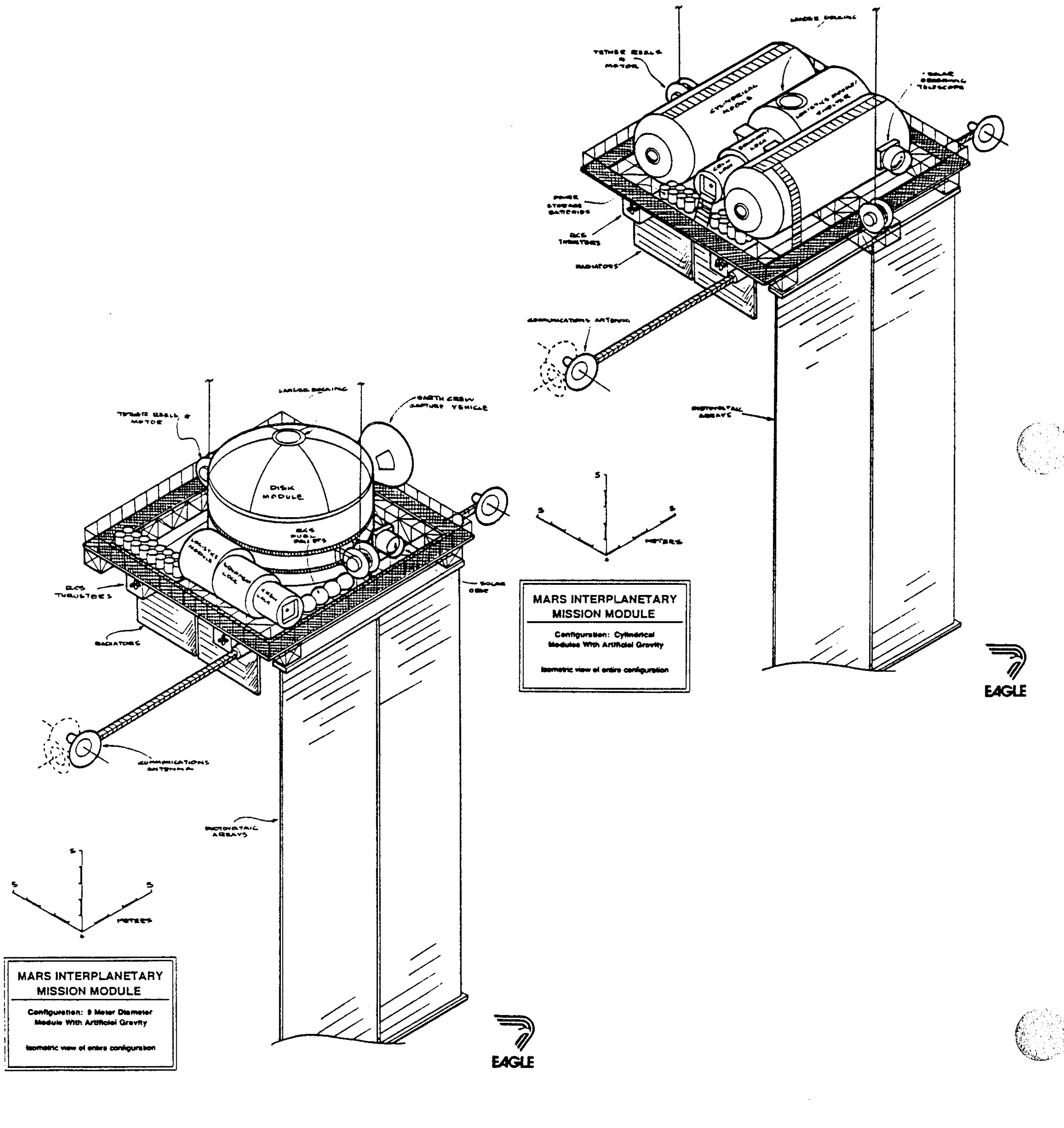


Table A **Summary of Module Configurations Comparison**

Summary Characteristics		Cylindrical Modules	Disk Modules
Mission Requirements			
	Crew Size	5	5
	Mission Duration	3	3
Module Sizing			
	Diameter, m	4.2	8.4
	Length, m	11.8	11
	Total Module Vol., m3	327	519
	Vol/person, m3/p	65	104
	Total Floor Area, m2	99	166
	Floor Area/person, m2/p	20	33.3
	Total Pressurized Vol, m3	425	581
Mass			
	Module Mass, kg	53,950	56,710
	Support System Mass, kg	15,384	14,509
	Total Configuration Mass, kg	69,334	71,219
	Total Mass/person, kg/p	13,867	14,244
Operations			
	Operating Power, kWe	17.4	17.4
	Operating Pressure, psi	14.7	14.7

Table B **System Characteristics Comparison**

		Cylindrical Modules	Disk Module
Life Support System			
	Mass, kg	5,548	5,548
	Power, kWe	4.1	4.1
	Volume, m3	18	18
Consumables			
	Mass, kg	19,194	20,075
	Volume, m3	46	46
Thermal Control System			
	Mass, kg	807	807
	Power, kWe	1.8	1.8
	Volume, m3	1	1
Module Interior			
	Mass, kg	6,948	6,735
	Power, kWe	6.7	6.7
	Volume, m3	77	75
Primary Structure			
	Mass, kg	7,333	6,262
Secondary Structure			
	Mass, kg	5,105	6,888
Radiation Shelter			
	Mass, kg	- - -	4,867
	Volume, m3	- - -	14
Logistics Module			
	Mass, kg	4,921	2,241
	Power, kWe	1.5	1.5
	Volume, m3	73	37
Airlock			
	Mass, kg	4,094	4,094
	Power, kWe	0.6	0.6
	Volume, m3	25	25
Power System			
	Mass, kg	1,120	1,120
	Area, m2	587	587
Energy Storage System			
	Mass, kg	2,494	2,494
	Volume, m3	2.5	2.5
External Thermal			
Control System	Mass, kg	649	649
	Area, m2	32	32
Attitude Control System			
	Mass, kg	3,138	3,138
Artificial G Equipment			
	Mass, kg	1,621	1,621
Communications			
	Mass, kg	334	334
	Power, kWe	0.2	0.2
Support Structure			
	Mass, kg	5,029	4,153
Solar Telescope			
	Mass, kg	1,000	1,000

1.0 Introduction

The Interplanetary Mission Module Study performed by Eagle Engineering provides Martin Marietta with preliminary conceptual designs of pressurized volumes for astronaut missions to Mars. The specified mission profile is characterized as a three year mission (conjunction class) for a crew of five persons. This study encompasses two module design approaches: 1) two cylindrical modules (sized in accordance with the Space Station Freedom habitation and laboratory modules), and 2) a single disk module with a diameter of 8.4 meters.

It is understood that the mission modules are part of a Mars Piloted Vehicle (MPV) consisting of two segments - the crew-occupied modules with supporting systems and the propulsion system mated to an aerobrake. The MPV segments are attached to each other by two structured tethers, as part of an artificial gravity system. The vehicle is intended to spin at a rate of 2 rpm to produce a gravity environment ranging from 3/8 g to 1 g. Periods of vehicle despin, particularly while in Mars orbit, necessitate module adaptability to 0-g conditions. Figure 1.0.1 illustrates the Martin Marietta concept for the vehicle segment attached to the mission modules. A Mars Crew Sortie Vehicle (MCSV) or lander is included at the center of the aerobrake, between the two extended tethers.

1.1 Definition of Mission Module Configurations

The cylindrical modules configuration shows two modules connected by an airlock and a logistics module, which also acts as a radiation storm shelter. The pressurized volumes are mounted to a support of truss structure. The retractable photovoltaic arrays extend from the platform on deployable booms. Radiators also extend from the platform perpendicular to the arrays. The two-module approach allows for the Earth Crew Capture Vehicle (ECCV) to be docked at the base of the logistics module throughout the duration of the mission. The MCSV is intended to dock at the top of the logistics module while the MPV segments are mated. Figure 1.1.1 shows an isometric view of the cylindrical modules configuration with supporting systems labeled. Figure 1.1.2 shows an elevation view of this configuration with docking capabilities highlighted.

The disk module configuration involves the one large module supported by truss structure. Two floors of the module extend above the platform, the other floor below the platform. External structure, such as the ECCV and logistics module/airlock, are mounted to a collar extending around the module. This configuration locates the radiation storm shelter inside the module rather than integrated with the logistics module. The retractable photovoltaic arrays extend from the platform on booms, while the radiators are positioned below the platform perpendicular to the arrays. The MCSV docks directly to the disk module at the top end of the module while the MPV segments are mated. Figure 1.1.3 illustrates an isometric view of the disk module configuration, and Figure 1.1.4 shows an elevation view.

1.2 Study Assumptions

1.2.1 Safe Haven Capability

The safe haven requirement for the Mars interplanetary crew is an important consideration for such a long duration mission. Currently for S.S. Freedom, safe haven has been established as a concept rather than a specific location. It is a distributed capability throughout the modules to sustain the crews following various emergencies, including the complete loss of any one module. Safe haven kits are designed with life-essential functions for up to 90 days. These kits include survival food, food heating means, bedding, survival clothing, trash stowage, medical kit, maintenance tools, and personal hygiene expendables.

For the conjunction-class mission to Mars, safe haven is defined as a location from which the crew can successfully execute and survive the mission. The alternative location must contain independent life support system equipment, galley provisions, and a command and control center. The cylindrical modules configuration satisfies the conservative safe-haven requirement with the capability to seal-off one module from the other, with two escape paths available - through the logistics module and the airlock. Although less accommodating, the crew of five could subsist, and even execute the mission, in one module with all the critical systems duplicated.

The disk module configuration also adheres to the Mars mission safe-haven requirement by dividing the module into two separate pressurized volumes. One volume includes level one and the bottom section of the hull; the other volume includes level three and the loft portion of the hull. Each pressurized volume contains the critical mission systems as well as access to the radiation storm shelter located in the middle level, which acts as a buffer between levels one and three. Level two can be included with either level one or three as part of the pressurized safe haven, or, if necessary, it can be depressurized, with the shelter acting as a crude airlock for access in suits to storage supplies.

1.2.2. Power Duty Cycles

Duty cycles were determined in order to translate the peak equipment power requirements into average power usage. The cycle time period was estimated at one week, with the five person crew activity restricted to single shifts. A duty cycle of 1 (power used 100% of the time) was assigned to life critical systems, such as air revitalization systems, temperature/humidity control systems, temperature/pressure sensors, fans, radiation monitoring equipment, and certain electrical systems. Duty cycles less than 1 imply periodic usage over a week's interval. For example, 5 crew members take 4 showers per week at 15 minutes per shower, resulting in a duty cycle of 0.03. Systems requiring power but having a duty cycle of 0 are not used during an average week. Compensation for such systems, when activated, are made or battery power is used. The majority of the health maintenance equipment falls in this category. Finally, the average power required is determined by multiplying the unit power by the duty cycle by the number of units "on."

1.3 Report Organization

The interplanetary mission modules study concentrates on producing two alternative designs for the habitation structure of the MPV. In addition to conceptualizing the cylindrical modules and the disk module configurations, the study focuses on two areas of analysis. Section 3 of this report details the pressurized mission modules analysis which includes the crew-related systems and the module structure. Definition of pressurized mission modules include life support system, mission consumables, internal thermal control system, module interior, and structures. Section 4 discusses the additional structure and systems associated with this segment of the MPV, including the radiation storm shelter, power and external thermal control systems, airlock, and logistics module. A summary comparison between the two configurations is presented in Section 5. References appear at the end of each topical subsection rather than at the end of the report. Appendix A contains a discussion of the 1/20 scale model constructed by Eagle to demonstrate the interior of a cylindrical mission module and the 1/72 scale model of an entire Mars vehicle as defined by Martin Marietta and Eagle Engineering. Appendix B includes the spreadsheet documentation of the volumetric analysis for each module configuration.

Some of the tables included in this report use a particular references format. The format of #/#/# means the first number is the reference for the mass value, the second for volume, and the third for power requirement. One number implies all three parameters were from that source. A dash (-) implies no value, no source. An "E" is an abbreviation for estimate, while "C" stands for calculation.

Certain additions to the module interior configurations were not defined in terms of mass, including the aerobrake-maneuver couches and the windows in both module concepts.

Figure 1.0.1 Martin Marietta Design of MPV Propulsion and Aerobrake System

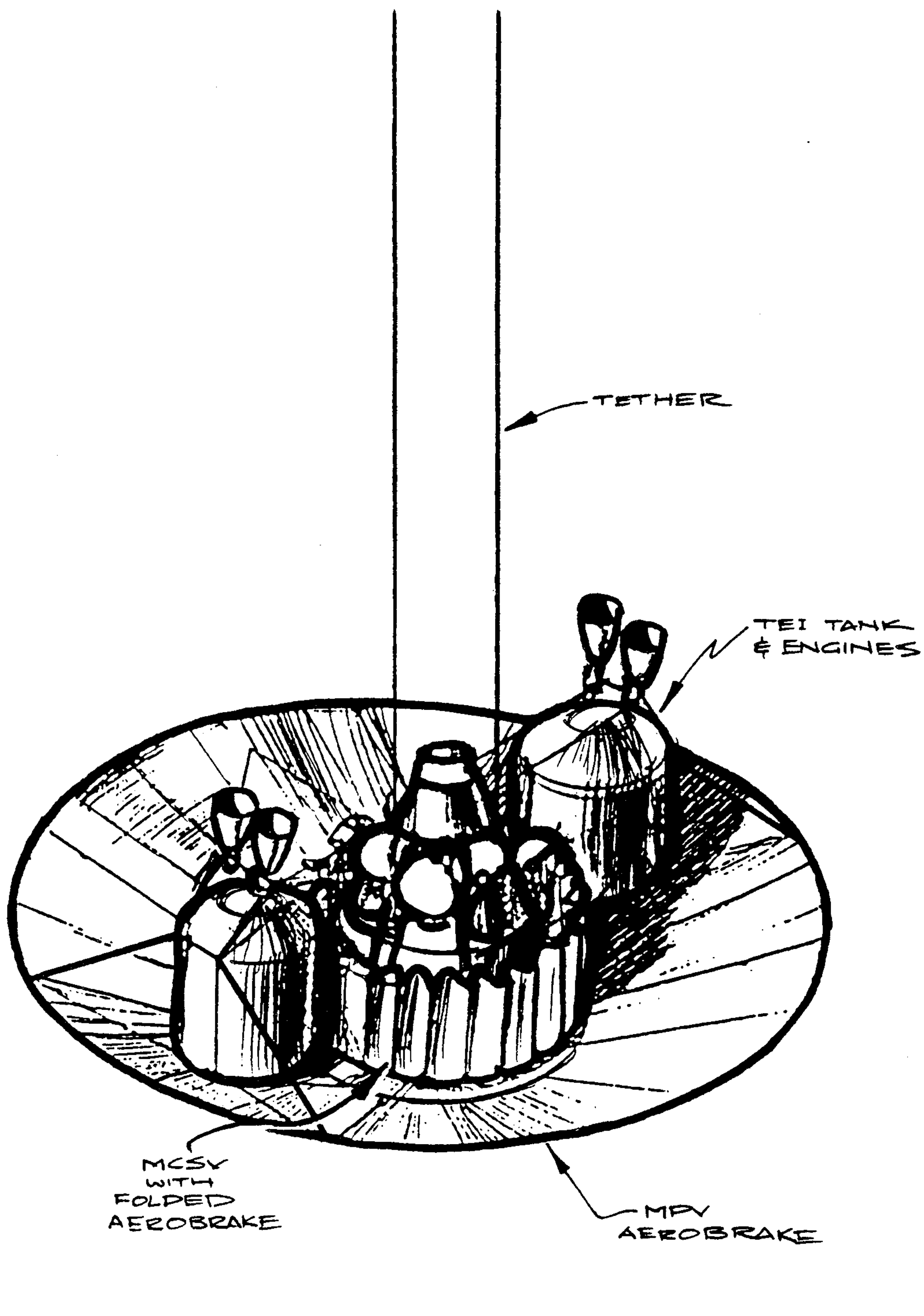
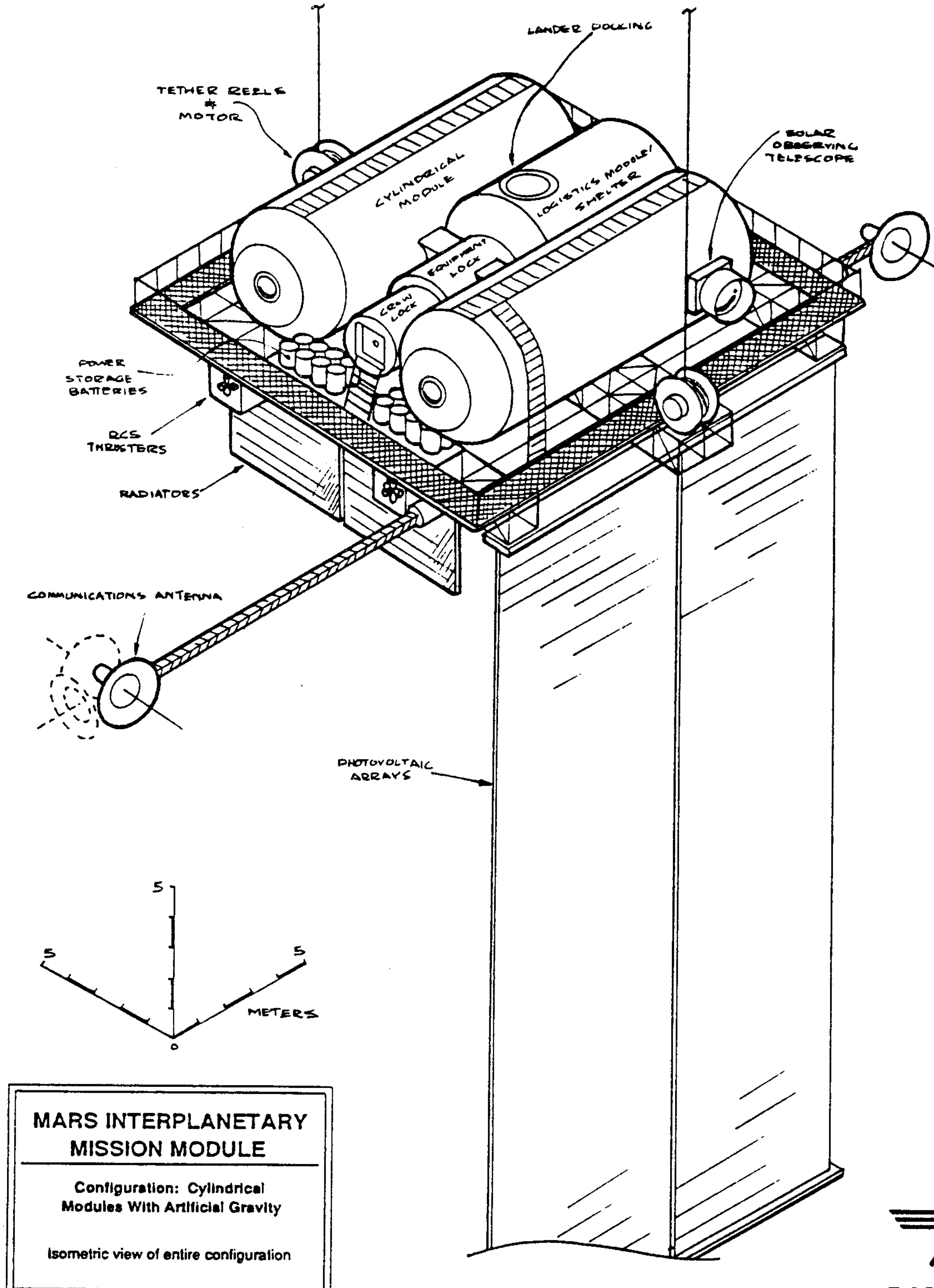


Figure 1.1.1 Cylindrical Modules Configuration, Isometric View



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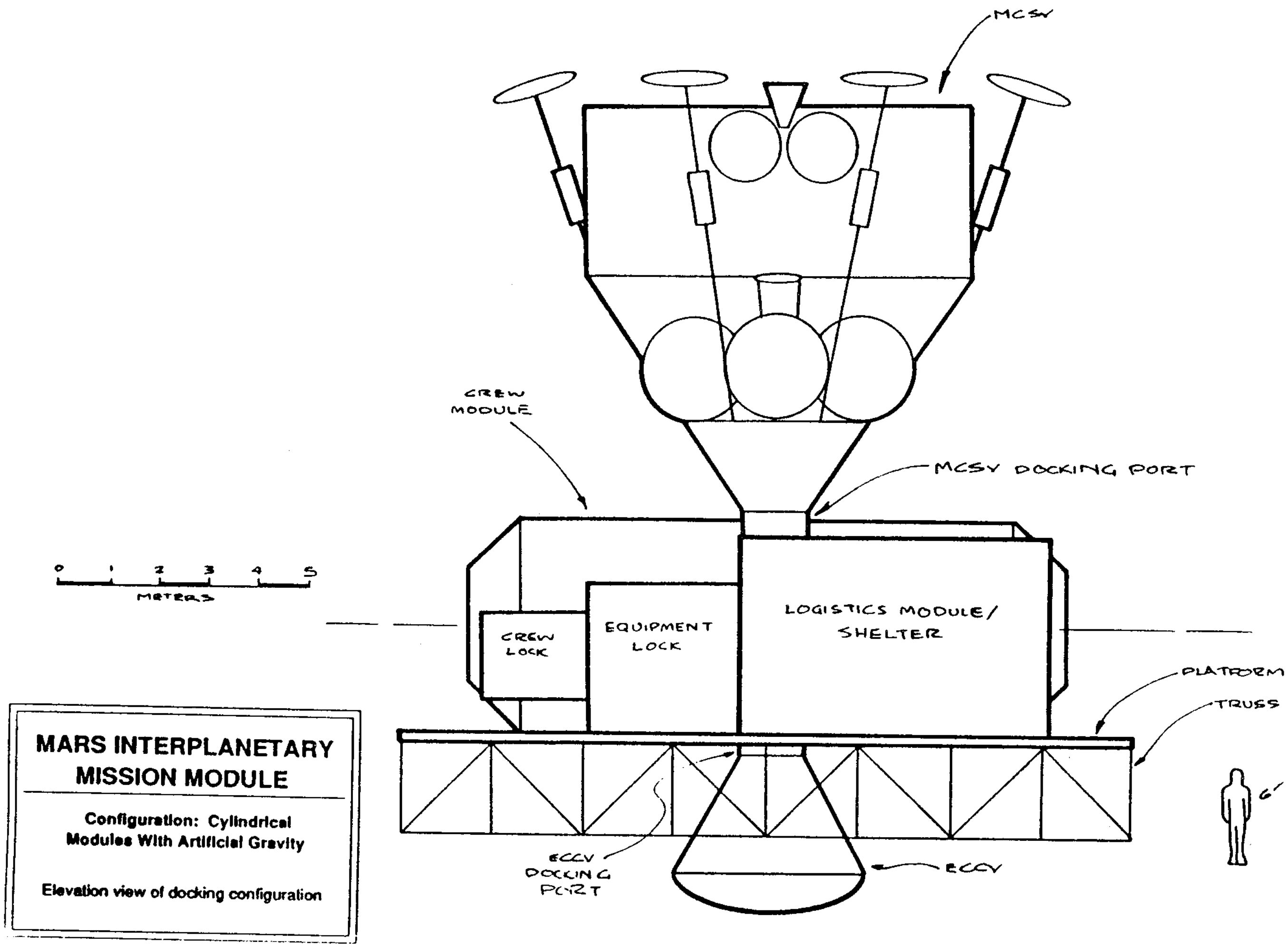

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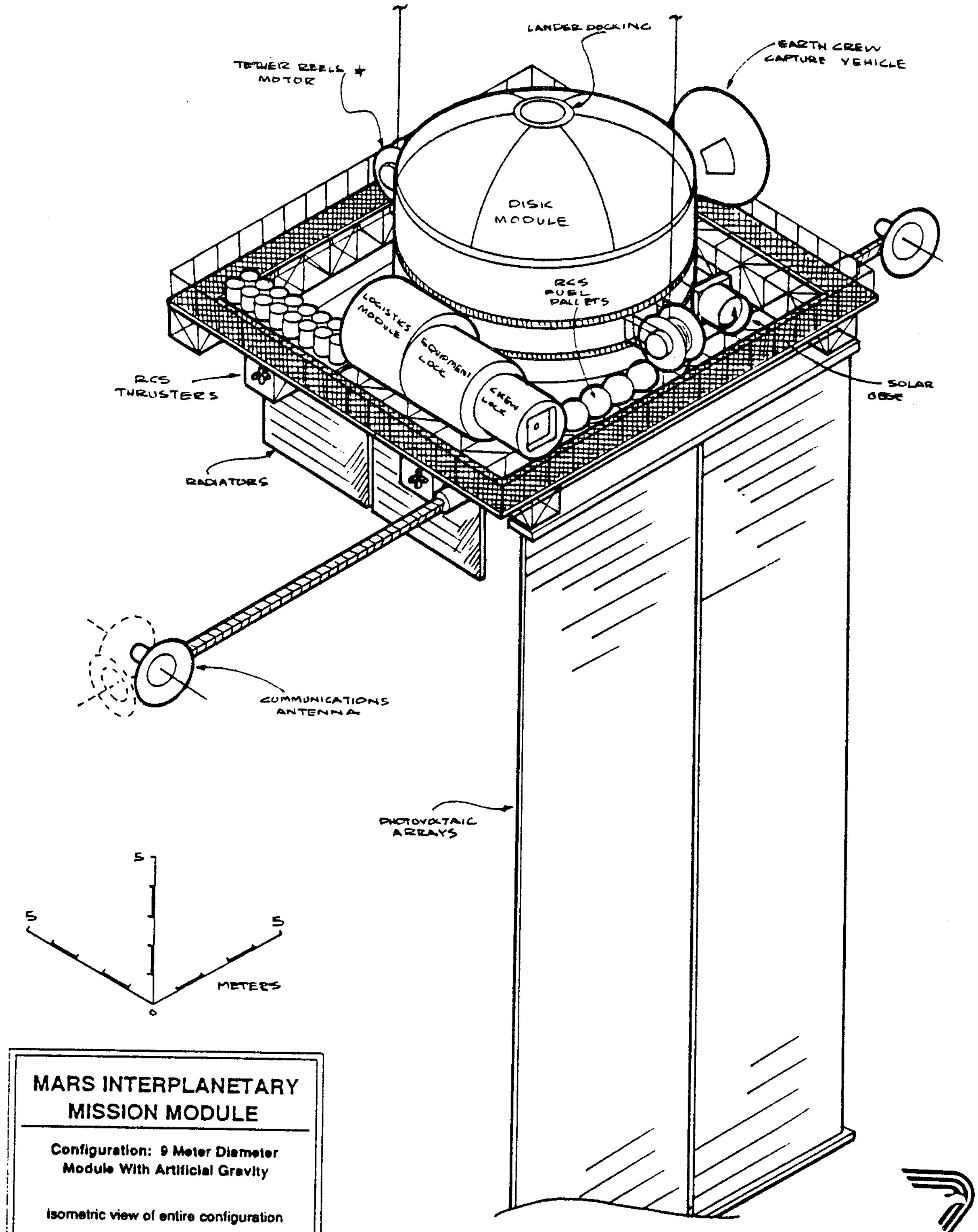
Figure 1.1.2 Cylindrical Modules Configuration, Orthographic Side View

Figure 1.1.3 Disk Module Configuration, Isometric View



MARS INTERPLANETARY MISSION MODULE

Configuration: Disk Module With Artificial Gravity

Elevation view of docking configuration

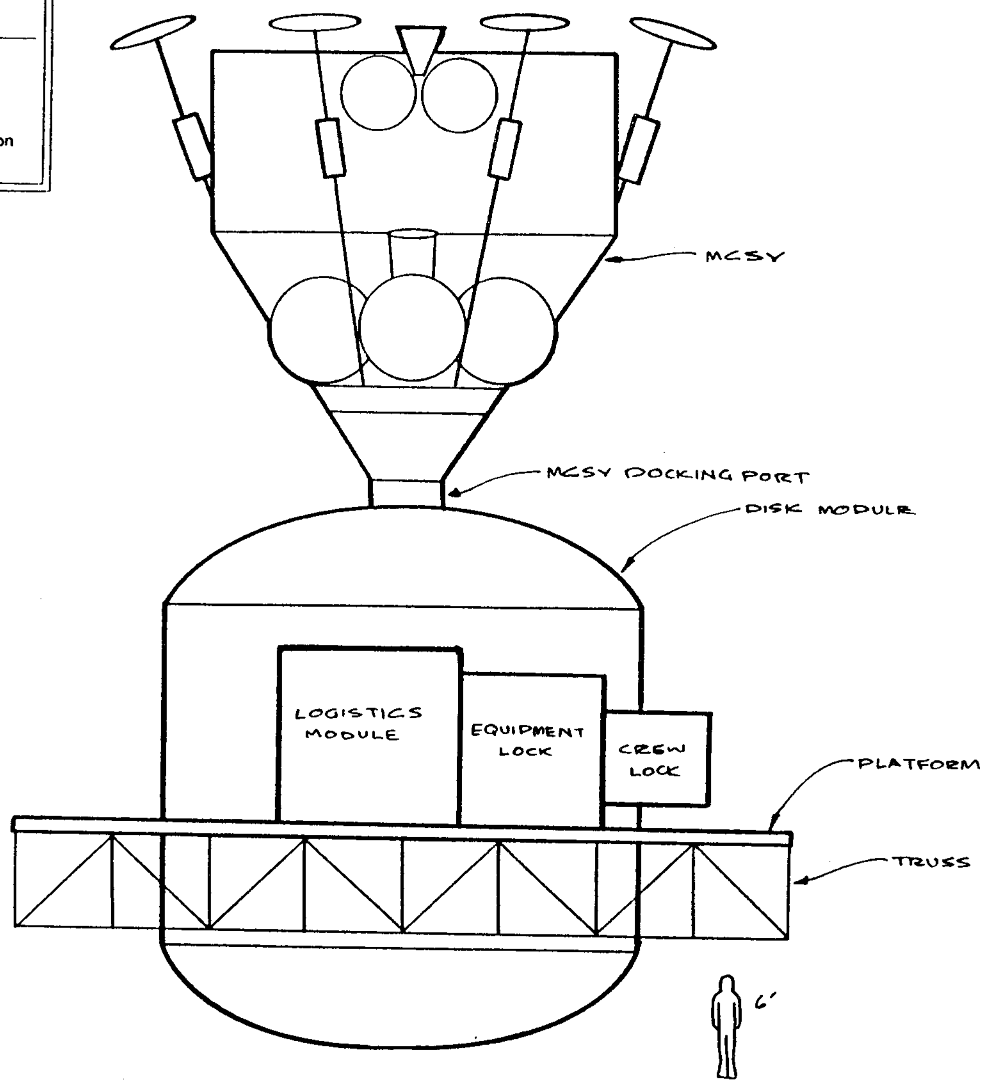


Figure 1.1.4 Disk Module Configuration, Orthographic Side View

2.0 Conceptual Designs of Mars Interplanetary Mission Modules

2.1 Interior of Cylindrical Modules Configuration

The following provides an itemized characterization of the interior of a cylindrical module according to each functional crew area. Figure 2.1.1 and Figure 2.1.2 illustrate the interior planviews of the modules. The cylindrical module interiors appear similar due to the safe haven requirement. Slight alterations between the modules, as depicted in the illustrations, include number of quarters, amount of lab space, and the inclusion/absence of a fitness area. Figure 2.1.3 shows an exemplary cross-sectional view of a cylindrical module with placement of floor, ceiling, racks and compartments highlighted.

General

- 2.4 m (8 ft.) ceiling height.
- Standoffs for cabling and plumbing exist in four corners of module.
- Radiation storm shelter/logistics module and airlock accessible from both modules.
- Equipment racks slide on tracks away from module hull for back access.
- Under-floor compartments used for life support system equipment and spares.
- Ceiling compartment contains full-body, form-fitting couches for crew restraint during aerobraking maneuvers.
- A 1 meter easement exists down the module center for crew passage and access.
- Racks include hand rails and chairs include foot restraints for crew mobility during 0 g periods.

Quarters

- Certain beds fold-away to provide additional space.
- Storage capability under bed.
- Soft storage and entertainment displays on walls.
- Accordion doors (sound-proof) for open view to module center.
- Terminal/desk space accessible with adjustable stool.

Galley/Wardroom

- Table has drop-leaf feature to add galley floor space.
- Galley equipment racks have deployable work surfaces and handrails.
- Galley consists of 4 racks for equipment and weekly food supply.
- Projection screen retracts from ceiling.
- Ceiling lowers to 2.1 m (7 ft.) to allow for loft configuration in one module.
- Second module has vaulted ceiling over galley area.
- Both modules have observation windows in the module endcap.

Personal Hygiene

- Crew has separate access to commode, handwash and shower.
- Commode is positioned for 0 g use too.
- Handwash area includes sink, stowage space for hygiene expendables, and dressing space.
- Shower is accessible through the handwash area or from the main corridor.

Command and Control Center

- Two seated stations provide capabilities for communications, computational support, flight simulation, and vehicle monitoring.
- Details include captain's chairs, hand-controllers, and deployable keyboards and workspaces.

Lab

- Lab equipment occupies two racks, with deployable keyboard, graphics pad, and workspace.
- Crew member can stand or sit on stool to monitor experiments.

Maintenance Area

- Maintenance area is only located in one module.
- Workbench provides space for hardware maintenance and repair plus stowage for tools and small spares.
- Window above workbench allows for an additional observation point, possibly for photography purposes.

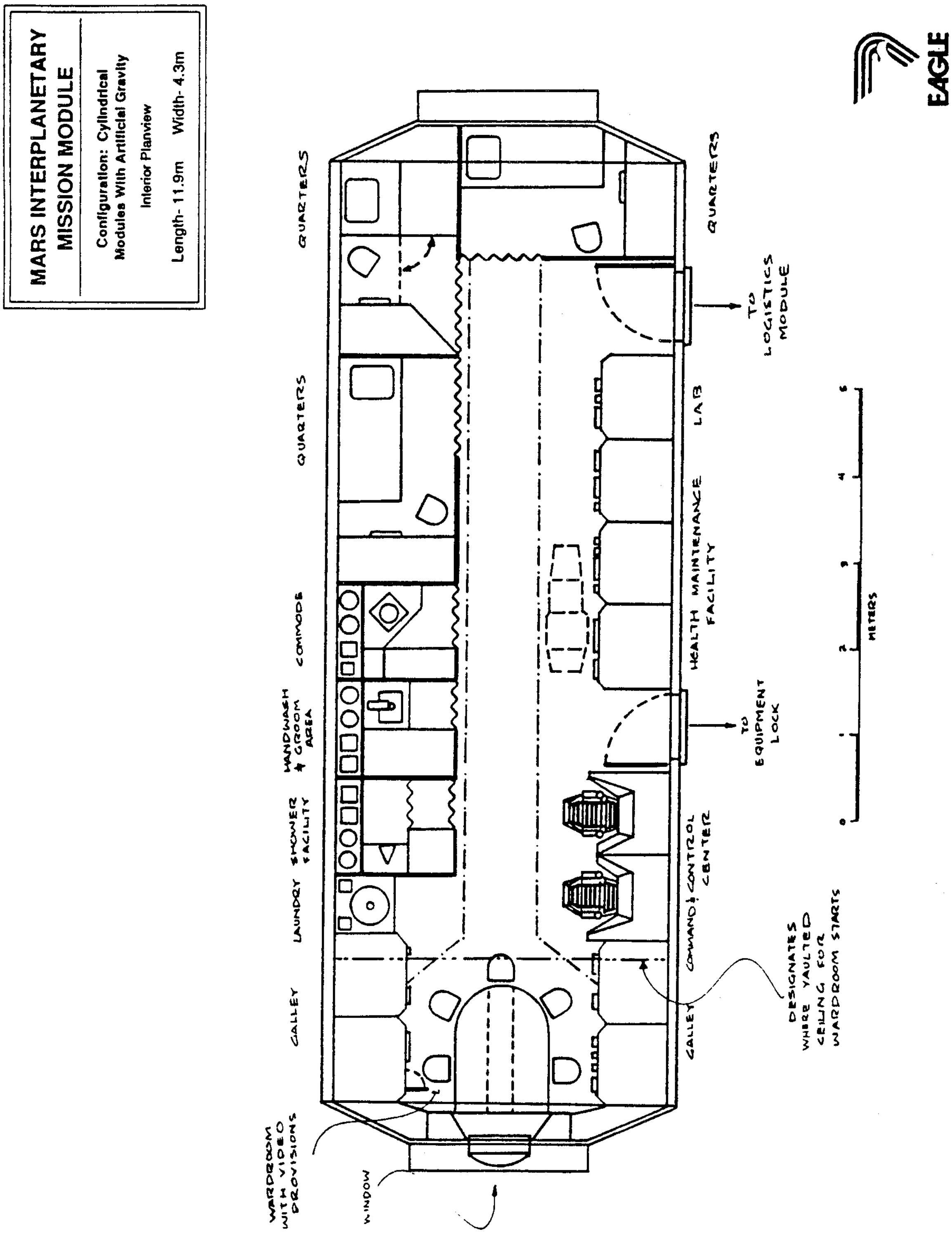
Health Maintenance Facility

- Medical treatment and surgical equipment located on three racks.
- Fitness area in one module is adjacent to medical monitoring equipment, with video provisions. Exercise machines are assumed stowed and easily accessible with aisle space used for fitness activities.
- Patient restraint collapses, stows, and deploys in aisleway adjacent to medical equipment.

Loft

- In one module the loft extends over the galley and provides the long-duration mission crew with a relaxation and solitude area. It also acts as a backup sleeping compartment if the second module is closed off.
- The loft is accessible via a ladder that stows above the wardroom along the ceiling which is lowered to 2.1 m (7 ft).
- The loft includes a large sitting cushion and cabinet stowage.

Figure 2.1.1 Cylindrical Module, Interior Planview



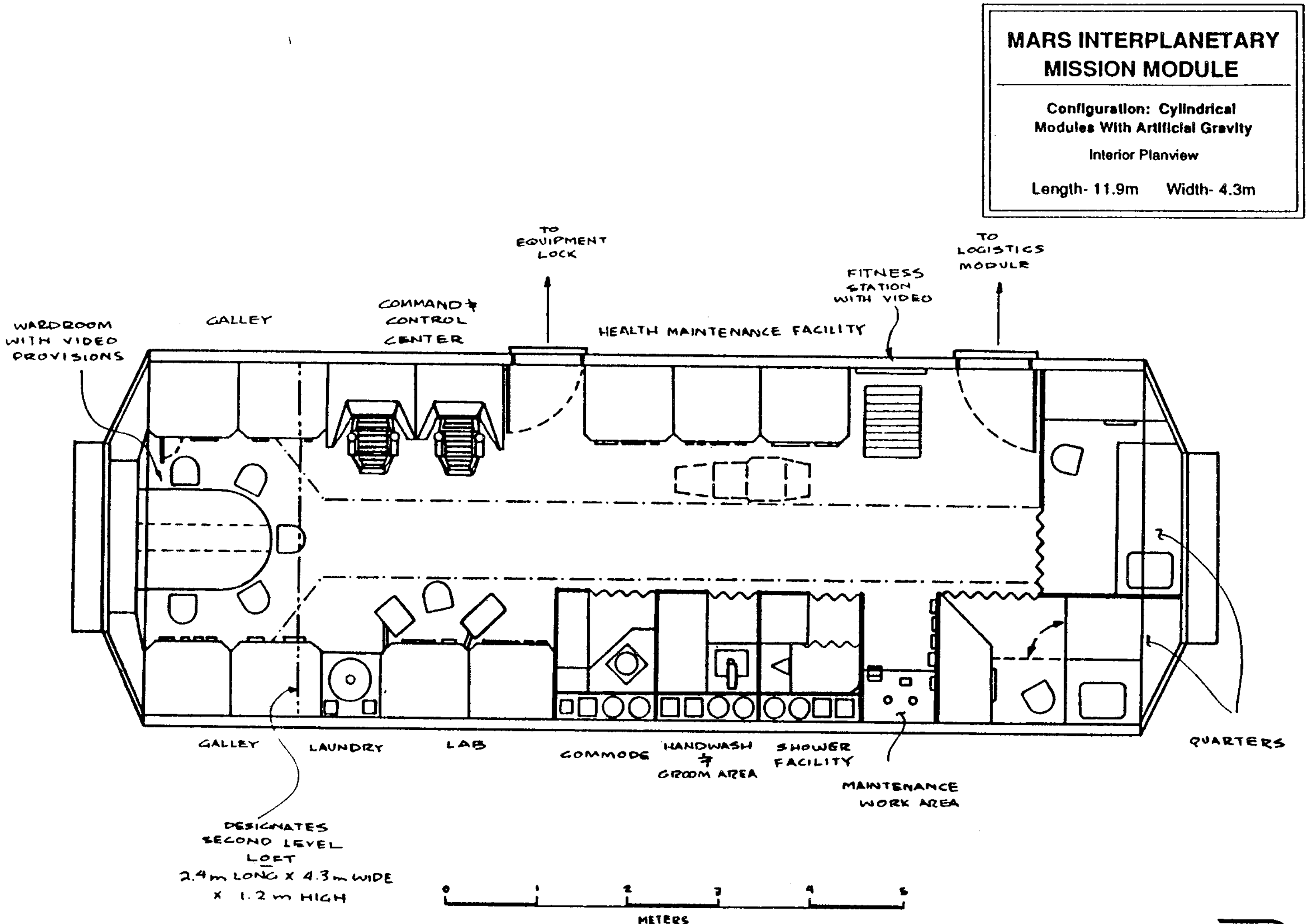


Figure 2.1.2 Cylindrical Module, Interior Planview

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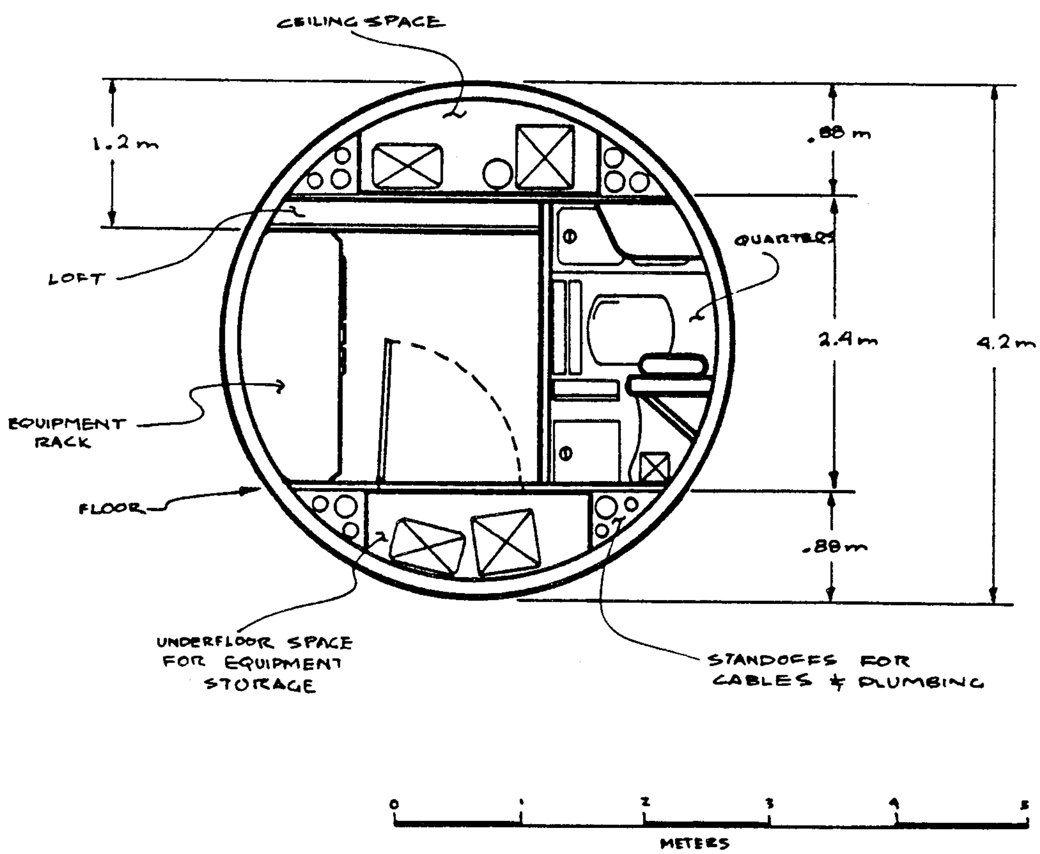
Figure 2.1.3 Cylindrical Module, Cross Section



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2.2 Interior of Disk Module Configuration

The following provides an itemized characterization of the interior of the disk module according to each functional crew area. Figure 2.2.1, Figure 2.2.2, and Figure 2.2.3 illustrate the interior planviews of each module level. Levels 1 and 3 appear similar due to the safe haven requirement. Slight alterations between the levels, as depicted in the illustrations, include number of quarters and the degree of health maintenance capability. Figure 2.2.4 shows a cross-sectional view of the disk module with placement of floors, storm shelter and compartments highlighted.

Levels 1 & 3

General

- 2.4 m (8 ft) ceiling height.
- Stand-off exists around circumference of module (0.3 m or 1 ft.) for cabling and plumbing.
- Racks and compartments mounted on tracks (to slide along wall or pull out from wall) for access to back of equipment or stand-off space.
- Ladder adjacent to center-floor hatch opening for crew access between levels.
- Center-floor hatch opening secured with safety net.

Health Maintenance Facility

- Level 1 medical treatment area provides for minimal needs, with physician's instruments and pharmaceuticals stowage.
- Level 3 has private compartment for surgical procedures and health monitoring, including patient restraint, medical laboratory, and surgical instruments.
- Adjacent medical treatment area provides for minimal needs, with physician's instruments and pharmaceuticals storage.

Personal Hygiene

- Separate access to commode, handwash and shower.
- Storage space for hygiene expendables.
- Space for grooming and dressing available.

Quarters

- Level 1 contains quarters for 3 crew members; level 3 for 2 crew members.
- Storage capability under bed.
- Soft storage and displays on walls.
- Accordion doors (sound-proof) for open view to module center.
- Terminal/desk space accessible with or without chair.

Command and Control Center

- Two seated stations provide capabilities for communications, computational support, flight simulation, and vehicle monitoring.
- Details include captain's chairs, hand-controllers, and deployable keyboards and workspaces.

Galley/Wardroom

- Galley equipment racks have deployable work surfaces and handrails.
- Table can be collapsed to provide large open area.
- Projection screen is retractable.
- Three windows (behind projection screen) allow for observation from wardroom.

Below Level 1

- Bottom level provides space for life support system equipment (to support level one pressurized volume), soft stowage, and secondary docking capability for the MCSV.

Above Level 3 - Loft

- Loft extends over crew quarters, command and control center, HMF, and part of galley and hygiene area.
- Loft provides space for life support system equipment (to support level three pressurized volume), soft storage, flight couches (mounted to top of dome) for crew positioning during Mars aerobraking maneuvers, and relaxation area.
- Windows (4) located in dome of loft provide for prox-ops viewing.
- Hatch at top of hull allows access to docked MCSV and provides EVA access to module exterior in event of emergency level 3 depressurization.

Level 2

General

- 2.4 m (8 ft) ceiling height.
- Stand-off exists around circumference of module (0.3 m or 1 ft.) for cabling and plumbing.
- Earth Crew Capture Vehicle (ECCV) docking hatch is accessible on this level.
- Logistics module/airlock hatch is accessible on this level.
- Ladder adjacent to center-floor hatch opening for crew access between levels.
- Center-floor hatch opening secured with safety net.

Stowage Racks

- Stowage racks slide on tracks similar to library shelf systems.
- Racks are configured for narrow-shelf and deep-shelf storage.
- Racks conveniently located next to the logistics module entrance.

Radiation Storm Shelter

- Radiation storm shelter located in the center of this level for maximum shielding.
- Shelter acts as passage between pressurized levels 1 and 3.
- Shelter has one hatch for access to the second level operations.
- Pressure hatches for levels 1 and 3 open into the storm shelter along tracks.
- To avoid massive hatch weight, the hatches are designed as hollow aluminum bladders to be filled with water in the event of a solar flare.
- Shelter contains 5 deployable hammocks, soft storage for consumables supplies, and tankage for water and emergency air supply.

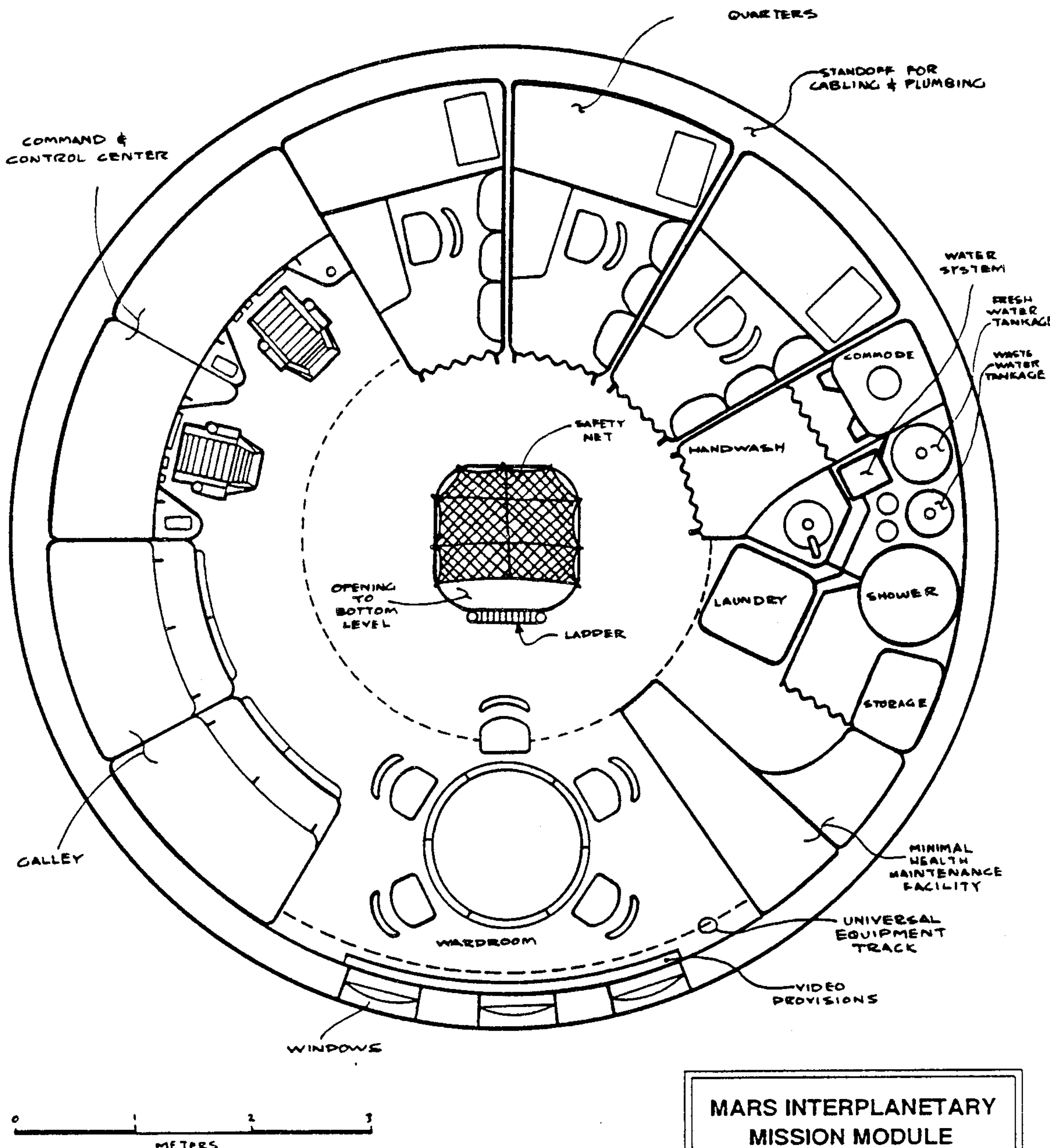
Lab

- Lab equipment occupies two racks with desk space and seating.
- Details include deployable keyboard, graphics pad, and workspace.
- Adjacent workbench is used for hardware maintenance and repair; table extension, in front of ECCV hatch, provides additional space.

Fitness and Entertainment Station

- Area includes floor-mounted treadmill, bike, and rowing machine.
- Entertainment includes video provisions and 3 observation windows.
- Health monitoring carts are also located at this station.

Figure 2.2.1 Disk Module - Level 1, Interior Planview



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MARS INTERPLANETARY MISSION MODULE

Configuration: Disk Module With Artificial Gravity

First Level, Interior Planview

Figure 2.2.2 Disk Module - Level 2, Interior Planview

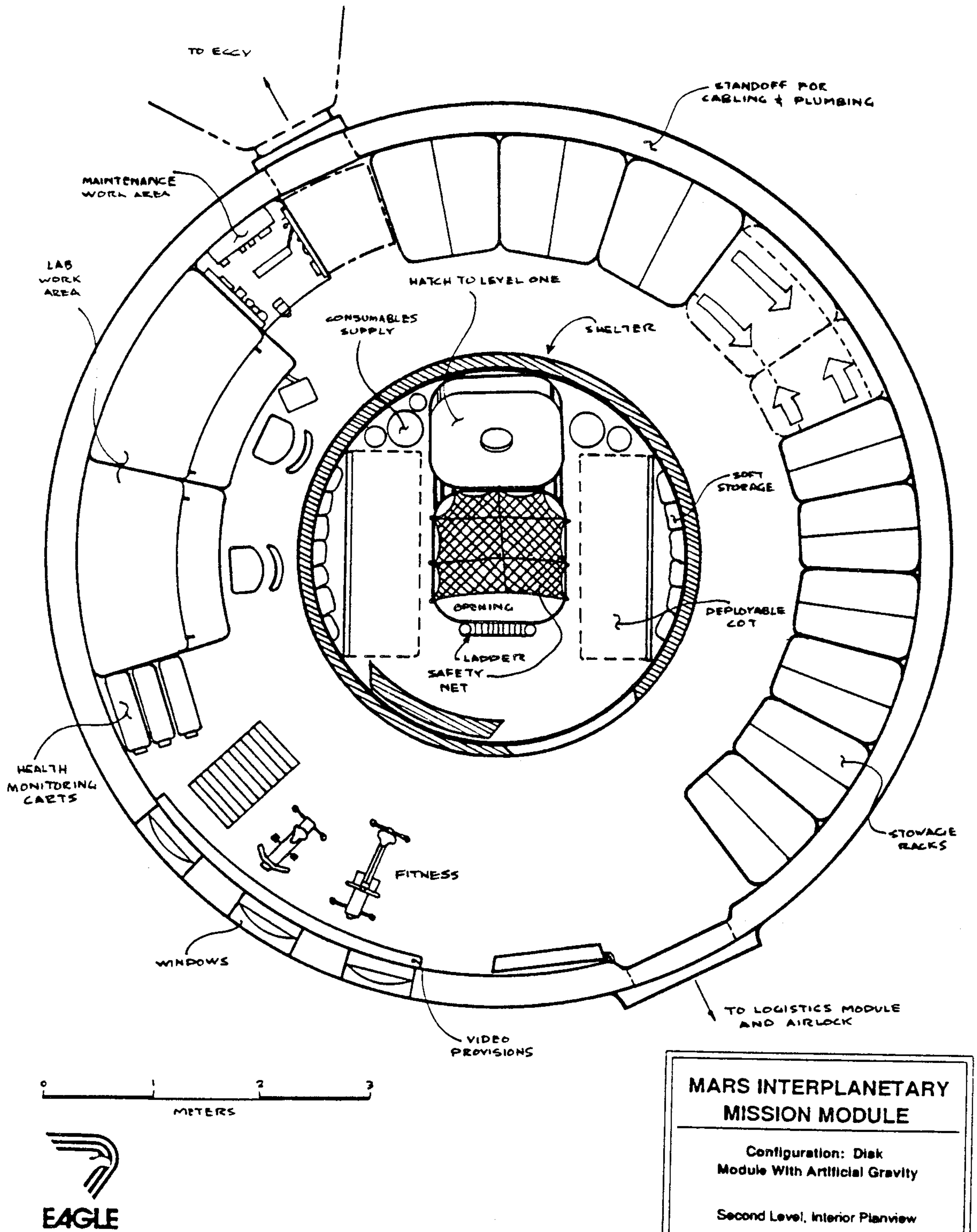
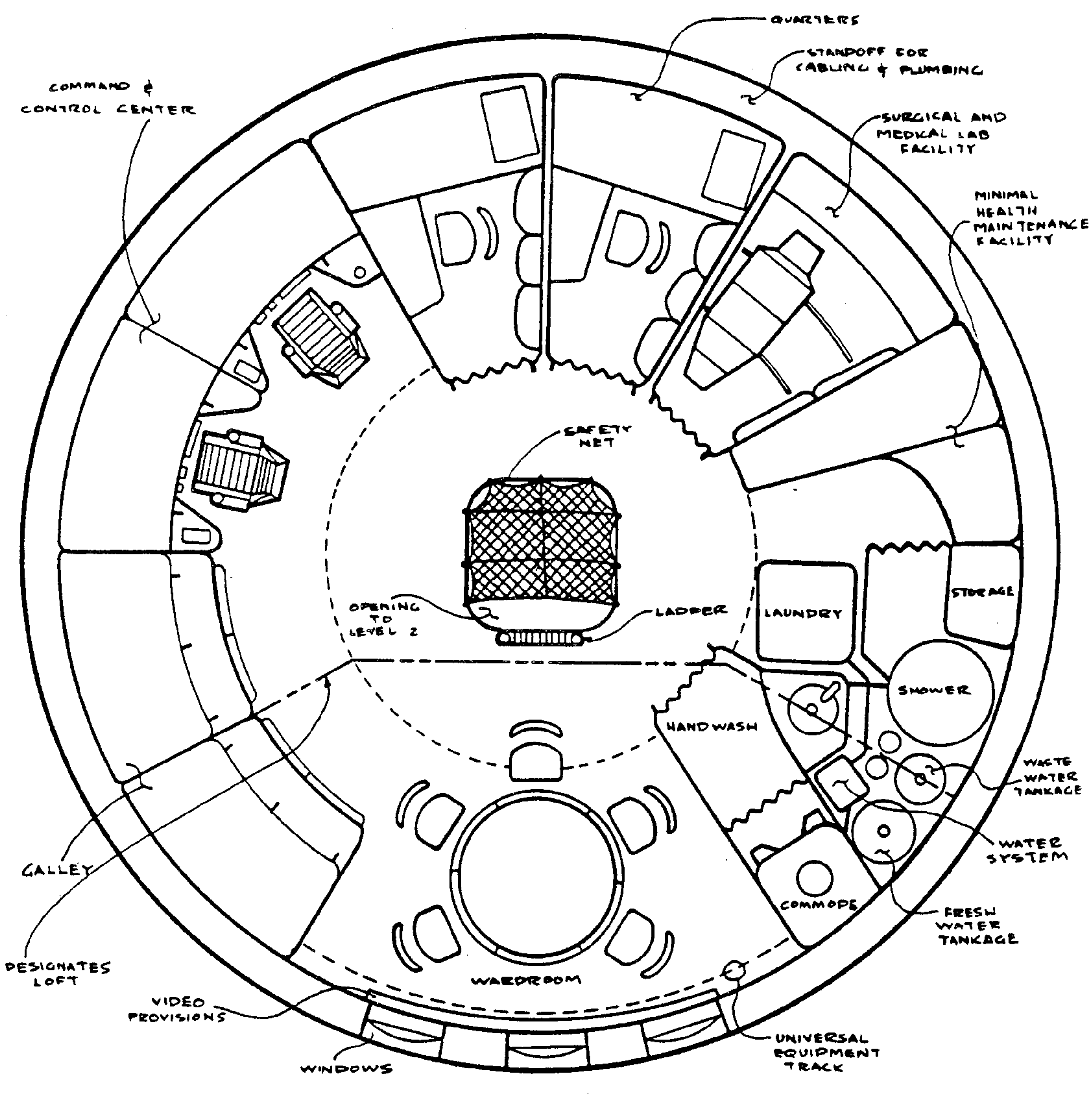


Figure 2.2.3 Disk Module - Level 3, Interior Planview



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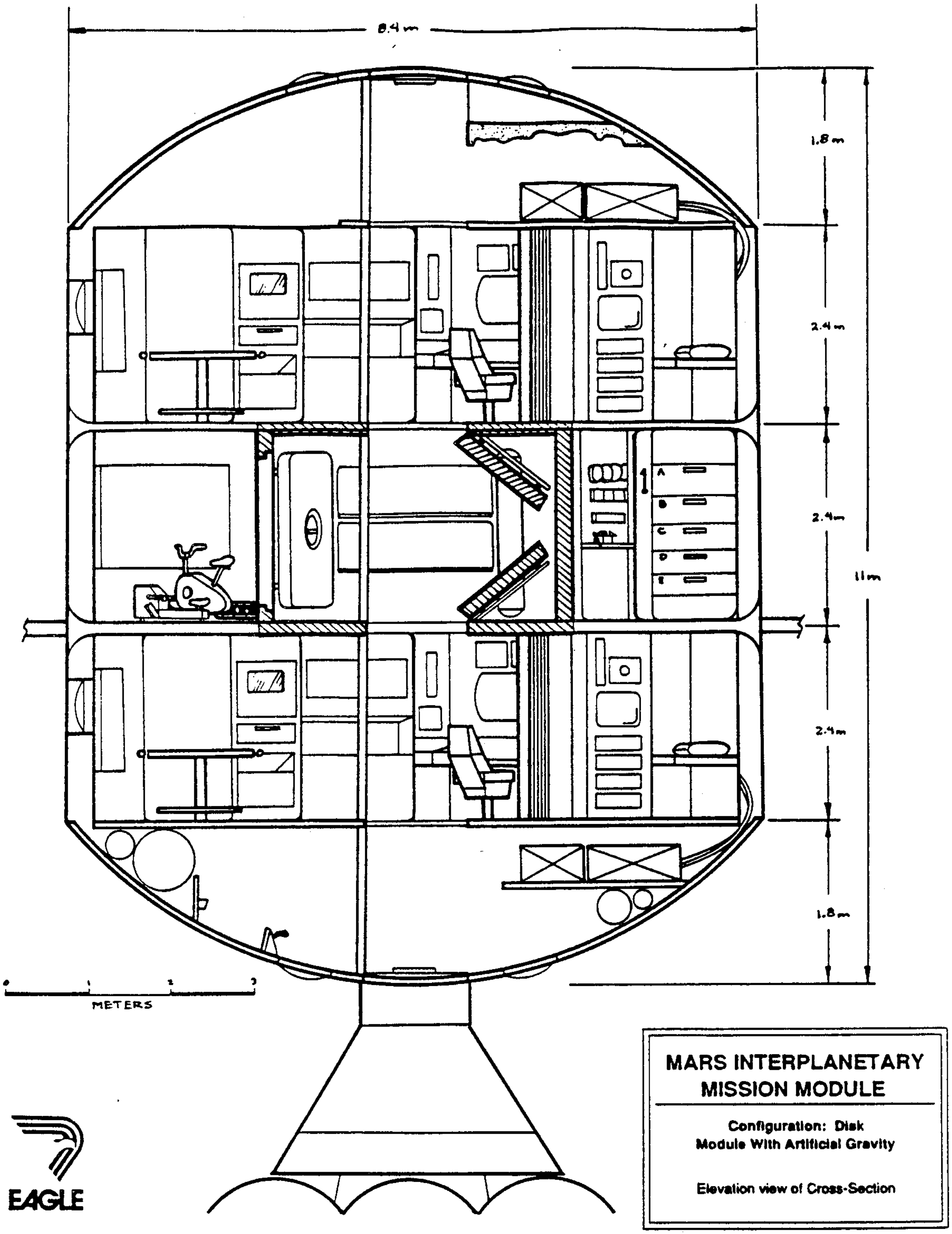
MARS INTERPLANETARY MISSION MODULE

Configuration: Disk Module With Artificial Gravity

Third Level, Interior Planview

Figure 2.2.4 Disk Module, Cross Section



3.0 Pressurized Mission Modules

3.1 Life Support System

The life support system is a physical/chemical system in which the air and water needs of the crew are revitalized at a 98% closure level. Loss of oxygen and nitrogen through cabin leakage is replenished through cryogenic liquified storage of make-up gas. The air revitalization system is separated into three main functions - CO_2 concentration and removal, CO_2 reduction, and oxygen regeneration. Trace contaminants in the cabin air are controlled by using a catalytic oxidizer and filters. The water recovery and management system involves the recovery and processing of waste, potable, and hygiene water. The life support system also includes means for controlling the module temperature and humidity levels. Trash management is accomplished through compaction and storage; solid waste management by a vacuum-drying process. The life support hardware is sized to accommodate up to seven persons (ref. 1).

The selection of the electrochemical depolarized cell (EDC) and the Sabatier reactor for air management functions is derived from recommendations in reference 1. The static feed electrolysis (SFE) system is chosen for its S.S. Freedom recommendation, minimal power consumption, small size, and use of non-pure water. The vapor compression distillation (VCD) process provides the best water quality, the lowest specific energy, and the highest percentage of water recovery (in comparison to TIMES and AES). Multifiltration is selected as a proven and effective process to be used on S.S. Freedom.

The redundancy of these five critical systems for both the cylindrical modules and disk module configurations is defined as two units in each pressurized volume (one operating and one back-up). Spares for these systems are estimated at 10% of the total equipment mass. Redundancy of other life support components in the pressurized volumes is based on the safe haven requirement, with each system duplicated. Values in the quantity column greater than 2 are determined by the recommended operating/spare units for S.S. Freedom (ref. 2).

The following pages provide an itemized account of the life support system in terms of mass, volume, and power requirements. Note that the water storage tanks included in the listing are designated for immediate consumables use (sized for one week's supply). The tanks required for complete mission storage are sized in the mission consumables section.

A summary of the life support system characteristics for the two module configurations appears below.

Summary

	Mass (kg)	Volume (m^3)	Power (kw)
Air Revitalization	1149.4	2.35	2.59
Water Recovery & Management	1587.2	5.21	0.11
Temperature & Humidity Control	1807.2	5.92	0.74
Additional Life Support Equipment	1004.4	4.89	0.63
Life Support Systems Total	5548.3	18.4	4.1

References

1. Hall, J., Ferebee, M., and K. Sage, "Environmental Control and Life Support Systems Technology Options for Space Station Application," SAE Technical Paper #851376, July, 1985.

2. NASA, Space Station Program System Engineering and Integration, "Weights Data Book - Revision 2.0," June, 1988.

3. Powell, F.T., and R.A. Wynveen, "Comprehensive Study of Environmental Control and Life Support Technology for Advanced Manned Space Missions," NASA Contract #NAS9-17531, July, 1987.

4. Gorensek, M.B., and D. Baer-Peckham, "Space Station Water Recovery Trade Study - Phase Change Technology," SAE Technical Paper #881015, July, 1988.

5. Fortunato, F.A., Kovach, A.J., and L.E. Wolfe, "Static Feed Water Electrolysis System for Space Station Oxygen and Hydrogen Generation," SAE Technical Paper #880994, July 1988.

6. NASA, Space Station Program System Engineering and Integration, "Power Data Book," August, 1988.

7. NASA, "Central Thermal Control System Properties for the Space Station," Thermal Systems Division, JSC, January, 1987.

Table 3.1.1 Life Support System Definition

Air Revitalization	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	#SS Unit "on"	Ref.	Mass (kg)	Volume (m^3)	Ave Power (w)
CO2 Concentration & Removal (EDC)	4	4	74.84	0.250	878	1	1	1/1/6	299.36	1.00	878
CO2 Reduction (Sabatier Reactor)	4	4	43.09	0.200	113	1	1	1/1/6	172.36	0.80	113
O2 Regeneration (SFE)	4	4	51.26	0.090	1010	1	1	1/1/6	205.04	0.36	1010
Trace Contaminant Monitor	2	2	125.24	0.096	295	1	2	2/3/6	250.48	0.19	590
Trace Contaminant Control Assembly	2	2	47.70	0	0	0	2	2/-/6	95.40	0	0
Regulators, Valves, Line	2	2	11.15	0	0	0	2	2/-/6	22.30	0	0
Spares @ 10% of total mass			35.33						104.49		
Subtotal									1149.4	2.4	2591

Water Recovery & Management	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref.	Mass (kg)	Volume (m^3)	Ave Power (w)
Urine Pretreat Unit	4	4	12.20	0.050	35	0.03	2	2/E/6	48.80	0.20	2.10
Urine Storage Tank (1 week supply)	2	2	12.20	0.020	0	0	0	2/C/-	24.40	0.04	0
Urine Recovery Processor (VCD)	4	4	51.94	0.143	156	0.03	2	1/1/6	207.76	0.57	9.36
Urine Water Quality Monitor	2	2	19.50	0.030	2	1.00	2	2/E/6	39.00	0.06	4.00
Disconnects, Valves, Line	2	2	55.79	0	0	0	0	2/-/-	111.58	0	0
Potable Storage Tank (1 week supply)	4	4	18.20	0.090	0	0	0	2/C/-	72.80	0.36	0
Potable Water Processor (Multifiltr.)	2	2	52.39	0.510	37	0.50	2	1/1/6	104.78	1.02	37.00
Potable Water Quality Monitor	2	2	19.50	0.030	2	1.00	2	2/E/6	39.00	0.06	4.00
Hygiene Stor. Tank (hdws-1 wk supply)	2	2	54.50	0.110	0	0	0	2/C/-	109.00	0.22	0
Hygiene Water Processor (Multifiltr.)	2	2	52.39	0.510	68	0.02	2	1/1/6	104.78	1.02	2.72
Hygiene Water Quality Monitor	2	2	19.50	0.030	2	1.00	2	2/E/6	39.00	0.06	4.00
Processor Cable Assemblies	2	2	19.73	0	2	0.01	2	2/-/6	39.46	0	0.04
H2O Sterilization/Distribution System	2	2	251.29	0.800	431	0.05	2	2/E/6	502.58	1.60	43.10
Spares @ 10% of total mass			63.91						144.29		
Subtotal									1587.2	5.2	106.3

Table 3.1.1 Life Support System Definition (cont.)

Temperature & Humidity Control	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Unit "on"	Ref.	Mass (kg)	Volume (m^3)	Power (w)
Air Temp & Humidity Control	2	2	228.16	5.45	862	1.0	2	3/3/6	456.32	10.90	1724
Equipment Air Cooling	2	2	36.38	0.14	1162	1.0	1	3/3/6	72.76	0.28	1162
Condensing Heat Exchanger	4	4	80.78	0.10	0	0	0	2/E/-	323.12	0.40	0
Cabin Air Fan	4	4	7.27	0.06	44	1.0	2	2/7/6	29.08	0.24	88
Water Separator	4	4	4.00	0.08	47	1.0	2	2/E/6	16.00	0.32	94
Ventilation Fan	12	12	1.00	0.06	20	1.0	6	2/7/6	12.00	0.72	120
Air Diffuser	16	16	1.36	0.08	30	1.0	8	2/E/6	21.76	1.28	240
Bacteria Filter	28	28	4.86	0.09	0	0	0	2/E/-	136.08	2.52	0
Avionics Cooling Fan	4	4	12.74	0.06	50	0.5	2	2/7/6	50.96	0.24	50
Refrigerator	2	2	86.20	0.10	150	0.5	2	2/E/6	172.40	0.20	150
Valves, Cables, Brackets	2	2	416.72	0	0	0	0	2/-/-	833.44	0	0
Spares @ 10% of total mass			87.95						212.39		
Subtotal									1807.2	5.9	742

Additional Life Support Equipment	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Unit "on"	Ref.	Mass (kg)	Volume (m^3)	Power (w)
Fire Detection & Suppression	2	2	97.07	0.200	245	1	2	2/3/6	194.14	0.400	490
Emergency Breathing Packs	7	7	7.98	0.113	0	0	0	2/3/-	55.86	0.791	0
O2/N2 Pressure Control	2	2	79.24	0.080	45	1	2	2/3/6	158.48	0.160	90
Waste Collection & Processing	2	2	105.24	0.930	0	0	0	3/3/-	210.48	1.860	0
O2/N2 Distribution	2	2	139.25	0.812	37	1	1	3/3/6	278.50	1.624	37
Vent & Relief	2	2	7.80	0.025	4	1	2	3/3/6	15.60	0.050	8
Spares @10% of total mass			43.66						91.30		
Subtotal									1004.4	4.9	625

Summary	Mass (kg)	Volume (m^3)	Power (kw)
Air Revitalization	1149.4	2.35	2.59
Water Recovery & Management	1587.2	5.21	0.11
Temperature & Humidity Control	1807.2	5.92	0.74
Additional Life Support Equipment	1004.4	4.89	0.63
Life Support Systems Total	5548.3	18.4	4.1

3.2 Mission Consumables

Mission consumables, although considered part of the life support system, are highlighted separately due to their significance in a long-duration mission. The consumables used throughout the interplanetary mission to Mars include water, oxygen, nitrogen, and dry goods such as food and clothing.

The water loop is assumed to be 98% closed over the entire mission requirements. In addition to the 2% supply for system makeup, an initial supply of water is necessary to initiate the loop, equivalent to one week's requirements. A contingency supply equal to 10% of mission drinking water and food prep water needs is estimated in case of contamination or leakage.

The atmospheric module loop is also assumed to sustain 98% closure. An initial air supply is calculated to activate the cabin air revitalization system. The stored air supply accounts for gas makeup from potential module leakage and a contingency amount, equivalent to two full repressurizations. The oxygen and nitrogen stored for the two module configurations also includes a supply for the logistics module.

Dry consumables are categorized in two ways - consumed and not replenished (such as food), and reusable (such as clothes). The amount of consumed items are calculated according to the number of crew members (5) and the mission duration (3 years), totaling 5475 person-days. Since some of the crew will be stationed on the Martian surface for approximately one year, this is a conservative estimate for consumed goods. However, in case of lander failure, the full crew would have sufficient consumables for the duration of the mission. The amount of reusable goods are determined by a standard supply that should endure the complete mission. A contingency supply of food for a 60 day period is added to offset shortages or contamination.

The storm shelter contains an independent supply of consumables (water and food) sufficient for 10 days. The airlock is also outfitted with a water supply to support EVA's and an air supply for EVA's and multiple pressurizations.

The following pages provide an itemized account of the mission consumables in terms of mass and volume. Additional assumptions regarding air composition, airlock requirements, and dry-goods quantities appear in the designated sections.

A summary of the mission consumables mass and volume for the two module configurations appears below.

Summary	Disk Mass (kg)	Cyl Mass (kg)	Disk Volume (m^3)	Cyl Volume (m^3)
Water Supply	5551.1	5551.1	5.76	5.76
Initial Air Pressurization	788.1	425.5	0	0
Module Oxygen Stored Supply	853.1	756.7	0.80	0.80
Module Nitrogen Stored Supply	2968.6	2546.0	3.50	3.50
Airlock Oxygen Supply	36.4	36.4	0.04	0.04
Airlock Nitrogen Supply	145.7	145.7	0.20	0.20
Dry Consumables	9732.3	9732.3	35.68	35.68
Total Consumables	20,075.3	19,193.7	46.0	46.0

Notes:

1. Tank Sizing

The following equation was used to size the storage tanks for oxygen, nitrogen, and water. All storage tanks are assumed spherical. The pressure vessels are assumed to be constructed of graphite/epoxy overwrap with a thin metal liner to save weight. All tanks will be covered with multilayer insulation for passive thermal protection. The oxygen and nitrogen tanks would require a compatible passive metallic material such as Inconel, while the water tanks liner would be aluminum alloy 2219-T87 since storage pressures would be low.

$$D_i = 2 * [3 * M * (1+f_u)/(\rho * N * 4\pi)]^{1/3}$$

where,

- D_i Inside tank diameter (m), if D_i>4 m, need to increase N
- M Mass of stored material (kg)
- f_u Ullage factor. For gaseous reactant tanks, f_u=0; for liquids, f_u=0.05.
- ρ Material density (kg/m^3). For water, 1000 kg/m^3; for liquid oxygen, 1140 kg/m^3 @-183°C; for liquid nitrogen, 819 kg/m^3 @-195°C.
- N Number of tanks.

2. Trash/Waste Produced

The solid wastes produced during the Mars mission will not be recycled. Facilities for human waste and trash compaction are included in the pressurized modules. Below is a brief listing of the mass incurred by these solid wastes. Whether the wastes would be stored in place of the consumables for the duration of the mission or ejected while in orbit is an issue to be resolved.

Waste	kg/person-day	kg	Reference
Trash	0.816	4467.6	2
Urine Solids	0.059	323.0	
Fecal Solids	0.032	175.2	
Sweat Solids	0.018	98.6	
Water solids (@13%)	0.709	3881.8	
Total		8946.2	

References

1. NASA, Space Station Program Office, "ECLSS Architectural Control Document - Revision B," SSP 30262, July 30, 1988.
2. Powell, F.T., and R.A. Wynveen, "Comprehensive Study of Environmental Control and Life Support Technology for Advanced Manned Space Missions," NASA Contract #NAS9-17531, July, 1987.
3. NASA, Space Station Program System Engineering and Integration, "Weights Data Book - Revision 2.0," June, 1988.
4. NASA, JSC - Crew Systems Division, Notes Regarding Space Station Crew Accommodations, Gary Kitmacher, January, 1989.

Table 3.2.1 Mission Consumables

Mission Requirements:

Crew Size	Duration (yrs)	Person-days
5	3	5475

WATER

Notes:
Assume 98% closure of crew water loop.

1) System Make-up Water Supply (@ 2%)

	Mass (kg/per-day)	Mass (Cyl & Disk) (kg) 2% of total supply for 3 yrs	Reference 1
Clothing wash water	12.47	1365.46	
Hand wash water	1.81	198.20	
Shower water	3.63	397.49	
Dishwater washing	0.68	74.46	
Drinking water	1.61	176.30	
Urinal flush water	0.49	53.66	
Food preparation water	0.79	86.50	
Subtotal		2,352.1	

2) Initial Water Supply (to start water loop)

	Mass (kg/per-day)	Mass (Cyl & Disk) (kg) 1 week supply	Reference 1
Clothing wash water	12.47	436.45	
Hand wash water	1.81	63.35	
Shower water	3.63	127.05	
Dishwater washing	0.68	23.80	
Drinking water	1.61	56.35	
Urinal flush water	0.49	17.15	
Food preparation water	0.79	27.65	
Subtotal		751.8	

3) Contingency Water Supply

	Mass (kg/per-day)	Mass (Cyl & Disk) (kg) 10% for mission
Drinking water	1.30	711.75
Food preparation water	1.27	695.33
Subtotal		1407.10

4) Miscellaneous Water

	Mass (Cyl & Disk) (kg)	Reference 3
Thermal control loop	91.80	
Fire suppression	43.90	
Module atmosphere	760.42	
Total	896.12	

5) Storm Shelter Water Supply

	Mass (kg/per-day)	Mass (Cyl & Disk) (kg/10 days)
Drinking water	1.3	65.0

6) EVA - Stored Water Supply

	Mass (kg/8 hr)	Mass (Cyl & Disk) (kg)	Reference 2
1 EVA per 2 months	4.39	79.02	

<u>Total Water Supply for Mission</u>

	Mass (kg)	No. tanks	Diameter (m)	Vol/tank (m^3)	Total Vol (m^3)
Both Module Configurations	5551.14	4	1.4	1.44	5.76

Table 3.2.1 Mission Consumables (cont.)

AIR

Notes:

Assume 98% closure of module atmospheric loop over duration of mission.

Metabolic oxygen	0.83 kg/person-day
Specific weight of air @ 14.7 psia and 0C:	1.29273 kg/m^3
Specific weight of air @ 0.5 psia and 21C:	0.04085 kg/m^3
Pressurized volume for disk module:	609.60 m^3
Pressurized volume for cylindrical modules:	329.18 m^3
Crew Lock volume @ 0.5 psia:	5.30 m^3
Equipment Lock volume @ 14.7 psia:	19.68 m^3
Logistics Module volume @ 14.7 psia:	73.43 m^3
Composition of air:	20% oxygen and 80% nitrogen

Modules

		Mass-Disk	Mass-Cyl	
1) System Make-up Oxygen Supply (@ 2%)	(kg/person-day)	(kg)	(kg)	Reference Calculation
Oxygen	0.83	90.9	90.9	
2) Initial Air Supply		(kg)	(kg)	Reference
Air		788.05	425.54	Calculation
Oxygen		157.61	85.11	
Nitrogen		630.44	340.43	
3) Contingency Supply-stored (2 repressurizations)		(kg)	(kg)	
Air		1576.10	851.08	
Oxygen		315.22	170.22	
Nitrogen		1260.88	680.87	
4) Makeup from module leakage-stored	(kg/day/module)	(kg)	(kg)	Reference
Oxygen	0.12	131.40	262.80	2
Nitrogen	0.45	492.75	985.50	
5) Subtotal Module Supply		Mass-Disk (kg)	Mass-Cyl (kg)	
Oxygen		695.13	609.02	
Nitrogen		2384.07	2006.80	

5) Logistics Module Supply	Mass (kg)
Initial Air Supply	111.02
Contingency Supply (2 repressurizations)	222.04
Total Oxygen	66.61
Total Nitrogen	266.45

Total Module Stored Supply for Mission

	Mass (kg)	No. tanks	Diameter (m)	Vol/tank (m^3)	Total Vol (m^3)
Disk module + Logistics module					
Oxygen (+10% boiloff, 2% unusable)	853.1	3	0.8	0.27	0.8
Nitrogen (+10% boiloff, 2% unusable)	2968.6	5	1.1	0.70	3.5
Cylindrical modules + Logistics module					
Oxygen (+10% boiloff, 2% unusable)	756.7	3	0.8	0.27	0.8
Nitrogen (+10% boiloff, 2% unusable)	2546.0	5	1.1	0.70	3.5

Airlock (Equipment & Crew Locks):

	Mass (kg)	Reference
Initial Air Supply	25.66	Calculation
EVA Losses (1 EVA/2 months & 1.09 kg/EVA)	19.62	2
Contingency Supply (4 repressurizations)	102.63	Estimate
Makeup for leakage (1 crew lock/2 months)	3.90	Estimate
EVA - Stored Oxygen Supply (0.6 kg/8hr)	10.80	2

Total Airlock Stored Supply for Mission

	Mass (kg)	No. tanks	Diameter (m)	Vol/tank (m^3)	Total Vol (m^3)
Air	162.60				
Oxygen (+10% boiloff, 2% unusable)	36.42	2	0.32	0.018	0.035
Nitrogen (+10% boiloff, 2% unusable)	145.69	2	0.57	0.098	0.196

Table 3.2.1 Mission Consumables (cont.)

DRY CONSUMABLES

Notes:

Crew Size	Contingency (days)	Person-days
5	60	300

1) Initial Supply	Mass (kg/per-day)	Volume (m³/per-day)	Mass (kg)	Volume (m³)	Reference
Food Supply (dehydrated)	0.62	0.0016	3394.50	8.760	2/4
Food Packaging	0.45	0.0002	2463.75	0.876	2/4
Liquids (soaps, biocides, etc)	0.27	0.0020	1478.25	10.950	2/E
Waste Disposal (filters, wipes, bags)	0.21	0.0020	1149.75	10.950	4/E
Solids:	Unit Mass (kg)	Unit Volume (m³)	Mass (kg)	Volume (m³)	Reference
Clothing (3 wk sply, not disposable)	6.51	0.016	683.55	1.68	4/4
Personal Hygiene (2 kits/person)	3.00	0.028	30.00	0.28	4/4
Kitchen Supply (3 sets/person)	3.56	0.028	53.40	0.42	4/4
Tools & Maintenance Supplies	49.21	0.700	49.21	0.70	4/4
Off-duty Supplies	55.34	0.450	55.34	0.45	4/4
Subtotal			9357.80	35.066	

2) Contingency Supply	Mass (kg/per-day)	Volume (m³/per-day)	Mass (kg)	Volume (m³)	Reference
Food Supply (dehydrated)	0.62	0.0016	186	0.480	2/4
Food Packaging	0.45	0.0002	135	0.048	2/4
Subtotal			321	0.528	

3) Storm Shelter Supply	Mass (kg/per-day)	Volume (m³/per-day)	Mass (kg/10 day)	Volume (m³/10 days)	Reference
Food Supply (dehydrated)	0.62	0.0016	31.0	0.080	2/4
Food Packaging	0.45	0.0002	22.5	0.008	2/4
Subtotal			53.5	0.088	

Total dry consumables for Mission	Mass (kg)	Volume (m³)
	9732.3	35.7

Mission Consumables Summary	Disk Mass (kg)	Cylindrical Mass (kg)	Disk Volume (m³)	Cylindrical Volume (m³)
Water Supply	5551.14	5551.14	5.76	5.76
Initial Air Pressurization	788.05	425.54	0	0
Module Oxygen Stored Supply	853.10	756.70	0.80	0.80
Module Nitrogen Stored Supply	2968.60	2546.00	3.50	3.50
Airlock Oxygen Supply	36.42	36.42	0.04	0.04
Airlock Nitrogen Supply	145.69	145.69	0.20	0.20
Dry Consumables	9732.25	9732.25	35.68	35.68
Total Consumables	20,075.30	19,193.70	45.98	45.98

3.3 Active Thermal Control System (ATCS)

The internal portion of the active thermal control system consists of a thermal bus located within the pressurized modules. The internal thermal bus provides for the acquisition and transport of cooling loads. The internal system consists of a closed loop maintained in circulation by a pump, with the working fluid as water. The thermal transport loops are separated into two temperature levels, 2°C and 21°C. The 2°C loop services the life support system and personal hygiene loads. This lower water temperature is required to effectively perform condensate recovery and humidity control as well as maintain thermally sensitive equipment. The 21°C loop services the higher temperature equipment located in the galley, workstations, and health facility. The ATCS is designed for the following capacities:

1) interface between the equipment and thermal bus at 10 kw for 2°C and at 20 kw for 21°C, and
2) pumps and thermal devices at 75 kw.

Passive thermal control measures are considered part of the temperature and humidity control system which appears in the life support systems definition. Passive thermal control equipment includes fans for avionics cooling, ventilation, and cabin air distribution. Other forms of passive temperature control, such as multilayer insulation or surface coatings, were not investigated for this study.

The following pages provide an itemized account of the two thermal loop buses and additional support equipment. Quantities of each piece of hardware are specified according to the configuration. All items are duplicated for the cylindrical modules and the disk module in order to provide independent loop capability in each pressurized volume. Additional mass for equipment spares is included at 10% of the total system mass.

A summary of the system characteristics for the two module configurations appears below.

Summary

	Mass (kg)	Volume (m^3)	Power (kw)
2° Internal Loop	431.7	0.41	0.38
21° Internal Loop	342.3	0.37	0.33
Miscellaneous Equipment	32.7	0.25	1.14
Thermal Control System Total	806.8	1.03	1.84

Reference

"Central Thermal Control System Properties for the Space Station," Thermal Systems Division, JSC, January, 1987.

Table 3.3.1 Active Thermal Control System

Internal Thermal Bus

2 C Loop Hardware	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Unit "on"	Mass (kg)	Volume (m^3)	Power (w)
Pump Package	2	2	14.51	0.036	212	0.5	2	29.02	0.072	212
Control Valves-primary	2	2	3.82	0.006	75	1	2	7.64	0.012	150
Control Valves-secondary	2	2	3.82	0.006	75	0	0	7.64	0.012	0
Heat Exchanger	2	2	6.12	0.007	0	0	0	12.24	0.014	0
Coldplate	2	2	20.32	0.024	0	0	0	40.64	0.048	0
Quick Disconnects	2	2	2.93	0.001	0	0	0	5.86	0.002	0
Working Fluid - water	2	2	2.83	0	0	0	0	5.66	0	0
Temperature Sensors	2	2	0.12	0	0	0	0	0.24	0	0
Pressure Sensor	2	2	0.17	0	2	1	2	0.34	0	4
Flow Meter	2	2	0.75	0.002	0	0	0	1.5	0.004	0
Controller	2	2	4.45	0.01	0	0	0	8.9	0.02	0
Readout Equipment	2	2	9.07	0.028	5	1	2	18.14	0.056	10
Thermal Storage	2	2	45.36	0.043	0	0	0	90.72	0.086	0
Plumbing:										
Liquid Line (0.2 in ID)	2	2	0.95	0.001	0	0	0	1.9	0.002	0
Vapor Line (1.85 in ID)	2	2	81.02	0.043	0	0	0	162.04	0.086	0
Spares @ 10% of total mass			19.62					39.25		
2 C Loop Subtotal								431.7	0.414	376

21 C Loop Hardware:	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Unit "on"	Mass (kg)	Volume (m^3)	Power (w)
Pump Package	2	2	7.26	0.036	212	0.5	2	14.52	0.072	212
Control Valves-primary	2	2	2.54	0.004	50	1	2	5.08	0.008	100
Control Valves-secondary	2	2	2.54	0.004	50	0	0	5.08	0.008	0
Heat Exchanger	2	2	6.12	0.007	0	0	0	12.24	0.014	0
Coldplate	2	2	20.32	0.024	0	0	0	40.64	0.048	0
Quick Disconnects	2	2	2.93	0.001	0	0	0	5.86	0.002	0
Working Fluid - water	2	2	2.83	0	0	0	0	5.66	0	0
Temperature Sensors	2	2	0.12	0	0	0	0	0.24	0	0
Pressure Sensor	2	2	0.17	0	2	1	2	0.34	0	4
Flow Meter	2	2	0.75	0.002	0	0	0	1.5	0.004	0
Controller	2	2	4.45	0.01	0	0	0	8.9	0.02	0
Readout Equipment	2	2	9.07	0.028	5	1	2	18.14	0.056	10
Thermal Storage	2	2	45.36	0.043	0	0	0	90.72	0.086	0
Plumbing:										
Liquid Line (0.2 in ID)	2	2	0.92	0.001	0	0	0	1.84	0.002	0
Vapor Line (1.85 in ID)	2	2	50.21	0.026	0	0	0	100.42	0.052	0
Spares @ 10% of total mass			15.559					31.12		
21 C Loop Subtotal								342.3	0.372	326

Table 3.3.1 Active Thermal Control System (cont.)

Miscellaneous Equipment:	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Unit "on"	Mass (kg)	Volume (m^3)	Power (w)
Water Thermal Conditioner	2	2	4.50	0.080	1500	0.25	2	9.00	0.160	750
Water Coolant Pump	2	2	7.30	0.040	335	0.50	2	14.60	0.080	335
Water Controller Valves	2	2	2.54	0.004	50	0.50	2	5.08	0.008	50
Bus Control Unit	2	2	0.54	0	3	1.00	2	1.08	0	6
Spares @ 10% of total mass			1.49					2.98		
Equipment Total								32.70	0.200	1141

<u>Active Thermal Control System Summary</u>

	Mass (kg)	Volume (m^3)	Power (kw)
2° Internal Loop	431.7	0.41	0.38
21° Internal Loop	342.3	0.37	0.33
Miscellaneous Equipment	32.7	0.25	1.14
System Total	806.8	1.03	1.84

3.4 Module Interior Definition

The pressurized habitation modules house the Mars crew of five in a shirt-sleeve environment. The interior design and definition was performed for artificial gravity conditions, assuming a gravity environment ranging from 3/8 g to 1 g for the majority of the mission duration. However, in order for the crew to function in the zero gravity mode during despin operations, the hardware defined is derived from S.S. Freedom data. Additional items, such as restraints and handrails, would allow the transition from gravity to zero gravity.

The disk module and the 2 cylindrical modules are divided into seven main functional areas, including

1. Quarters
2. Galley/Wardroom
3. Command and Control Center
4. Lab and Maintenance Work Areas
5. Personal Hygiene
6. Health Maintenance Facility
7. Electrical Systems - Data Management, Audio-Video, Power Distribution.

Each crew member is furnished his/her own private quarters, outfitted with a bed, workstation, storage space, and seating. The galley provides the basic capabilities for food preparation, trash management, and housekeeping. The wardroom, with a full crew seating capacity, serves the needs of dining room and all-purpose meeting space. The command and control center contains work space for two individuals to monitor the vehicle, practice simulations, perform communications, and transfer data. The lab allows for scientific analysis during the mission, with basic equipment included for observing the solar radiation environment. In addition to the lab is a maintenance work area for equipment repair and stowage for tools and small spares. The personal hygiene area provides the crew with facilities (shower, handwash, and commode) for maintaining hygiene. The health maintenance facility includes extensive provisions for fitness, surgery, dental care, pharmacy, and overall medical treatment. The electrical systems defined satisfy three requirements - data management system, audio-video equipment, and power distribution.

The following pages provide an itemized account of the module interior elements in terms of mass, volume, and power requirements. The redundancy of each element is designated by the quantity columns listed. Interior divisions that do not require redundancy include quarters and lab/maintenance work areas. Otherwise, all equipment is duplicated in the cylindrical modules configuration in order to fulfill the safe haven requirement. Equivalent redundancy exists in the disk module except for equipment associated with the health maintenance facility (note the variation in the quantity column). Surgical capability and some supporting fluid therapy are not contained on both pressurized levels; furthermore, the fitness provisions are solely located on the middle level, accessible from both safe haven floors. Quantities for most of the electrical systems are derived from those specified for S.S. Freedom (ref. 4).

A summary of the module interior characteristics for the two module configurations appears below.

Summary

	Cyl Mass (kg)	Disk Mass (kg)	Cyl Volume (m^3)	Disk Volume (m^3)	Cyl Ave Power (w)	Disk Ave Power (w)
Quarters	658.3	658.3	23.0	23.0	0.3	0.3
Galley/Wardroom	1218.1	1218.1	19.2	19.2	0.8	0.8
Command and Control Center	293.5	293.5	8.9	8.9	0.5	0.5
Lab/Maintenance Work Areas	788.2	788.2	2.9	2.9	0.7	0.7
Personal Hygiene	616.7	616.7	8.4	8.4	0.7	0.7
Health Maintenance Facility	1209.1	996.0	10.3	8.6	1.4	1.4
Data Management Systems	1285.0	1285.0	1.4	1.4	1.3	1.3
Audio/Video Equipment	370.3	370.3	0.7	0.7	0.4	0.4
Electrical Power Distribution	508.9	508.9	2.3	2.3	0.6	0.6
Module Interior Totals	6948.1	6735.0	77.1	75.4	6.7	6.7

References

1. NASA Document 1, "Space Station Health Maintenance Facility Systems Requirements," Medical Sciences Division, Johnson Space Center, July, 1986.

2. Boeing Proposal, Space Station Work Package 1, Vol II, Technical Part 1, Systems Engineering and Integration, July 21, 1987, Submitted to MSFC.

3. NASA Technical Memorandum 89604, "Reference Mission Operational Analysis Document (RMOAD) for the Life Sciences Research Facilities," February, 1987.

4. NASA, Space Station Program System Engineering and Integration, "Weights Data Book - Revision 2.0," June, 1988.

5. NASA, Space Station Program System Engineering and Integration, "Power Data Book," August, 1988.

6. NASA, JSC - Crew Systems Division, Notes Regarding Space Station Crew Accommodations, Gary Kitmacher, January, 1989.

Table 3.4.1 Module Interior Definition

Quarters (per crew member)	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	Ave Power (w)	Reference
Bed	49.96	0.330	0	0	0	Calculation
Desk/terminal space	59.00	1.770	175	0.2	35.0	Calculation
Personal stowage	20.00	2.500	0	0	0	Calculation based on ref. 2
Lighting - general	1.81	0.002	19	0.5	9.5	2/6/5
Vent Fan	0.90	0	20	1.0	20.0	4/-/5
Quarters Unit Subtotal	131.70	4.610	194		64.5	
Quarters for 5 Subtotal	658.30	23.030	970		322.5	

Galley/Wardroom	Quantity Cyl	Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m^3)	Ave Power (w)
Oven/microwave	2	2	20.17	0.384	2370	0.10	1	4/6/5	40.34	0.768	237.00
Dishwasher	2	2	199.58	0.568	1510	0.08	1	4/6/5	399.16	1.136	120.80
Trash compactor	2	2	54.89	0.176	365	0.01	1	4/6/5	109.78	0.352	3.65
Water dispenser	2	2	39.92	0.083	700	0.03	2	4/6/5	79.84	0.166	42.00
Hand washer	2	2	45.36	0.206	600	0.01	1	4/6/5	90.72	0.412	6.00
Clothing washer/dryer	2	2	67.59	1.140	1260	0.17	1	4/6/5	135.18	2.280	214.20
Housekeeping Supplies	2	2	6.80	0.280	0	0	0	4/2	13.60	0.560	0
Vacuum	4	4	3.07	0.127	500	0.03	2	4/6/5	12.28	0.508	30.00
Stowage Miscellaneous	2	2	33.00	1.700	0	0	0	E	66.00	3.400	0
Video/VCR	2	2	6.23	0.050	40	0.10	1	4/6/5	12.46	0.100	4.00
Wardroom display provtns	4	4	8.34	0.800	0	0	0	4/6/-	33.36	3.200	0
Inventory monitor	2	2	9.07	0.083	100	0.10	1	E/6/5	18.14	0.166	10.00
Table	2	2	49.90	1.700	0	0	0	2/C	99.80	3.400	0
Seating (for 5)	2	2	17.50	1.350	0	0	0	E	35.00	2.700	0
Lighting - task	8	8	1.81	0.002	30	0.40	8	2/6/5	14.48	0.016	96.00
Spares @ 5% of subtotal mass			28.16						58.01		
Galley/Wardroom Subtotal									1218.1	19.2	763.7

Table 3.4.1 Module Interior Definition (cont.)

Command and Control Center	Quantity Cyl	Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m^3)	Ave Power (w)
Desk/seating	4	4	33.00	1.270	0	0	0	E	132.00	5.080	0
Computer terminals	4	4	18.14	0.224	200	0.5	4	3	72.56	0.896	400
Ancillary provisions	4	4	15.00	0.710	0	0	0	E	60.00	2.840	0
Lighting - task	8	8	3.62	0.010	30	0.4	8	2/2/5	28.96	0.080	96
Command and Control Center Subtotal									293.5	8.9	496

Lab/Maintenance Work Areas	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	Ave Power (w)	Reference
Proton/heavy ion spectrometer	9.07	0.008	100.0	1.00	100.0	3
Spectrophotometer (visual,UV,IR)	40.00	0.100	300.0	1.00	300.0	3
Radiation dosimeter	5.00	0.008	0	0	0	3
Ion selective chromatograph	12.69	0.024	0	0	0	3
Expmt control & data interface	11.34	0.091	120.0	1.00	120.0	3
Terminal, computer	9.07	0.112	100.0	0.50	50.0	3
Digital multimeter	0.79	0.001	0	0	0	4/2/2
Recording oscilloscope	44.97	0.068	40.0	1.00	40.0	4/3/5
Digital thermometer	0.60	0.001	0	0	0	4/2/2
Microscope system	48.08	0.231	150.0	0.25	37.5	2
Dosimeter, passive	63.01	0.040	1.0	1.00	1.0	4/2/2
Mass measurement - small	53.00	0.068	1.0	1.00	1.0	4/2/2
Incubator	113.76	0.164	60.0	0.05	3.0	4/3/5
Freeze dryer	109.00	0.258	200.0	0.05	10.0	4/3/5
Cleaning equipment	30.31	0.125	0	0	0	4/2
Fluid handling tools	40.00	0.100	0	0	0	4/2
Maintenance workstation	197.54	1.473	275.5	0.20	55.1	2
Lighting - task (2)	3.62	0.010	30.0	0.40	24.0	2/2/5
Lab/Maintenance Subtotal	788.20	2.870	1347.5		717.6	

Personal Hygiene	Quantity Cyl	Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m^3)	Ave Power (w)
Shower-pump, heater, blower	2	2	90.72	2.100	1400	0.03	2	4/6/5	181.44	4.200	84.00
Commode	2	2	88.00	1.780	360	0.07	2	4/6/5	176.00	3.560	50.40
Hand wash/grooming	2	2	40.82	0.226	1846	0.08	2	2/6/5	81.64	0.452	295.36
Air/water separator	8	8	18.13	0.020	47	1.00	4	4/E/5	145.04	0.160	188.00
Lighting - general	6	6	5.43	0.006	19	0.30	6	2/2/5	32.58	0.036	34.20
Personal Hygiene Subtotal									616.7	8.4	652.0

Table 3.4.1 Module Interior Definition (cont.)

Health Maintenance Facility	Quantity Cyl	Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	#Cyl Units "on"	#Disk Units "on"	Ref	Cyl Mass (kg)	Disk Mass (kg)	Cyl Volume (m^3)	Disk Volume (m^3)	Ave Cyl Power (w)	Ave Disk Power (w)
1) Fitness															
Treadmill	2	1	11.79	0.310	25	0.40	1	1	3	23.58	11.79	0.620	0.310	10.0	10.0
Rowing machine	2	1	17.24	0.220	0	0	1	1	3	34.48	17.24	0.440	0.220	0	0
Anaerobic exercise device	2	1	21.46	0.170	105	0.40	1	1	3	42.92	21.46	0.340	0.170	42.0	42.0
Bicycle ergometer	2	1	13.61	0.120	10	0.40	1	1	3	27.22	13.61	0.240	0.120	4.0	4.0
2) Dental	2	2	4.54	0.040	0	0	0	0	1	9.08	9.08	0.080	0.080	0	0
3) Fluid therapy															
Fluid infusion module	2	1	22.68	0.020	0	0	0	0	1	45.36	22.68	0.040	0.020	0	0
Formulation module	2	1	2.27	0.020	0	0	0	0	1	4.54	2.27	0.040	0.020	0	0
IV pump module	2	2	3.00	0.010	250	0	0	0	1	6.00	6.00	0.020	0.020	0	0
Tubing sets/accessories	2	1	2.27	0.020	0	0	0	0	1	4.54	2.27	0.040	0.020	0	0
IV catherization tray mod.	2	2	2.27	0.030	0	0	0	0	1	4.54	4.54	0.060	0.060	0	0
IV drugs & fluid salts	2	2	4.54	0.020	0	0	0	0	1	9.08	9.08	0.040	0.040	0	0
Urine collection system	2	1	18.14	0.140	50	0	0	0	3	36.28	18.14	0.280	0.140	0	0
Blood storage	2	1	6.80	0.010	0	0	0	0	1	13.60	6.80	0.020	0.010	0	0
Nutritional substrates mod.	2	1	22.68	0.040	0	0	0	0	1	45.36	22.68	0.080	0.040	0	0
4) Imaging, ultrasound	2	2	90.72	0.179	600	0	0	0	3	181.44	181.44	0.358	0.358	0	0
5) Medical analytical Lab															
Blood/urine chemistry	2	2	2.27	0.080	360	0.05	1	1	1	4.54	4.54	0.160	0.160	18.0	18.0
Blood dissolved gases	2	2	2.27	0.040	280	0.05	1	1	1	4.54	4.54	0.080	0.080	14.0	14.0
Hematology	2	2	9.10	0.040	110	0.05	1	1	1	18.20	18.20	0.080	0.080	5.5	5.5
Microbiology/urinalysis	2	2	4.54	0.100	700	0.05	1	1	1	9.08	9.08	0.200	0.200	35.0	35.0
Centrifuge	2	2	22.68	0.020	200	0.05	1	1	1	45.36	45.36	0.040	0.040	10.0	10.0
Fluid mixer	2	2	6.80	0.010	150	0.05	1	1	1	13.60	13.60	0.020	0.020	7.5	7.5
Workstation/medical comp	2	2	24.04	0.660	700	0.25	2	2	1	48.08	48.08	1.320	1.320	350.0	350.0
6) Ptnt restraint (collps.)	2	1	30.00	0.500	0	0	0	0	E	60.00	30.00	1.000	0.500	0	0
7) Pharmacy															
Pharmacy storage - freezer	2	2	38.68	0.730	400	1.00	2	2	3	77.36	77.36	1.460	1.460	800.0	800.0
Pharmacy supplies	2	2	49.89	0.380	0	0	0	0	1	99.78	99.78	0.760	0.760	0	0
8) Physician's instruments															
Human Calorimeter	2	2	6.78	0.042	0	0	0	0	3	13.56	13.56	0.084	0.084	0	0
Heart rate monitor	2	2	0.95	0	10	0.40	2	2	3	1.90	1.90	0	0	8.0	8.0
Diagnostic instruments	2	2	4.54	0.010	0	0	0	0	1	9.08	9.08	0.020	0.020	0	0
Eye tray	2	2	2.72	0.010	0	0	0	0	1	5.44	5.44	0.020	0.020	0	0
Emergency life support kit	2	2	13.61	0.750	100	0	0	0	3	27.22	27.22	1.500	1.500	0	0
9) Power (AC/DC modules)	2	2	13.60	0.100	0	0	0	0	1	27.20	27.20	0.200	0.200	0	0
10) Surgery															
Anesthesia/analgesia	2	1	4.54	0.030	0	0	0	0	1	9.08	4.54	0.060	0.030	0	0
Surgical instrument kit	2	1	6.80	0.020	0	0	0	0	1	13.60	6.80	0.040	0.020	0	0
Air/fluid separator system	2	1	22.68	0.060	480	0	0	0	1	45.36	22.68	0.120	0.060	0	0
11) Ventilator/respiratory															
Mechanical ventilator	2	2	18.14	0.070	320	0	0	0	1	36.28	36.28	0.140	0.140	0	0
O2 consumption monitor	2	2	18.14	0.080	200	0	0	0	1	36.28	36.28	0.160	0.160	0	0
12) Lighting - task	8	8	7.24	0.020	30	0.30	8	8	2/2/5	57.92	57.92	0.160	0.160	72.0	72.0
Spares @ 5% of subtotal mass			27.70							57.57	47.43				
Health Maintenance Facility Subtotal										1209.1	996.0	10.3	8.6	1376.0	1376.0

Table 3.4.1 Module Interior Definition (cont.)

ELECTRICAL SYSTEMS

Data Management System (DMS)	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m^3)	Ave Power (w)
Bridge	4	4	15.870	0.007	100	1.00	2	4/2/5	63.48	0.028	200.00
NIU Assembly	2	2	6.530	0.007	87	0.25	2	2/2/5	13.06	0.014	43.50
MDM, EDP, I/O cards	20	20	14.520	0.015	20	0.50	15	4/2/5	290.40	0.300	150.00
Standard Data Processor	4	4	16.780	0.015	104	0.50	2	4/2/5	67.12	0.060	104.00
Magnetic disk assembly	2	2	27.220	0.028	40	0.50	1	2	54.44	0.056	20.00
Multipurpose console	2	2	12.750	0.015	117	0.50	2	2	25.50	0.030	117.00
Optical disk assembly	2	2	31.620	0.065	130	0.25	1	2	63.24	0.130	32.50
Multipurpose console CRT	4	4	38.100	0.094	180	0.50	2	4/2/2	152.40	0.376	180.00
Multipurpose panel display	2	2	9.070	0.003	75	0.50	2	4/2/5	18.14	0.006	75.00
Multips caution/warn panel	2	2	9.070	0.007	2	1.00	2	4/2/5	18.14	0.014	4.00
Multipurpose printer	2	2	24.950	0.035	285	0.05	1	4/2/5	49.90	0.070	14.25
Multipurpose keyboard	4	4	0.910	0.007	5	0.50	2	4/2/5	3.64	0.028	5.00
Multipurpose handcontroller	2	2	1.360	0.001	20	0.25	1	2	2.72	0.002	5.00
High-rate data patchboard	2	2	13.610	0.007	140	1.00	1	2/2/5	27.22	0.014	140.00
Portable multips console	2	2	9.070	0.008	47	0.20	1	2	18.14	0.016	9.40
Portable multips ports	2	2	4.540	0	0	0	0	2	9.08	0	0
Mass Storage Unit	5	5	69.000	0.060	160	0.30	4	4/E/5	345.00	0.300	192.00
Electrical Cabling	60	60	0.005	0	0	0	0	4	0.30	0	0
Optical Fiber - Single	87	87	0.005	0	0	0	0	4	0.44	0	0
Optical Fiber - Dual	166	166	0.010	0	0	0	0	4	1.66	0	0
Spares @ 5% of subtotal mass			15.250						61.20		
DMS Subtotal									1285.2	1.4	1291.7

Audio-Video Equipment	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m^3)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m^3)	Power (w)
Speaker microphone unit	15	15	3.31	0.003	18	0.40	10	4/2/5	49.65	0.045	72.0
Audio recorder unit	2	2	11.34	0.022	35	0.40	1	4/2/5	22.68	0.044	14.0
Audio interface unit	2	2	2.72	0.003	19	0.40	1	4/2/5	5.44	0.006	7.6
Voice recognition unit	2	2	2.72	0.003	20	0.20	1	4/2/5	5.44	0.006	4.0
Audio bus interface adapter	2	2	0.50	0.001	2	1.00	1	4/2/5	1.00	0.002	2.0
Crew wireless unit	10	10	1.13	0.001	0	0	0	4/2	11.30	0.010	0
Battery charger	15	15	0.68	0.001	7	0.40	10	4/2/5	10.20	0.015	28.0
Video switching unit	2	2	9.07	0.020	31	1.00	1	4/2/5	18.14	0.040	31.0
Sync & control generator	2	2	10.89	0.022	50	0.80	1	4/2/5	21.78	0.044	40.0
Pan-tilt unit	4	4	3.58	0.006	20	0.10	4	4/2/5	14.32	0.024	8.0
Video recorder	2	2	31.75	0.050	190	0.40	1	4/2/5	63.50	0.100	76.0
Video control adapter	2	2	0.50	0.001	2	1.00	1	4/2/5	1.00	0.002	2.0
Camera body	4	4	4.08	0.009	25	0.20	4	4/2/5	16.32	0.036	20.0
Viewfinder Monitor	2	2	1.77	0.002	7	0.20	1	4/2/5	3.54	0.004	1.4
Audio bus assembly	2	2	12.47	0	20	0.35	1	4/2/5	24.94	0	7.0
Video bus assembly	2	2	5.44	0	0	0	0	4/2	10.88	0	0
Video Monitor	4	4	18.14	0.080	150	0.12	2	4/E/5	72.56	0.320	36.0
Spares @ 5% of subtotal mass			6.00						17.63		
Audio-Video Subtotal									370.3	0.7	349.0

Table 3.4.1 Module Interior Definition (cont.)

Electrical Power Distribution	Quantity Cyl	Quantity Disk	Unit Mass (kg)	Unit Volume (m³)	Unit Power (w)	Duty Cycle	# Units "on"	Ref	Mass (kg)	Volume (m³)	Power (w)
Bulk convertor	2	2	21.32	0.004	100	1	1	4/2/2	42.64	0.008	100
28V dc power	2	2	0.36	0.001	8	1	2	2	0.72	0.002	16
Power control unit	20	20	1.36	0.110	39	1	10	4/2/5	27.20	2.200	390
Power protection assembly	2	2	2.00	0.002	3	1	2	2	4.00	0.004	6
Light switches (20)	2	2	10.00	0.020	0	0	0	2	20.00	0.040	0
Emergency light assembly	2	2	4.00	0.008	33	1	2	2	8.00	0.016	66
Bulkhead feedthrough	2	2	0.68	0.001	0	0	0	2	1.36	0.002	0
Utility outlets (50)	2	2	4.50	0	0	0	0	2	9.00	0	0
Bulk Converter Cable	2	2	74.84	0	0	0	0	4	149.68	0	0
Distribution cables (15% DMS/EPD)	1	1	-----	-----	0	0	0	C	232.00	0.033	0
Spares @ 5% of subtotal mass			7.18						14.36		
Electrical Power Subtotal									508.9	2.3	578

Summary	Cyl Mass (kg)	Disk Mass (kg)	Cyl Volume (m³)	Disk Volume (m³)	Cyl Ave Power (w)	Disk Ave Power (w)
Quarters	658.3	658.3	23.0	23.0	0.3	0.3
Galley/Wardroom	1218.1	1218.1	19.2	19.2	0.8	0.8
Command and Control Center	293.5	293.5	8.9	8.9	0.5	0.5
Lab/Maintenance Work Areas	788.2	788.2	2.9	2.9	0.7	0.7
Personal Hygiene	616.7	616.7	8.4	8.4	0.7	0.7
Health Maintenance Facility	1209.1	996.0	10.3	8.6	1.4	1.4
Data Management Systems	1285.0	1285.0	1.4	1.4	1.3	1.3
Audio/Video Equipment	370.3	370.3	0.7	0.7	0.4	0.4
Electrical Power Distribution	508.9	508.9	2.3	2.3	0.6	0.6
Module Interior Totals	6948.1	6735.0	77.1	75.4	6.7	6.7

3.5 Structures

Cylindrical Modules

The shield and pressure hull sizes are based on one year in LEO exposed to debris/meteoroid environment and three years exposed to interplanetary meteoroid flux (shield = 1.5 mm Al 6061-T6, pressure hull = 2.1 mm Al 2219-T87). Other primary structures are based on Boeing Space Station Definition and Preliminary Design DR-02 WP-01 data package, NAS8-36536, June 1986. Although the inclusion of windows in the module design is suggested, a mass estimate for windows is not included.

Table 3.5.1 Cylindrical Module Specifications

Dimensions (pressure shell):	
Length (inside), m	11.79
Diameter (inside), m	4.22
Volume/module, m^3	164.59
Total Volume, m^3	326.69
Surface Area, m^2	156.16

Table 3.5.2 Structural Mass Requirements for Cylindrical Modules

Structural Component	Mass, kg	Reference
Primary (for 1 module):		
Pressure shell	1065.9	Calculation
Ring frames (4)	539.0	Boeing
Hatches (2)	675.0	Boeing
Meteoroid shield & supports	895.3	Calculation
Trunnion Support Provisions	139.0	Boeing
Grapple Fixture Provisions	19.0	Boeing
Contingency (10%)	333.3	Calculation
Primary subtotal	3666.5	
Secondary (for 1 module):		
Standoffs & cable trays (4)	1116.0	Boeing
Floor/ceiling (horizontal)	402.0	Calculated using graphite/epoxy (t=1.0/0.5 cm).
Equipment rack - (11) Double: 2h x 1.1w x 0.9d	523.9	Boeing
Miscellaneous (@ 25%) (storage bins, internal walls, etc.)	510.5	Calculation
Secondary subtotal	2552.4	
Total	6,218.9	
Total for 2 modules	12,437.8	

Note: Additional Information for Module Shield and Pressure Hull Calculations

- An external surface area of 200 m^2 was calculated for each cylindrical module. It was assumed that 50% of this area was shielded by adjacent structures. Therefore, 100 m^2 of exposed surface was used in the debris/meteoroid impact assessment.

- The new debris environment was used (which is to be issued soon in a revised SSP 30425, "Space Station Program Natural Environment Definition for Design"). The new debris environment includes a 5% per year growth in debris mass, and accounts for the effects of the solar cycle on atmospheric density and drag. Thus, while the old environment was static, the new debris environment generally worsens with time. The debris environment in low-Earth orbit (LEO) is more severe than the LEO or interplanetary meteoroid environment.

- One year in LEO exposed to debris and meteoroid flux is assumed (starting circa 2002), and 3 years exposed to the interplanetary meteoroid flux.

- A 0.9955 probability of no failure from meteoroid or debris impact (while in LEO) is selected as the basis of the shielding calculation for each module. This corresponds with the accepted level of reliability for the Space Station pressurized modules as defined in the requirements document (SSP 30000, Sec.3, Rev.F, "Space Station Program Definition and Requirements, Section 3: Space Station Systems Requirements," May 6, 1988).

- A 1.5 mm (0.059") thick Al 6061-T6 bumper shield was determined to provide the required level of reliability, given that a 0.75 cm debris particle is calculated (from the probability, time in orbit, and exposed area) as the critical particle size which the shield must stop. Based on hypervelocity impact shock theory, a 0.2 ratio of aluminum bumper thickness to projectile diameter is calculated as the optimum to fully shock the projectile at the average debris velocity (10 km/sec). Explanation of the calculational procedure is beyond the scope of this note but can be found in Eagle Report No. 87-163, "Evaluation of Space Station Meteoroid/Debris Shielding Materials," September 30, 1987.

- The mass of shielding support structure (graphite/epoxy composite) was calculated as 10 percent of the aluminum bumper mass, which is a ratio generally similar to that proposed by both WP-01 contractors (Boeing and Martin Marietta) for the S.S. Freedom common modules.

- The pressure hull mass was calculated from the thickness, determined to be 2.1 mm (0.081"). This was the maximum of two different calculations: (1) thickness required to contain the pressure differential, and (2) thickness required to sustain the blast loading and fragmented particles generated by a debris impact of the "design" particle size on the shield.

  For #1 above, the standard equations for pressure vessels were used assuming the pressure hull is made of Al 2219-T87 (same as S.S. Freedom modules) with a yield stress of 358.5 MPa, a factor of safety of 2 (same as S.S. Freedom pressurized modules), and a pressure level of 1 atm. A thickness of 1.19 mm (0.047") was found by this approach as needed to contain the pressure differential with the required safety factor. The module thickness was found to be sized for #2 above (i.e. debris/meteoroid protection was the design driver). The backwall sizing equation of Cour-Palais (B.G. Cour-Palais, "Space Vehicle Meteoroid Shielding Design," pp.85-92, ESA SP-153, October 1979) was used with a spacing of 11.4 cm (4.5", same as S.S. Freedom modules) between shield and backwall.

Disk Module

The shield and pressure hull sizes are based on one year in LEO exposed to debris/meteoroid environment and three years exposed to interplanetary meteoroid flux (shield = 1.5 mm Al 6061-T6, pressure hull = 2.5 mm Al 2219-T87). Other primary structures modified from Boeing Space Station Definition and Preliminary Design DR-02 WP-01 data package, NAS8-36536, June 1986. Although the inclusion of windows in the module design is suggested, a mass estimate for windows is not included.

Table 3.5.3 Disk Module Specifications

Dimensions (pressure shell):

Length (inside), m	11.0
Diameter (inside), m	8.4
Total Volume, m^3	609.6
Surface Area, m^2	401.1

Note:
Capable of meeting ALS payload requirements.

Table 3.5.4 Structural Mass Requirements for Disk Module

Structural Component	Mass, kg	Reference
Primary:		
Pressure shell	1956.3	Calculation
Ring frames (4)	709.4	Calculation
Hatches (3)	1012.5	Estimate from Boeing
Meteoroid shield & supports	1795.6	Calculation
Trunnion Support Provisions	192.3	Estimate from Boeing
Grapple Fixture Provisions	26.3	Estimate from Boeing
Contingency (10%)	569.2	Calculation
Primary subtotal	6261.6	
Secondary:		
Standoffs & cable trays (5)	1543.9	Estimate from Boeing
Floor, pressure vessel (2)	2106.7	Calculation using graphite/epoxy (t=1.3 cm)
Floor, lower level (1)	567.2	Calculation using graphite/epoxy (t=1 cm)
Ceiling (1)	81.0	Calculation using graphite/epoxy (t=0.1 cm)
Equipment Rack (12)	858.2	Derived fr Boeing (1.5x S.S. Freedom double rack)
Stowage Rack - double (3)	143.0	Derived from Boeing
Stowage Rack - single (6)	210.0	Derived from Boeing
Miscellaneous (@ 25%) (internal walls, tracks, handrails, etc.)	1377.5	Calculation
Secondary subtotal	6887.5	
Total	13,149.1	

Note: Additional Information for Module Shield and Pressure Hull Calculations

A similar sizing procedure was used to size the disk module walls with the following differences or changes:

- The surface area of the disk module was determined to be 401 m^2. Assuming 33% is shielded by adjacent structures (less because of the geometry of the disk module), 274 m^2 of surface is exposed to the debris and meteoroid flux.
- The LEO no-failure probability was decreased to 0.991 (the same as two S.S. Freedom modules at 0.9955 each). The critical particle size remains at 0.75 cm and thus the Al 6061-T6 shield thickness remains at 1.5 mm. Shield mass per module changes due to the size difference.
- The pressure level containment criteria dictates pressure hull thickness of 2.58 mm (0.10").

4.0 Additional Structures and Systems

4.1 Radiation Storm Shelter

The long duration of the Earth-Mars transit (3 years) increases the chances of the interplanetary crew encountering a solar flare and thus warrants the need for radiation protection. Although the crew will also be exposed to galactic cosmic rays (GCR), shielding is not provided for GCR protection. The shelter concepts defined for both module configurations provide protection from solar proton events while the crew is stationed in the interplanetary vehicle.

At present, no radiation exposure limits have been established for astronauts on interplanetary class missions. Therefore, S.S. Freedom proposed guidelines are assumed for this study (Ref. 1). According to the ionizing radiation exposure limits for S.S. Freedom astronauts, the dose equivalent to the blood-forming organs for an exposure interval of 30 days equals 25 rem. This 30-day dose is the most stringent and appropriate for limiting overall risk to the Mars-mission crew from solar flares.

Note that the storm shelter consumables are accounted in the consumables calculations. Ample flare caution and warning devices are included in the mission modules with support from solar activity monitoring equipment and an external solar observing telescope.

Cylindrical Modules

The shelter specified for the cylindrical modules configuration is contained in the logistics module, situated between the habitation modules with pressurized access from either module. Rather than configuring a singular protective structure of water or aluminum, this design utilizes the stowed consumables as the radiation shielding medium. The logistics module, with a length of 5.3 meters and a diameter of 4.2 meters, has a total volume of 73.4 m^3. Stowage racks 0.9 m deep are placed along the walls, above the ceiling and below the floor (see Section 4.2 for details). The free space available in the outfitted logistics module equals 44 m^3, thus providing 8.8 m^3/person. With the minimal volume per person estimated at 1.2 m^3/person in a gravity environment (Ref. 2), the logistics module offers an extremely spacious geometry.

Reference 3 advocates a similar concept of using a pantry as a radiation shelter. A wall thickness of 35 gm/cm^2 is recommended for the pantry walls, equivalent to 0.7 m of stowed consumables and/or trash. With the rack depth measured at 0.9 m, a 77% concentration of goods stowed in the racks is feasible. These specifications would allow a dose equivalent well under the 25 rem limit, approximately 17 rem for the February 1956 solar flare event. The cylindrical modules configuration also contributes to radiation protection with the modules' bulk mass on either side of the logistics module and the airlock at one end. More dense consumables could be positioned along less-protected sides to provide the optimum shielding.

Mass estimates for the logistics module structures and support appear in Section 4.2 and the mass allocation for the mission consumables and trash are itemized in Section 3.2. No additional mass is designated particularly for radiation protection.

Disk Module

The disk module's expansive volume is sufficient to contain the radiation storm shelter. Located on the second level, the shelter is situated in the center of the floor acting as the passageway between levels one and three. The interior diameter measures 2.7 meters and the height 2.4 meters, thus allowing 13.7 m^3 or 2.7 m^3/person. The shelter contains five deployable bunk beds, soft stowage for consumable supplies, and tankage for a ten-day supply of water and emergency air.

The shelter is constructed as a thin-walled cylindrical bladder containing water (shell double-wall thickness = 2.5 mm, Al 2219-T87). Water shielding thickness, as derived in reference 4, is defined as 20 gm/cm^2, or 20 cm which affords 25 rem to the blood-forming organs during a 30-day exposure interval. These specifications are referenced to the February 1956 solar flare event using a conservative geometric model. Water is selected as the radiation protection medium to reduce shield mass in addition to providing stowage for mission water consumables and back-up supply. The cylindrical wall structure would constantly contain water, while the hatches would remain empty for easy closing operations. In the advent of a radiation emergency the hatches would be pumped full once the shelter had been sealed.

The following table itemizes the mass requirements for the disk module shelter including the water and the aluminum shell.

Table 4.1.1 Disk Module Shelter Mass Requirements

	Water, kg	Aluminum, kg	
Cylindrical wall	2110.0	431.7	
Two 1.1 m square hatches	484.0	35.5	
Remaining floor/ceiling area	1806.0	--	(Part of interiors)
Total Shelter Mass, kg	4,867		

References

1. "Space Station Program Definition and Requirements," Space Station Program Office, Washington, D.C., SSP 30000, Section 3, Rev. E, 1988.

2. Gill, Bill, et al: "Lunar Storm Shelter Conceptual Design," Eagle Engineering, Inc., Report No. 88-189, NASA Contract NAS9-17878, May, 1988.

3. "Transportation IA, FY89 Case Studies, Cycle 2, WGW #3," Martin Marietta Presentation, April 24, 1989, page 308.

4. Townsend, L.W., et al, "Large Solar Flare Radiation Shielding Requirements for Manned Interplanetary Missions," Journal of Spacecraft and Rockets, Volume 26, Number 2, page 126.

4.2 Logistics Module

The three year duration of the Mars interplanetary vehicle requires a substantial supply of mission consumables and system spares and replaceable units. The volume of such an extensive supply warrants the need for a logistics module, separate from the pressurized habitation modules.

The cylindrical modules configuration employs a logistics module derived directly from the S.S. Freedom definition. The cylindrical modules do not contain adequate stowage volume to house the complete supply of dry consumables. The under-floor volume is devoted to life support system equipment and consumables tankage. Stowage volume is located in the quarters, the galley, personal hygiene area, and partially in the ceiling. The volume of dry consumables for the entire mission equals approximately 36 m^3, 30% of which is stored inside the habitation modules.

The logistics module provides a pressurized stowage facility with a total volume of 73.43 m^3, with a diameter of 4.2 m and length of 5.3 m (Ref. 1). The logistics module for the Mars mission is outfitted with 16 racks sized for the S.S. Freedom (1.89h x 1.05w x 0.89 d). Five racks are positioned along the floor and the ceiling, two along each horizontal side, and one on each module end. The 16 racks occupy a volume of 29 m^3 and provide a usable volume of 25 m^3, an adequate amount of space for the 70% dry consumables. The 44 m^3 of free space in the logistics module provide the crew of five with storm shelter capability as well as adequate space for mobility and passage between the two modules.

The disk module configuration uses a smaller logistics module to store approximately 50% of the mission dry consumables. Most of the stowage requirements are satisfied by the disk module's expansive volume. The second level of the module is outfitted with extensive stowage racks providing about 20 m^3 of volume (6 racks at 2.3h x 0.7w x 1.1d and 3 racks at 2.3h x 1.1w x 1.1d). The loft and bottom hull of the disk module are used for life support equipment, consumables, and spares.

The logistics module for the disk configuration provides a pressurized stowage facility with a total volume of 36.6 m^3, with a diameter of 3.6 m and length of 3.6 m. This logistics module is outfitted with 12 S.S. Freedom-sized racks: 3 along the ceiling, floor and one side, 2 along the side with hatch, and 1 at the end of the module. A total of 19 m^3 of stowage volume exists in this configuration, or 53% of the mission dry consumables. The free space enclosed in the smaller logistics module allows for crew access to the airlock, but is not required to hold all five crew members in case of an emergency.

Logistics Module Specifications

	Total Volume (m^3)	Stowed Volume (m^3)	Diameter (m)	Length (m)
For Cylindrical Config	73.4	29	4.2	5.3
For Disk Config	36.6	19	3.2	3.6

The power required to maintain either logistics module is estimated at 1.5 kWe (ref. 1).

Table 4.2.1 provides a brief estimate of the logistics module mass breakdown. The values for the cylindrical modules' stowage facility are derived from reference 2. The mass estimates for the disk configuration logistics module are calculated at 40% of the referenced values. The percentage corresponds to the volume comparison between the two logistics modules.

References

1. Boeing Proposal, Space Station Work Package 1, Vol II, Technical Part 1, Systems Engineering and Integration, July 21, 1987, Submitted to MSFC.

2. NASA, Space Station Program System Engineering and Integration, "Weights Data Book - Revision 2.0," June, 1988.

Table 4.2.1 Logistics Module Mass Requirements

	Cyl Mass (kg)	Disk Mass (kg)
Primary Structure	1768.6	707.4
Secondary Structure	945.3	378.1
Stowage Racks (cyl:16, disk:12)	777.9	583.4
Mechanisms	306.6	122.6
Life Support System	576.5	230.6
Thermal Control System	290.3	116.1
Electrical Power	255.8	102.3
Total	4921.0	2240.5

4.3 Airlock

An airlock is included in the Mars interplanetary vehicle design to allow EVA access to external systems and structures on the manned platform. The airlock provides depressurization, egress, ingress, and repressurization for two EVA crew members. The configuration selected for the Mars vehicle corresponds to the current design selection for S.S. Freedom. This configuration is referred to as the "in-line." The in-line design consists of two structural sections - the equipment lock and the crew lock. The equipment lock is outfitted for donning and doffing of two EMU's, stowage of EVA equipment, and maintenance of EMU's. The crew lock provides additional volume for final crew repressurization.

The operation of the in-line airlock is as follows:

- Crew enters equipment lock at original pressure of 14.7 psia,
- Two crew members don EMU's and close module hatches,
- Pressure is equalized between equipment lock and crew lock (originally at 0.5 psia),
- Equalized pressure becomes 10.7 psia,
- Crew enters crew lock closing connecting hatch to equipment lock,
- Air is pumped from crew lock to equipment lock to achieve pressure of 0.5 psia,
- Exterior hatch is opened, allowing crew to exit and venting 0.5 psia to space.

The in-line airlock selected for the Mars vehicle has standard rather than hyperbaric capabilities. The mass estimates for the airlock include structural mass (equipment and crew locks) at **2218.6 kg** and outfitting equipment mass at **1875.2 kg**, for a total of **4093.8 kg** (ref. 1). The volume and dimensions for the equipment and crew locks are summarized in the following table (ref. 1). Note that 33% of the equipment lock volume is dedicated for donning EMU's.

Airlock Specifications

	Volume (m^3)	Diameter (m)	Length (m)
Equipment Lock	19.7	2.9	3.0
Crew Lock	5.3	1.8	2.1
Total	25.0		

The air requirements for airlock usage are determined in the mission consumables section. Assumptions include 1 EVA every two months, a contingency supply worth 4 repressurizations, and leakage equivalent to 1 crew lock every 2 months.

The power requirements for maintaining airlock functions are derived from reference 2. The total power required, duty cycle, and average power usage for each piece of airlock equipment are itemized in Table 4.3.1.

Reference

1. Wong, Willson, McDonnell Douglas, "Airlock Configuration Trade Study," Report to the Airlock Review Board, October 25, 1988.

2. NASA, Space Station Program System Engineering and Integration, "Power Data Book,"August, 1988.

Table 4.3.1 Airlock Equipment Power Summary

	Power (w)	Duty Cycle	Ave Power (w)
Ventilation Fan	22	1	22.00
Air Temp & Humidity Control	431	1	431.00
Temp. Controller/Valve	19	1	19.00
O2/N2 Press. Control	5	1	5.00
Fire Detection/Supp.	19	1	19.00
H2O Storage & Distr.	3	1	3.00
Airlock Instruments	2	1	2.00
Wireless Battery Charger	7	1	7.00
Audio - Speaker Phone	18	1	18.00
Video - Camera	25	0.05	1.25
Video - Camera	25	0	0.00
Video - Pan-tilt unit	20	0.05	1.00
DMS/Airlock MDM Control	80	0.02	1.60
Lights - General (4)	19	0.5	38.00
Lights - Emergency (1)	30	1	30.00
Airlock Depress/Repress	2000	0.001	2.00
Crew Lock Repress/Depress	5	0.001	0.01
Equipment Lock Ventilation	300	0.001	0.30
Equipment Lock Repress.	50	0.001	0.05
EMU O2 Regenerator Air	100	0.001	0.10
Checkout & Servicing	150	0.01	1.50
EMU Battery Charger	167	0.01	1.67
EMU CO2 Regenerator	712	0.001	0.71
Thermal Sink Wax Regen.	180	0.001	0.18
EMU Drying Unit	100	0.001	0.10
EMU IV Umbilical Ops.	200	0.001	0.20
Tool Battery Recharger	168	0.01	1.68
Portable Contamination	200	0.001	0.20
Tool Box Light (2)	20	0.04	0.80
Hatch Light (2)	20	0.04	0.80
Total	5097		608.1

4.4 Power and External Thermal Control Systems

The power requirements for the two Mars interplanetary mission module configurations are listed in the table below. These values represent the average power load for each manned system considered. A fifteen percent contingency factor is included to account for underestimates or overlooked power needs. Certain exterior mechanisms (tether reel motors, solar telescope, lighting, communication antennas) located on the platform and requiring power were not considered for the definition of the power and thermal control systems.

Module Power Requirements

	kWe
Module Interior	6.7
Life Support System	4.1
Thermal Control System	1.8
Airlock	0.6
Logistics Module	1.5
Communications (max)	0.4
Contingency @ 15%	2.3
Total	17.4

The power requirements are met by a combination of photovoltaic solar arrays and nickel-hydrogen rechargeable batteries. Each photovoltaic power increment contains two solar array wings and associated equipment and one battery for energy storage. The illustrations of the two module configurations depict two increments located on the "sun-viewing" side of the vehicle. Each power increment is intended to operate independently.

The active thermal control system for this analysis is defined as the heat rejection element. The radiator is configured as two double-sided, emitting panels. The amount of heat to be rejected is estimated as equivalent to the amount of power required by the manned systems. A more complete sizing of the thermal control system would include an external thermal bus, a pump mechanism, and a condenser.

A summary of the sizing results for the power and thermal control systems appears below.

Summary

	Unit	Total (2)
Photovoltaic System		
Area, m^2	293.7	587.4
Mass, kg	559.8	1119.5
Energy Storage System		
Volume, m^3	1.2	2.5
Mass, kg	1247.0	2493.9
Thermal Control System		
Area, m^2	16.2	32.4
Mass, kg	324.3	648.6

The following sections outline the equations for sizing the photovoltaic power system, energy storage system, and thermal control system as well as determining the power load.

Photovoltaic (PV) Power System (Ref.7-10)

The Interplanetary Mission Module's (IMM) PV solar arrays are sized by conditions in Mars orbit because: a) Lowest solar insolation of mission, b) Occultation periods while in Mars orbit requires PV power to recharge energy storage systems, and c) Period of greatest crew activity and presumably highest power usage will be in Mars vicinity.

Mass of all PV equipment including array blankets, array structures, power conversion (PC) and PC thermal control equipment, and electrical bus to mission module interface is estimated from:

$$\text{PV Mass (kg)} = P_L\ (\text{kWe})/f_M \qquad \text{Eqn.1}$$

where,

- P_L = Required PV power load which is the sum of the power for mission module equipment (P_{MM}) and the power to recharge the energy storage system (P_{ES}).
- f_M = Specific power factor for PV system = 0.0255 kWe/kg (Ref.7,10) which is the ratio of gross power (before energy storage) to system mass which includes the solar array blankets, array structures, power conversion and thermal control equipment, and electrical bus to mission module interface, but excludes the energy storage system mass.

The dimensions of the PV arrays can be determined from the area of the arrays, A (m^2), which is calculated by:

$$A = P_L * 1000/\{F_S * n * (1 - f_d) * \cos\Theta * [1 - (T-28)*f_t] * f_p\} \qquad \text{Eqn.2}$$

where,

- F_S = Mean solar insolation @ 1.52 AU = 590 W/m^2 (Ref.1)
- n = Cell efficiency at 28°C = 0.115 (Si cells = 11.5% conversion efficiency now, 13.3% with development, GaAs = 20.5% with development required)
- f_d = Degradation factor = 0.1 (assume 10% over 3 years)
- Θ = Sun angle = 6.5° from normal
- T = Operating temperature = 50°C
- f_t = 0.005 = 0.5% efficiency loss per °C for silicon cells (0.0025 for GaAs cells)
- f_p = Packing factor = 0.9 (90% solar cell area)

Power Load

The PV solar arrays are sized to provide the required user power load, P_L (kWe), which is the sum of the power for mission module equipment (P_{MM}) and the power to recharge the energy storage system (P_{ES}). P_{ES} is determined based on Mars orbit conditions, specifically the fraction of each orbital period the IMM spends in the occulted region at Mars, f_C, and the efficiency of recharging the energy storage system, n_{ES}, as given by:

$$P_{ES} = P_{MM} * f_C/[(1-f_C) * n_{ES}] \qquad \text{Eqn.3}$$

where f_C is found from the orbital altitude above the mean martian surface, h (km), (assuming a circular orbit) and the radius of Mars, r_M = 3397.2 km (equatorial), by:

$$f_C = \{\arc\sin [r_M / (r_M + h)]\}/\pi \qquad \text{Eqn.4}$$

For a 500 km orbit, f_C = 0.337, or 41.5 min in the occulted region for each 123.1 min orbit. Orbital period, P (min), is found by:

$$P = 2\pi/60 * (r_M + h)^{1.5} * \mu_M^{-0.5} \qquad \text{Eqn.5}$$

where for Mars, $\mu_M = 42{,}828.32\ km^3/s^2$.

Required electrical power to be generated by the solar arrays, P_L (kWe), becomes:

$$P_L = P_{MM} * \{1 + f_C/[(1-f_C) * n_{ES}]\} \qquad \text{Eqn.6}$$

<u>Energy Storage System</u>

The following information is derived from several recent Eagle studies (2-6) which described certain aspects of energy storage systems. These references should be consulted for more detailed information on these system.

Nickel-Hydrogen (NiH_2). S.S. Freedom will use NiH_2 rechargeable batteries. At 80% depth-of-discharge, this battery is rated at 35 Wh/kg (2, p.11). Energy storage mass, M_{ES}, for this option becomes:

$$E_{ES}\ (kWh) = P_{ES} * P/60 * f_C \qquad \text{Eqn.7a}$$
$$M_{ES}\ (kg) = E_{ES} * 1000/ef \qquad \text{Eqn.7b}$$

where,

E_{ES} is the energy storage requirements in kilowatt-hours (kWh) including inefficiencies
P_{ES} is found from Eqn.3
P is found from Eqn.5
f_C is found from Eqn.4
ef = specific energy factor = 35 Wh/kg

Cycle efficiency is about 80%; thus, the energy storage efficiency for Eqn.3 and 6, n_{ES} = 0.8.

An alternative approach is to define a storage time period, say 4 hours, which would be the time the IMM's solar arrays are retracted during de-spin operations. In this case, the PV array is still sized based on the occultation period method given in Eqns 3-6 (because the energy storage system is fully-charged prior to despin). However, the energy storage requirements must be increased, i.e.:

$$E_{ES}\,(kWh) = P_{MM} * \tau/n_{ES} \qquad \text{Eqn.7c}$$

where,

τ = Energy storage time period, 4 hrs
n_{ES} = Energy storage cycle efficiency = 0.8 for NiH_2 batteries

Battery mass is found by Eqn.7b. Battery volume is scaled based on 0.0285 m^3/kWh, i.e.:

$$V_{ES} = E_{ES} * f_V \qquad \text{Eqn.7d}$$

where,

V_{ES} = Volume of NiH_2 batteries (m^3)
f_V = 0.0285 m^3/kWh (derived from data in Ref.7)

Thermal Control System

Waste heat from the Mars mission modules is rejected by a radiator positioned perpendicular to the solar arrays. Heat rejection from both sides of the radiator is assumed. However, the radiator area calculation also includes an efficiency factor to compensate for the thermal radiosity emitted by other surfaces which are in view of the radiator:

$$A = Q/(2 * n * \sigma * \varepsilon * T^4) \qquad \text{Eqn.8}$$

where,

A = Projected radiator area (m^2)
Q = Heat rejection load (kWe)
n = Efficiency of heat rejection = 0.75 (assumed good view of space)
σ = Stefan-Boltzmann constant = 5.67×10^{-11} kW/m^2 - K^4
ε = Radiator surface emissivity = 0.8
T = Rejection temperature = 298°K

The mass of all external thermal control systems, M_{TCS} (kg), is estimated based on a 20 kg/m^2 scaling factor:

$$M_{TCS} = 20 * A$$

References

1. Smith, R.E. and West, G.S.: "Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development, 1982 Revision (Volume 1)," NASA Technical Memorandum 82478, 1983.

2. McBryar, H. et al.: "Conceptual Design of a Lunar Base Solar Power Plant," Eagle Engineering, Inc., Report No. 88-199, NASA Contract NAS9-17878, August 14, 1988.

3. Davidson, W.L. et al.: "Lunar Surface Transportation Systems Conceptual Design," Eagle Engineering, Inc., Report No. 88-188, NASA Contract NAS9-17878, July 7, 1988.

4. Stump, W.R. et al.: "Lunar Lander Conceptual Design," Eagle Engineering, Inc., Report No. 88-181, NASA Contract NAS9-17878, March 30, 1988.

5. Christiansen, E.L. et al.: "Conceptual Design of a Lunar Oxygen Pilot Plant," Eagle Engineering, Inc., Report No. 88-182, NASA Contract NAS9-17878, July 1, 1988.

6. Stump, W.R., Christiansen, E.L. et al.: "Technology Required for New Space Initiatives," Eagle Engineering, Inc., Report No. 87-166, NASA Contract NAS-17900, November 13, 1987.

7. Reppucci, G.: "Spacecraft Power Systems," Federal Systems Division, TRW Space & Technology Group, Presentation at the AIAA Fundamentals of Spacecraft Design Lecture Series, Houston, Texas, April 15, 1986.

8. Nored, D.L. and Halinan, G.J.: "Electrical Power System for the U.S. Space Station," NASA LaRC, A87-16138, IAF Paper No.86-37, 37th International Astronautical Congress, Innsbruck, Austria, October 4-11, 1986.

9. Brandhorst, H.W., Juhasz, A.J., and Jones, B.I.: "Alternative Power Generation Concepts for Space," N87-28961, NASA LaRC, Proceedings of 5th European Symposium on Photovoltaic Generators in Space, Scheveningen, The Netherlands, September 30 - October 2, 1986, ESA SP-267, November 1986.

10. Stump, W.R. et al.: "Transportation Node Space Station Conceptual Design," Eagle Engineering, Inc., Report No. 88-207, NASA Contract NAS9-17878, September 30, 1988.

4.5 Attitude Control System

The cylindrical and disk module configurations require an Attitude Control System (ACS) to stabilize the platform structure. The ACS is not sized or intended to provide propulsion for the artificial gravity system. The ACS consists of four thruster assemblies placed at each corner of the platform. Two tanks each for the fuel and oxidizer are included to service the assemblies (four tanks total). The ACS fuel selected for this mission is storable Monomethylhydrazine ($CH_3N_2H_3$), and the oxidizer, also storable, is Nitrogen Tetroxide (N_2O_4). The physical properties of these propellants are listed below (Ref. 1):

	N_2O_4	$CH_3N_2H_3$
Molecular Weight	92.02	46.072
Density (mt/m^3)	1.448	0.870
@ temp. (°C)	20	25
Normal Boiling Point (°C)	21.3	87.7
Melting Point (°C)	-9.3	-52.4
Heat of Vaporization @ BP (W-hr/kg)	88.9	
Heat of Fusion @ MP (W-hr/kg)	69.98	

The ACS is sized for the module platform portion of the vehicle only, using an approximate mass of 65,000 kg. The total mass requirements for the ACS include the storage tanks, the propellant and the four thruster assemblies. The tank masses include the shell, insulation, shielding and an additional 10% for miscellaneous structure.

Table 4.5.1 Attitude Control System Mass Requirements

	Mass (kg)
Propellant:	
$CH_3N_2H_3$	642.7
N_2O_4	1347.1
Tanks:	
$CH_3N_2H_3$ (2)	54.3
N_2O_4 (2)	63.1
Thruster Assemblies (4):	
(257.6 kg/mod) (Ref. 2)	1030.6
Total	3,137.8

Table 4.5.2 Propellant Tank Specifications

Tank Shape	spherical
Inside Tank Diameter ($CH_3N_2H_3$), m	0.90
Inside Tank Diameter, (N_2O_4), m	0.98
Tank Wall Thickness, mm	0.64
Insulation Thickness, cm	1.00
Shielding Thickness, cm	0.23

Calculations for Sizing Tanks

Assumptions:

Fuel: Monomethylhydrazine ($CH_3N_2H_3$)
Oxidizer: Nitrogen Tetroxide (N_2O_4)

Ratio (Oxidizer:Fuel): 2.1:1 (Ref. 3)

Mass Estimate of Vehicle (Module Platform Portion) = 65,000 kg
ACS = 3% of Spacecraft Mass = 1,950 kg
Propellant = ACS Mass = 1,950 kg

Tank Equations:

First calculate the inside diameter of the tanks:

$$D_i = 2 * [3 * M * (1+f_u)/(\rho * N *4\pi)]^{1/3}$$

where,

D_i = Inside tank diameter (m)
M = Mass of stored material (kg)
Mass is Usable Propellant/Recovery Factor, where recovery factor is 98%.
f_u = Ullage factor, fraction of tank volume not filled @ 100% capacity.
For liquid reactant tanks, $f_u = 0.05$.
ρ = Material density (kg/m^3)
For $CH_3N_2H_3$, ρ=870.
For N_2O_4, ρ=1448.
N = Number of Tanks

Assuming the wall thickness (t) is 0.635 mm, and the pressure vessel material is Al 2219-T87, calculate the tank shell mass:

$$M_u = \rho * 4\pi/3 * [(D_i/2 + t/1000)^3 - (D_i/2)^3]$$

where,

M_u = Tank shell mass (kg)
ρ = Shell density = 2824 kg/m^3 for Aluminum

Calculate the mass of the surrounding thermal insulation, using multilayer insulation (MLI) for passive thermal protection:

$$M_i = \rho_i * 4\pi/3 * [(D_i/2 + t/1000 + t_i/100)^3 - (D_i/2 + t/1000)^3]$$

where,

M_i = Mass of MLI (kg)
ρ_i = Insulation density = 120 kg/m^3 for MLI
t_i = Insulation thickness = 1 cm

Due to exposure on the platform during the interplanetary mission, additional tank shielding is included. The mass of the required shielding is calculated as:

$$M_s = \rho_s * 4\pi/3 * [(D_i/2 + t/1000 + t_i/100 + t_s/100)^3 - (D_i/2 + t/1000 + t_i/100)^3]$$

where,

M_s = Mass of Al 6061-T6 shielding (kg)
ρ_s = Shielding density = 2712.6 kg/m^3
t_s = Shielding thickness = 0.23 cm

Adding 10% for the miscellaneous structure mass (M_{ms}), the mass of each tank equals:

$$M_t = M_u + M_i + M_s + M_{ms}$$

References

1. Transportation Node Space Station Conceptual Design, NAS9-17878, Prepared for NASA-JSC by Eagle Engineering, Houston, Texas, September 1988

2. NASA, Space Station Program System Engineering and Integration, "Weights Data Book - Revision 2.0," June, 1988.

3. Rocketdyne Division Brochure, "A Past to Build a Future On," Rockwell International, Canoga Park, California, 1988.

4.6 Artificial Gravity Equipment

The artificial gravity method employed by both the cylindrical modules and disk module MPV configurations during the conjunction class mission is a tether system. Two tethers are used to attach the MPV module platform to the propellant/aerobrake structure, such that the two masses revolve about a common center of mass (C.M.). The propellant/aerobrake structure acts as a counter weight to the manned portion of the vehicle in order to produce the desired gravity level. It is assumed that a 3/8 g environment will be provided for the astronauts during the outward-bound leg of the mission, and a 1 g environment during the return trip. This variation will allow the astronauts to acclimate to the gravity they will experience at their destination. To avoid most of the physiological side effects of rotating the spacecraft, a relatively slow rotation rate of 2 revolutions per minute is assumed. At that rate a large radius is necessary in order to obtain 1 g.

Phillystran® Working Rope PSWR-260 is used to size the tether system. This 5 cm diameter rope has a breaking strength of 120,000 kg, which affords a factor of safety of approximately 3.7 (Ref. 1). The rope weighs only 1.6 kg/m because it is constructed of Kevlar® aramid fiber. The Kevlar® aramid construction also provides the cable with low stretch characteristics normally associated with steel cables (Ref. 2).

Table 4.6.1 Tether System Mass Requirements

	Mass (kg)
Tethers (2)	1299
Spools (2)	52
Support Structure (Fasteners,Spoked Flanges,etc.) (10% of mass)	135
Motors (10% of mass)	135
Total for 2 components	1,621

Calculations for Sizing Tether

Assumptions:

Size tethers for maximum conditions with respect to gravity level and mass.
Maximum artificial gravity level = 1 g

ω = 2 rev/min, or
$\omega = 2 \text{ rev/min} \times 2\pi \text{ rad/rev} \times 1 \text{ min/60 sec} = \pi/15 \text{ rad/s}$

m_H = estimated mass of module platform = 65,000 kg
m_T = estimated mass of propellant/aerobrake platform = 100,000 kg

r_H = radius of module platform to C.M.
r_T is the radius of propellant/aerobrake platform to C.M.

Calculate r_H:

$r_H\omega^2 = 1\ g = 9.81\ m/s^2$

$r_H = 9.81\ m/s^2 / (\pi/15)^2\ rad/s^2 = 223.6\ m$

Summing moments about the C.M.:

$r_H m_H = r_T m_T$

Calculate r_T:

$r_T = r_H(m_H/m_T) = 223.6\ m \times (65{,}000\ kg/100{,}000\ kg) = 145.3\ m$

Total tether length (l) is the sum of r_H and r_T plus 10%:

$l = r_H + r_T + .1(r_H + r_T) = 406\ m$

Calculations for Sizing Reel

r_s = radius of spool
r_r = radius of spool & wrapped tether
l = length of tether
t = tether diameter
w = length of spool
i = number of layers of tether
t_s = thickness of spool

For one revolution the total tether length reeled is

$$l_1 = 2\pi r_s$$

The length of tether reeled for first layer is

$$l_2 = 2\pi r_s(w/t)$$

For i number of wraps

$$l \approx 2\pi(w/t)[r_s + (r_s + t) + (r_s + 2t) + ...+(r_s + ti)]$$

$$l \approx 2\pi(w/t)[ir_s + (i+1)(i/2)t]$$

$$l \approx 2\pi(w/t)i[r_s + (i+1)(t/2)]$$

The radius of the spool plus the wrapped tether is

$$r_r = r_s + ti$$

Assumptions

$r_r = 0.9$ m
$r_s = 0.25$ m
$t = 0.05$ m
$l = 406$ m
$t_S = 0.0125$ m

Solving for i yields

$$i = (r_r - r_s)/t$$
$$i = 13$$

Solving for w yields

$$w = lt/\{2\pi i[r_s + (i+1)(t/2)]\}$$
$$w = 0.41 \text{ m}$$

Adding a 20% contingency to the width gives (to account for non-uniform wrapping)

$$w = 0.5 \text{ m}$$

The volume of the structural volume of the spool is

$$V_S = \pi[r_S^2 - (r_S - t_S)^2]w$$
$$V_S = 0.0096 \text{ m}^3$$

The mass of the spool is

$$M_S = \rho_{Al} V_S$$
$$M_S = 26 \text{ kg}$$

The radius of each end plate is equal to the radius of the spool plus the wrapped tether plus 2 times the thickness of the cable.

$$r_E = r_R + 2t$$
$$r_E = 1 \text{ m}$$

The mass of the tether is calculated by:

$$M_C = \beta l$$
$$M_C = 650 \text{ kg}$$

where,

β = mass per unit length, 1.6 kg/m (Ref. 1)
l = tether length, 406 m

References

1. Phillystran® Marketing Brochure, "Industrial Products," United Ropeworks (U.S.A.) Inc., Montgomeryville, Pennsylvania, 1989.
2. Dupont Marketing Brochure, "Cable with Dupont Kevlar Aids Satellite Deployment," E.I. duPont de Nemours & Co., Wilmington, Delaware, 1989.

4.7 Communications

The specifications for the Mars interplanetary communication antennas are given below. The design pertains to both the cylindrical module configuration and the disk module configuration. Two antennas for each vehicle are specified for redundancy, as well as the capability of communicating with crewmembers on the Martian surface and with Earth simultaneously. The bit rates, power, and dish diameter are assumed to be the same as those given in Reference 1 and summarized below. The specified data rates will permit TV transmission and data transference to Earth. The maximum transmitting range cited in Ref. 1 is 1.8 AU. If the range of this mission exceeds 1.8 AU the dish diameter and/or the transmitting power will need to be increased. It is assumed that the parabolic dishes have a thickness of 5 mm and are machined of aluminum 6061-T6. Aluminum was chosen for micrometeorite and debris protection.

Communications Requirements Summarized (extracted from Ref. 1)

1. Conditions and Assumptions

   - Range from 0 to 1.8 AU
   - 5 Mbps nominal, 15 Mbps at 10% duty cycle (daily), 40 Mbps special event and emergency
   - One 34 m Deep Space Network (DSN) antenna, Ka band reception
   - 1 dB pointing loss

2. Results

   - Single 5 m diameter dish antenna
   - 1 dB pointing requires 0.035° aiming accuracy
   - For 10 Mbps, 145 W radiated power and for 5 Mbps, 72 W radiated power
   - For 15 Mbps power would be ramped to 215 W for DSN 34 m antenna or use the DSN 70 m antenna.
   - 40 Mbps special/emergency is achieved with one 70 m or quad 34 m, with 145 W radiated

Table 4.7.1 Communication Specifications

	Mass (kg)	Diameter (m)	Max Power (W)
Communication Dish	139	5	215
Supporting structure (boom, motors, etc.) (20% of dish mass)	28	-	-
Total (2 antennas)	334	-	430

Reference

1. "Transportation IA, FY89 Case Studies, Cycle 2, WGW #3", Martin Marietta Presentation, April 24, 1989, p 38.

4.8 Support Structure

The cylindrical and disk module configurations require support structure to connect external systems (solar arrays, tether reels, etc.) and the pressurized volumes. The additional structure prevents distributing the loads through the modules and also provides stability during tether operations. The support structure for the cylindrical module configuration consists of a lattice of truss-work and a 1 meter walkway encircling the structures for crew mobility during potential EVA. The frame of truss-work is assumed sufficient to support the platform. Further assumptions include the diameter of the truss members at 4 cm and each "box" of truss at 1m x 1m x 1m. The truss members are manufactured of solid aluminum 6061-T6 rather than as hollow tubes in order to support the structural loads in an artificial gravity environment. Figure 4.8.1 illustrates the truss arrangement with respect to the cylindrical module configuration. The mounted walkway is an aluminum grating (approximately 60% solid) placed over the truss frame.

The support structure for the disk module configuration does not require as large a truss frame as the cylindrical configuration; however, a ring around the module provides necessary load-bearing support. The disk module ring facilitates structural connections for the tether reels, telescope, ECCV, and logistics module/airlock. The ring is assumed to be constructed with a box beam cross section. As before, the diameter of the truss members is 4 cm and each truss "box" measures 1m x 1m x 1m. An aluminum grating is also used as a walkway on this configuration. Figure 4.8.2 shows the layout of the truss with respect to the pressurized volumes for the disk module configuration.

Table 4.8.1 Support Structure Mass Requirements

	Cyl. Modules Mass (kg)	Disk Module Mass (kg)
Truss frame	4116	2638
Grating	456	407
Ring	-	730
Miscellaneous (fasteners, etc.) (10% of calc. mass)	457	378
Total Mass	5029	4153

Calculations

The mass of the truss members is calculated by:

$$M_T = \rho_{Al}\pi r^2 l$$

where,

ρ_{Al} = Density of Al 6061-T6 = 2716 kg/m^3
r = Radius of a truss member = .02 m
l = Total length of all truss members = 1206 m (for cylindrical configuration)
l = Total length of all truss members = 773 m (for disk configuration)

The mass of the walkway grating is calculated by:

$$M_G = \rho_{Al}*0.6[(l*w)-(l-2d)*(w-2d)]*t$$

where,

ρ_{Al} = Density of Al 6061-T6 = 2716 kg/m^3
l = Length of the walkway = 15 m for cyl. config, 14 m for disk config
w = Width of the walkway = 15 m for cyl. config, 13 m for disk config
d = Depth of walkway = 1 m for both configurations
t = Thickness of grating = 0.005 m for both configurations

The mass of the ring is calculated by:

$$M_R = \rho_{Al}2\pi r\{[w*h] - [(w-t)*(h-t)]\}$$

where,

ρ_{Al} = Density of Al 6061-T6 = 2716 kg/m^3
r = Radius of disk = 4.2 m
w = Width of ring, assumed to be 0.15 m (6 in)
h = Height of ring, assumed to be 2.4 m (8 ft)
t = Thickness of beam walls, assumed to be 4 mm (.15 in)

Figure 4.8.1 Cylindrical Modules Configuration - Support Structure Layout

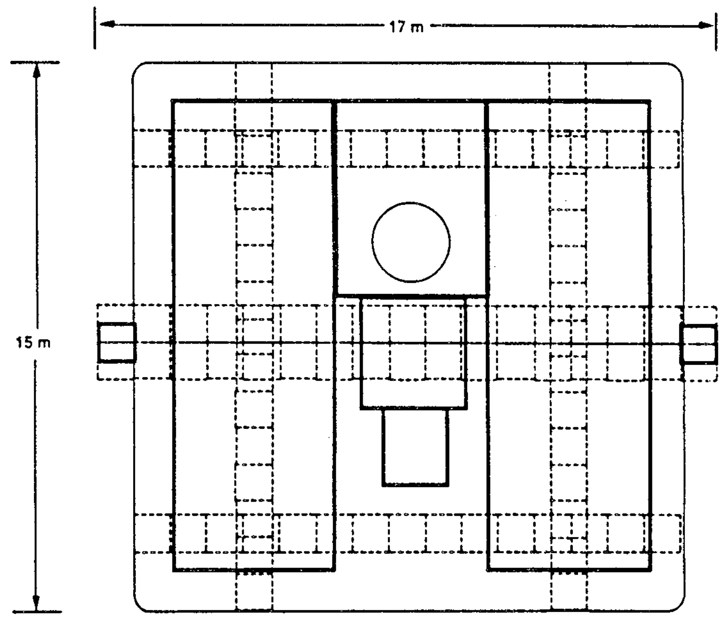
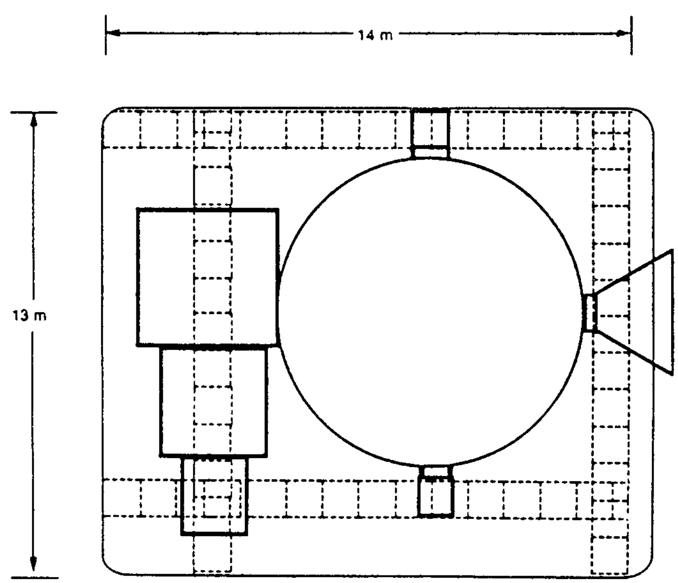


Figure 4.8.2 Disk Module Configuration - Support Structure Layout



4.9 Solar Telescope

A solar observing telescope is included on the MPV to provide the crew with solar flare observational capability and astronomical experimentation during the three year mission. The interplanetary mission crew will be concerned with their exposure to highly penetrating and damaging space radiations, particularly solar particle events. The onboard telescope would allow the crew to monitor and track solar activity and, in conjunction with observations performed on Earth, be forewarned of solar flares. As a scientific tool, the telescope would offer the astronauts an opportunity for solar research, including plasma physics and the morphology and development of active regions. The mission duration of three years offers an extensive period for performing observations of slowly varying phenomena over periods of days, months, and years.

The solar observing telescope is mounted on the exterior of the pressurized mission modules - on the central ring encircling the disk module and above the crew lock in the cylindrical modules configuration. The telescope is oriented, as are the photovoltaic arrays, in the direction of the sun during the majority of the mission. Experimental controls - pointing system, monitoring instrumentation, thermal control readouts, alert indicators - would be located inside the mission modules in the allotted lab rack space.

In order to include a mass estimate for the solar telescope, similar orbiting telescope systems were identified. The specifications for these exemplary telescopes are listed in the table below. The launch mass of each telescope appears high because it includes auxiliary systems such as solar arrays. An approximate mass of 1,000 kg is selected for the MPV telescope (similar to the Skylab instrument mass) since the support systems are supplied by the vehicle.

Table 4.9.1 Reference Orbiting Telescopes

Apollo Telescope Mount (Skylab)

Includes: Cannister assembly, 8 solar telescopes, sun sensors, and auxiliary systems.

Length	3.4 m (11 ft)
Diameter	2.1 m (7 ft)
Overall Mass	9,979 kg (22,000 lbs)
Instrument Mass	998 kg (2,200 lbs)

Reference: Belew, Leland F., and Ernst Stuhlinger, Skylab, A Guidebook, NASA, Washington, D.C.

Solar Optical Telescope

Includes: Optical telescope assembly, scientific instruments, and support systems module.

Length	13.1 m (43.5 ft)
Diameter	4.3 m (14 ft)
Launch Mass	11,000 kg (25,500 lbs)

References:
1. McRoberts, Joseph J., "Space Telescope," NASA, Division of Public Affairs, Washington, D.C.
2. "Space Telescope, A New Look in Astronomy," NASA/Lockheed, Sunnyvale, California, 1980.

Hubble Space Telescope

Length	13 m (43 ft)
Diameter	4 m (14 ft)
Launch Mass	11,431 kg (25,200 lbs)

Reference: Sam Ballard (408-743-0284) of Lockheed, Sunnyvale, CA, 1989.

5.0 Results

The conceptual design analysis performed for this study provides two alternative yet viable configurations for Mars mission modules. Both configurations adhere to the mission requirements of supporting a crew of five for three years in an artificial gravity environment with a conservative safe-haven capability.

Table 5.0.1 presents a summary comparison of the two design options with respect to module sizing, mass, and operations. With a diameter of 8.4 meters, the disk module affords a volume 1.5 times that of the two cylindrical modules, and thus a greater volume per person ratio. A similar comparison in terms of floor area indicates the three levels of the disk module offer more space than the two cylindrical module floors. The total pressurized volume listed in the table includes the crew module(s) plus the logistics module and airlock. The overall mass comparison favors the cylindrical modules design by approximately 2 metric tons. The module mass value accounts for the systems associated with the pressurized volumes including the life support system, active thermal control system, consumables, module interior, primary and secondary structure, radiation shelter, logistics module, and airlock. The support system mass involves the systems external to the pressurized modules such as the power system, energy storage system, external thermal control system, attitude control system, artificial gravity equipment, communications, support structure, and solar telescope.

Table 5.0.2 presents a more detailed comparison of the two designs with an itemization of mission systems and their parameters of interest. System areas of noticeable mass differences include structures (primary, secondary, and support), logistics module, radiation shelter, and consumables. Figure 5.0.1 illustrates the module mass distribution in terms of percentage, with a comparable spread for both configurations. The consumable supplies overshadow the distribution, with the other systems more evenly apportioned.

Both module configurations operate under equivalent power requirements due to the similarity in equipment specifications and operating duty cycles. Figure 5.0.2 shows the power usage distribution for the Mars mission systems. The operation of the module interior equipment (i.e. HMF, DMS, galley, etc.) requires the largest draw on the power supply. The life support system consumes approximately a quarter of the total power usage. A contingency of 15% is included to cover the miscellaneous needs, such as the tether motors and communication antenna boom motors.

In addition to the overall system comparison between the module designs, a volumetric analysis of the cylindrical modules and the disk module was performed. The spreadsheets containing the module specifications and volume calculations appear in Appendix A. The total volume of the two cylindrical modules equals 327 m^3 and 519 m^3 for the disk module. Table 5.0.3 details the volume occupied by functional areas and the percent distribution, while Figure 5.0.3 graphically compares the volume distributions. The functional areas are defined as work (CCC, lab and maintenance work areas, and HMF), personal (quarters and personal hygiene), social (galley, wardroom, and fitness), free space (corridors and hatch openings), standoffs, shelter, and stowage. The cylindrical modules devote the most volume to personal space; whereas, the disk module attributes the majority of the volume to social space and free space. The distributions between the designs also vary in terms of areas required for volume allocation, with the disk module providing space for the radiation shelter and stowage. Note that the ceiling/floor volumes in the cylindrical modules and the bottom/top hull volumes in the disk module are not included in this volume distribution. These unique volume compartments provide similar stowage capability for life support system equipment, loft accommodations, and aerobrake-couch restraints.

The module volume analysis extends to floor area utilization since the mission profile identifies an artificial gravity environment for a predominate part of the trip-time. Table 5.0.4 shows the floor area occupied by the functional areas and the percent distribution; Figure 5.0.4 graphically compares the floor area distributions. The floor area spread for the cylindrical modules design follows a distribution trend similar to the volume except for the deletion of the standoffs, which are not positioned along the floors. The floor area distribution for the disk module is identical to the volume distribution for that design.

Table 5.0.5 shows the volume breakdown for the two individual cylindrical modules with respect to the same functional areas listed above. The variation in personal volume is due to the uneven number of quarters, two in one module and three in the other module. The personal volume distribution in turn influences the differences in the other areas of each module, particularly in the work and social areas. Table 5.0.6 reflects a similar comparison between the two cylindrical modules in terms of floor area allocations.

The disk module is divided according to the three levels in order to determine the volume and floor area distributions, as detailed in Table 5.0.7 and Table 5.0.8. Levels 1 and 3 provide identical functional areas for safe-haven capability. Level 1 includes an additional crew quarter at the expense of extensive health maintenance facilities, while level 3, with only two crew quarters contains additional volume for medical care. Level 2 allows for a predominate amount of free space for hatch openings to access the ECCV and logistics module and for mobility around the shelter and amongst the stowage racks. The floor area distribution accounted in Table 5.0.8 shows a similar pattern for each level.

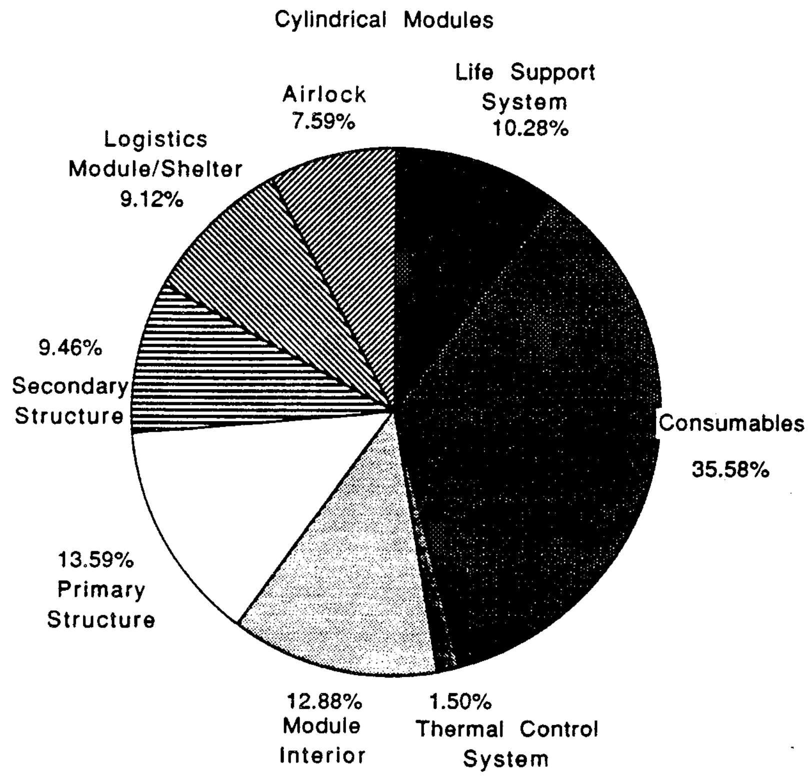
Table 5.0.1 Summary of Module Configurations Comparison

Summary Characteristics		Cylindrical Modules	Disk Modules
Mission Requirements			
	Crew Size	5	5
	Mission Duration	3	3
Module Sizing			
	Diameter, m	4.2	8.4
	Length, m	11.8	11
	Total Module Vol., m3	327	519
	Vol/person, m3/p	65	104
	Total Floor Area, m2	99	166
	Floor Area/person, m2/p	20	33.3
	Total Pressurized Vol, m3	425	581
Mass			
	Module Mass, kg	53,950	56,710
	Support System Mass, kg	15,384	14,509
	Total Configuration Mass, kg	69,334	71,219
	Total Mass/person, kg/p	13,867	14,244
Operations			
	Operating Power, kWe	17.4	17.4
	Operating Pressure, psi	14.7	14.7

Table 5.0.2 System Characteristics Comparison

		Cylindrical Modules	Disk Module
Life Support System			
	Mass, kg	5,548	5,548
	Power, kWe	4.1	4.1
	Volume, m3	18	18
Consumables			
	Mass, kg	19,194	20,075
	Volume, m3	46	46
Thermal Control System			
	Mass, kg	807	807
	Power, kWe	1.8	1.8
	Volume, m3	1	1
Module Interior			
	Mass, kg	6,948	6,735
	Power, kWe	6.7	6.7
	Volume, m3	77	75
Primary Structure			
	Mass, kg	7,333	6,262
Secondary Structure			
	Mass, kg	5,105	6,888
Radiation Shelter			
	Mass, kg	- - -	4,867
	Volume, m3	- - -	14
Logistics Module			
	Mass, kg	4,921	2,241
	Power, kWe	1.5	1.5
	Volume, m3	73	37
Airlock			
	Mass, kg	4,094	4,094
	Power, kWe	0.6	0.6
	Volume, m3	25	25
Power System			
	Mass, kg	1,120	1,120
	Area, m2	587	587
Energy Storage System			
	Mass, kg	2,494	2,494
	Volume, m3	2.5	2.5
External Thermal			
Control System	Mass, kg	649	649
	Area, m2	32	32
Attitude Control System			
	Mass, kg	3,138	3,138
Artificial G Equipment			
	Mass, kg	1,621	1,621
Communications			
	Mass, kg	334	334
	Power, kWe	0.2	0.2
Support Structure			
	Mass, kg	5,029	4,153
Solar Telescope			
	Mass, kg	1,000	1,000

Figure 5.0.1 Module Mass Distribution



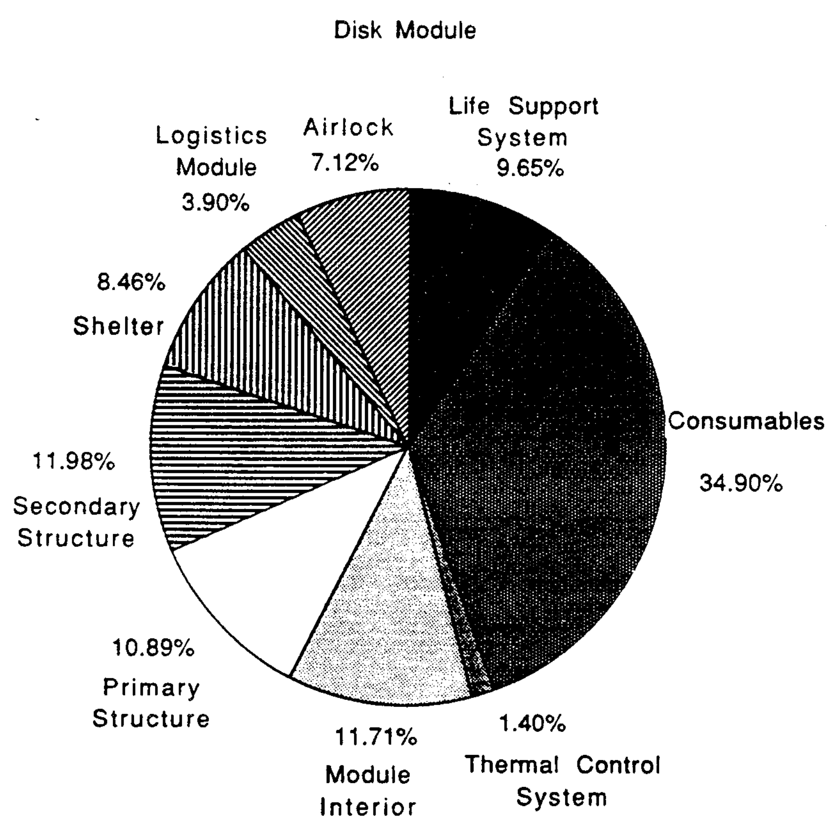


Figure 5.0.2 Average Power Usage Distribution

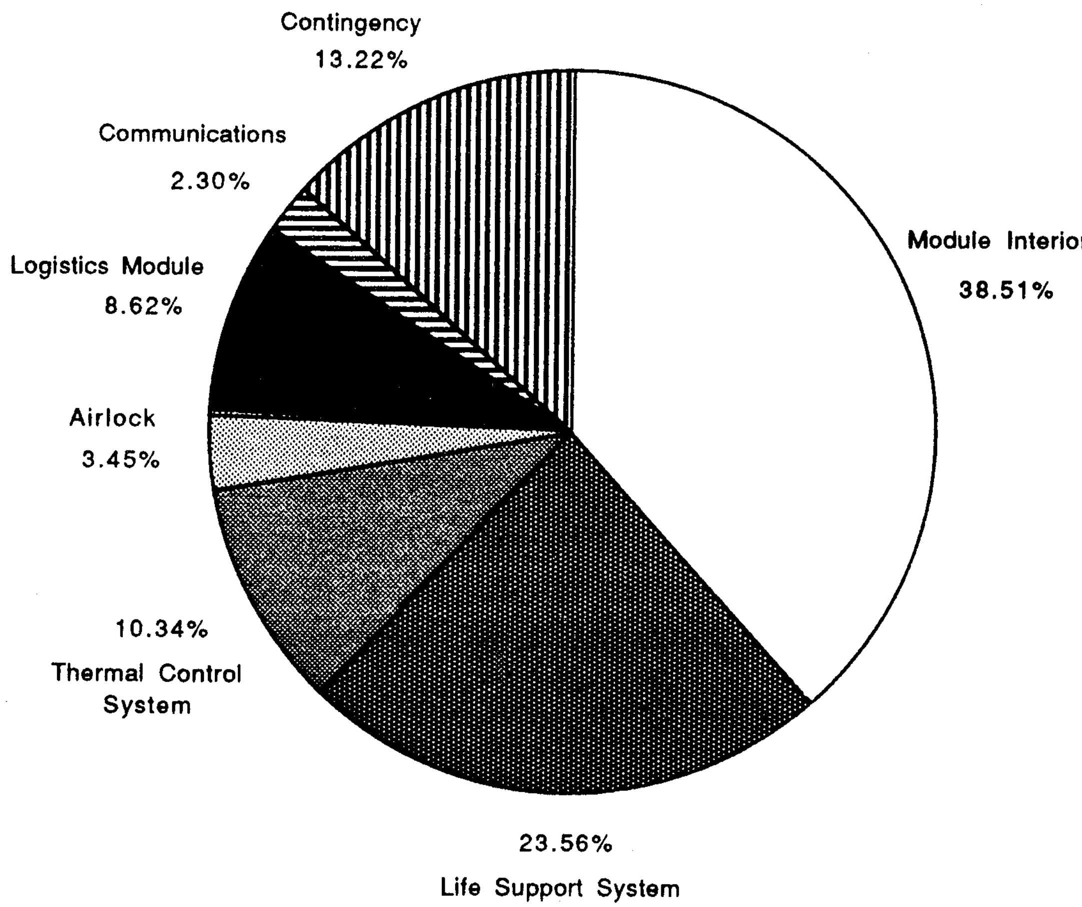


Table 5.0.3 Volume Distribution Comparison

Functional Area		Cylindrical Modules	Disk Module
Work	Volume, m3	57.3	55
(CCC, lab, HMF,	% Total Volume	23%	14%
maintenance)			
Personal	Volume, m3	79.4	64.8
(quarters	% Total Volume	31%	16%
personal hygiene)			
Social	Volume, m3	44.6	96.9
(galley, wardroom,	% Total Volume	15%	24%
fitness)			
Standoffs	Volume, m3	33	46.5
	% Total Volume	13%	12%
Free Space	Volume, m3	50.1	96.7
(corridors, hatch	% Total Volume	20%	24%
openings)			
Shelter	Volume, m3	0	15.9
	% Total Volume	0%	4%
Stowage	Volume, m3	0	19.8
	% Total Volume	0%	5%

Figure 5.0.3 Volume Distribution Comparison

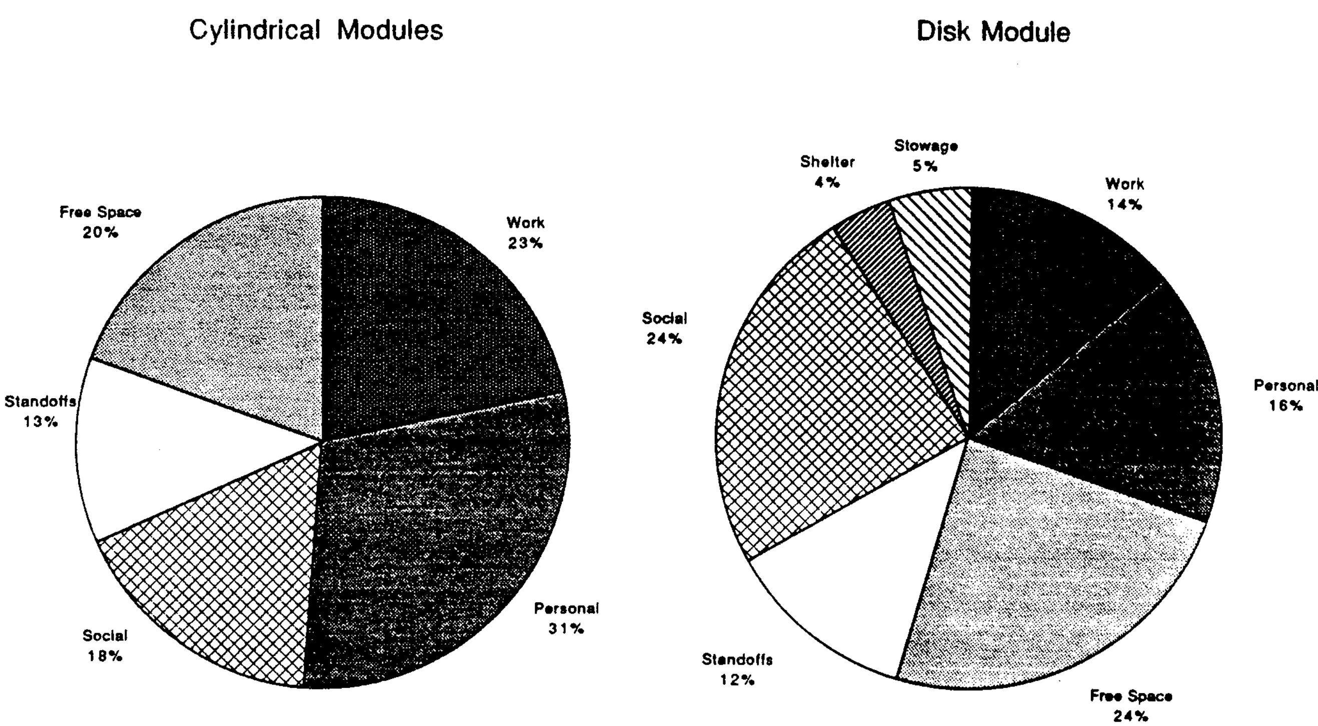


Table 5.0.4 Area Distribution Comparison

Functional Area		Cylindrical Modules	Disk Module
Work	Area, m2	26.5	23
(CCC, lab, HMF,	% Total Area	27%	14%
maintenance)			
Personal	Area, m2	34.9	27
(quarters	% Total Area	35%	16%
personal hygiene)			
Social	Area, m2	16.9	40.4
(galley, wardroom,	% Total Area	17%	24%
fitness)			
Standoffs	Area, m2	0	19.4
	% Total Area	0%	12%
Free Space	Area, m2	21	40.3
(corridors, hatch	% Total Area	21%	24%
openings)			
Shelter	Area, m2	0	6.6
	% Total Area	0%	4%
Stowage	Area, m2	0	8.3
	% Total Area	0%	5%

Figure 5.0.4 Area Distribution Comparison

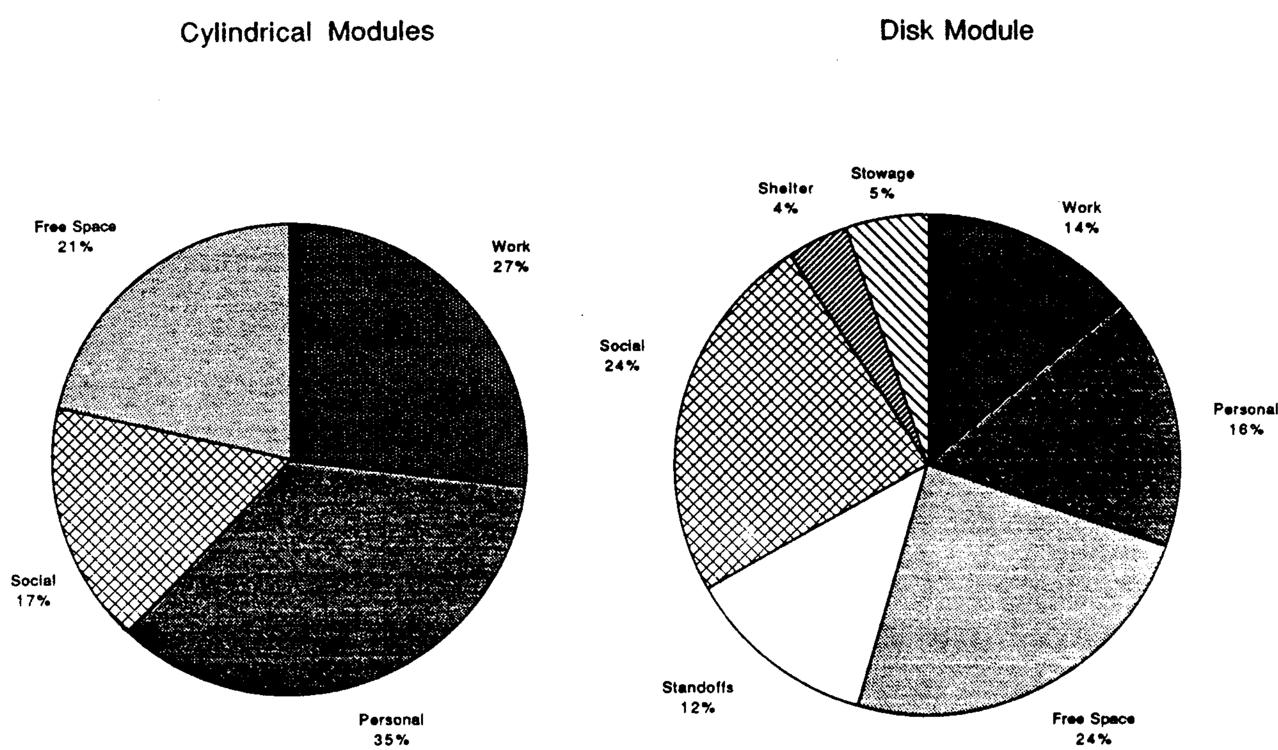


Table 5.0.5 Volume Distribution of Cylindrical Modules

Functional Area		Cylindrical Module #1	Cylindrical Module #2
Work	Volume, m3	32.1	25.1
(CCC, lab, HMF,	% Total Volume	25%	20%
maintenance)			
Personal	Volume, m3	34.3	45.1
(quarters	% Total Volume	27%	28%
personal hygiene)			
Social	Volume, m3	27.4	17.4
(galley, wardroom,	% Total Volume	17%	11%
fitness)			
Standoffs	Volume, m3	16.5	16.5
	% Total Volume	13%	10%
Free Space	Volume, m3	22.1	28
(corridors, hatch	% Total Volume	17%	17%
openings)			

Table 5.0.6 Floor Area Distribution of Cylindrical Modules

Functional Area		Cylindrical Module 1	Cylindrical Module 2
Work	Area, m2	14.8	11.7
(CCC, lab, HMF,	% Total Area	30%	24%
maintenance)			
Personal	Area, m2	15.2	19.6
(quarters	% Total Area	31%	40%
personal hygiene)			
Social	Area, m2	8.9	7.9
(galley, wardroom,	% Total Area	18%	16%
fitness)			
Free Space	Area, m2	10.6	10.3
(corridors, hatch	% Total Area	21%	21%
openings)			

Table 5.0.7 Volume Distribution of Disk Module Levels

Functional Area		Level 1	Level 2	Level 3
Work	Volume, m3	17.5	12.6	24.8
(CCC, lab, HMF,	% Total Volume	13%	9%	19%
maintenance)				
Personal	Volume, m3	36.1	0	28.8
(quarters	% Total Volume	27%	0%	22%
personal hygiene)				
Social	Volume, m3	42	12	42.9
(galley, wardroom,	% Total Volume	32%	9%	32%
fitness)				
Standoffs	Volume, m3	15.5	15.5	15.5
	% Total Volume	12%	12%	12%
Free Space	Volume, m3	20	56.6	20
(corridors, hatch	% Total Volume	15%	43%	15%
openings)				
Shelter	Volume, m3	0	15.9	0
	% Total Volume	0%	12%	0%
Stowage	Volume, m3	0	19.8	0
	% Total Volume	0%	15%	0%

Table 5.0.8 Floor Area Distribution of Disk Module Levels

Functional Area		Level 1	Level 2	Level 3
Work	Area, m2	7.3	5.3	10.3
(CCC, lab, HMF,	% Total Area	13%	9%	19%
maintenance)				
Personal	Area, m2	15	0	12
(quarters	% Total Area	27%	0%	22%
personal hygiene)				
Social	Area, m2	17.5	5	17.9
(galley, wardroom,	% Total Area	32%	9%	32%
fitness)				
Standoffs	Area, m2	6.5	6.5	6.5
	% Total Area	12%	12%	12%
Free Space	Area, m2	8.4	23.6	8.4
(corridors, hatch	% Total Area	15%	43%	15%
openings)				
Shelter	Area, m2	0	6.6	0
	% Total Area	0%	12%	0%
Stowage	Area, m2	0	8.3	0
	% Total Area	0%	15%	0%

Appendix C

Manned Mars Mission Design
(Mile High L5 Study Group)

M^{3}SG-02

PROPOSED CONCEPT

FOR A

MANNED MARS MISSION (M3) PROGRAM

FINAL

May 1990

Produced by:

Manned Mars Mission Study Group

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This document was prepared by an independent volunteer group at Martin Marietta Astronautics Group, Denver, Colorado in support of the Manned Mars Transportation Facility Infrastructure Study under contract to NASA/Marshall Space Flight Center. Most of the members of the Manned Mars Mission Study Group are employees of Martin Marietta Astronautics Group. The study group has been assisted by contributors that are not associated with Martin Marietta. A majority of the members of the study group are also members of the Mile High L5 chapter of the National Space Society.

The study group was organized under the auspices of the Mile High L5 chapter in order to provide lateral thinking and innovative ideas in support of the contract and as a parallel effort to the contract in support of the objectives of the National Space Society. The members of the study group have donated their personal time toward this effort at no cost to the government or Martin Marietta.

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This study is dedicated to building upon the visions of

Werhner Von Braun

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Department of Aerospace Engineering Sciences, University of Colorado at Boulder and the NASA/University Advanced Space Mission Design Program

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ABSTRACT

The Manned Mars Mission program is proposed as a practical approach for developing systems to implement long term expansion of the US space program. The concept involves incremental installation of permanent facilities on the planet Mars and its satellites to support the construction of a space station orbiting Mars. Space Station Freedom could be the launch platform for commercial flights that would establish a transportation infrastructure for exploitation of resources in the asteroids and exploration of the outer planets. Mars would be the way station providing refueling, resupply and crew exchange for exploration missions and commercial ventures in the outer solar system.

The first phase of the program includes three unmanned, autonomous material prepositioning flights and two manned flights. A mission would be launched every three years beginning in the year 2004. The initial unmanned mission would position robotic water extraction units on Phobos and Mars to stockpile water for propellant production. Each subsequent mission would utilize the water extracted on Phobos and Mars as propellent for the return flight to Earth. Any excess supply of water would be transported back to LEO for conversion to propellant for the OTV and lunar spacecraft. During this phase the research station would not be continually manned. Robotic servicers would be used to maintain the station between the manned missions, primarily for stockpiling life support materials and propellant. The objectives of the first phase are to (a) demonstrate water extraction and propellant production capabilities, (b) sustain self-sufficient survival in an extraterrestrial closed environment, (c) establish a research station that can be permanently manned and (d) initiate manned exploration on the surface of Mars.

The concept for the first phase consists of designs for an interplanetary spacecraft, lander spacecraft and the surface habitat. The Mars research station would attempt self-sufficiency in life support materials and consumables. Supplemental quantities of the essential elements to sustain life, primarily water, would be extracted from the Martian atmosphere, soil and polar caps. Propellant would be produced from Martian resources to expand the capability of the transportation systems. The efficiency in utilizing Martian resources presents significant advantages in comparison to the cost of supporting a large manned presence on the moon with supplies from Earth and limited lunar resources.

The Manned Mars Mission concept is presented as a long range national objective and is not intended to interfere with short range national objectives relating to the space station and lunar activity, but in fact, can support those objectives and should be implemented concurrently.

Table of Contents

ACRONYMS

CELSS	Closed (Controlled) Ecological (Environment) Life Support System
FTS	Flight Telerobotic Servicer
ET	External Tank (NSTS)
HLLV	Heavy Lift Launch Vehicle
IOC	Initial Operational Capability
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LOX	Liquid Oxygen
L1	LaGrange Point between Moon and Earth
L5	LaGrange Point to the Left of the Moon
MCL	Mars Cargo Lander Spacecraft
MEV	Mars Exploration Vehicle
MLI	Multi-Layered Insulation
MOI	Mars Orbital Injection
MUSC	Mars Utility Space Craft
MUV	Mars Utility Vehicle
M^3	Manned Mars Mission
M^3SG	Manned Mars Mission Study Group
NEA	Near Earth Asteroid
NERVA	Nuclear Engine for Rocket Vehcile Applications
NSTS	National Space Transportation System
NTR	Nuclear Thermal Rocket
OTV	Orbital Transfer Vehicle
TCE	Tethered Canyon Explorer
TMI	Trans Mars Injection

INTRODUCTION

The search for indigenous life forms or evidence of their existence on Mars continues to drive the scientific community toward exploration of the planet. The key to finding evidence of life probably lies beneath the surface of Mars in the permafrost or in the ice of the polar cap. The water on Mars is the key element to survival in that harsh environment and is a prime source for propellant. The water is the resource that makes Mars the obvious target for human exploration in space. Because Mars is the next most habitable planet beyond Earth, it provides the best opportunity for experimental production of food and the most potential for supporting a permanently manned presence. The orbit of Mars offers a strategic location for a way station to support commercial exploitation of the resources in Near Earth Asteroids (NEA) and the main asteroid belt. The resources on and near Mars are the key to the economic feasibility for establishing a space colony. As the population on Earth continues to increase and terrestrial resources are expended at increasing rates, the political survival of a nation may depend upon the ability and effort to extend its economy into space.

The concept for a Manned Mars Mission (M^3) program proposed by the study group was developed as a volunteer project directed by the technical committee of the Mile High L5 chapter of the National Space Society. The scope of the project evolved into an exercise in system engineering and mission concept planning. To develop the concept the study group used materials from the three "Case for Mars" Conferences, NASA publications and other sources too numerous to list individually.

Objective. The objective of the study group is to develop a lateral concept to the one proposed by the Case for Mars Conference[4]. The concept for an M^3 program is proposed by the study group as a more feasible and practical approach. The theme of the concept focuses on what is necessary to get to Mars and how to organize the effort. Practical ideas from various sources were incorporated into what is considered by the study group to be a concept that could be implemented as a national effort. At the beginning of the study effort in 1986, Soviet ambitions toward Mars were being revealed publicly and were perceived as a potential threat to the U.S. position in the space race. The failures of the Soviet probes to Phobos in 1989 have weakened the Soviet position but the potential threat has not diminished. A cooperative effort between the Soviet and U.S. space programs to reach Mars is an admirable goal but may not be feasible for some years to come.

This concept report describes the mission, operational elements and support requirements of the proposed M^3 program. It also describes the functions and characteristics of the transportation and facility systems involved in the program. The report is the result of the effort by the Manned Mars Mission Study Group (M^3SG). It represents the product of an extensive exercise in basic system engineering and mission planning to develop a technically practical and economically feasible concept for a Manned Mars Mission. Emphasis was focused on reducing risks in order to ensure human safety and mission success and reducing program cost in order to minimize impact on the budget for NASA.

The purpose of the Manned Mars Mission program is to establish continuous human presence on the planet Mars with the ultimate objective of constructing a space colony to support deep space activities. In order to accomplish this objective, five mission scenarios for the first phase and element systems of the program infrastructure are described in this report.

OVERVIEW OF PROPOSED MARS PROGRAM

The concept for the M^3 program attempts to summarize the planning and development of the systems necessary to implement phased expansion of the US space effort. U.S. space policy should allow operation and maintenance of the infrastructure systems by commercial ventures. NASA should lease facilities and services to support research and exploration activities. Commercialization should not be constrained by competition for resources between activities controlled by NASA. The opportunity to exploit the resources of the main asteroid belt and the moons of Jupiter suggests significant long term economic benefit in the construction of a space colony to support deep space activities by humans. Construction of space colonies requires a sophisticated infrastructure capable of marshalling resources from the most efficient and cost effective sources. The concept for the M^3 program presents a phased approach that would provide expansion of the infrastructure during each successive phase through use of extraterrestrial resources.

Short Range Objectives. The following short range objectives are proposed for the early phases of the Manned Mars Mission program:

1. Establish a permanently manned base on Mars,

2. Explore Mars and conduct scientific experiments,

3. Develop a self-sufficient infrastructure through in-situ production of volatiles and materials on Mars, Phobos and NEA.

Long Range Objectives. The following long range objectives are proposed for the later phases in expansion of the Manned Mars Mission program:

1. Exploit the resources of NEA[1], the main asteroid belt and the moons of other planets,

2. Construct a space colony in orbit around Mars.

Concept Features. The following features are key elements of the proposed concept for a Mars program:

1. The spacecraft and facilities that constitute the infrastructure of the program are designed for repeated use on successive missions without major modifications.

2. NERVA technology is employed in the propulsion subsystems of the spacecraft in order to overcome the limited performance and sources of hypergolic and cryogenic fuels and reduce operational costs.

3. Materials to establish an infrastructure are prepositioned during autonomous preparatory missions in order to reduce risk for the first manned mission.

4. The program objectives and infrastructure systems are structured to accommodate commercial enterprises and support scientific pursuits.

Phase One Summary. The first phase of the program includes three unmanned material prepositioning missions and two manned missions. A mission would be launched approximately every three years beginning in mid year 2004. The initial unmanned mission would position robotic water extraction units on Phobos to stockpile water or ice for production of liquid hydrogen and oxygen to be used as propellent. Each subsequent mission would use the liquid hydrogen produced on Phobos as propellent for the return flight to Earth and could transport a supply of liquid oxygen back to LEO or L1 for use in the propellant mixture for the Orbital Transfer Vehicle (OTV) and lunar shuttle spacecraft. During the first phase of the M^3 program the facility on the surface of Mars would not be continually manned. Robotic servicers would be used to maintain the Mars base between the manned missions.

The first phase of the concept involves implementation of an interplanetary spacecraft design, Mars shuttle spacecraft designs and surface habitat designs. The surface station would attempt self-sufficiency in life support materials and consumables. The essential elements to sustain life, primarily water, would be extracted from the Martian atmosphere, soil and polar caps. Propellant will be extracted from Martian resources to provide an expanded transportation capability. The efficiency in utilizing Martian resources presents significant advantages in comparison to supporting a large manned presence on the moon using the limited lunar resources. The following objectives of first phase are identified by priority:

1. Establish water extraction and propellant production capabilities,

2. Demonstrate self-sufficient survival in an extraterrestrial closed environment,

3. Construct a research station on the surface of Mars,

4. Initiate manned exploration and scientific experiments.

Phase Two Summary. The second phase of the program includes regular manned flights launched every 2.1 years. During the second phase the configuration of the Mars base would be expanded to support an increased population and permanent manning. A sprint mission flight profile would be used and the interplanetary spacecraft would not be maintained in Mars orbit for long periods. Propellant production on the surface of Mars and expanded water extraction operations on NEA will be necessary to compensate for the depletion of resources on Phobos and support increasing transportation requirements. The capacity of the propellent depot on Phobos would be expanded to support exploration flights to the main asteroid belt. Two additional interplanetary spacecraft would be needed to support the second phase activities. The following objectives of the second phase are identified by priority:

1. Expand the facilities on the surface of Mars using indigenous materials,

2. Expand the water extraction capability at the polar cap of Mars and construct a mass driver to launch containers of water or ice into Mars orbit,

3. Explore the surface of Mars extensively and expand the scientific experiment activities,

4. Launch unmanned robotic missions to autonomously explore other NEA and the main asteroid belt.

Phase Three Summary. The third phase of the program includes establishment of mining and manufacturing operations. Outpost stations are established and resource extraction is accelerated to support the increased population and transportation systems involved in production activities. The following objectives of the third phase are identified by priority:

1. Construct outpost stations to support specific manned and robotic activities,
2. Expand the infrastructure by establishing mining and manufacturing operations to exploit indigenous resources,
3. Launch manned missions to explore large asteroids and establish robotic mining operations on those asteroids rich in minerals and subsurface ice.

Phase Four Summary. The fourth phase of the program includes regular manned flights to the asteroid belt to expand mining operations. Raw and processed materials from the asteroids are transported to Mars orbit for construction of an orbiting space station. The population on Mars decreases when the space station becomes operational. The space station would provide multi-level artificial gravity. The use of robots on the surface of Mars is expanded to replace humans who were moved to the orbiting space station. The following objectives of the fourth phase are identified by priority:

1. Expand the mining operations on the asteroids,
2. Construct a space station in Mars orbit,
3. Launch unmanned robotic missions to autonomously explore the moons of Jupiter.

Phase Five Summary. The fifth phase of the program includes expanded mining operations on the asteroids. Materials from the asteroids are transported to Mars orbit for expansion of the orbiting space station into a space colony. The population on Mars decreases as surface activity is increasingly performed by robots. Significant transportation economy and environmental control can be achieved by concentrating human activities at a space colony. The following objectives of the fifth phase are identified by priority:

1. Expand the Mars space station to a space colony,
2. Construct a space station in or near the main asteroid belt,
3. Launch manned missions to explore the moons of Jupiter and establish robotic mining operations on those moons rich in resources,
4. Launch unmanned robotic missions to autonomously explore the moons of Saturn.

MISSION CONCEPTS FOR THE FIRST PHASE

The first phase of the Manned Mars Mission program involves five flights to Mars, three unmanned and two manned. The first three missions will be described in more detail than the last two missions of the phase which only expand the activity started in the first three missions.

The study group proposes that the extraction of water on Phobos could be undertaken as a commercial venture. A government sponsored holding company could loan a joint venture of aerospace companies the necessary funds to construct an interplanetary spacecraft and water extraction system. Funding for the loan could be raised through the sale of "space bonds" to the public and private investment of venture capital. The loan would be repaid by leasing to NASA facilities, transportation resources and support services provided by the M^3 program. The loan could also be repaid by selling liquid hydrogen and oxygen to NASA upon delivery to the propellant depot at LEO. NASA could lease space on the interplanetary spacecraft to deliver material and personnel to Mars orbit. Liquid hydrogen produced at Phobos could be used as propellent for the return flight to Earth in order to decrease the propellent requirement at launch from Earth and decrease the total mission cost. Such a venture would maximize commercial opportunities and expedite NASA missions by reducing the burden on the NASA budget and mission development staff.

The extraction of water and metals on NEA could be undertaken as an additional commercial venture that could be profitable within a relatively short period of time. An aerospace company could borrow the necessary funds to construct metal extraction and processing systems. Space could be leased on the unmanned freighter missions to transport the extraction and processing systems to an NEA and then deliver the products to LEO. During the return flight from Mars the interplanetary spacecraft could autonomously rendevous with an NEA and extract its valuable resources with the aid of robotic servicers. The potential revenue from the sale of water and metals on-orbit is limited only by the cost of transporting such material from Earth to LEO. NEA in the carbonaceous chondrite class, such as, 1979 VA, 1986 JK, 1986 RA and 1988 TA are potential candidates for water extraction operations during the first phase of the Mars program. Other NEA in the carbonaceous chondrite class, such as, 1580 Betulia and 1983 SA are less accessible but are good candidates during the second phase. NEA in the metallic class, such as, 3554 Amum and 1986 DA are good candidates for metal extraction operations during the second phase of the Mars program.

<u>First Mission</u>. The first mission would be an unmanned freighter flight and would occur between the years 2004 and 2006. The objective for the first autonomous freighter mission is to establish the capability to extract water on Phobos[2] autonomously.

<u>Second Mission</u>. The second mission would be another unmanned freighter flight and would occur between the years 2007 and 2009. The following objectives are identified for the second autonomous freighter mission:

1. Select the location of the Mars base using a robotic rover and extract water near the location autonomously,

2. Preposition components of the base on the surface of Mars,

3. Expand the capability of the water extraction system on Phobos,

4. Rendevous with a NEA and extract water.

Third Mission. The third mission would be the first manned flight and would occur between the years 2010 and 2012. The following objectives are identified for the first manned mission:

1. Assemble the first increment of the Mars base and establish the Initial Operational Capability (IOC) of the research station,

2. Explore key areas of interest near the Mars base, such as, Olympus Mons and Vallis Marineris.

Fourth Mission. The fourth mission would be the third unmanned freighter flight and would occur between the years 2013 and 2015. The following objectives are identified for the third autonomous freighter mission:

1. Extract water at the polar cap of Mars autonomously,

2. Transport additional components of the Mars base,

3. Rendevous with a NEA and extract water.

Fifth Mission. The fifth mission would be the second manned flight and would occur between the years 2014 and 2016. The following objectives are identified for the second manned mission:

1. Assemble the second increment of the Mars base and expand the capabilities of the research station,

2. Explore key areas of interest further away from the Mars base, such as, the polar cap. Areas briefly visited during the first manned mission would be revisited and examined more closely.

Infrastructure Facilities. The following facilities are key elements of the infrastructure for the proposed Mars Program:

1. A small on-orbit servicing facility would be positioned in Low Earth Orbit (LEO) or at L1 as a flying dry dock to support initial assembly and between flight servicing of the interplanetary spacecraft. The servicing facility could be attached to Space Station Freedom or positioned as a stand-off, free-flying platform. The servicing facility would be physically separate from the cryogenic propellent storage depot.

2. A facility, referred to as the Mars Base, would be constructed on the surface of Mars as a research station and would provide habitat for installation personnel. The facility would be expanded later to support robotic materials processing and other commercial ventures.

3. Water extraction systems and a cryogenic propellent storage depot would be positioned on Phobos as a refueling station. Additional water extraction systems and cryogenic storage tanks could be positioned on suitable NEA to supplement the capability on Phobos.

4. A water extraction system would be positioned on the surface of Mars near the polar cap to tap the water resources in the ice.

Infrastructure Transportation Systems. The following spacecraft and vehicles are key elements of the infrastructure for the proposed Mars Program:

1. Two interplanetary spacecraft would be assembled in LEO supported by Space Station Freedom and would fly between LEO or L1 and Mars orbit. One interplanetary spacecraft would be configured as a freighter to carry large quantities of cargo and the other would be configured with habitat units to transport personnel and limited quantities of cargo.

2. A non-man rated spacecraft, referred to as the Mars Cargo Lander (MCL), would be used to shuttle cargo containers between Mars orbit and the surface of Mars. An MCL would be used to transport equipment and supplies to remote exploration sites on the surface of Mars.

3. A man rated spacecraft, referred to as Mars Utility Spacecraft (MUSC), would be used to shuttle up to five crew members between Mars orbit and the surface of Mars. The MUSC would be used to transport personnel to remote exploration sites on the surface of Mars. The MUSC would also be used as a space tug to ferry cargo containers between the interplanetary spacecraft in Mars orbit and the propellent depot on Phobos.

4. A tracked utility vehicle, referred to as the Mars Utility Vehicle (MUV), would be used on the surface of Mars to assist construction activity and provide transportation for exploration in the immediate vicinity of the base.

5. A wheeled exploration vehicle, referred to as the Mars Explorer Vehicle (MEV), would be used on the surface of Mars to provide a mobile habitat for several personnel and transportation in remote areas away from the base. The MEV would be air-lifted to the exploration area in an MCL.

Program Sizing. Roughly $^2/_3$ of the mass at launch from LEO is propellent if the interplanetary spacecraft uses a NERVA engine for propulsion. Use of NERVA technology significantly increases the payload capacity because of its higher ISP as opposed to the less efficient conventional propulsion systems. The ability to replenish the propellent at Phobos after the first mission also significantly increases the payload capacity. Even with these advantages the concept for the first phase of the M^3 program reflects substantial requirements for transporting large quantities of mass to Mars. The following tables describe the projected materials manifest for each mission of the first phase.

Table 1-a Interplanetary Spacecraft Chassis Sizing Estimates

Materials Manifest	**1st Freighter Mission**	**2nd Freighter Mission**	**3rd Freighter Mission**
Interplanetary Spacecraft Chassis			
	(all new)		
Command Center Module	18,000 kg	18,000 kg (reused)	18,000 kg (reused)
Service & Support Module	18,000 kg	18,000 kg (reused)	18,000 kg (reused)
Electrical Power Module (4 SP100)	18,000 kg	18,000 kg (reused)	18,000 kg (reused)
NERVA Unit	13,000 kg	13,000 kg (reused)	13,000 kg (reused)
Interstage Connecting Structures	3,600 kg (4)	3,600 kg (reused)	3,600 kg (reused)
Airlock/MUSC Docking Port	5,400 kg	5,400 kg (reused)	5,400 kg (reused)
Aerobrake Shield Support Structure	21,000 kg	21,000 kg (reused)	21,000 kg (reused)
Aerobrake Shield Exterior Structure	11,000 kg	11,000 kg (reused)	11,000 kg (reused)
External Tank LOX Tanks (empty)	36,000 kg (6)	36,000 kg (reused)	36,000 kg (reused)
Tank Insulation & Cryogenic Coolant	19,200 kg (6)	19,200 kg (reused)	19,200 kg (reused)
External Tank LH_2 Tanks (empty)	96,000 kg (6)	96,000 kg (5 new)	96,000 kg (5 new)
Tank Insulation & Cryogenic Coolant	40,800 kg (6)	40,800 kg (5 new)	40,800 kg (5 new)
ET LH_2 Tank Mounts (200 kg each)	6,000 kg (30)	6,000 kg (reused)	6,000 kg (reused)
Propellent for Interplanetary Transit			
(at Launch from LEO)			
LH_2 Propellent in ET LH_2 Tanks	595,800 kg (6)	595,800 kg (6 new)	595,800 kg (6 new)
LH_2 Propellent in ET LOX Tanks	181,250 kg (5)	181,250 kg (5 new)	181,250 kg (5 new)
Subtotal	777,050 kg	777,050 kg	777,050 kg
(at Arrival in Mars Orbit)			
LH_2 Propellent in ET LH_2 Tanks	99,300 kg (1)	minimum reserve	minimum reserve
LH_2 Propellent in ET LOX Tanks	72,500 kg (2)	only	only
Subtotal	171,800 kg	(refuel at Phobos)	(refuel at Phobos)
(at Launch from Mars Orbit)			
LH_2 Propellent in ET LH_2 Tanks	99,300 kg (1)	99,300 kg (1)	99,300 kg (1)
LH_2 Propellent in ET LOX Tanks	72,500 kg (2)	72,500 kg (2)	72,500 kg (2)
Subtotal	171,800 kg	171,800 kg	171,800 kg
Roundtrip Total	777,050 kg	948,850 kg	948,850 kg
Assumed Benefit from Water Extraction on Phobos (Decrease in Required Propellent or Increase in Payload at Launch from LEO)	0	- 198,600 kg or + 170,000 kg	- 198,600 kg or + 170,000 kg

Table 1-b Interplanetary Spacecraft Chassis Sizing Estimates

Materials Manifest	**1st Manned Mission**	**2nd Manned Mission**
Interplanetary Spacecraft Chassis		
	(all new)	
Command Center Module	18,000 kg	18,000 kg (reused)
Service & Support Module	18,000 kg	18,000 kg (reused)
Electrical Power Module (4 SP100)	18,000 kg	18,000 kg (reused)
NERVA Unit	13,000 kg	13,000 kg (reused)
Interstage Connecting Structures	3,600 kg (4)	3,600 kg (reused)
Airlock/MUSC Docking Port	5,400 kg	5,400 kg (reused)
Aerobrake Shield Support Structure	21,000 kg	21,000 kg (reused)
Aerobrake Shield Exterior Structure	11,000 kg	11,000 kg (reused)
External Tank LOX Tanks (empty)	36,000 kg (6)	36,000 kg (reused)
Tank Insulation & Cryogenic Coolant	19,200 kg (6)	19,200 kg (reused)
External Tank LH_2 Tanks (empty)	96,000 kg (6)	96,000 kg (3 new)
Tank Insulation & Cryogenic Coolant	40,800 kg (6)	40,800 kg (3 new)
ET LH_2 Tank Mounts (200 kg each)	6,000 kg (30)	6,000 kg (reused)
Elevator/Airlock Units	8,175 kg (3)	8,175 kg (reused)
Propellent for Interplanetary Transit		
(at Launch from LEO)		
LH_2 Propellent in ET LH_2 Tanks	595,800 kg (6 new)	595,800 kg (6 new)
LH_2 Propellent in ET LOX Tanks	181,250 kg (5 new)	181,250 kg (5 new)
Subtotal	777,050 kg	777,050 kg
(at Arrival in Mars Orbit)		
LH_2 Propellent in ET LH_2 Tanks	minimum reserve	minimum reserve
LH_2 Propellent in ET LOX Tanks	only	only
Subtotal	(refuel at Phobos)	(refuel at Phobos)
(at Launch from Mars Orbit)		
LH_2 Propellent in ET LH_2 Tanks	198,600 kg (2)	198,600 kg (2)
LH_2 Propellent in ET LOX Tanks	72,500 kg (2)	72,500 kg (2)
Subtotal	271,100 kg	271,100 kg
Roundtrip Total	1,048,150 kg	1,048,150 kg
Assumed Benefit from Water Extraction on Phobos (Decrease in Required Propellent or Increase in Payload at Launch from LEO)	- 198,600 kg or + 170,000 kg	- 198,600 kg or + 170,000 kg

Table 2 Interplanetary Spacecraft Payload Sizing Estimates

Materials Manifest	1st Freighter Mission	2nd Freighter Mission	3rd Freighter Mission
Payload			
	(all new)	(all new)	
Mars Utility Spacecraft (empty)	14,000 kg (1)	14,000 kg (1)	14,000 kg (1 new)
LH_2 Propellent in ET LOX Tanks (for local flights at Mars)	36,250 kg (1)	36,250 kg (1)	36,250 kg (1 new)
Phobos Water Extraction Unit	36,000 kg (2)		
Phobos Power Plant (2 SP100)	18,000 kg (2)	9,000 kg (1)	
Phobos Electrolysis Unit	9,000 kg (2)	4,500 kg (1)	
Phobos Cryogenic Cooling Unit	9,000 kg (2)	4,500 kg (1)	
Phobos Robotic Servicer & Fuel	4,500 kg (1)	4,500 kg (1)	
Mars Water Extraction Unit		36,000 kg (2)	18,000 kg (1 new)
Mars Base Habitat Modules		95,400 kg (6)	95,400 kg (6 new)
Mars Base Airlock Unit		22,720 kg (4)	17,040 kg (3 new)
Mars Utility Vehicle		5,800 kg (1)	5,800 kg (1 new)
Mars Robotic Servicer		1,000 kg (1)	
Mars Cargo Lander (empty)		37,000 kg (2)	18,500 kg (1 new)
Mars Cargo Lander Mounts		1,350 kg (10)	675 kg (reused)
Mars Base Power Plant (4 SP100)		36,000 kg (2)	18,000 kg (1 new)
Mars Base Greenhouse Units			36,000 kg (4 new)
Mars Base Waste Processing Unit			4,500 kg (1 new)
Subtotal	126,750 kg	308,020 kg	227,915 kg
LIMIT	144,000 kg	314,000 kg	314,000 kg

	1st Manned Mission	2nd Manned Mission
Payload		
	(all new)	
Mars Utility Spacecraft (empty)	14,000 kg (1)	14,000 kg (1 new)
LH_2 Propellent in ET LOX Tanks (for local flights at Mars)	36,250 kg (1)	36,250 kg (1 new)
Consumable Water for Crew	45,000 kg	38,000 kg (reused) 7,000 kg (new)
Auxiliary Water Tanks (empty)	1,000 kg (6)	1,000 kg (reused)
Consumable Oxygen for Crew	20,000 kg	20,000 kg (new)
Auxiliary Oxygen Tanks (empty)	1,000 kg (6)	1,000 kg (reused)
Food Stores	18,000 kg	18,000 kg (new)
Waste Processing Units	2,250 kg (3)	2,250 kg (reused)
Interplanetary Spacecraft Habitat Modules	95,400 kg (6)	95,400 kg (reused)
Habitat Unit Support Structure	4,050 kg (3)	4,050 kg (reused)
Mars Explorer Vehicle	18,000 kg (1)	
Mars Robotic Tethered Explorer	3,000 kg (1)	
Mars Utility Vehicle	11,600 kg (2)	
Mars Base Greenhouse Units	18,000 kg (2)	18,000 kg (2 new)
Mars Base Waste Processing Unit	4,500 kg (1)	4,500 kg (1 new)
Subtotal	292,050 kg	259,450 kg
LIMIT	305,825 kg	305,825 kg

Table 3 Mission Sizing Summary

Manifest Summary	**1st Freighter Mission**	**2nd Freighter Mission**	**3rd Freighter Mission**
at Launch from LEO			
Chassis (dry weight) Mass Subtotal	306,000 kg	306,000 kg	306,000 kg
Total Propellent Mass	813,300 kg	813,300 kg	813,300 kg
Maximum Payload (dry weight) Mass	107,750 kg	277,750 kg	277,750 kg
Interplanetary Spacecraft Total	1,227,050 kg	1,397,050 kg	1,397,050 kg
at Launch from Mars Orbit			
Chassis (dry weight) Mass Subtotal	192,000 kg	192,000 kg	192,000 kg
Total Propellent Mass	171,800 kg	171,800 kg	171,800 kg
Total Payload (dry weight) Mass	0	0	0
Interplanetary Spacecraft Total	363,800 kg	363,800 kg	363,000 kg
Total Mass Reused (from previous mission)	0	192,000 kg	192,675 kg
Total New Mass	1,227,050 kg	1,205,050 kg	1,204,375 kg
HLLV Flights Required (using NERVA on the interplanetary spacecraft)	14	14	14
HLLV Flights Required (if conventional Liquid Hydrogen & Oxygen Propulsion System is used on the interplanetary spacecraft)	25	25	25

	1st Manned Mission	**2nd Manned Mission**
at Launch from LEO		
Chassis (dry weight) Mass Subtotal	314,175 kg	314,175 kg
Total Propellent Mass	813,300 kg	813,300 kg
Maximum Payload (dry weight) Mass	269,575 kg	269,575 kg
Interplanetary Spacecraft Total	1,397,050 kg	1,397,050 kg
at Launch from Mars Orbit		
Chassis (dry weight) Mass Subtotal	245,775 kg	245,775 kg
Total Propellent Mass	271,100 kg	271,100 kg
Total Payload (dry weight) Mass	161,700 kg	161,700 kg
Interplanetary Spacecraft Total	678,575 kg	678,575 kg
Total Mass Reused (from previous mission)	0	387,475 kg
Total New Mass	1,397,050 kg	1,009,575 kg
HLLV Flights Required (using NERVA on the interplanetary spacecraft)	16	12
HLLV Flights Required (if conventional Liquid Hydrogen & Oxygen Propulsion System is used on the interplanetary spacecraft	27	23

INTERPLANETARY SPACECRAFT

The design of the interplanetary spacecraft employs a linear concept that provides considerable adaptability to various configurations. Many of the components of the interplanetary spacecraft are similar in shape and function to those used on Space Station Freedom. Component commonality with existing spacecraft was used as a design criteria and cost reduction strategy in developing the concept for the Mars program. The main modules of the vehicle are space station modules redesigned internally to withstand the structural stresses of interplanetary flight. The structural strength provided by the linear design is needed to accommodate the thrust required to accelerate and decelerate the entire vehicle as one unit and to tolerate the stress of the aerocapture maneuvers. The same basic chassis of the interplanetary spacecraft is common to all missions. The interplanetary spacecraft is designed for use on repeated flights without major modifications and employs a flexible configuration that can support missions beyond the scope of the first phase of the Mars program and throughout the 21st century. The concept provides a flexible and durable spacecraft design that is intended to support various mission scenarios throughout the future phases of the Mars program. The interplanetary spacecraft would be assembled in four stages at the on-orbit servicing facility.

Autonomous Freighter Requirements. The design configuration of the unmanned freighter version of the interplanetary spacecraft must implement the following requirements.

1. Propulsive capability is required to accelerate up to 1.5 million kilograms from LEO at greater than minimum delta velocity. The propulsion system must provide specific impulse of at least 1000 ISP and 20,000 kilograms of thrust.

2. Adequate radiation shielding is required to protect critical electronic systems from radiation levels of at least 5000 Rems during multiple periods of up to 5 days each.

3. Adequate active and passive measures are required to detect and counteract micrometeorite collisions and punctures at the rate of one puncture per major subsystem during each 60 day period.

4. The spacecraft is required to withstand severe levels of vibration during aerocapture, acceleration and deceleration maneuvers.

5. The spacecraft is required to withstand severe levels of structural stress during aerocapture, acceleration and deceleration maneuvers. Adequate active and passive measures are required to detect and counteract hazardous structural stress.

6. The onboard computer systems are required to provide adequate information processing and spacecraft control capability for autonomous operations; such as, navigation and guidance. A capability for teleoperation is also required to provide remote intervention.

7. Mechanisms and structures are required in order to attach the Mars Cargo Lander (MCL) spacecraft and additional cargo containers to the aerobrake shield support trusses.

8. Two fault tolerance is required of onboard control subsystems.

Manned Spacecraft Requirements. The design configuration of the manned version of the interplanetary spacecraft must implement the following requirements in addition to the requirements identified for the unmanned freighter.

1. Accommodations are required for a minimum crew size of 10 and a optimum crew size of 12. At least 27 cubic meters should be allocated per crew member for private quarters. Common living areas should provide at least another 30 cubic meters per crew member.

2. The spacecraft is required to continuously rotate to generate centrifugal force and provide artificial gravity during the transit periods of the mission between maneuvers. An adjustable rotational speed between 3.5 rpm and 5.3 rpm is required in order to provide artificial gravity levels between 0.4 g and 0.9 g.

3. Adequate radiation shielding is required in at least the command center module of the spacecraft to protect crew members and sensitive electronics from radiation levels up to 5000 Rems during multiple periods of up to 5 days each. Safe haven supplies and accommodations are required within the shielded module.

4. Independent life support systems are required for each habitat module. A completely Closed Ecological Life Support System (CELSS) is required to support a crew of 12 for durations up to two years continuously with a dormant period of up to 18 months in the middle of the mission.

5. Seal designs for pressurized modules are required to maintain pressure for up to five years with no more than one percent loss. Intra-module seals must be able to isolate any module posing a life threatening situation.

6. Adequate active and passive measures are required to detect and counteract harmonic vibration created by crew movement during rotation of the aerobrake shield and habitat modules and periods of artificial gravity.

7. The design of the command center module is required to support human as well as autonomous operations. The control stations are required to accommodate human operations during acceleration, deceleration and aerocapture maneuvers.

8. Mechanisms and structures are required in order to attach the habitat modules and cargo containers to the aerobrake shield support trusses.

9. Adequate safety measures are required to prevent or decrease hazards and life threatening situations, such as, fire, toxic contamination, depressurization, malfunction of protective subsystems and moving components and damage to mechanical subsystems or structural components.

<u>Servicing Facility</u>. The servicing facility would be composed of six space station like modules and a central airlock compartment as shown in Figure 1. The facility would be an expansion of Space Station Freedom or a separate free-flying platform nearby. Spare parts, tools and other equipment used in assembling and servicing the interplanetary spacecraft would be stored in the modules. As many as two modules may be habitat modules to house the assembly or maintenance crew members. An auxiliary electrical power supply, such as batteries or a fuel cell, would be located in one of the modules. Panels of photovoltaic cells could be attached to provide electrical power. The facility would be activated only when attached to the chassis of the interplanetary spacecraft. The facility would use the SP100 units in the chassis of the interplanetary spacecraft as its primary electrical power supply.

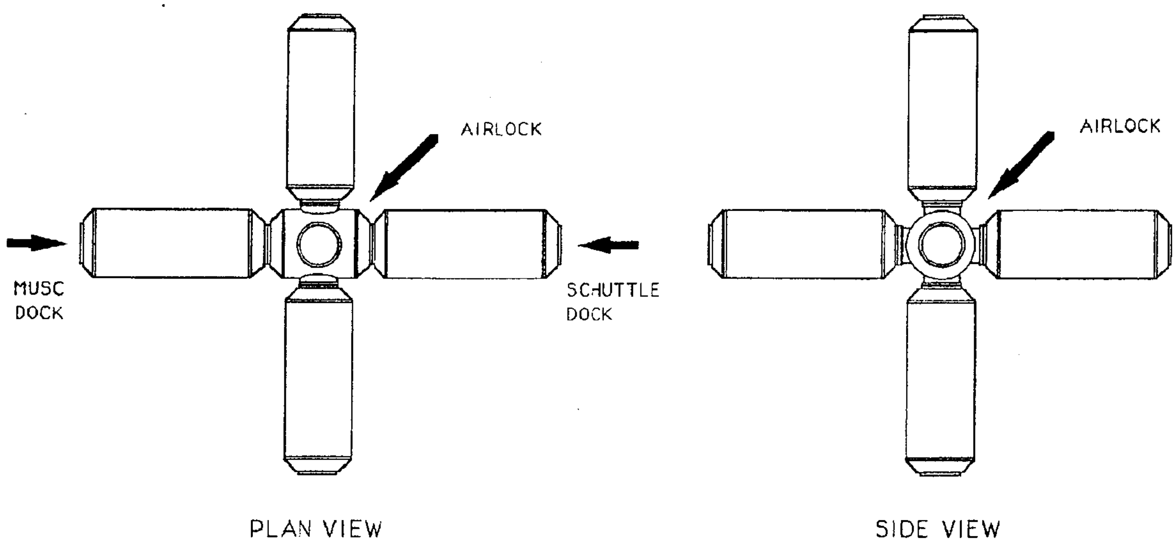

Figure 1 Interplanetary Spacecraft Servicing Facility

First Stage of Assembly. During the first stage of the assembly sequence the following chassis sections would be connected in a linear configuration:

1. An airlock compartment would be used as a docking port for the Mars Utility Space Craft (MUSC). The airlock would have three side doors to the elevators, a rear door to the command module and a forward door to the MUSC.

2. The command module would contain key electronic equipment, such as the primary computers for navigation and spacecraft subsystems control. The command module is divided into four levels or stacked compartments to accommodate the direction of force during acceleration and deceleration of the spacecraft. The walls of the command module are heavily lined for protection against radiation. The command module serves as the emergency "storm" shelter for the crew during periods of dangerous solar flare activity.

3. The service and support module would contain spare parts, supplies, maintenance equipment and cryogenic cooling pumps for the propellent tanks. A centralized second stage waste processing unit may also be located in the service and support module.

4. The power generator module would contain four SP100 nuclear fueled electrical power generation units,

5. The nuclear rocket engine (NERVA) unit would be attached at the end of the chassis assembly. The required engine performance is considered to be 1000 seconds specific impulse yielding approximately 22,725 kilograms of thrust. The reactor core made of uranium loaded graphite fuel elements heats the hydrogen to a temperature between five and six thousand degrees. The existing NERVA engine design specifies a pressure vessel of approximately 56 inches in diameter with a maximum envelope of approximately 8 feet across the pressurization spheres. The NERVA engine includes the following major subsystems:

   a. Propellant feed system including turbopump, valves and fuel lines,

b. Nuclear subsystem including reactor, reflector and radiation shield,

c. Thrust chamber assembly including thrust structures, pressure vessel, gimbal for thrust vector control and the main engine exhaust nozzle,

d. Engine control system,

e. Pneumatic system.

f. Radiator panels attached to the exterior of the NERVA unit.

The exterior design of the three main modules is similar in appearance and size to that of Space Station Freedom modules, approximately 14 meters long and 5 meters in diameter. The same configuration of the interplanetary spacecraft chassis would be used for both freighter and manned missions.

Second Stage of Assembly. During the second stage of the assembly sequence the following components would be used to construct the aerobrake shield and connected to the forward end of the chassis of the interplanetary spacecraft:

1. Six short, tower-like truss structures would be attached diagonally to the airlock compartment and reinforcing bulkhead structure to support the center of the shield and form the docking area for the MUSC.

2. Six medium length, tower-like truss structures would be attached diagonally to the airlock compartment and reinforcing bulkhead structure to support the middle area of the shield.

3. Six long, tower-like truss structures would be attached perpendicular to the airlock compartment and reinforcing bulkhead structure to support the outer edge of the shield. Three of the truss structures will contain elevators and support the three habitat areas during manned missions. The truss structures would be aligned to divide the shield into six equal sections.

4. Six liquid oxygen (LOX) tanks reused from expended NSTS External Tanks (ET) would be attached between the six sets of truss structures and adjacent to the MUSC docking port. The tanks would be used to store liquid hydrogen (LH_2) or consumables, such as, water or oxygen for life support during manned missions. A tank may also be used to store methane produced in the waste processing units. The methane could be used as an additional propellent for the MUSC.

5. A wide, thin truss structure shaped like a bowl and similar to that used to support a geodesic dome would be assembled and attached to the ends of the other truss structures to support the shape of the exterior shell of the shield.

6. A flexible bladder or skin would be draped over the outer truss structure and would harden to form the exterior shell of the shield. Another flexible bladder would be draped over the exterior shell and a silica based foam would be injected between the two layers, would harden and form the heat shield. The foam would be somewhat similar to the material used to make the heat shield tiles on the NSTS.

The truss structure components would be somewhat similar to those used on Space Station Freedom. The aerobrake shield would be shaped roughly like a flattened bowl with a hole in the center and function as a heat shield to protect the chassis and payload during the aerocapture maneuver into a planet's atmosphere. The heat shield on the bottom of the MUSC would fill the hole in the center of the aerobrake shield. The aerobrake shield would be approximately 62 meters in diameter and 16.5 meters deep.

Third Stage of Assembly. During the third stage of the assembly sequence the six primary propellant tanks would be attached to the chassis of the interplanetary spacecraft as shown in Figure 2. Payload containers and MCL spacecraft would be attached to the back of the aerobrake shield and the MUSC would be docked in the center of the aerobrake shield. The propellant tanks would be reused from expended NSTS External Tanks, but modified internally to provide separate compartments for load leveling. The tanks provide cryogenic storage of the liquid hydrogen (LH_2) propellant. Several inches of Multi-Level Insulation (MLI) purged with liquid nitrogen would be required in addition to at least two inches of rigid insulation in order to prevent loss of propellant through boil-off. Cryogenic cooling pumps in the operational support module would provide refrigeration to maintain the temperature of the tanks. Radiator units for the thermal conversion cycle of the power generation system would be mounted on the tanks to provide additional shielding. The length of the tanks would be standardized according to average propellant requirements. Internal partitioning would be required to create independent compartments in order to minimize loss in case of punctures in the tank walls. An internal baffle system would be required to suppress fluid slosh. The tanks could be used on the return flights from Mars to transport water or ice extracted from Phobos or the polar cap on Mars.

Fourth Stage of Assembly. A fourth stage of the assembly sequence would be required prior to the manned mission. The following habitat components would be connected to the interplanetary spacecraft chassis as shown in Figure 3:

1. Three elevators would be inserted into three of the tower-like truss structures supporting the edge of the aerobrake shield and would be used to provide transit for the crew between the habitats and the airlock compartment located forward of the command module.

2. Six habitat modules would be positioned on three platforms attached to three of the tower-like truss structures supporting the edge of the aerobrake shield.

The design of the habitat, storage and greenhouse modules is derived from the design of Space Station Freedom modules. The size of each module is approximately 14 meters long and 5 meters in diameter.

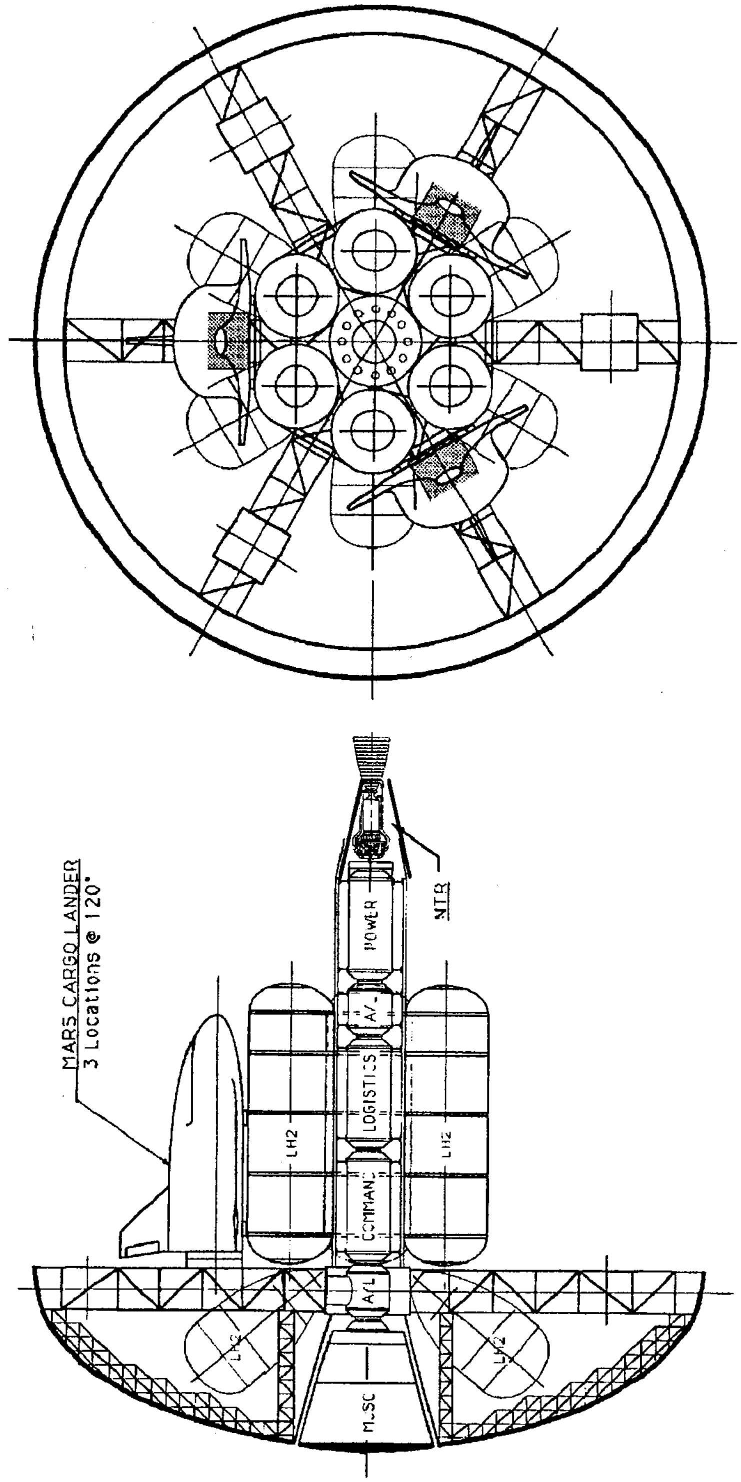

Figure 2 Cutaway View of Unmanned Freighter Configuration for the Interplanetary Spacecraft

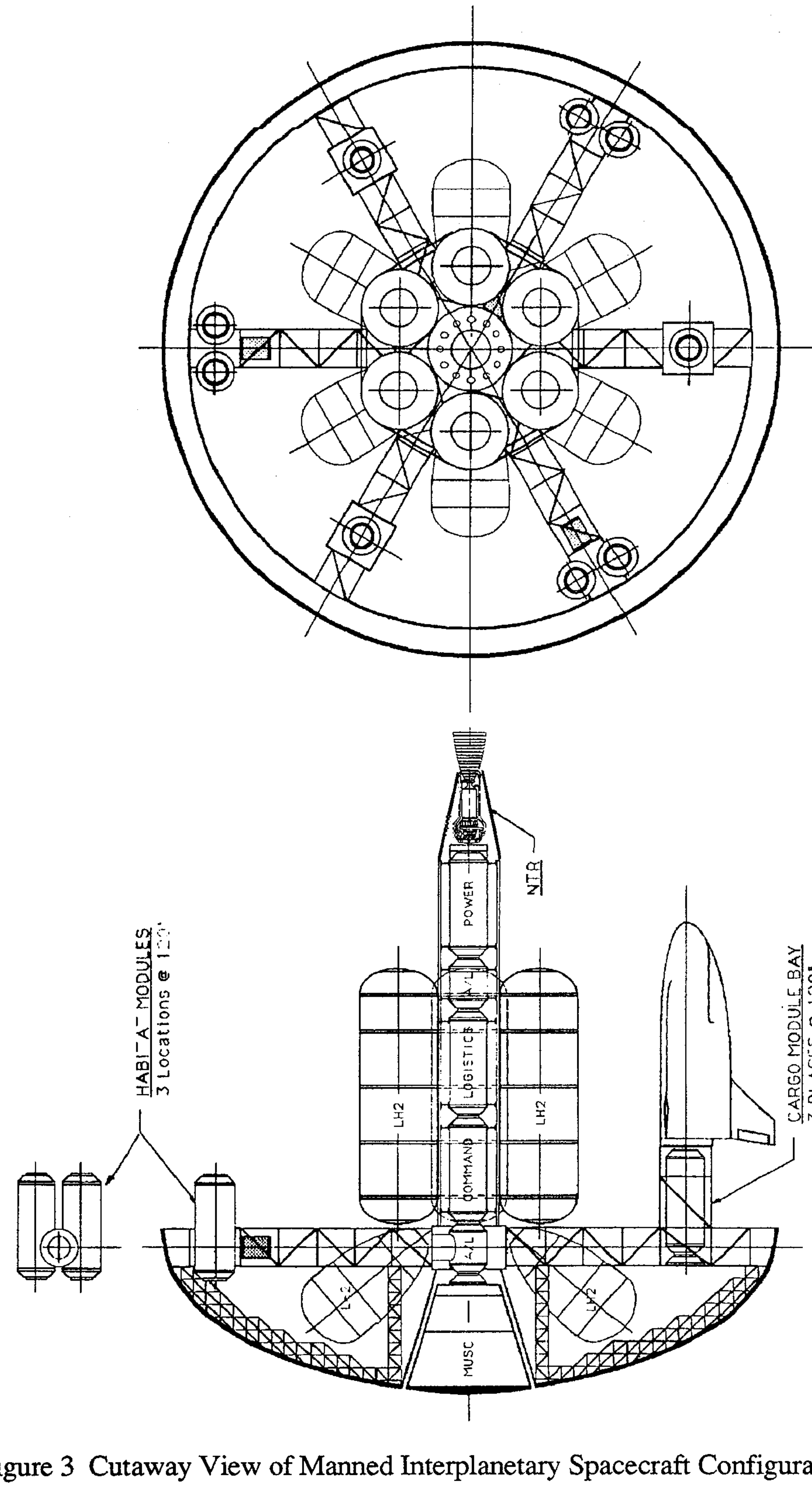

Figure 3 Cutaway View of Manned Interplanetary Spacecraft Configuration

Future Enhancements. The modular design of the interplanetary spacecraft supports interchangeability of component units to accommodate integration of technology improvements or expanded capabilities such as the following potential enhancements.

1. When reliable ion engine technology becomes available, several ion engines could be integrated into the design with only minor modifications; such as additional shielding. An additional power generation module may be required and could easily be integrated into the linear design of the chassis of the interplanetary spacecraft. The continuous thrust of the ion engines would enable use of sprint mission profiles.

2. The interior of the habitat modules could be reconfigured to accommodate up to 18 crew members after completion of the Mars base.

3. Micrometeorite deflection system

INTERPLANETARY FLIGHT TRAJECTORIES

Three different interplanetary flight trajectories have been chosen for the various flights in the Manned Mars Mission Concept as shown in Table 4. All of the flights use low energy ballistic trajectories due to anticipated propulsion limitations during the mission time frame. For this study, it was determined that the more energetic propulsion systems required for short duration Sprint class missions may not be available during the first phase of the Mars program. This forces the the use of lower energy ballistic trajectories with longer flight times. The longer flight times make it necessary to provide artificial gravity for the manned missions. More advanced, continuous high thrust propulsion systems could be used with the basic interplanetary spacecraft chassis as they become available.

Table 4 Flight Scenarios

	UNMANNED FREIGHTER MISSIONS			FIRST MANNED MISSION	SECOND MANNED MISSION
TRAJECTORY	Combination of Venus Swingby and Opposition			Opposition Class	Conjunction Class
VELOCITY	Minimum Δ			Maximum attainable Δ	Maximum attainable Δ
TRANSIT TIME					
Earth to Mars	approx 10 months (Venus Swingby)			7-9 months	8-9 months
Mars to Earth	approx 8 months (Opposition)			7-8 months	6-7 months
PERIOD IN MARS ORBIT	60 days			approx. 2-3 months	approx. 13 months
	1st Flight	2nd Flight	3rd Flight		
EARTH to MARS					
Depart (approx.)	June 2004	Sept. 2007	Dec. 2013	Nov. 2010	Dec. 2013
Arrive (approx.)	May 2005	April 2008	Sept. 2014	Sept. 2011	Oct. 2014
MARS to EARTH					
Depart (approx.)	July 2005	June 2008	Nov. 2014	Nov. 2011	Oct. 2015
Arrive (approx.)	March 2006	March 2009	Sept. 2015	Aug. 2012	June 2016

Unmanned Freighter Missions. A Venus swingby type trajectory would be used to minimize the fuel required and maximize the cargo delivered on station to Mars park orbit. The inbound Venus swingby trajectory could be used for flights returning to earth that exceed the original launch mass but that situation is not anticipated in the mission scenarios. Since the freighter vehicles will need to return to Earth as soon as their cargo is delivered at Mars, it will not be possible to use a Venus swingby on both the outbound and inbound legs of the journey.

The close approach to Venus gives a boost in inertial velocity due to the planet's gravity. The increase in velocity is achieved without large expenditures of onboard propellants. This is the same technique used by the unmanned Voyager missions to the outer planets. A diagram of the outbound Venus swingby class trajectory is shown in Figure 4. The transit time is between eight and eleven months each way, with a stay at Mars of around sixty days.

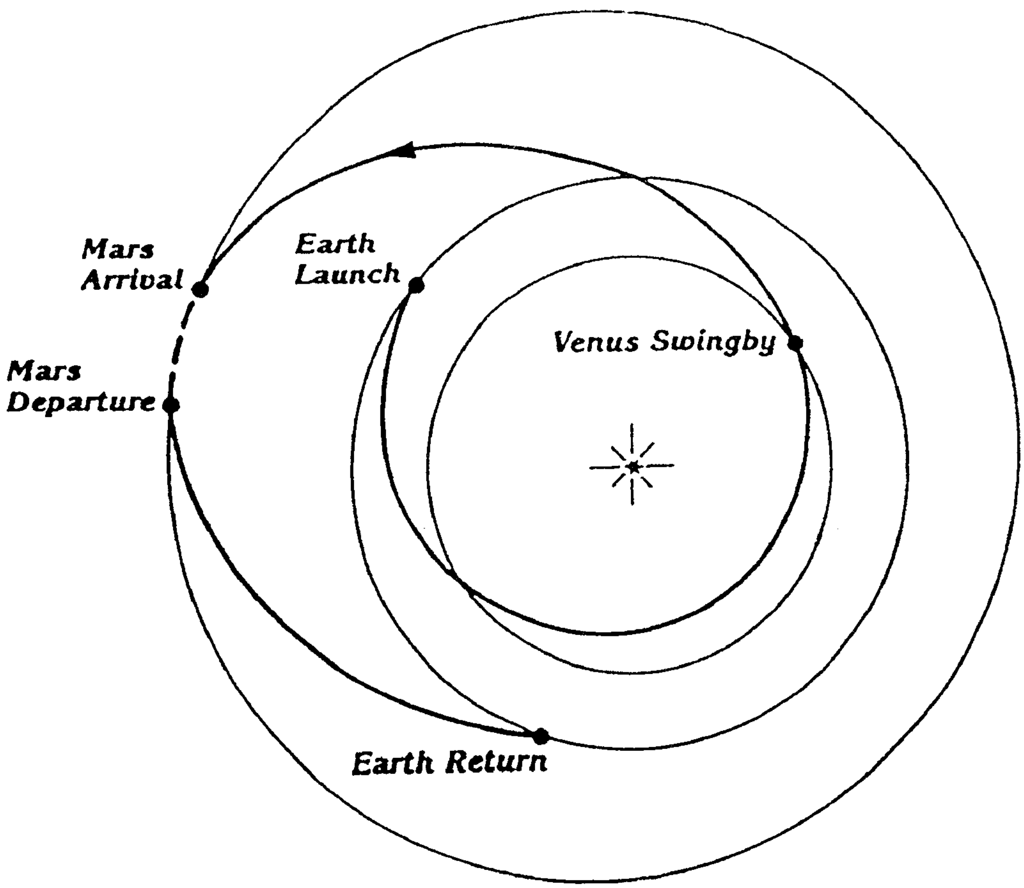

Figure 4 Outbound Venus Swingby Class Trajectory

The Venus swingby was ruled out for the manned missions because of the increased radiation hazard to the crew during the close approach to the Sun. Additional heavy radiation shielding would have to be provided for all the habitable areas of the interplanetary spacecraft before this type of trajectory could be safely used for manned missions.

First Manned Mission. An opposition class trajectory would be used for the first manned mission. A diagram of the opposition class trajectory is shown in Figure 5. The opposition transit time is the shortest of all the purely ballistic trajectories. The twelve person crew would be exposed to space for seven to ten months each way, depending on the exact departure and arrival dates chosen.

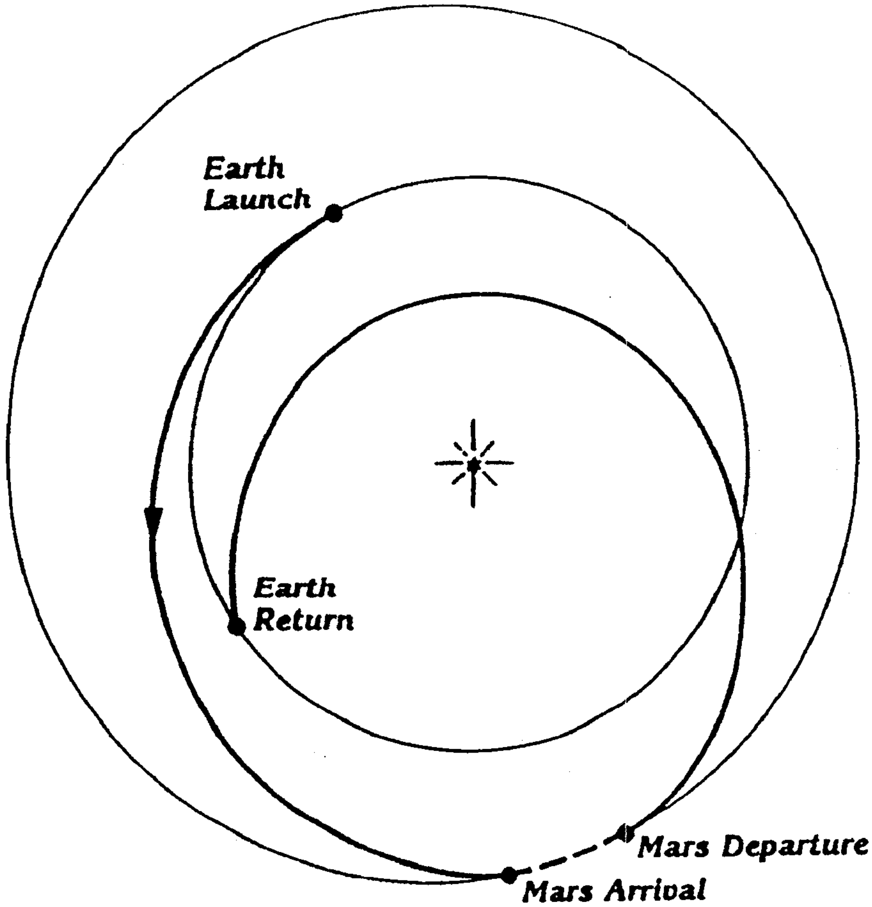

Figure 5 Opposition Class Trajectory

The crew of the first manned mission will stay at Mars for two to three months to assemble a station and conduct initial exploration activity. By the time of the first manned mission, only two unmanned freighter missions will have reached Mars. Only a minimum of supplies and equipment will be available to the crew; therefor, two to three months is the maximum practical period of activity at Mars.

<u>Second Manned Mission</u>. A conjunction class trajectory would be utilized for the second manned mission to provide an extended stay on Mars of approximately thirteen months. A diagram of the conjunction class trajectory is shown in Figure 6. The longer stay allows more time for science experimentation, base expansion, and exploration sorties on the Martian surface. By the time of this mission, additional freighter missions will have delivered more base equipment and supplies. The materials processing units and greenhouses set up by the crew of the first mission will have begun to produce usable propellants, water, air, and food.

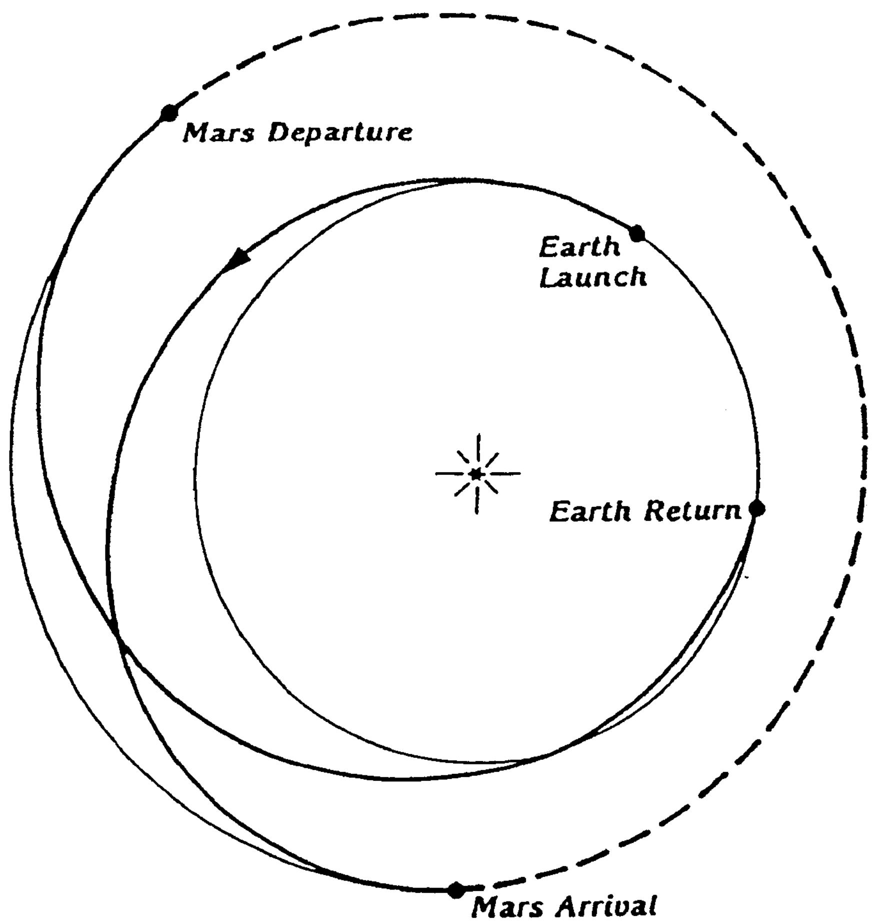

Figure 6 Conjunction Class Trajectory

The conjunction class trajectory will take eight to eleven months both ways, requires lower departure energies at Earth and Mars, and results in lower arrival velocities. This mission will have a crew of twelve and would initiate the era of a permanently manned base on the surface of Mars.

FREIGHTER MISSION SCENARIO

Assembly of the freighter version of the interplanetary spacecraft at Space Station Freedom in Low Earth Orbit (LEO) would precede the first unmanned mission. The launch operations and sequence of activities necessary to prepare the spacecraft are described in Appendix I. Prior to departure for Mars, a thorough shakedown cruise around the Moon and Earth would be conducted to checkout the spacecraft systems and verify autonomous flight control systems.

Interplanetary Flight Profile. Departure from Earth orbit begins with the Trans-Mars Injection (TMI) burn. Main engine cutoff of the NERVA nuclear thermal propulsion unit occurs after thirty to sixty minutes of constant acceleration at 0.2 g's. After the TMI burn, the spacecraft turns the aerobrake shield towards the Sun. The aerobrake shield then serves a dual purpose as a radiation shield during the ballistic cruise to Mars and assists in cooling the primary propellent tanks that are in the shade behind the aerobrake shield.

On approach to Mars the interplanetary spacecraft is secured for the deceleration thrust which will slow it down to the velocity necessary for the aerocapture maneuver into Mars park orbit. The duration of the Mars Orbital Injection (MOI) burn depends on the specific geometry of the planetary approach, and on the amount of velocity that can be safely lost during the aerocapture maneuver. A heavily loaded interplanetary spacecraft will not be able to depend on the aerocapture as much for MOI because of the g forces experienced during the maneuver. Use of a propulsive deceleration approach to Mars is less stressing on the spacecraft structure than the aerocapture manuever into Mars orbit. The aerobrake shield is then pointed forward and used to slow the spacecraft by passing through the Martian atmosphere like a meteor. This maneuver is a high g pass which slows the spacecraft into an elliptical capture orbit. After the interplanetary spacecraft has stabilized its orbit around Mars, the autonomous on-orbit activities would commence.

The inbound return to Earth mission sequence is the reverse of the one described above, with a second aerocapture into Earth orbit. Use of the aerocapture maneuver reduces the requirements for onboard propellants and allows more usable payload weight delivered to the destination planet. Aerocapture technology is not yet considered reliable, but the necessary materials, structures, guidance and control techniques, and computer technologies should available by the year 2004.

First Freighter On-Orbit Activity. After the interplanetary spacecraft arrives in orbit around Mars communication satellites would be deployed into stationary orbits. The Mars Utility Spacecraft (MUSC) would exit the docking port in the center of the aerobrake shield, attach to individual payload elements and push the cargo containers and five empty LH_2 propellant tanks to the surface of Phobos. After completion of the cargo off loading activity the MUSC would park in orbit on Phobos in order to refuel for activity during the next mission.

Phobos Activity. After the cargo containers and propellant tanks have been positioned on the surface of Phobos by the MUSC, two water extraction units would deploy and begin operations. Several concepts for the water extraction units have been proposed by other parties[5]. The descriptions of the proposed water extraction concepts indicate a production capability sufficient to support the concept of a manned Mars program proposed by the M^3SG. The M^3SG has not examined the technical merit of the concepts in detail or adopted any particular concept. Phobos is classified as a carbonaceous chondrite meteorite of which up to 20% of the mass of its surface material may be recovered as water. A proposed concept for a water extraction unit on Phobos is shown in Figure 7. The robotic water extraction system is autonomously mobile and uses microwave or low power lasers to vaporize the hydrated phyllosilicate components in the surface material. The resulting water vapor is captured in the chambers surrounding the heat sources and pumped into the onboard water storage tanks. Once filled, the water extraction unit would

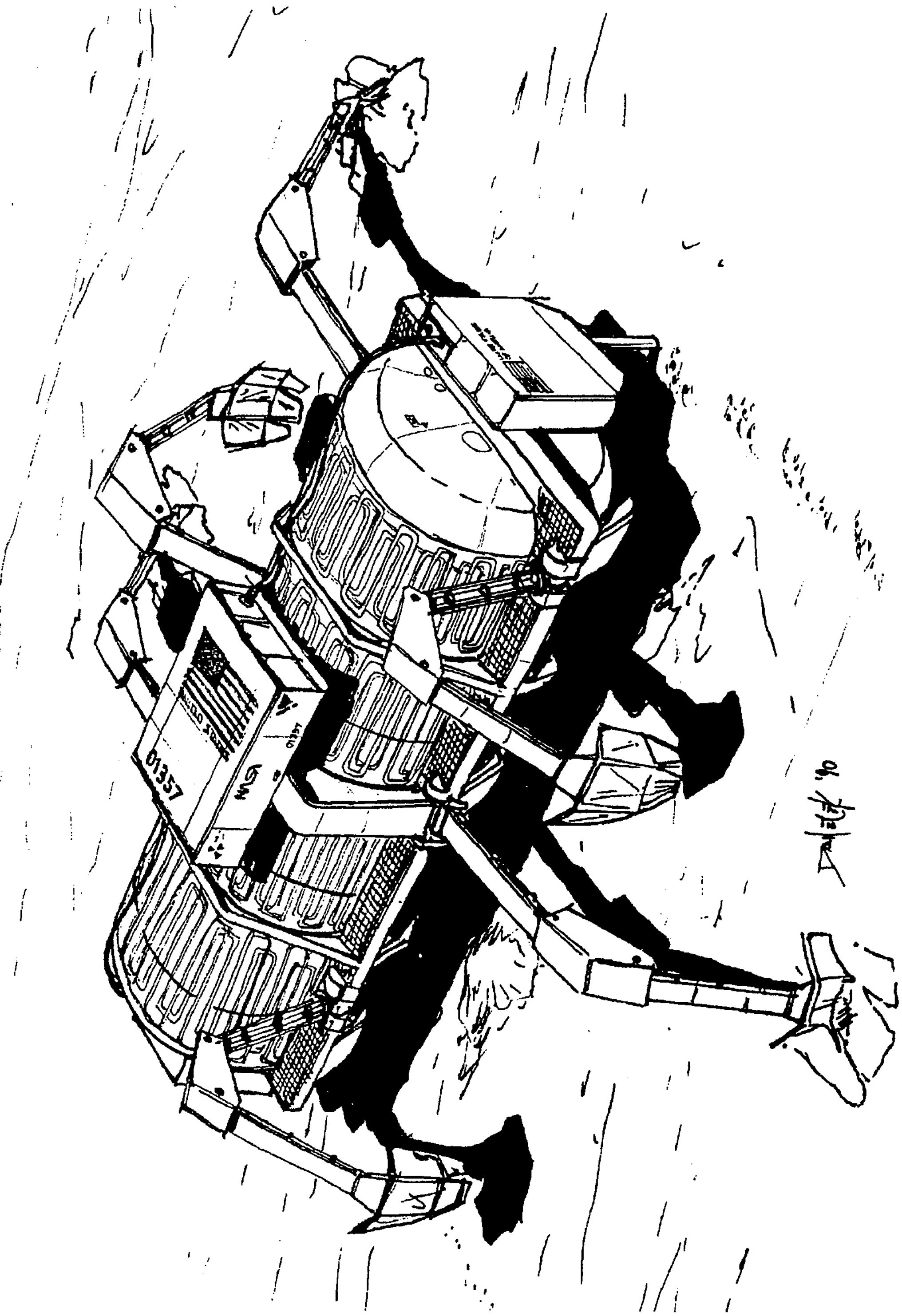

Figure 7 Mobile Water Extraction Unit on Phobos

transport the water back to the propellent depot, transfer the water into a storage tank and then resume operations in a different area. One of the cargo containers would be configured as an electrolysis plant to separate the water into hydrogen and oxygen. Cyrogenic cooling pumps would liquefy the gases and transfer them into storage tanks. SP100 units would generate the electrical power for the electrolysis unit and cooling pumps. Heat from the SP100 units would be used to keep the water from freezing in the tanks. This activity would continue until all tanks are full at the propellent depot. The empty LH_2 propellent tanks on the interplanetary spacecraft that were expended during earth escape and Mars capture would be removed and used as storage tanks for water/ice or LH_2 and LOX extracted on Phobos. The cargo containers would also be configured for reuse as water storage tanks. Each item in the system would generate telemetry for transmission back to earth in order to monitor operational status. The in-situ production and storage of propellant and the capability to refuel the interplanetary spacecraft at Phobos must be confirmed prior to launch of the next mission.

<u>Second Freighter On-Orbit Activity</u>. After arrival in orbit around Mars the two Mars Cargo Lander (MCL) spacecraft would detach from the interplanetary spacecraft park in orbit on Phobos for later deployment to the Mars base during the first manned mission. The Mars Utility Spacecraft (MUSC) would exit the docking port in the center of the aerobrake shield, attach to individual payload elements and push the cargo containers and five empty LH_2 propellant tanks to the surface of Phobos. After completion of the cargo off loading activity the MUSC would park in orbit on Phobos in order to refuel for activity during the next mission.

MANNED MISSION SCENARIO

Assembly of a second interplanetary spacecraft at Space Station Freedom in Low Earth Orbit (LEO) or at a construction facility located at the L1 LaGrange point would precede the first manned mission. The launch operations and sequence of activities necessary to prepare the spacecraft are described in Appendix I. Prior to departure for Mars, a thorough shakedown cruise around the Moon and Earth would be conducted to checkout the spacecraft systems and verify operational procedures of the flight and ground crews.

<u>Interplanetary Flight Profile</u>. Departure from the Earth Orbit begins with the Trans-Mars Injection (TMI) burn. The entire crew would man the control workstations inside the command module during periods of acceleration and deceleration. As a safety precaution, the crew would not leave the command center during periods of NERVA activity. Main engine cutoff of the NERVA nuclear thermal propulsion unit occurs after thirty to sixty minutes of constant acceleration at 0.2 g's. After the TMI burn, the interplanetary spacecraft rotates in order to position the aerobrake shield towards the Sun. The interplanetary spacecraft is then spun about the core axis to create artificial gravity for the crew now in the habitat modules. Figure 8 shows the configuration of the interplanetary spacecraft during the manned flight. Elevators are used for transport between the habitats and the command module in the center. Fluid masses are moved to counterbalance the precessional motions induced when the elevators move within the support trusses. The gravity can be gradually decreased from 0.9 g during the cruise to Mars by changing the spin rate of the vehicle in order to acclimatize the crew to the 0.38 g Martian gravity.

After the transit phase of the flight the spin of the interplanetary spacecraft is gradually slowed down to zero velocity and the spacecraft is oriented for deceleration. Once the spacecraft is secured for thrust, it is brought to the velocity necessary for the aerocapture maneuver into a stable orbit around Mars. The duration of the Mars Orbital Injection (MOI) burn depends on the specific geometry of the planetary approach, and on the amount of velocity that can be safely lost during the aerocapture maneuver. The aerobrake shield is then pointed forward and used to slow the spacecraft by passing through the Martian atmosphere like a meteor. This maneuver is a high g pass which slows the spacecraft into an elliptical capture orbit. After the interplanetary spacecraft has stabilized its orbit around Mars, the crew can enter the MUSC and depart for the Mars surface. The inbound return to Earth mission sequence is the reverse of the one described above, with a second aerocapture into Earth orbit.

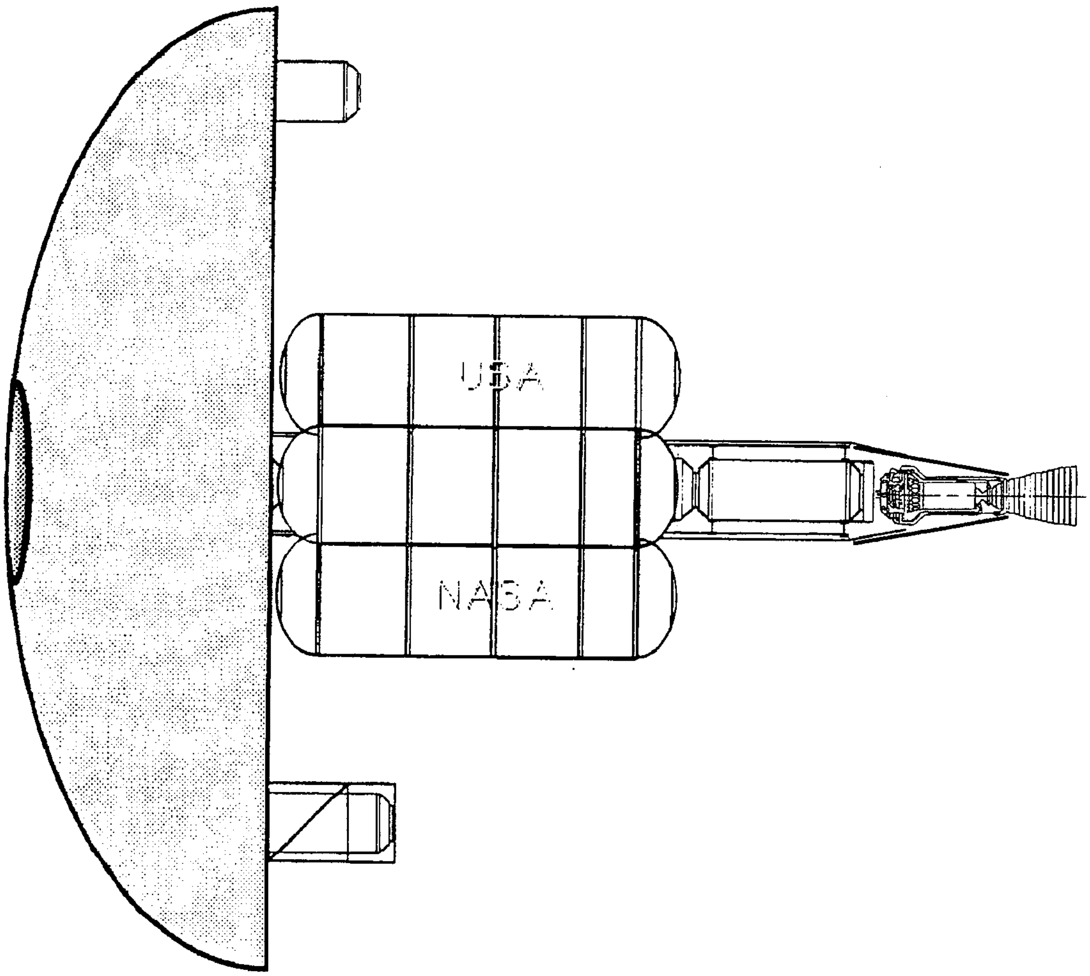

Figure 8 Side View of Interplanetary Spacecraft During Flight

Two crew members would remain onboard the interplanetary spacecraft in Mars orbit to provide backup support and maintain spacecraft functions. The availability of the interplanetary spacecraft in Mars orbit provides a safety measure in case of a mishap at the Mars Base. The crew could be swiftly evacuated to safety on the interplanetary spacecraft. During the second phase of the M^3 program after the Mars Base operations stabilize and are considered reliable, the interplanetary spacecraft would not remain in Mars orbit for long periods.

Crew Requirements. A crew consisting of at least ten and preferrably twelve members is considered appropriate for the mannned missions during the first phase of the Mars program. Analysis of activities on board the interplanetary spacecraft and on the surface of Mars in conjunction with analysis of cross training capabilities determined a crew size slightly larger than previously assumed. At least seven personnel with support and service type skills are required to operate and maintain the interplanetary spacecraft and Mars base systems. At least three personnel with scientific research type skills are needed to perform the numerous investigations. A suggested scheme for cross-training is described in Table 5. Manning requirements for around the clock watch on system controls would involve a schedule for shift work. Other psychological factors were also considered in determining a minimum crew size. For example, a variety in social interactions with more diverse personalities appears to be an essential ingredient to surviving long periods in confined quarters.

Table 5 Minimum Crew Requirements and Proposed Cross-Training

	Number	Primary Skill	Secondary Skill
Support and Services	2	Mechanical Specialist	Pilot
	2	Electronics Specialist	Computer Specialist
	1	Doctor	Organic Chemist
	1	Dentist	Medical Technician
	1	Horticulture Specialist	Chef
Scientific Research	1	Geologist	Mineralogist
	1	Meterologist	Astronomer
	1	Physicist	Inorganic Chemist
	2	Unassigned	

ACTIVITY ON THE SURFACE OF MARS

After achieving a stable orbit around Mars, the crew members of the first and second manned missions would begin to move material to the surface of Mars by teleoperating the MCL spacecraft from the orbiting interplanetary spacecraft. Later the crew members would descend in groups in the MUSC to the location on the surface where the materials for assembly of the Mars base were offloaded. The first priority would be to assemble the first increment of the Mars base configuration in order to provide shelter until the Mars base can be completed. During the first few days the Mars Explorer Vehicle would provide temporary housing until the first few habitat modules became operational.

Surface Activity Tasks During First Manned Mission. During the two months of activity on the surface of Mars in the first manned mission, the following tasks would be undertaken to accomplish mission objectives and prepare for future missions:

1. Test a limited capability in Closed Ecological Life Support Systems (CELSS) which would reclaim air and water, supply some food and process waste material;

2. Set up sensor stations for meteorology studies;

3. Monitor the health of the astronauts in the Martian environment and perform medical experiments;

4. Explore key areas of interest near the Mars base in order to study geological features and find sources of water, such as permafrost and underground aquiferous formations;

5. Establish autonomous water extraction operations to supply the Mars Base. Two water extraction systems would be mounted on specially configured MUVs.

Surface Activity Tasks During Second Manned Mission. During the thriteen months of activity on the surface of Mars in the second manned mission, the following tasks would be undertaken to accomplish mission objectives and prepare for future missions:

1. Start materials processing experiments which would operate autonomously between manned missions and produce metals, chemicals and volatiles;

2. Explore key areas of interest farther from the Mars base in order to study geological features and to find mineral-rich ore deposits, from which useful materials can be extracted.

Local Shuttle Spacecraft. Two different shuttle spacecraft concepts are considered necessary to support activity on the surface of Mars. Dependence on a single common design to provide all of the required functions would compromise the potential efficiency in propellent consumption and increase operational risks. The long term benefits from the flexibility in the uses of the two spacecraft concepts appears to out weigh the savings in development costs for a single spacecraft concept. The following concepts would provide maximum reusability of the spacecraft well into the second phase of the Mars program.

1. The design of the Mars Cargo Lander (MCL) is derived from that of the NSTS but with major differences to support its role on Mars. The MCL is a non-man rated spacecraft in order to maximize its cargo capability and only operates between Mars orbit and the surface of Mars. The MCL could be teleoperated from the interplanetary spacecraft or could also operate in autonomous flight control mode. The MCL would be powered by three small NERVA engines which could also use carbon dioxide extracted from the Martian atmosphere as propellent. A carbon dioxide extraction unit on board the MCL would be used to refill the propellent tanks. The NERVA engines must be isolated from each other by distance and shielding to prevent harmful effects on the control electronics. The cargo compartment of the MCL would be sized and shaped to accommodate the habitat modules of the Mars base, approximately 14 meters in length, 5 meters in width and 5 meters in heighth. The MCL must be capable of carrying mass of up to 18,500 kg from Mars orbit to the surface. The MCL would fly in a manner similar to the NSTS on Earth, but the wings and rudder are not effective during subsonic flight due to the thin atmosphere of Mars. The MCL would operate primarily as a Vertical Take-Off and Landing (VTOL) vehicle but also has horizontal flight capability. At least four and possibly six vertical thrust nozzles would be located on the underside of the MCL and only one horizontal thrust nozzle at the rear. A Mars Utility Vehicle (MUV) would be unloaded from an MCL as shown in Figure 9.

2. The Mars Utility Spacecraft (MUSC) would be used primarily to transport astronauts between the interplanetary spacecraft and the Mars base. The MUSC is a man rated spacecraft and requires extensive backup controls and life support subsystems. The MUSC operates not only between Mars orbit and the surface of Mars but also has a trans-orbit capability in order to reach Phobos and Deimos. The MUSC could be teleoperated from the interplanetary spacecraft, flown independently by the astronaut crew or could also operate in autonomous flight control mode. The MUSC would be powered by a small NERVA engine and would also carry a carbon dioxide extraction unit in order to refill its propellent tanks on the surface of Mars. The NERVA engine must be isolated from the crew compartment by heavy shielding to prevent harmful effects on the crew and control electronics. The top of the MUSC is configured somewhat like the Apollo capsules but could seat up to five crew members. The MUSC would stand approximately 12 meters high and about 9 meters wide. The MUSC would take-off and land vertically as shown in Figure 10. At least six and possibly eight vertical thrust nozzles would be located on the bottom of the MUSC.

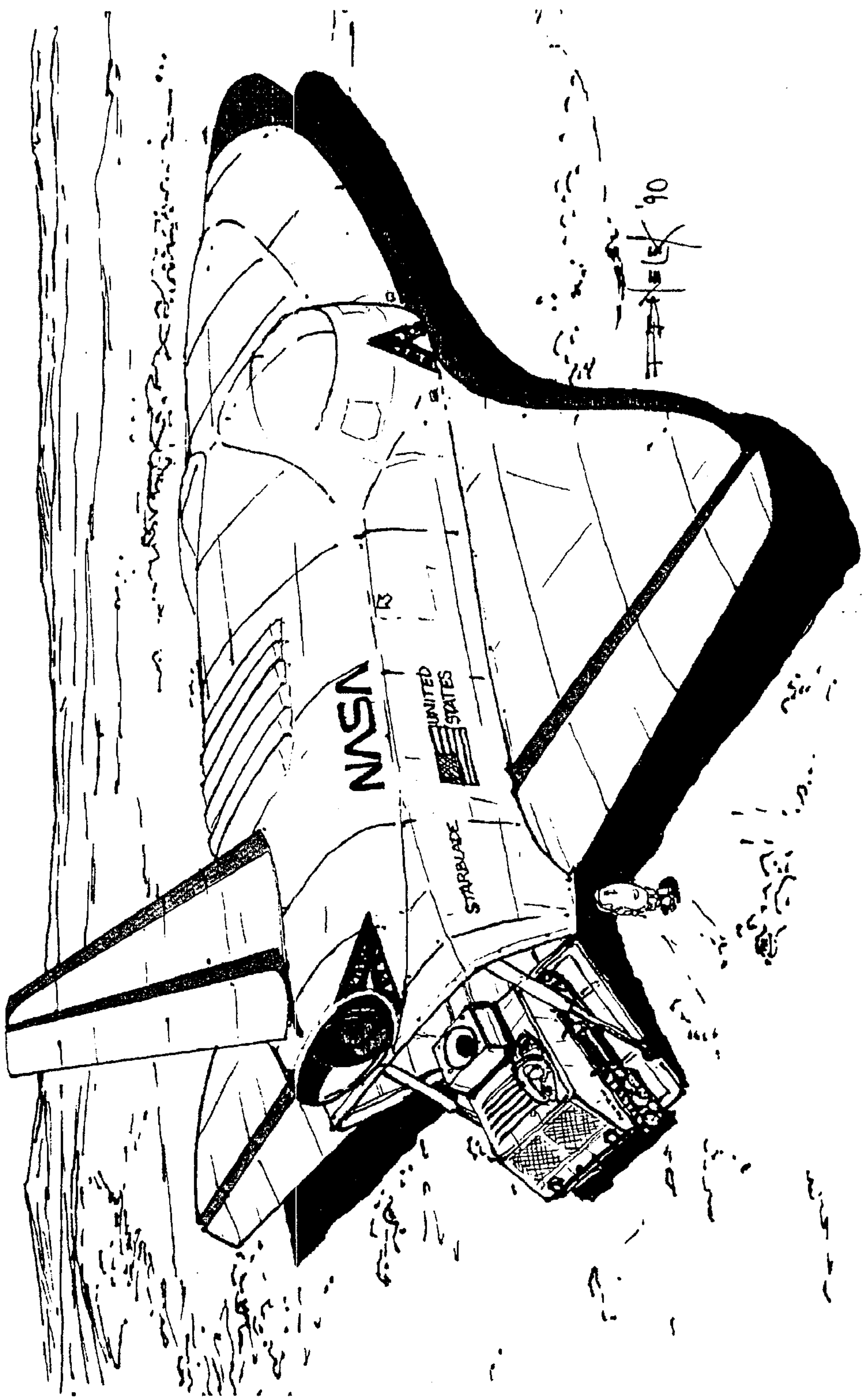

Figure 9 MUV Exiting a Mars Cargo Lander

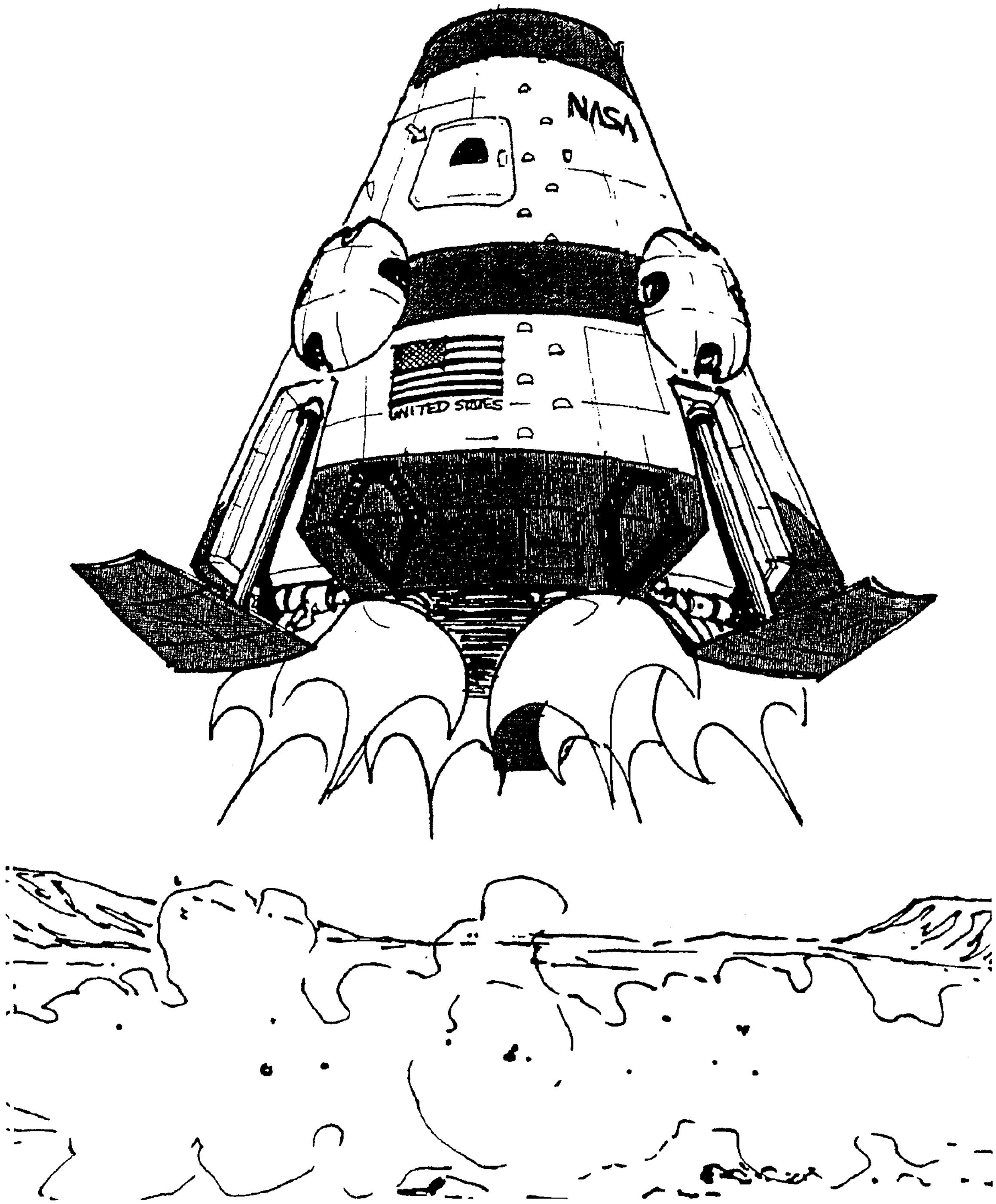

Figure 10 MUSC Landing at the Mars Base

Vehicular Systems. Two different surface vehicle concepts are considered necessary to conduct activity on the surface of Mars. Dependence on a single common design to provide all of the required functions would compromise the potential efficiency in fuel consumption and increase maintenance problems. The long term benefits from the flexibility in the uses of the two vehicle concepts appears to out weigh the savings in development costs for a single vehicle concept. The following concepts would provide maximum reusability of the vehicles well into the second phase of the Mars program.

1. The design of the Mars Utility Vehicle (MUV) is derived from that of the NSTS but with major differences to support its role on Mars. The MUV functions as a construction tractor and robotic servicer platform. The MUV could be teleoperated from a control workstation inside the Mars Base or on the interplanetary spacecraft. It could also operate autonomously or under the control of a human operator. The MUV is a man rated system and requires adequate backup controls and life support subsystems. Two astronauts could be accommodated in the MUV in a shirt sleeve environment. The astronauts must be able to enter or exit their spacesuits easily in the small crew compartment of the MUV. The MUV would be powered by fuel cells and batteries providing electricity for the various motors and actuators. The front and rear mount assemblies on the MUV must accommodate the interchange of a variety of tool attachments, such as, a scraper blade, backhoe and telerobotic servicer. The MUV would crawl on tracks in a manner similar to a bulldozer on Earth, but must be designed for reliable operation and low maintenance. An MUV would clear the site for the Mars Base as shown in Figure 11.

2. The Mars Explorer Vehicle (MEV) would be used primarily to transport astronauts and provide mobile quarters during relatively long exploration expeditions away from the Mars base. The MEV is a man rated system and requires adequate backup controls and life support subsystems. Teleoperation and autonomous control modes are not considered necessary for the MEV. As many as four astronauts could be accommodated in the MEV in a shirt sleeve environment. An airlock compartment would be required to permit entry and exit for the crew. The MEV would be powered by fuel cells and batteries providing electricity for the various motors and actuators. The front and rear mount assemblies on the MEV must accommodate the interchange of a variety of tool attachments, such as, a scraper blade, backhoe and manipulator. The MEV would operate on wheels similiar to those of the lunar rover , but must be designed for reliable operation and low maintenance in remote areas of Mars. An MEV is shown in Figure 12.

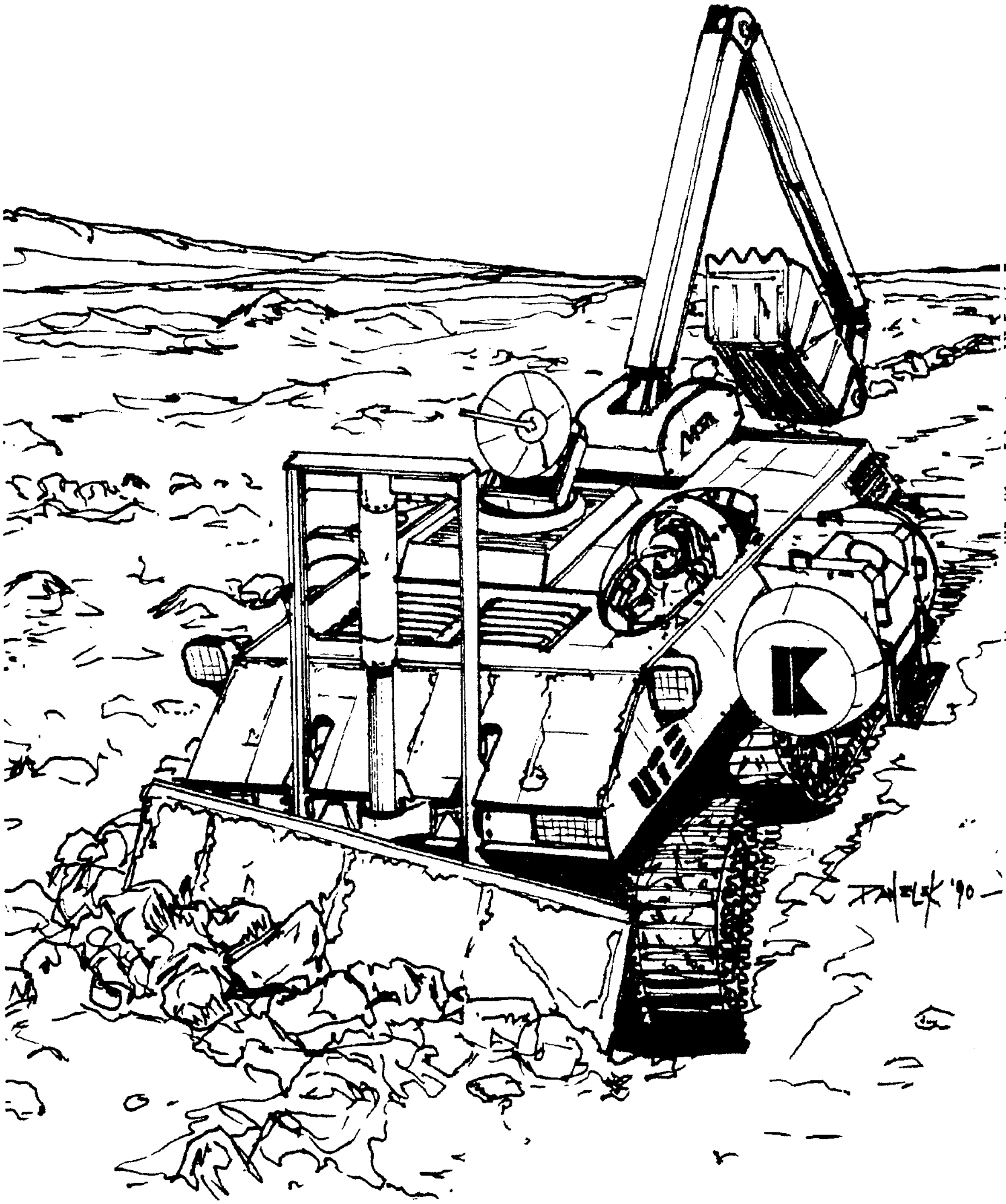

Figure 11 MUV Preparing Site for Base Construction

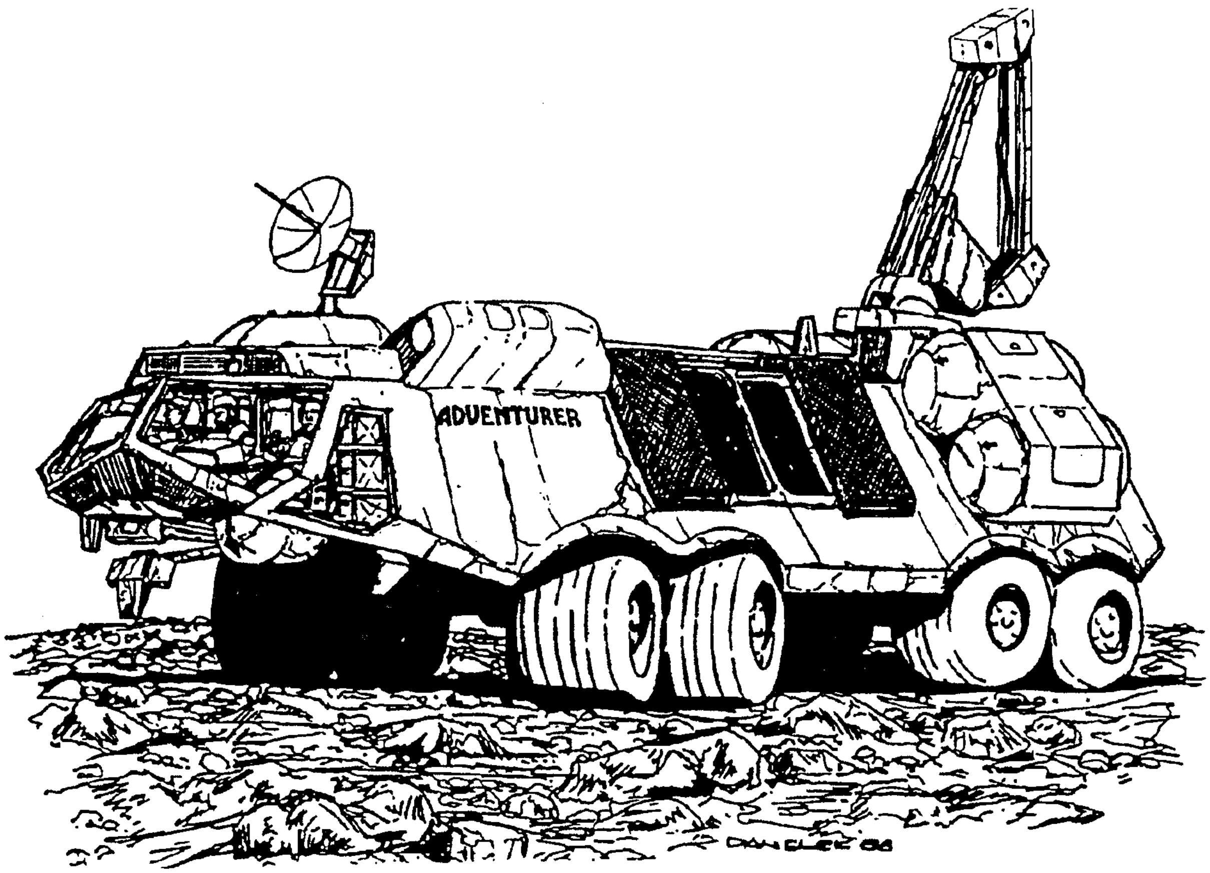

Figure 12 Mars Exploration Vehicle (MEV)

Base Assembly. The first manned mission would layout the Mars base at the location surveyed by robotic rovers during precursor missions. The site of the base should be free of large obstacles, relatively flat and in the vicinity of Olympus Mons and Vallis Marineris. The following locations are considered possible sites for the Mars Base:

1. North of Tithonium Chasma at approximately 0° North/South and 90° West,
2. West of Biblis Patera at approximately 5° North and 140° West,
3. Between Syria, Sinai and Solis Planum at approximately 20° South and 95° West,
4. North of Jovis Tholus at approximately 25° North and 120° West.

The MUV would assist in extracting the payload from the MCL near the base site, positioning the modules into the base configuration and then covering the modules with Martian soil as shown in Figure 13. The SP100 electrical power generation units would be positioned at least 200 meters away from the base elements. The MUV would construct a berm of Martian soil between the power plant and the base to help shield the base from any radiation emitted from the SP100 units. The waste processing units would also be located a short distance from the base elements to avoid the risk of methane and other contaminants leaking into the habitat modules. The first MCL to land at the site of the Mars Base would carry an MUV which would clear obstacles from the landing area to prepare the area as a shuttle port for the MUSC and MCL spacecraft. The second MCL to land at the site of the Mars Base would carry the MEV in which the crew members would live during the first few days until enough of the base components are installed to achieve an IOC.

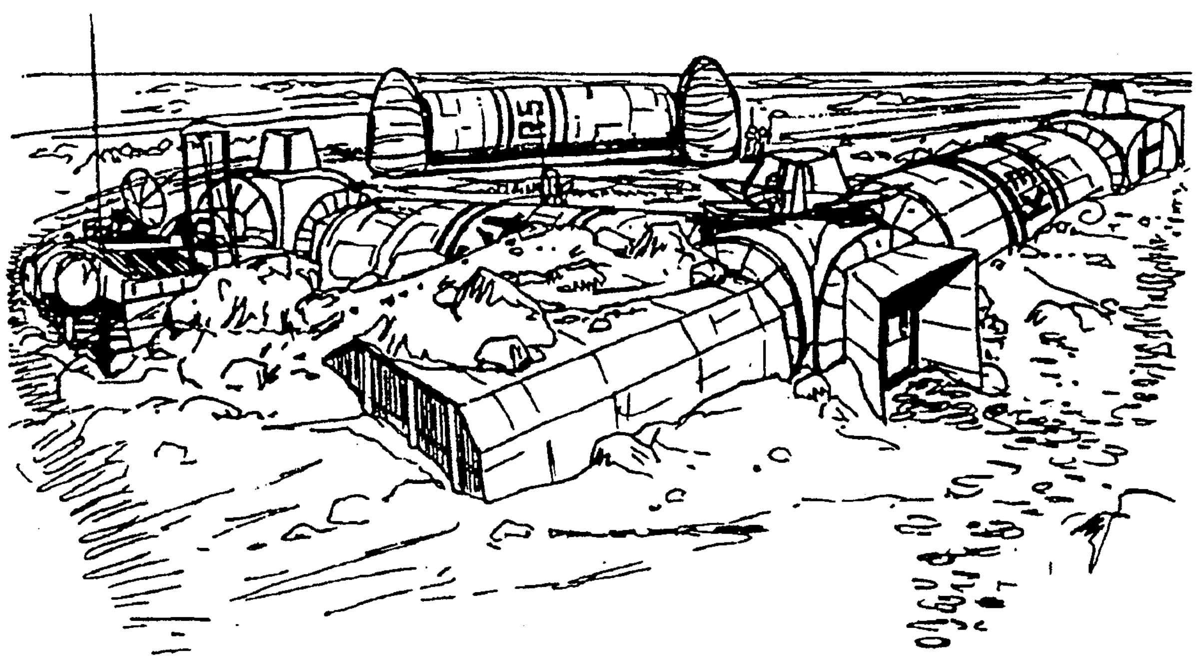

Figure 13 Construction of the Mars Base

Base Configuration. The design of the habitat modules is derived from the design of Space Station Freedom modules. The habitat modules are the main operational elements of the initial base configuration. Six habitat modules would be rolled into the partial configuration of a hexagon using large inflatable tires embedded in the ends of the modules. The hexagon layout of the base shown in Figure 14 is expandable and satisfies the safety requirement for two exits in each module. The six habitat modules, experimental greenhouse modules, waste recycling unit and power generation module of the base configuration would provide comfortable living and work quarters in addition to all of the life support requirements for the surface crew of ten astronauts during the first manned mission. During the second manned mission six additional habitat modules would be integrated into the hexagon shaped layout in order to complete the first phase configuration of the base. During the second phase of the Mars program the base could be expanded by erecting inflatable modules between and above the first layer of habitat modules. In-situ produced materials could be used later to construct additional habitable areas.

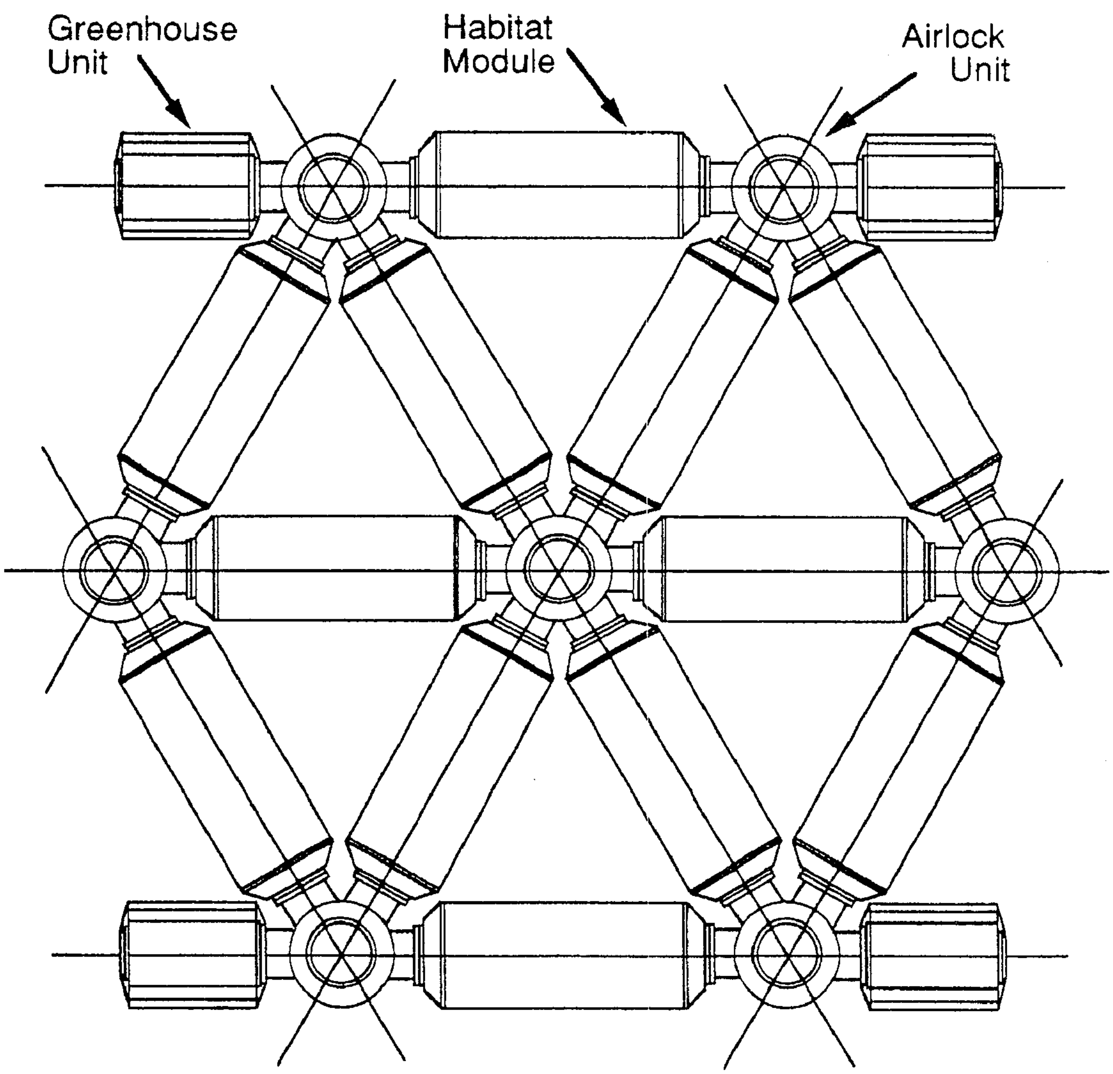

Figure 14 Layout of Mars Base Habitat Modules

Manned Exploration. Manned exploration activity would involve the following scenarios:

1. The MUV would be used by a two man crew for one day excursions in the area of the base site.

2. During the second manned mission the Mars Exploration Vehicle (MEV) shown in Figure 14 would be used by a three man crew for extended excursions to more remote areas from the base site. An MCL could be used to position the MEV in the area for exploration. Another MCL carrying supplies for the MEV could be parked in orbit and would be deployed to the exploration site when needed. A MUSC could provide transportation for crew changeover enroute.

3. The MUSC would provide transportation to remote areas inaccessible to vehicles, such as the volcano Olympus Mons and the basin Hellas Planitia in the southern hemisphere.

Unmanned Exploration. For situations of higher risk and hazard, unmanned exploration activity would involve the following scenarios:

1. An MUV could be teleoperated from a remote workstation at the base or onboard the interplanetary spacecraft in orbit. Communications with the MUV would be relayed through satellites or the interplanetary spacecraft. An MCL could transport a MUV to remote locations.

2. Several MUVs would be configured as autonomous water extraction systems and would operate near the area of the Mars Base. Well drilling booms or other devices suitable for extracting water from permafrost would be attached to the MUVs or trailers. The MUV would pull a trailer with a water storage tank and detachable SP100 electrical power plant. A telerobotic servicer mounted on the front of the MUV may also be required to operate and maintain the system.

3. An MCL would transport the Tethered Canyon Explorer (TCE) and MUV shown in Figure 15 to the rim of the canyon Vallis Marineris. The six legged walking robot would be attached by a tether deployed from a reel mounted on a specially configured MUV. The MUV would transport the TCE to various departure points along the rim of the canyon. Fiber optic cable in the tether would provide communications for teleoperation from the base or interplanetary spacecraft. The tether would also provide a recovery capability if autonomous operation or teleoperation failed to avoid impassable terrain or prevent a mishap.

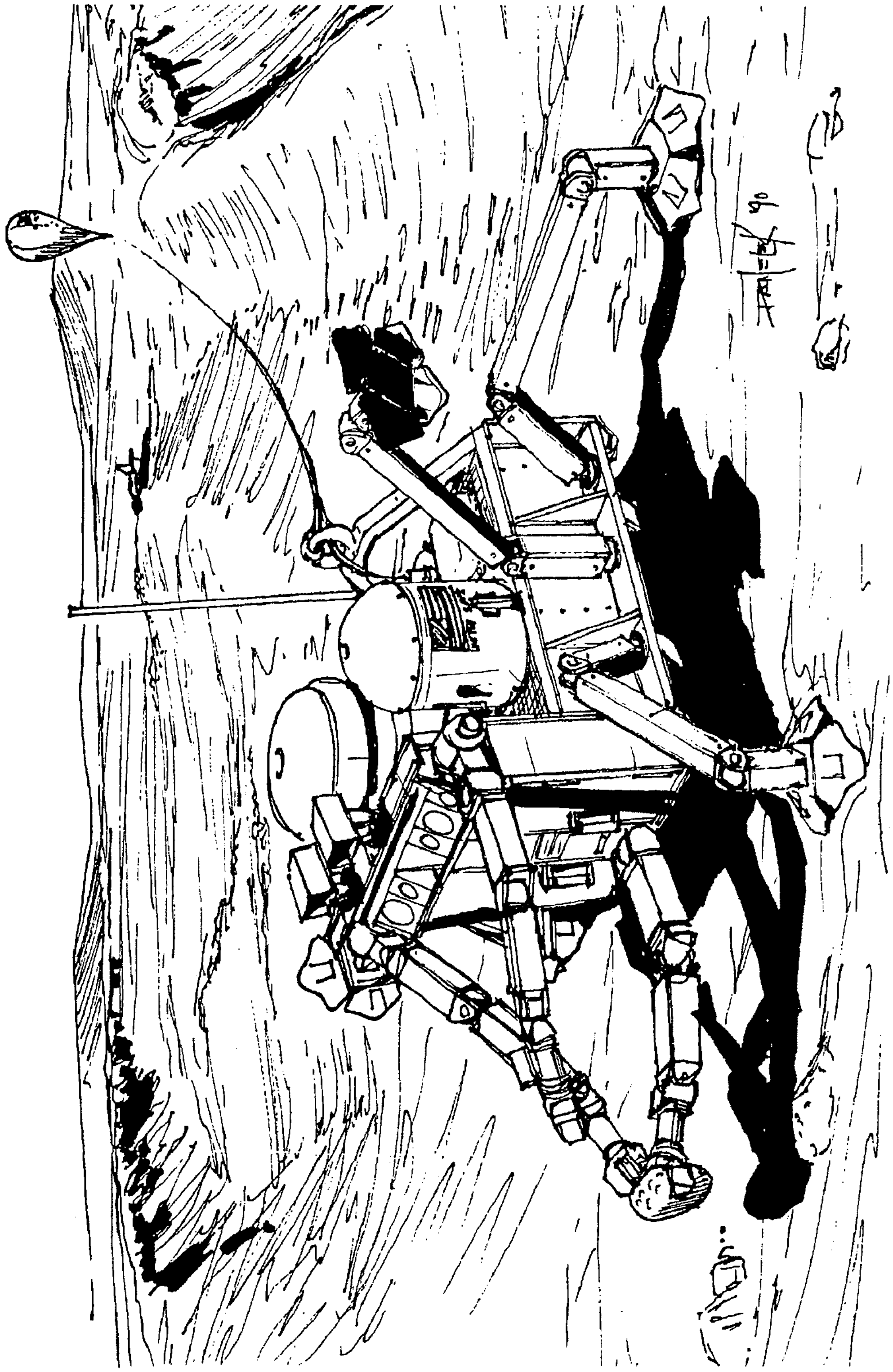

Figure 15 Tethered Canyon Explorer (TCE)

LIFE SCIENCES

The crew is the most important payload on manned missions to the planet Mars. The foremost consideration of the biologists, psychologists, engineers and physicians in the Life Sciences Team of the Manned Mars Mission Study Group is to ensure that the design of each system and the mission planning provide for the safety, well-being, survival and healthy return of the men and women in the crew.

Microgravity. Weightlessness during prolonged manned missions in space leads to fluid and electrolyte imbalances, general physical deconditioning and physiological deconditioning of cardiovascular, musculoskeletal, metabolic and neuroendocrine systems. One of the first issues the team confronted was how to deal with the microgravity of space and the 0.38 g gravity on Mars. There are more unknown than known facts about the effects of extended microgravity. Published information is very limited about alternatives and methods of management. The proposed scenario for the first manned mission would require a crew to spend a total of approximately 14 to 22 months in spaceflight traveling to and from Mars and up to 13 months on the surface of Mars. The primary known effects of exposure to prolonged microgravity over long periods are bone demineralization, muscular atrophy, circulatory stasis and cardiac degeneration. None of these effects would cause serious discomfort while remaining in space because adaptation to microgravity is similar to the sensations of walking or running downhill. The problems appear when the crew is required to adapt to living on the surface of Mars. The problems worsen when the crew returns to the surface of Earth. The stress of adapting to increased gravity is like walking or running uphill and can be exhausting, even debilitating.

Immunosuppression. Another possible effect of microgravity on humans may be immunosuppression postulated during Spacelab missions. Spaceborne studies of microgravity effects on T-Lymphocytes demonstrated inconsistent instances of T-Lymphocytes growth which suggests immunosuppression. Immunosuppression may present a serious effect on humans. If severe, it could allow even friendly and vital microorganisms to become deadly to the crew. This effect is magnified by the potential growth of pathogenic microorganisms in closed environments. These potential pathogens in the microgravity environment will form free-floating aerosols which can be easily inhaled and tend to collect and concentrate in air filtration and water vapor condensation components of life support systems.

Alternative Measures. Man's capacity for adaptation coupled with appropriate countermeasures may make long-duration missions tolerable. The debilitating effects of microgravity need to be counteracted while adequately adapting the crew to each phase of activities during the flight to Mars and the return flight to Earth. Four known alternative measures for counteracting the effects of microgravity are exercise, artificial gravity, electromagnetic stimulation and drug therapies. The only tried measure is exercise, but that has proven only marginally effective. All of the methods have negative aspects, but artificial gravity provided by centrifugal force seems to create the least detrimental effects. The most significant limitation of centrifugal force is the sensitivity to inertia on the endolymph in the semicircular canals of the human inner ear. This coriolis effect is experienced when moving perpendicular to the axis of rotation as shown in Figure 16 or when turning the head in a rotating environment as shown in Figure 17. Artificial gravity can provide effective prophalaxis against all of the effects of microgravity and is currently the most understood measure. The study group determined that artificial gravity produced by centrifugal force would be not only practical but also essential to the functional design of the interplanetary spacecraft in order to maintain the health of the crew during the long spaceflights. During activity on the surface of Mars, limited deconditioning of the crew's health could be avoided by providing adequate countermeasures to the lower level of gravity. Exercise, electromagnetic stimulation during sleep and potential drug therapies may suffice to maintain crew fitness for the return trip.

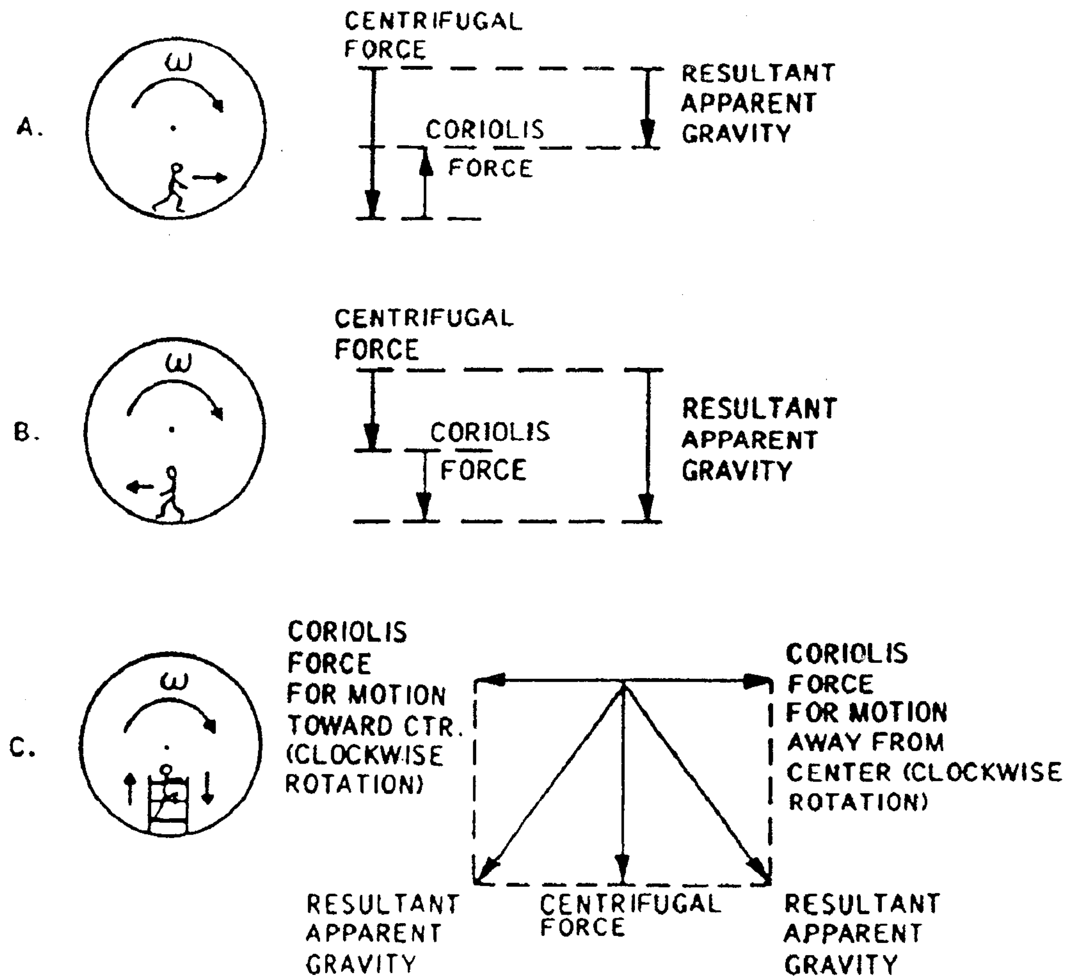

Figure 16 Coriolis Effect When Moving Perpendicular to the Axis of Rotation

Artificial Gravity. The interplanetary spacecraft would rotate about the center axis of its chassis to produce centrifugal force as shown in Figure 18. The crew would be housed in six pods paired at the edge of the aerobrake shield. The rotation of the structure would be capable of producing up to one g gravity at 6 rpm. The gravity level is determined by the rate of rotation and the length of the arms as shown in Figure 19. For optimal adaptation during the flight to Mars, the level of artificial gravity would decrease gradually from one g down to 0.6 g, but if that scenario proves to be impractical, a static level of 0.6 g could avoid the long-term effects of microgravity and prepare the astronauts for the 0.38 g gravity on Mars. For the return flight to Earth adequate adaptation could be achieved at a static level of 0.6 g. The ideal scenario for readapting the crew to Earth's level of gravity would increase the level of gravity gradually from 0.38 g to one g either on board the interplanetary spacecraft or on a space station providing artificial gravity. In order to reduce the additional cost and complexity of the artificial gravity structure on the interplanetary spacecraft, the following would be required:

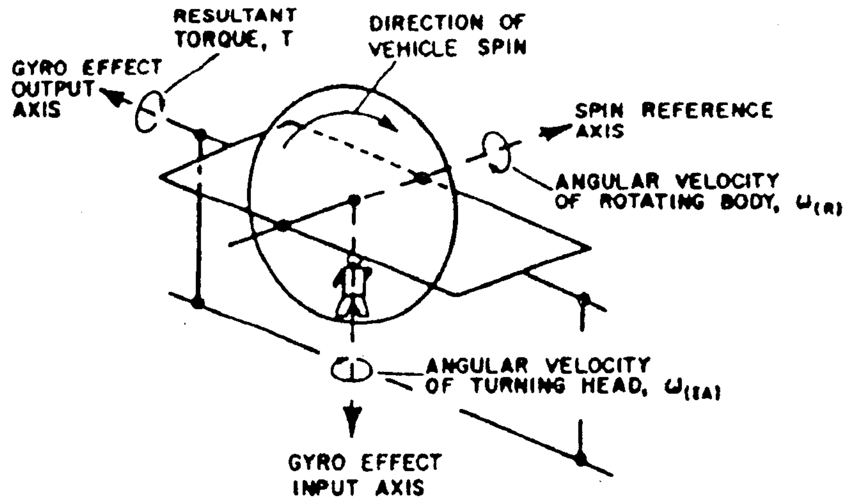

Figure 17 Gyroscopic Torque Effect on Semi-circular Canals When Turning Head in a Rotating Environment

1. More efficient continuous propulsive capability for sprint missions, probably requiring nuclear power,
2. More knowledge of the forces which control gravity,
3. More effective methods for adapting astronauts to other levels of gravity.

Radiation Protection. Onboard the interplanetary spacecraft extra shielding material would be embedded in the walls of the command module in order to provide protected shelter from periods of hazardous radiation caused by solar particle events. On the surface of Mars at least one of the preconfigured habitat modules will be buried under 5 to 10 feet of Martian soil in order to provide protected shelter from periods of hazardous radiation. Procedures have been suggested for astronauts to improvise emergency shelter from hazardous radiation during exploration on the surface of Mars. The most favored scenarios involve employment of the mass of the MUV and MEV to provide substantial shielding.

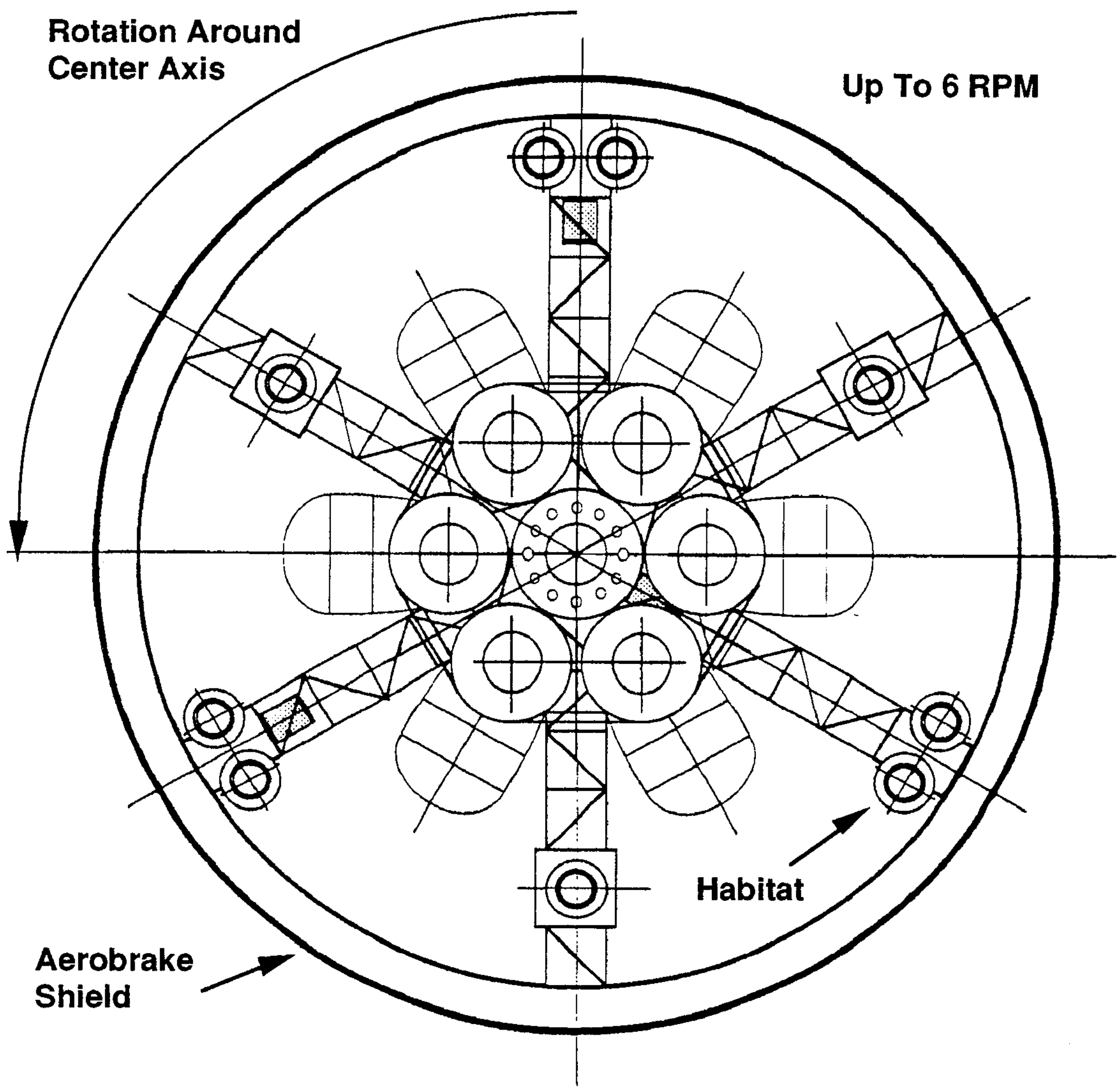

Figure 18 Rotation to Generate Artificial Gravity
(View from the Rear of the Interplanetary Spacecraft)

Psychological Factors Affecting Crew Performance. Cramped crew quarters, confined living spaces, limited recreational opportunities, the interminable distances and impossibility of rescue are all factors expected to place a great deal of psychological stress upon the crew members. These stresses negatively affect work performance and crew interactions. Moreover, the effects of these stresses will increase over the duration of the voyage. Therefore, psychologists and human factors engineers must be intimately involved in the overall system design to ensure the minimization of physical and psychological stressors in the entire system, from designing the hardware to planning crew member interactions. The design of living quarters must ensure privacy to allow crew members to "get away" from each other when desired. Activity centers must provide the greatest variety of both mental and physical activity possible within the constraints of spacecraft weight and size. In addition to entertaining the crew members, some activities will incorporate elements of training to keep their skills sharp, others will promote cooperation and teamwork, while still others will offer an outlet for competitiveness. The design of crew stations, tools and even the computer software must also involve the cooperation of psychologists and human factors engineers. Every

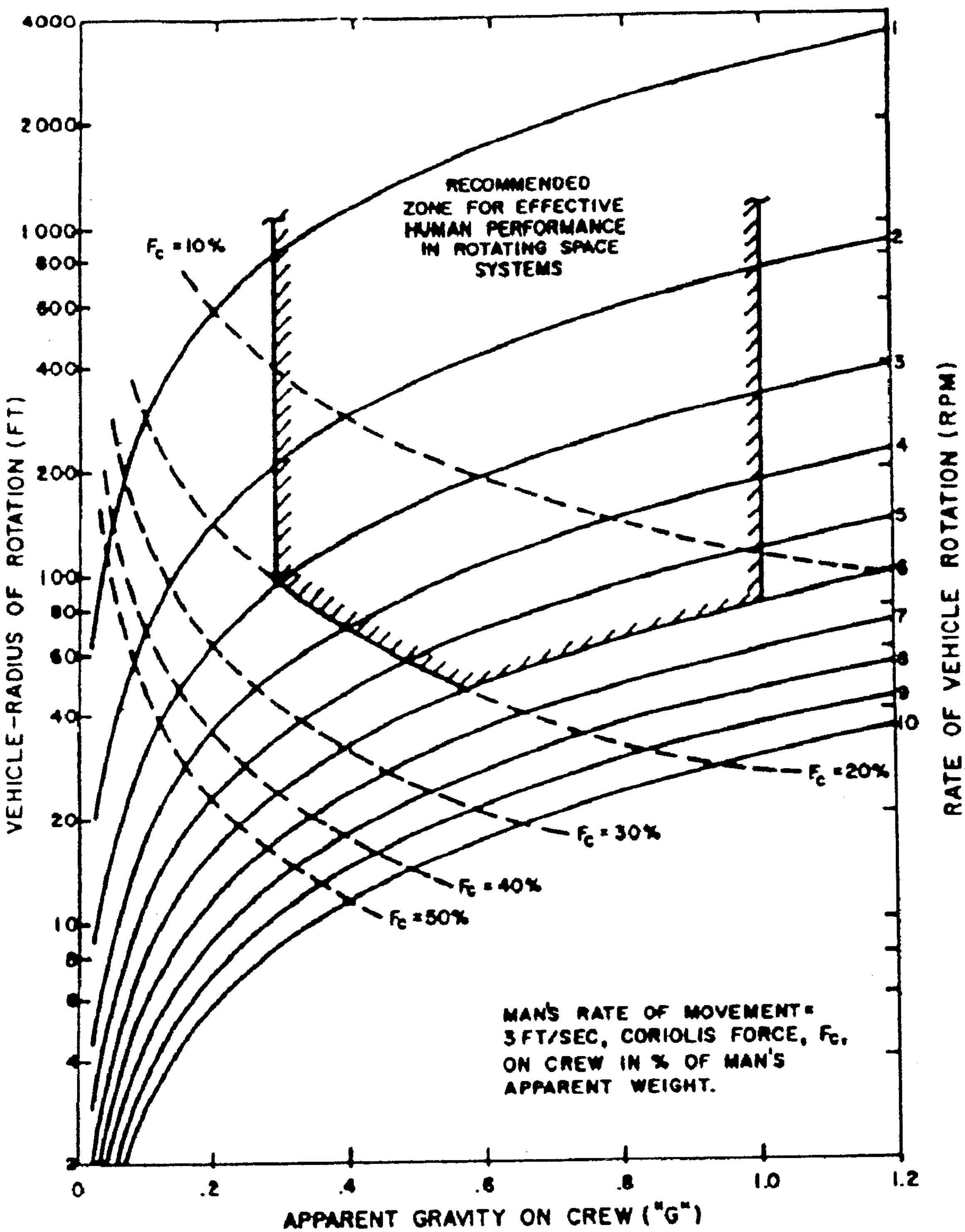

Figure 19 Design Criteria for Effective Human Performance in Rotating Space Systems

part of the overall system must operate easily and in the way the crew members expect, thereby producing as little frustration with the system as possible. Some design considerations include compatibility between task requirements and crew member abilities, the reduction of task related stress, optimal distribution of physical and mental workloads over time and between crew members and elimination of sources of human error.

Possible Losses. The age of the crew members on the first manned missions to Mars will probably range from 35 to 45 years old. Statistically, one or two crew members could die during the period of the mission even if they did not leave Earth. The inherent hazards of a trip to more than 60 million miles from Earth increases the risk. We must not only be willing to face the possible loss of astronauts but also be dedicated to making the journey as safe as reasonably possible.

Contamination. No direct contact with the Martian environment will be tolerated in order to protect the astronauts against contamination by yet unknown elements on Mars and to protect Mars against contamination from the astronauts. Quarantine procedures would be implemented during early missions until research provides assurance of safe environmental crossover and relaxation of protection requirements. Waste products would be decomposed, sterilized and prepared for reuse in a recycling system. Contaminants and toxins would have to be closely monitored and actively filtered from the enclosed environments in which the astronauts live and work. Potential changes in the microbial ecosystem could adversely impact food production systems, life support systems, waste processing systems, hygiene facilities and crew health.

Advanced Spacesuit. To efficiently perform activities on Mars a more durable and lightweight spacesuit will be required in comparison to the spacesuit used on the moon or in space. The Mars spacesuit will need to withstand abrasion and provide greater freedom of movement and superior dexterity for the astronauts working and exploring outside the habitat modules. The helmet visor must withstand violent dust storms and protect the eyes from lasing of the atmosphere. The spacesuit must be pressurized to avoid caisson's disease and must maintain a hermetic seal against contamination even during rugged use. The suit must be relatively easy to enter and exit. A concept for the Mars spacesuit is shown in Figure 20.

Medical Facilities. Both the interplanetary spacecraft and Mars Base would be equipped with medical clinics capable of dealing with emergencies. The clinics would also be equipped for performing major surgical procedures and some dental procedures. A concept for the medical clinic is shown in Figure 21. The crew should be adequately trained to perform and assist in these procedures. A communication link to Earth and a global resource network of medical teleconsultants could assist when necessary. Diagnostic equipment comparable in function to that on Earth but much more compact is required to analyze body fluids and perform hard and soft tissue body scans. The monitoring and maintenance of the health and safety of the crew is crucial to the success of the mission.

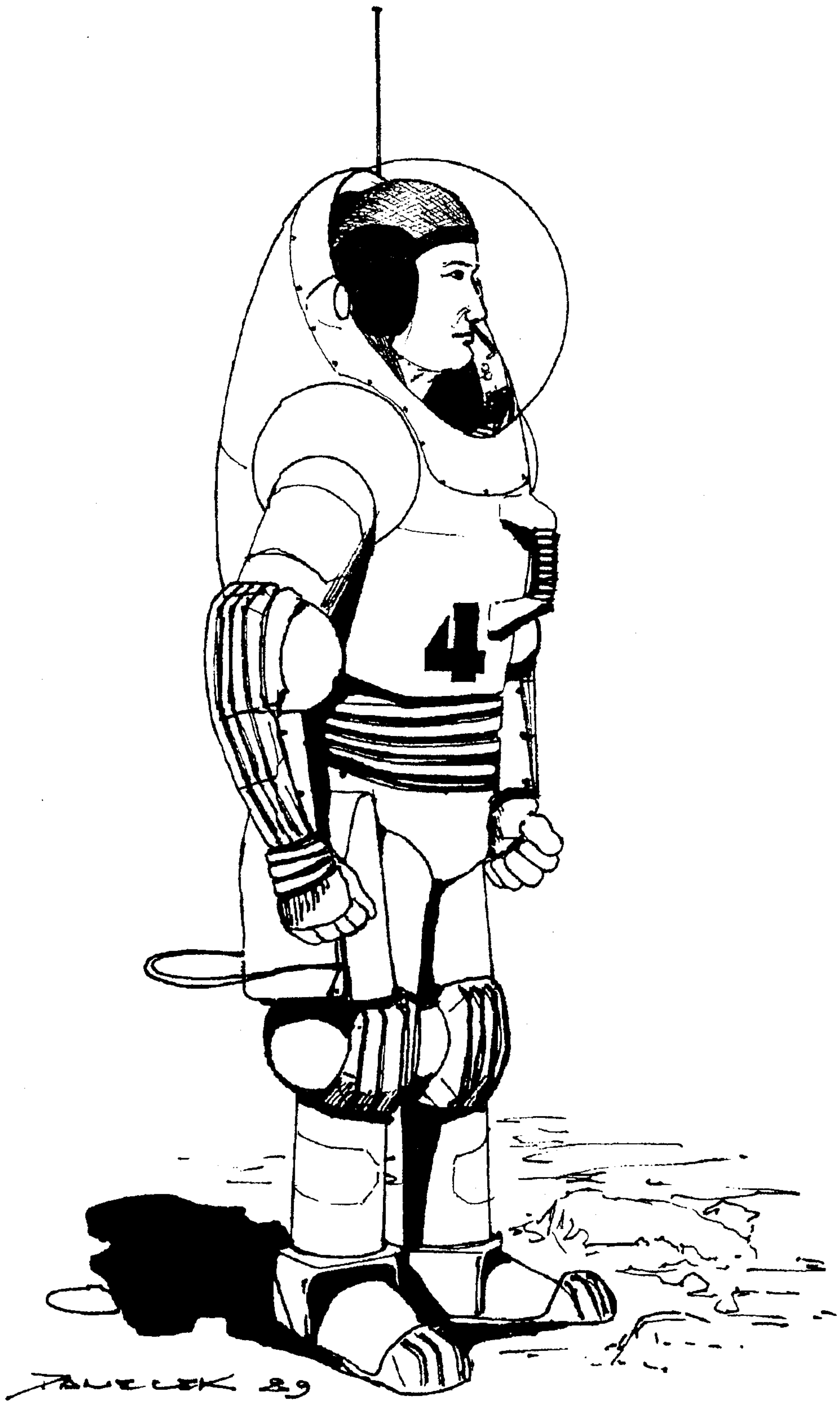

Figure 20 Mars Spacesuit Concept

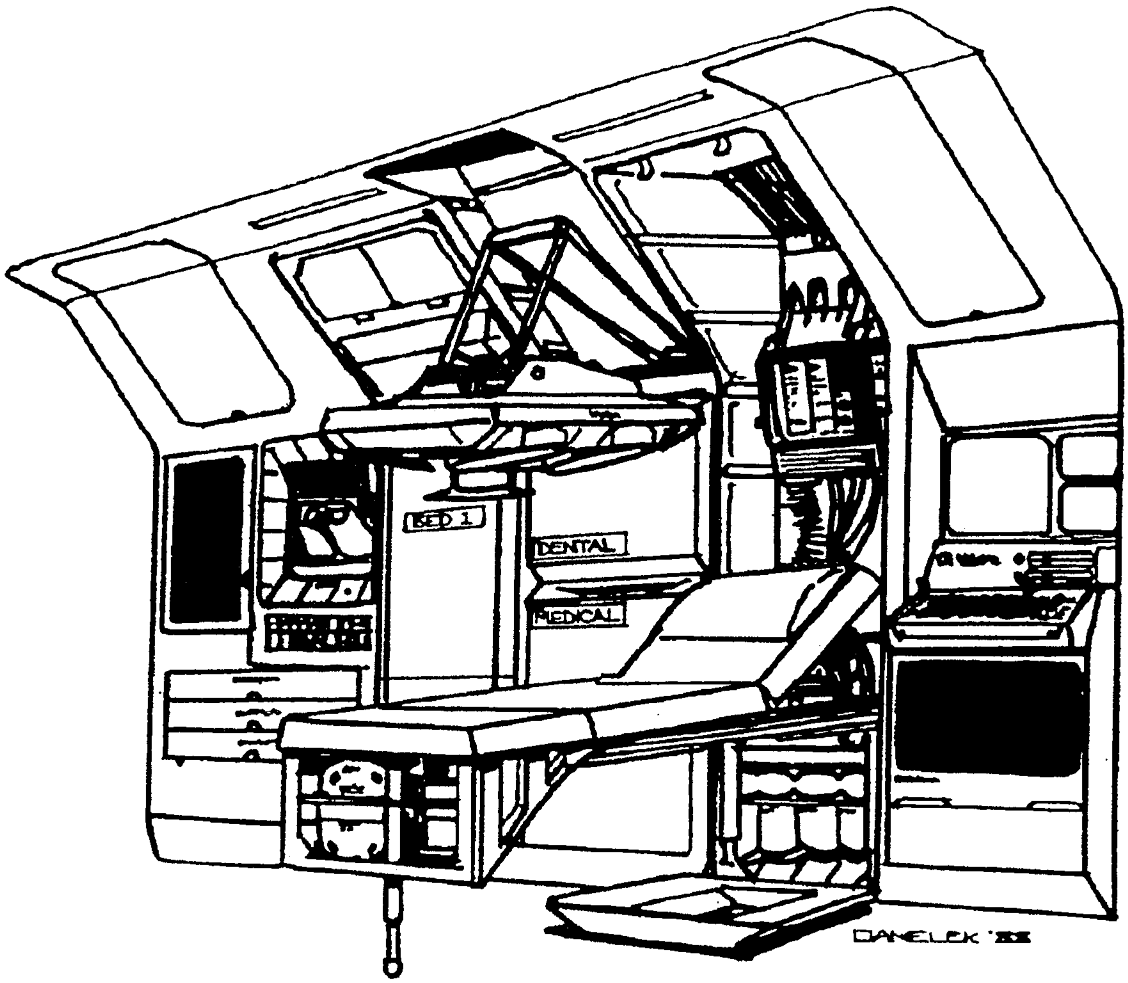

Figure 21 Medical Clinic Concept

Food Production. The feasibility of permanently manned extra- terrestrial bases is dependent upon development of Closed Ecological Life Support System (CELSS) technology. Technology is required to provide the ability to recycle almost 100% of consumable materials, the use of in-situ resources to maintain or expand systems and transition beyond plant based systems to microbiologically based bioregenerative systems. The concept for a plant food production system shown in Figure 22 involves autonomous operation and robotic tending, artificial rather than ambient environment, hydroponics rather than solid root medium for better control and automated monitoring employing artificial intelligence.

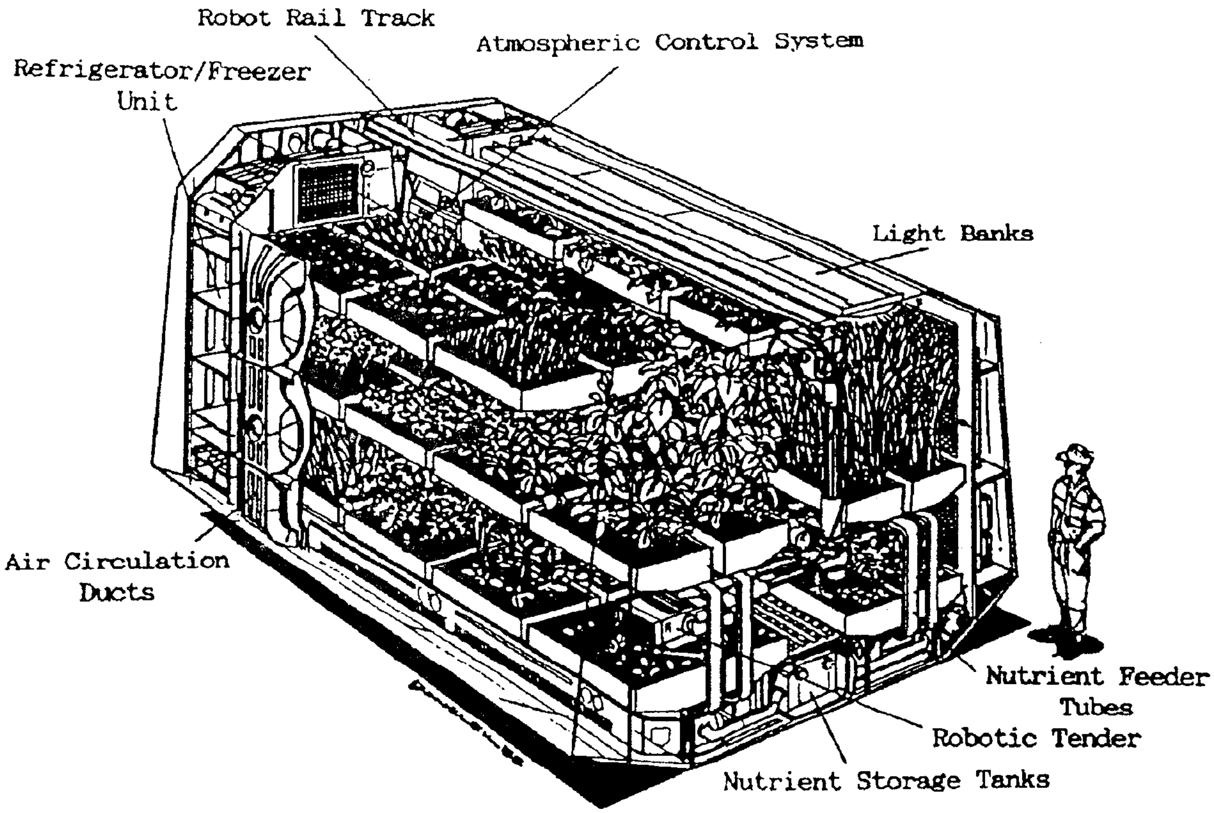

Figure 22 Concept for a Food Production Unit

CONCLUSION

The following features distinguish the proposed concept for a Mars program from other previous concepts:

1. The concept is structured around specific long range objectives that go far beyond just a couple of flights for U.S. astronauts to plant the flag, collect a few rock samples and perform a few experiments.

2. The transportation systems are dependent on NERVA technology to overcome the limited performance and sources of hypergolic or cryogenic propellents and reduce overall cost of the program. The nuclear propulsion technology proposed in this concept currently exists and has been perfected to safety requirements.

3. Materials to establish an infrastructure are prepositioned during autonomous preparatory missions in order to reduce risk for the first manned mission.

4. Commercialization of the support systems and the infrastructure is an integral part of the concept. Commercial ventures do not develop new systems for every trip for the same function; they maximize reuse of existing systems to minimize cost. The proposed concept employs systems designed for repeated use and maximum flexibility in their application beyond the first phase of the Mars program.

Prerequisite Technology. The analysis conducted by the study group revealed that the proposed Mars program would be too costly or incur too many serious risks if the following critical technologies are not developed and incorporated into the design of the systems:

1. More efficient advanced propulsion systems are required in order to implement the sprint mission profile. Only modest improvement is currently under development on the NERVA engine design tested in the late 1960's. Ion engine technology appears promising but requires much more research and development funding to perfect a flight rated engine. A concept for a laser assisted NERVA engine also deserves consideration for research funding.

2. In order to utilize aerocapture maneuvers to reduce propellent requirements, advanced designs must be developed for flexible structures within large spacecraft that can withstand the forces from severe vibration.

3. Efficient and less labor intensive techniques are needed for on orbit construction of a seamless heatshield for the aerobrake shield. Development is needed of advanced materials that will cure in the vacuum of space with nontoxic outgasing.

4. Efficient, autonomous water extraction systems must be developed for use on Phobos and the surface of Mars. The production capability must be sufficient to warrant the investment. The ability to replenish propellent at Mars is a critical element contributing to the cost effectiveness of the M^3 program.

5. Reliable Closed Ecological Life Support System (CELSS) technology is needed for the long duration flights of deep space missions.

6. Heavy Lift Launch Vehicles are needed to handle payloads in excess of 91,000 kilograms with reduced operating costs. Other alternative systems, such as the piston driven Livermore mass driver or the electromagnetically driven Sandia mass driver, that provide an equivalent function should be implemented as soon as testing proves their feasibility. The ability to transport large quantities of mass to LEO within reasonable cost is a critical element in the concept for the M^3 program.

7. Medical support systems must be developed for use in reduced gravity environments.

8. Spacecraft shielding measures must be developed to provide effective tolerance to micrometeorite impacts and peak periods of solar radiation .

9. Even though the technology may exist, a sophisticated and effective system is needed for early detection and warning of solar flare activity in order to provide sufficient time for securing the crew members in adequate protective shelter.

LAUNCH OPERATIONS

EARTH ORBITING LAUNCH COMPLEX

UNMANNED INTERPLANETARY SPACECRAFT PROCESSING

General. This section outlines typical recurring receipt through launch activities for the interplanetary spacecraft and depicts the major events to be accomplished.

10.0 Receiving Operations

10.10 Receipt of Spacecraft Elements and Hardware
- o Heat Shield Elements, Support Structure and Hardware
- o Core Vehicle Elements (Command, Logistics, Electrical Power and Airlock Modules), Support Structure and Hardware
- o Main Propellant Tanks, Support Structure and Hardware
- o Secondary Propellant Tanks, Support Structure and Hardware
- o Nuclear Thermal Rocket (NTR), Support Structure and Hardware
- o MUSC and Docking Pad
- o MCLS and Docking Pad
- o Flight Telerobotic Servicer (FTS)

10.20 Receiving Inspection
- o Perform Visual Inspections and Accounting of Spacecraft Elements and Hardware

10.30 Pre - Assembly Preparations
- o Prepare Orbiting Assembly Platform, Support Equipment and Test Equipment for the Assembly and Checkout of the Spacecraft

20.0 Assembly and Test Operations

20.10 Heat Shield Assembly. Perform the following assembly operations:
- o Install the Hub, Support Structure and Airlock System (Musc Docking Interface)
- o Erect and install the six (6) Main Trusses and interconnecting structure
- o Erect and install the six (6) Secondary Trusses and interconnecting structure normal to the Main Trusses
- o Install the support structure and six(6) Hydrogen Propellant Tanks (ET LOX Tanks)
- o Install the propellant feed lines, valves, pressurization system and associated electrical system
- o Install the three (3) MCL Docking Pads and associated Launch System
- o Install the Geodesic structure
- o Install the Exterior Shell (Thermal Shield)
- o Install the Inert Gas System and Storage Tanks
- o Install Environmental Oxygen System and Storage Tanks
- o Install the Reaction Control System
- o Install Spacecraft Spin System (Verify for manned vehicle)
- o Install Instrumentation Systems
- o Install the Electrical Power and Control Cables
- o Install the Flight Telerobotics System (FTS) Docking Pad

20.20 Heat Shield Systems Test. Install Simulators, Checkout Equipment and conduct tests on the following systems:
- o Structural Systems
- o Geodesic structure
- o MCL Docking Pads and associated Launch System
- o Musc Docking Pad
- o Propellant Tanks and Pressurization Systems
- o Electrical and Ordnance Systems
- o Instrumentation Systems
- o Thermal Shield
- o Inert Gas System and Storage Tanks
- o Environmental Oxygen System and Storage Tanks
- o Reaction Control System
- o Spacecraft Spin System
- o Instrumentation Systems
- o Electrical Power and Control Cables
- o Flight Telerobotic Servicer (FTS) Docking Pad

30.0 Spacecraft Elements Systems Test. Install Checkout Equipment and perform a Systems Test on each of the following Elements:

30.10 Command Module

30.20 Logistics Module

30.30 Aft Airlock

30.40 Electrical Power Module

30.50 Flight Telerobotic Servicer

30.60 ET Hydrogen Tanks (6) and pressurization system

30.70 Nuclear Thermal Rocket (NTR) Propulsion System

40.0 Core Vehicle Assembly and Combined Systems Test (CST)

40.10 Assembly of Core Vehicle
Perform the following assembly operations:
- o Install Core Vehicle Support Structure
- o Align and install the Command Module onto the Forward Airlock
- o Align and install the Logistics Module onto the Command Module
- o Align and install the Aft Airlock onto the Logistics Module
- o Align and install the Electrical Power Module onto the Aft Airlock
- o Align and Install the six (6) ET Hydrogen Tanks and support structure. Also install the insulation blankets
- o Align and install the NTR Propulsion System and support structure
- o Install Shielding around the Command Module
- o Install Shielding around specific areas of the other Modules
- o Connect all Electrical Power and Control Cables

40.20 Core Vehicle Combined Systems Test (CST). Install Checkout Equipment and perform a CST on the following systems to verify the Core Vehicle operation:
- o Electrical
- o Guidance
- o Flight Control
- o Instrumentation
- o CELSS
- o Communications
- o Module Pressurization
- o Hydraulic System
- o ET Hydrogen Tanks (6) and Pressurization System
- o NTR Propulsion System
- o Shielding
- o Structure

Note: The Launch Operations represents an Unmanned Mission, but the Core Vehicle will be man-rated to verify the system for Manned Missions

50.0 Payload Combined Systems Test (CST)

50.10 Mars Cargo Landers CST. Install Checkout Equipment and perform a CST on the following Systems of the Cargo Landers:
- o Electrical
- o Guidance
- o Flight Control
- o Instrumentation
- o Communications
- o Vehicle Pressurization
- o Inert Gas System and Storage Tanks
- o Hydraulic System
- o CO_2 Processor
- o Vehicle Health Monitor
- o Propellant Tanks and Pressurization System
- o Rocket Engine Propulsion System
- o Reaction Control System
- o Shielding
- o Structure

50.20 MUSC CST. Install Checkout Equipment and perform a CST on the following MUSC Systems:
- o Electrical
- o Guidance
- o Flight Control
- o Instrumentation
- o CELSS
- o Communications
- o Vehicle Health Monitor
- o Vehicle Pressurization
- o Hydraulic System

50.20 MUSC CST (Continued)
- o Inert Gas System and Storage Tanks
- o Propellant Tanks and Pressurization System
- o Rocket Engine Propulsion System
- o Reaction Control System
- o Shielding
- o Structure

Note: The Launch Operations represents an Unmanned Mission, but the MUSC will be man-rated to verify the system for Manned Missions and as a Rescue Vehicle

60.0 Mating of Spacecraft Elements

60.10 Mate Core Vehicle with Heat Shield
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the Core Vehicle and the Heat Shield
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.20 Mate Cargo Landers with Docking Pads (3 places) on the Heat Shield
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the Cargo Lander and Docking Pad
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.30 Mate MUSC with Docking Pad in the Heat Shield
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the MUSC and Docking Pad
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.40 Mate the Flight Telerobotic Servicer with Docking Pad on the Heat Shield
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the FTS and Docking Pad
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

70.0 Launch CST and Launch Countdown

70.10 Launch CST. The Launch CST provides a final launch readiness verification of the total Spacecraft airborne electrical and electronic systems, and the Launch Complex Systems with a minimum of system interface violations. The test consists of an Automatic Vehicle Verification (AVV), Automatic Countdown Sequence (ACS) and a Simulated Flight Sequence (SFS). Discretes are issued verifying all flight functions. This test is performed on simulated internal power, with arm/safe switches armed and with the Spacecraft guidance system issuing a simulated flight sequence steering profile. All ordnance functions shall be simulated using ordnance simulators.

70.20 Launch Countdown

A. Readiness Countdown. Perform the following readiness countdown functions after review and establishment of a successfull Launch CST by data review.
- o Perform Spacecraft power-on stray voltage test
- o Perform installation of ordnance
- o Check Systems Integrities
- o Load other propellants (Reaction Control System, Vehicle Spin,etc.)
- o Load the Cargo Lander propellants (3)
- o Load the MUSC propellants
- o Load the Main Liquid Hydrogen Tanks (NTR propellant)
- o Load the Secondary Liquid Hydrogen Tanks (NTR propellant)
- o Perform Command Control checks
- o Safe and Arm installation
- o Pressurize propellant tanks
- o Start Countdown

B. Master Countdown. Perform the following Countdown Functions:
- o Cargo Landers tests
- o MUSC tests
- o FTS tests
- o Remove safe and arm pins
- o Open loop Telemetry checks
- o Vehicle verification
- o Clear the Launch Pad
- o Command and Control Checks
- o Terminal Countdown

80.0 Post-Launch Operations

80.10 Trans Mars Insertion. The Δ v required to escape Earth orbit from a Space Station type orbit is approximately 3.2 Km/Sec. Additional Δ v to achieve TMI (i.e. hyperbolic excess velocity) will be 2.9 Km/ Sec for a typical optimal transfer (actual value can vary between 2.3 Km/Sec when Mars is near perihelion and 3.513 Km/Sec when Mars is near aphelion).

80.20 Acceleration to Mars

80.30 Mid Course Correction. Computation of fuel requirements was based on a total course correction Δ v requirement of 1.0 Km/Sec. This is very conservative.

80.40 Cruise Speed

80.50 Aerobraking. Roughly 3.0 Km/Sec (actual value will be between 2.0 Km/Sec and 3.3 Km/Sec) will be trimmed during aerocapture on the first pass through the atmosphere. Further aerobraking passes will lower the spacecrafts apoapsis to an altitude of 485 Km above the surface of Mars.

80.60 Insertion into Mars Orbit. A Δ V of approximately 0.11 Km/Sec will be required to circularize the orbit at 485 Km altitude.

80.70 Descent Preparation. The Cargo landers will separate from the Spacecraft and decelerate by approximately 110 meters per second to lower its periapsis inside the Martian atmosphere. Further deceleration will be achieved through aerobraking. Actual propellant requirements for control and landing are TBD.

80.80 Surface Activities. See text of study for details.

LAUNCH OPERATIONS

EARTH ORBITING LAUNCH COMPLEX

MANNED INTERPLANETARY SPACECRAFT PROCESSING

General. This section outlines typical recurring receipt through launch activities for the interplanetary spacecraft and depicts the major events to be accomplished.

10.0 Receiving Operations

10.10 Receipt of Spacecraft Elements and Hardware
- o Heat Shield Elements, Support Structure and Hardware
- o Core Vehicle Elements (Command, Logistics, Electrical Power and Airlock Modules), Support Structure and Hardware
- o Main Propellant Tanks, Support Structure and Hardware
- o Secondary Propellant Tanks, Support Structure and Hardware
- o Nuclear Thermal Rocket (NTR), Support Structure and Hardware
- o MUSC and Docking Pad
- o Manned Habitat Modules and Support Structure (3 Pairs Reqd)
- o Cargo Modules and Support Structure (3 Reqd)
- o Flight Telerobotic Servicer and Docking Pad

10.20 Receiving Inspection
- o Perform Visual Inspections and Accounting of Spacecraft Elements and Hardware

10.30 Pre - Assembly Preparations
- o Prepare Orbiting Assembly Platform, Support Equipment and Test Equipment for the Assembly and Checkout of the Spacecraft

20.0 Assembly and Test Operations

20.10 Heat Shield Assembly. Perform the following assembly operations:
- o Install the Hub, Support Structure and Airlock System (Musc Docking Interface)
- o Erect and install the six (6) Main Trusses and interconnecting structure
- o Erect and install the six (6) Secondary Trusses and interconnecting structure normal to the Main Trusses
- o Install the support structure and six(6) Hydrogen Propellant Tanks (ET LOX Tanks)
- o Install the propellant feed lines, valves, pressurization system and associated electrical system
- o Install the three (3) Habitat Modules Support Pads
- o Install the three (3) Cargo Modules Support Pads
- o Install the Geodesic structure
- o Install the Exterior Shell (Thermal Shield)
- o Install the Inert Gas System and Storage Tanks
- o Install Environmental Oxygen System and Storage Tanks
- o Install the Reaction Control System
- o Install Spacecraft Spin System

20.10 Heat Shield Assembly (Continued)
- o Install Instrumentation Systems
- o Install the Electrical Power and Control Cables
- o Install the Flight Telerobotics System (FTS) Docking Pad

20.20 Heat Shield Systems Test. Install Simulators, Checkout Equipment and conduct tests on the following systems:
- o Structural Systems
- o Geodesic structure
- o Habitat Modules Support Pads Interface
- o Cargo Modules Support Pads Interface
- o Musc Docking Pad
- o Propellant Tanks and Pressurization Systems
- o Electrical and Ordnance Systems
- o Instrumentation Systems
- o Thermal Shield
- o Inert Gas System and Storage Tanks
- o Environmental Oxygen System and Storage Tanks
- o Reaction Control System
- o Spacecraft Spin System
- o Instrumentation Systems
- o Electrical Power and Control Cables
- o Flight Telerobotic Servicer (FTS) Docking Pad

30.0 Spacecraft Elements Systems Test . Install Checkout Equipment and perform a Systems Test on each of the following Elements:

30.10 Command Module

30.20 Logistics Module

30.30 Aft Airlock

30.40 Electrical Power Module

30.50 Flight Telerobotic Servicer

30.60 ET Hydrogen Tanks (6) and pressurization system

30.70 Nuclear Thermal Rocket (NTR) Propulsion System

40.0 Core Vehicle Assembly and Combined Systems Test (CST)

40.10 Assembly of Core Vehicle. Perform the following assembly operations:
- o Install Core Vehicle Support Structure
- o Align and install the Command Module onto the Forward Airlock
- o Align and install the Logistics Module onto the Command Module
- o Align and install the Aft Airlock onto the Logistics Module
- o Align and install the Electrical Power Module onto the Aft Airlock
- o Align and Install the six (6) ET Hydrogen Tanks and support structure. Also install the insulation blankets
- o Align and install the NTR Propulsion System and support structure
- o Install Shielding around the Command and Habitat Modules

- o Install Shielding around specific areas of the other Modules
- o Connect all Electrical Power and Control Cables

40.20 Core Vehicle Combined Systems Test (CST). Install Checkout Equipment and perform a CST on the following systems to verify the Core Vehicle operation:
- o Electrical
- o Guidance
- o Flight Control
- o Instrumentation
- o CELSS
- o H_2O Processing
- o Communications
- o Module Pressurization
- o Hydraulic System
- o ET Hydrogen Tanks (6) and Pressurization System
- o NTR Propulsion System
- o Shielding
- o Structure

50.0 Payload Combined Systems Test (CST)

50.10 Habitat Modules CST. Install Checkout Equipment and perform a CST on the following Systems of the Habitat Modules:
- o Electrical
- o Communications
- o CELSS
- o H_2O Processing
- o Inert Gas System and Storage Tanks
- o Instrumentation
- o Module Health Monitoring System
- o Module Pressurization
- o Hydraulic System
- o Shielding
- o Structure

50.20 Cargo Modules CST. Install Checkout Equipment and perform a CST on the following Systems of the Cargo Modules:
- o Electrical
- o Communications
- o CELSS
- o Instrumentation
- o Module Health Monitoring System
- o Module Pressurization
- o Hydraulic System
- o Shielding
- o Structure

50.30 MUSC CST. Install Checkout Equipment and perform a CST on the following MUSC Systems:
- o Electrical
- o Guidance
- o Flight Control
- o Instrumentation
- o CELSS
- o Communications
- o Vehicle Pressurization
- o Vehicle Health Monitoring System
- o Hydraulic System
- o Inert Gas System and Storage Tanks
- o Propellant Tanks and Pressurization System
- o Rocket Engine Propulsion System
- o Reaction Control System
- o Shielding
- o Structure

60.0 Mating of Spacecraft Elements

60.10 Mate Core Vehicle with Heat Shield
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the Core Vehicle and the Heat Shield
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.20 Mate Habitat Modules with their Support Pads on the Heat Shield (3 Places)
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the Habitat Modules and Support Pads (3Places)
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.30 Mate Cargo Modules with their Support Pads on the Heat Shield (3 Places)
- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the Cargo Modules and Support Pads (3Places)
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.40 Mate MUSC with Docking Pad on the Heat Shield

- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the MUSC and Docking Pad
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

60.50 Mate the Flight Telerobotic Servicer with Docking Pad on the Heat Shield

- o Remove protective covers from mating interfaces
- o Inspect and verify mating interfaces
- o Align and perform a mechanical connection between the FTS and Docking Pad
- o Connect electrical cables
- o Connect fluid lines
- o Perform interface checkouts
- o Perform functional checkouts
- o Perform subsystem preps for the Launch CST

70.0 Launch CST and Launch Countdown

70.10 Launch CST. The Launch CST provides a final launch readiness verification of the total Spacecraft airborne electrical and electronic systems, and the Launch Complex Systems with a minimum of system interface violations. The test consists of an Automatic Vehicle Verification (AVV), Automatic Countdown Sequence (ACS) and a Simulated Flight Sequence (SFS). Discretes are issued verifying all flight functions. This test is performed on simulated internal power, with arm/safe switches armed and with the Spacecraft guidance system issuing a simulated flight sequence steering profile. All ordnance functions shall be simulated using ordnance simulators.

70.20 Launch Countdown

A. Readiness Countdown. Perform the following readiness countdown functions after review and establishment of a successfull Launch CST by data review.

- o Perform Spacecraft power-on stray voltage test
- o Perform installation of ordnance
- o Check Systems Integrities
- o Load other propellants (Reaction Control System, Vehicle Spin,etc.)
- o Load the Main Liquid Hydrogen Tanks (NTR Propellant)
- o Load the Secondary Liquid Hydrogen Tanks (NTR Propellant)
- o Load the MUSC propellants
- o Perform Command Control checks
- o Safe and Arm installation
- o Pressurize propellant tanks
- o Start Countdown

B. Master Countdown. Perform the following Countdown Functions:

- o MUSC tests
- o FTS tests
- o Remove safe and arm pins
- o Open loop Telemetry checks
- o Vehicle verification
- o Clear the Launch Pad
- o Command and Control Checks
- o Terminal Countdown

80.0 Post-Launch Operations

80.10 Trans Mars Insertion. The Δ v required to escape Earth orbit from a Space Station type orbit is approximately 3.2 Km/Sec. Additional Δ V to achieve TMI (i.e. hyperbolic excess velocity) will be 2.9 Km/ Sec for a typical optimal transfer (actual value can vary between 2.3 Km/Sec when Mars is near perihelion and 3.513 Km/Sec when Mars is near aphelion).

80.20 Acceleration to Mars

80.30 Mid Course Correction. Computation of fuel requirements was based on a total course correction Δ v requirement of 1.0 Km/Sec. This is very conservative.

80.40 Spin Spacecraft About its Axis. The spinning of the spacecraft is primarily for manned missions to simulate the various levels of gravity specified in the study. Fuel consumption will be negligible compared with other mission activities.

80.50 Cruise Speed

80.60 De-Spin the Spacecraft. De- Spin of the Spacecraft occurs prior to aerobraking. Fuel consumption will be negligible compared with other mission activities.

80.70 Aerobraking. Roughly 3.0 Km/Sec (actual value will be between 3.3 Km/Sec and 2.0 Km/.Sec) will be trimmed during aerocapture on the first pass through the atmosphere. Further aerobraking passes will lower the spacecrafts apoapsis to an altitude of 485 Km above the surface of Mars.

80.80 Insertion into Mars Orbit. A Δ v of approximately 0.11 Km/Sec will be required to circularize the orbit at 485 Km altitude.

80.90 Descent Preparation. The Cargo landers will separate from the Spacecraft and decelerate by approximately 110 meters per second to lower its periapsis inside the Martian atmosphere. Further deceleration will be achieved through aerobraking. Actual propellant requirements for control and landing are TBD.

80.100 Surface Activities. See text of study for details.

Appendix D

Nuclear Electric Performance Generator

APPENDIX D: NUCLEAR ELECTRIC PERFORMANCE GENERATOR (NEPG)

Nuclear Electric Performance Generator (NEPG) is an Excel spreadsheet program written by Science Applications International Corporation (SAIC). The user's guide is being submitted separately.

NEPG enables rapid systematic evaluations of mass performance for various robotic, low-thrust, lunar or Mars missions. The program allows the user to perform trade studies for various mission scenarios by varying inputs such as propulsion system parameters or NEP spacecraft payload mass changes at any node. NEPG obtains optimal trajectory data for Earth-to-Mars and Mars-to-Earth transfers from a low-thrust trajectory data base. This data base contains the dates and least squares curve fit coefficients that enable highly accurate computation of parameters such as initial power to initial mass ratio and final mass to initial mass ratio.

NEPG generates spacecraft mass performance and key event calendar date summaries for user-defined, low-thrust, lunar and Mars missions. It also computes mass performance for a trans-Mars injection stage (TMIS) that propels the NEP spacecraft to Earth escape, and/or for a space transfer vehicle (STV) that transports payload or the fully loaded NEP spacecraft between Earth nodes.

NEPG has inputs allowing not only addition or subtraction of payload mass at any node, but also choice of thruster type. In addition to spacecraft system masses and NEP spacecraft calendar dates, NEPG outputs include ΔV, final mass to initial mass ratio, and initial acceleration for each low-thrust spiral or transfer. NEPG best demonstrates its versatility by constructing customized output labels to describe each propulsive event required to perform the user-specified mission scenario.

To demonstrate the features of this spreadsheet program, a 2015 conjunction class mission is analyzed. It uses a TMIS stage for Earth escape, a NEP spacecraft to transfer from LEO to Mars and back to Earth (Nuclear Safe Orbit (NSO)) via the moon, and a STV to transfer the NEP cargo from NSO to LEO. The following two pages depict the input file, and the next two pages show the results of the analysis.

Nuclear Electric Performance Generator (NEPG)

INPUT

Case Title: 2015 Conjunction Class Earth-Mars-Moon-Earth Mission with TMI Stage Recovery

Allowable Nodes

Node ID	Description	Body	Inclination(°)	Altitude (km)
LEO	Low Earth orbit	Earth	28.5	500
NSO	Nuclear safe orbit	Earth	28.5	1,000
HEO	High Earth orbit	Earth	0.0	35,787
ML1	Earth-Moon libration point 1	Earth	23.5	320,006
LLO	Low lunar orbit	Moon	- -	100
DEI	Deimos orbit	Mars	0.0	20,060
HMO	High Mars orbit	Mars	0.0	17,030
PHO	Phobos orbit	Mars	0.0	5,977
LMO	Low Mars orbit	Mars	30.0	500

Vehicle Definition

Vehicle ID	Description
STV	Space transfer vehicle (chemical propulsion spacecraft operating within Earth's sphere of influence)
TMIS	Trans Mars injection stage (chemical propulsion spacecraft for Earth escape to C3 = 0)
NEP	Nuclear electric propulsion (may operate at any node except LEO)

Node Sequence

Transfer Number	Initial Node ID	Final Node ID	Vehicle ID	Payload Mass Change (t) Initial Node	Final Node
1	LEO	LEO	TMIS	0	0
2	LEO	LMO	NEP	100	-75
3	LMO	LLO	NEP	50	0
4	LLO	NSO	NEP	0	-50
5	NSO	LEO	STV	50	0
6					
7					
8					
9					
10					
11					
12					

NEP System Parameters

Thruster type (ION or MPD)	ION
Specific impulse (sec)	7000
Specific mass (kg/kW)	20.0
Initial power (MW) Enter "0" if calculated.	0.000
Propellant reserve and tankage factor (kg/kg propellant)	0.10
Payload structure factor (kg/kg payload)	0.05
Aerobrake shield factor (kg/kg entry mass)	0.15
Initial mass estimate (lower value in metric tons)	200
Initial mass estimate (upper value in metric tons)	230

Chemical Propulsion System Parameters

Specific impulse for large ΔVs (sec)	480
Specific impulse for small ΔVs (sec)	340
Large ΔV propellant reserve and tankage factor (kg/kg propellant)	0.070
Small ΔV propellant reserve and tankage factor (kg/kg propellant)	0.150
Fixed pre-NEP stage inert mass.--propellant tanks and engines (metric tons	0.000
Fixed pre-NEP stage maximum propellant mass (metric tons)	0.000
Fixed post-NEP stage inert mass--propellant tanks and engines(metric tons	0.000
Fixed post-NEP stage maximum propellant mass (metric tons)	0.000
Engine mass for large ΔVs (kg)	1588
Engine mass for small ΔVs (kg)	910

ΔV margin (%)	0.0
Intervehicle adapter factor (kg/kg payload)	0.02
Aerobrake shield factor (kg/kg entry mass)	0.15
STV orbital plane change required (deg)	0.00

Heliocentric Transfer Data

Transfer Direction	Julian Dates Start	Julian Dates Venus Swingby	Julian Dates End	Thrust-On Time (days)	MF/MO	PO/MO (kW/kg)	K3
Earth->Mars	2457365	0	2457755	388.2	0.9134	8.770	0.2307
Mars->Earth	2458125	0	2458475	251.0	0.9227	12.114	0.2374

Launch year	2015
Outbound transfer time (days)	390
Inbound transfer time (days)	350
Mission class (1=conjunction, 2=opposition)	1

Options (1=Yes, 0=No)

TMI stage recovery?	1
UpdateTMIS/STV performance **only?**	0
Aerobraking at Mars?	0
NEP node sequence unchanged?	0
Output mass performance debugging data?	1

Planetary Constants

Body	Gravitational Parameter (μ) (km^3/sec^2)	Equatorial Radius (km)
Earth	3.9860E+05	6,378.1
Moon	4.9028E+03	1,738.0
Mars	4.2828E+04	3,397.2

Mean Earth-Moon distance (km)	384,404

OUTPUT

Run date/time	27-Feb-90	2:43 AM

Run time =	3.05	minutes

Case Title: 2015 Conjunction Class Earth-Mars-Moon-Earth Mission with TMI Stage Recovery

TMIS Mass Summary (Metric Tons)	
Initial Mass	269.873
Propellant Loading	242.026
Propulsion Inerts	23.513
Recovery Aerobrake	4.334
Payload	214.658
Initial Mass in LEO (Metric Tons)	490.718

STV Mass Summary (Metric Tons)	1st ST	2nd STV
Initial Mass	6.186	
Propellant Loading	3.364	
Propulsion Inerts	2.823	
Payload	50.000	
NEP initial power (MW)	1.883	

NEP PERFORMANCE SUMMARY BY EVENT

EVENT	TIME (days)	THRUST (days)	J (m^2/s^3)	ΔV (km/sec)	MF/M0	a0 (m/s^2)
Earth to Mars	390.000	388.247	1.263	6.218	0.9134	1.7723E-04
Mars Capture	179.206	179.206	0.736	3.072	0.9562	1.9403E-04
Mars stay time	42.833					
Mars Escape	147.961	147.961	0.766	3.060	0.9564	2.3413E-04
Mars to Earth	350.000	251.028	1.527	5.526	0.9227	2.4481E-04
Earth Capture	24.296	24.296	0.150	0.559	0.9919	2.6533E-04
Moon Capture	63.353	63.353	0.409	1.480	0.9787	2.6750E-04
Moon Escape	61.968	61.968	0.418	1.479	0.9787	2.7333E-04
Earth spiral from Moon to NSO	251.053	251.053	3.370	6.342	0.9118	2.7928E-04
Total NEP Trip Time	**1510.670**					

KEY EVENTS (NEP)	DATE
Earth Escape	8-Dec-15
Mars Capture	1-Jan-17
Arrive at LMO	29-Jun-17
Depart from LMO	11-Aug-17
Mars Escape	6-Jan-18
Earth Capture	22-Dec-18
Moon Capture	15-Jan-19
Arrive at LLO	20-Mar-19
Depart from LLO	20-Mar-19
Moon Escape	21-May-19
Arrive at NSO	27-Jan-20

NEP MASS SUMMARY (Metric Tons)

Initial Mass		214.658
Payload Interface		5.000
Power/Propulsion System		37.651
Propellant Loading		65.460
Tankage and Reserve		6.546
Aerobrake(s)		0.000
Final Payload		25.001
Earth to Mars		214.658
	Propellant Loading	18.590
	Tankage and Reserve	1.859
	Aerobrake	0.000
	Final Mass	196.068
Mars Capture		196.068
	Propellant Loading	8.581
	Tankage and Reserve	0.858
	Aerobrake	0.000
	Final Mass	112.487
Mars Escape		162.487
	Propellant Loading	7.085
	Tankage and Reserve	0.708
	Aerobrake	0.000
	Final Mass	155.403
Mars to Earth		155.403
	Propellant Loading	12.020
	Tankage and Reserve	1.202
	Aerobrake	0.000
	Final Mass	143.383
Earth Capture		143.383
	Propellant Loading	1.163

	Tankage and Reserve	0.116
	Aerobrake	0.000
	Final Mass	142.220
Moon Capture		142.220
	Propellant Loading	3.033
	Tankage and Reserve	0.303
	Aerobrake	0.000
	Final Mass	139.186
Moon Escape		139.186
	Propellant Loading	2.967
	Tankage and Reserve	0.297
	Aerobrake	0.000
	Final Mass	136.219
Earth spiral from Moon to NSO		136.219
	Propellant Loading	12.021
	Tankage and Reserve	1.202
	Aerobrake	0.000
	Final Payload	25.001

Appendix E

Acronyms

Appendix E: Acronyms, Abbreviations, and Definitions

Acronym	Definition	Comments
Ab	Aerobrake	
AC or A/C	Aerocapture	
ACC	Aft Cargo Carrier	(bottom compartment added to ET; 7.7x7.7m dia)
ACS	Attitude Control System	
ACS	Atmosphere Control and Supply	
AFE	Aeroassist Flight Experiment	(aerocapture brake test program)
AI	Artificial Intelligence	
AL	Airlock	(see also HAL)
ALARA	As Low As Reasonably Achievable	(applies to shielding to lower radiation dose)
ALS	Advanced Launch System	(USAF HLLV)
ALSPE	Anomalously Large Solar Particle Event	(maximum solar flare)
AMA	Atmosphere Monitor Assembly	
AMTEC	Alkali Metal Thermoelectric Converter	(advanced RTG power converter; static)
ANRE	Advanced Nuclear Rocket Engine (NTR)	
AR	Atmosphere Revitalization	
ARD	Ascent, Rendezvous, and Docking	
ARS	Air Revitalization System	(purification; oxygen, nitrogen supply)
ASOA	Advanced State of the Art	
A/V	Audio/Video	
base	permanently human-occupied facility -- Martian surface, Mars orbit, surface of moons	
BER	Bit error rate	
BFO	Blood Forming Organs	(bone marrow)
BOL	Beginning of Life	
bps	bits per second	
B/W	Band Width	
Byps	Bytes per second	
CAD	Computer-Aided Design	
CAE	Computer-Aided Engineering	
CAI	Computer-Aided Instruction	
CAM	Computer-Aided Manufacturing	
CAT	Computer-Aided Training	
CCTV	Closed-circuit television	
CELSS	Controlled Ecological Life Support System	

CERV	Crew Emergency Return Vehicle	
cg	center of gravity	
CH4	methane	
CO	carbon monoxide	
C/O	Checkout	
CO2	carbon dioxide	
ComSat	Communications satellite	
ComSciSat	Communication/Science satellite	
COS	Co-orbiting satellite	
CQ	Crew quarters	
CR	Conclusions/Recommendations	
CRS	Carbon dioxide Reduction System	(Bosch, Sabatier, or other process)
CSTI	Civil Space Technology Initiative	(advanced technology for LEO access)
d	day	
DIPS	Dynamic Isotope Power System	
DMS	Data Management System	
DSC	Differential Scanning Calorimeter	(instrument for water, minerals detection in soil)
DSM	Deep Sapce Maneuver	(broken-plane or other major interplanetary propulsive maneuver)
DSN	Deep Space Net	(NASA earthbased interplanetary communications system)
ECCV	Earth Crew Capture Vehicle	(small vehicle for crew EOC)
ECLSS	Environmental Control and Life Support System	
EDC	Electrochemical Dipolarized Cell	(Electrochemical carbon dioxide concentrator)
EELS	Earth Entry & Landing System	
EGA	Evolved Gas Analyzer	(instrument for water, organics detection in soil)
element	a system that plays a major role in performing a mission activity (a rover, MDV,orbiter)	
ELV	Expendable Launch Vehicle	
EOC	Earth Orbital Capture	
EOCS	Earth Orbital Capture System	(Earth aerobrake + retro-propulsion, if required, plus G&C)
EOL	End of Life	
EPS	Electrical Power System	
ES	Emergency shower	
ESA	European Space Agency	
ET	External Tank	
ETO	Earth-to-Orbit	(vehicles such as STS, HLLV, etc.)
ETV	Earth Transfer Vehicle	(MSS configuration during flight to Earth)
ETX	Earth Transfer Expendables	(propellant and other consumables during flight to Earth)
EVA	Extra-vehicular Activity	(any human activity outside protective shirtsleeve environment and requiring a spacesuit)
F/A	Failure Analysis	

FDS	Fire Detection and Suppression	
FMEA	Failure Modes and Effects Analysis	
FTS	Flight Telerobotic Servicer	(teleoperated robot for SS)
g	acceleration of gravity at the surface of the Earth	
GCR	Galactic Cosmic Rays	(cosmic rays, from outside the solar system)
GEO	Geosynchronous Earth Orbit	(geostationary orbit about Earth; 34,500 km circ)
GMO	Geosynchronous Mars Orbit	(geostationary orbit about Mars; 17,097 km circ)
GP	Guidance Package	(G&C, guidance and control, equipment)
h	hour	
HAB	Habitability	(HAB module; where the astronauts live)
HAL	Hyperbaric Airlock	(see AL)
HBC	Hyperbaric Chamber	(for treating decompression sickness)
Hc	Hydrocarbon	(propellant; methane (CH4) or other)
HEO	High Earth Orbit	
HLLV	Heavy-Lift Launch Vehicle	(SDVs and other advanced launchers)
HM	Habitation Module	
HMF	Health Maintenance Facility	(diagnosis and treatment of illness and trauma)
IFF	In-Flight Fabrication	(on-board shop tools and supplies)
IFM	In-Flight Maintenance	(tools, parts, unscheduled, prev. maintenance)
IFSE	Interplanetary Flight Science Equipment	
IMLEO	Initial Mass in Low Earth Orbit	
IMM	Interplanetary Mission Modules	(Hab/Lab/Log modules for crew in space)
IMS	Inventory Management System	
in-space	as in "in-space assembly"	
IOC	Initial Operational Capability	
IR	Infrared	
IRU	Inertial Reference Unit	
ISA	In-space Assembly	(see also OOA)
I_{sp}	Specific impulse	(units of Ns/kg or lb_f-s/lb_m)
ISS	International Space Station	
ISRU	*in situ* Resources Utilization	
ISXP	*in situ* X Production	(e.g., X is: P=propellant, W=water, F=food, R=resources, C=consumables)
K	Kelvin	(temperature)
kW	kilowatt	
LAB	Laboratory	(LAB module; where the astronauts work)
LEO	Low Earth Orbit	
LH2	Liquid hydrogen	

LLOX	Lunar Liquid Oxygen	(propellant grade; manufactured on the moon)
LM	Logistics Module	
LMO	Low Mars Orbit	
LN2	liquid nitrogen	
LOG	Logistics	(LOG module; consumables/equip. stowage)
LOX	liquid oxygen	(also, LO2)
LRB	Liquid Rocket Booster	
L_S	Areocentric longitude	(position of Mars around sun -- seasonal index)
LSI	Life Systems, Inc.	
LSS	Life Support System	
m	meter	(Note: "m" as a prefix indicates "milli")
M	mega	(one million)
MAP	Mission Activity Plan	
MAV	Mars Ascent Vehicle	(the vehicle which is launched to Mars orbit)
MCC	Mission Control Center OR, Mid-course correction	
M/CDA	ballistic coefficient	
MCV	Mars Cargo Vehicle	(logistics vehicle sent for cargo staging)
MDV	Mars Descent Vehicle	(the vehicle which de-orbits to land on Mars)
MECS	Mars-Earth Cycling Station	(cyclers)
MELS	Mars Entry & Landing System	(de-orbit propulsion + aerobrake + parachute + terminal propulsion + G & C)
MeV	Million Electron Volts	
MHD	Magnetohydrodynamic	(electric propulsion engine technique)
MITG	Modular Isotope Thermoelecic Generator	
MLI	Multi-layer Insulation	
MLMM	Mars Landed Mission Module(s)	(Hab/Lab/Log modules for the surface of Mars)
MLOE	Mars Landed Operations Equipment	(Science, Transportation, Construction, Manufacturing equipment -- substitue S, T, C, M for O)
MLOX	Mars Liquid Oxygen	(propellant grade; manufactured on Mars)
MMA	Martin Marietta Astronautics	
MMH	Monomethyl hydrazine	(propellant)
MMM	Manned Mars Mission	
mmm	minimum mass mission	
MMMPA	Manned Mars Missions and Program Analyses	(the study performed by SRS)
MMTFI	Manned Mars Transportation and Facility Infrastructure	(study performed by MMA)
MMSS	Manned Mars Systems Study	(same as MMTFI)
MMU	Manned Maneuvering Unit	
MO	Mars Observer	(polar orbiter mission to Mars, planned for 1992 launch)

MO	Mars Orbit	
MOC	Mars Orbital Capture	
MOCS	Mars Orbital Capture System	(Mars aerobrake + retro-propulsion, if required + G&C)
MOS	Mars-Orbiting Satellite	(satellites in Mars orbit, independent of the MOV) (e.g., communications satellites)
MOSE	Mars Orbit Science Equipment	(Instruments for studies from Mars orbit)
MOV	Mars Orbiting Vehicle	(MSS configuration in Mars orbit)
MPS	Mars Propulsion System	(propulsion stages (S1, S2, etc.) and PAP)
MR, O/F	Mass Ratio, Oxygen to Fuel	
MRSR	Mars Rover Sample Return	(combined rover and sample return mission)
MSD	Meteoroid and Space Debris	
MSIS	Man-Systems Integration Standards	(NASA-STD-3000)
MSS	Mars Spaceship	(the spaceship that is assembled in LEO;
mtls	materials	
MTV	Mars Transfer Vehicle	(MSS configuration during flight to Mars)
N2H4	Hydrazine	(monopropellant; N_2H_4)
NCOS	National Commission on Space	(commissioned by the President)
NCRP	National Council on Radiation Protection	
NEP	Nuclear Electric Propulsion	(ion drive; nuclear reactor)
NERVA	Nuclear Engine for Rocket Vehicle Application	(nuclear thermal rocket program)
NSSTM	NASA Space Systems Technology Model	
NTO	Nitrogen tetroxide	(N_2O_4, biprop oxidizer)
NTR	Nuclear Thermal Rocket	
OAST	Office of Aeronautics and Space Technology	
OMS	Orbital Maneuvering System	
OMV	Orbital Maneuvering Vehicle	
OOA	On-orbit Assembly	(see also ISA)
ORU	Orbital Replacement Unit	
OSHA	Occupational Safety and Health Administration	
OSSA	Office of Space Science and Applications	
OTV	Orbital Transfer Vehicle	
outpost	permanent, but only occassionally manned facility	
PAP	Propulsion Avionics Package	
PCDA	Power Conditioning and Distribution Assembly	
PhD	Phobos/Deimos	(natural satellites of Mars)
Ph-Tele	Phobos Teleoperator	(remotely operated free-flyer to Phobos)
PhSE	Phobos Science Equipment	(instruments for studies of Phobos from a PhEM)
P/L	payload	(means different thing to different people)

PLSS	Primary Life Support System	
PMC	Permanently Manned Capability	(SS)
PPL	Preferred Parts List	(see QPL)
PRFE	Planetary Return Flight Experiment	(hypervelocity aerobrake test; compare with AFE)
prox ops	proximity operations	
PVPA	Photovoltaic Power Array	(solar cells)
PWS	Portable Workstation	
px	person-x	(where x is h (hour), d (day), sol, or y (year))
QPL	Qualified Parts List	(see PPL)
RCAM	Remote Computer Aided Manufacture	
RCS	Reaction Control System	
RHU	Radioisotope Heater Unit	
RL-10	(LH2/LOX engine, mfg. by Pratt & Whitney)	
RMS	Remote Manipulator System	(Shuttle robot arm)
RN	Resource Node	
RTG	Radioisotope Thermoelectric Generator	
RVR	Rover	
SAIC	Science Applications International Corporation	
SAS	Space Adaptation Syndrome	
SAWD	Solid Amine Water Desorbed	
SDAS	Steam Desorbed Amine Subsystem	(carbon dioxide concentrator)
SDV-3R	Shuttle Derived Vehicle	(3 reusable SSMEs)
SE	Science Equipment	(see also IFSE, MOSE, and MLSE under MLOE above)
SEP	Solar Electric Propulsion	(ion drive; solar power)
SFE	Static Feed Electrolyzer	(water electrolysis to hydrogen and oxygen)
SI	International System of Units	
SOA	State-of-the-Art	
sol	sol	(the martian day)
SPAS	Space Power Architecture Study	
SPF	Specific Pathogen Free	
SRB	Solid Rocket Booster	
SRS	SRS Technologies	
SS	Space Station	(phase 1)
SSA	Spacesuit Assembly	
SSME	Space Shuttle Main Engine	
STAS	Space Transportation Architecture Study	
STBE	Space Transportation Booster Engine	(Hc/LOX)
STME	Space Transportation Main Engine	(LH2/LOX; lower performance than SSME)
STS	Space Transportation System	(Shuttle)

t	metric ton	(tonne, 1000 kg, or 1 Mg)
TBD	To Be Determined	
TCS	Thermal Control System	
TCCS	Trace Contaminant Control System	(air purification system)
TEI	Trans-Earth Injection	(Mars orbital escape and trans-Earth maneuver)
TEIS	Trans-Earth Injection System	(propulsion and guidance system for TEI)
THC	Temperature and Humidity Control	
TMI	Trans-Mars Injection	(Earth orbital escape and trans-Mars maneuver)
TMIS	Trans-Mars Injection System	(propulsion and guidance system for TMI)
TPS	Thermal Protection System	
TSE	Transfer Science Equipment	("cruise" science)
T/W	Thurst-to-Weight Ratio	
UDMH	Unsymmetrical dimethyl hydrazine	(propellant)
USL	U.S. Laboratory	(SS module)
UV	Ultraviolet	
VCD	Vapor Compression Distillation	(water purification technique)
vehicle	a combination of one or more system elements that are physically integrated and operate together in a particular mission phase.	
VGAM	Venus Gravity Assist, Marsbound	
VGAE	Venus Gravity Assist, Earthbound	
VGRF	Variable Gravity Research Facility	(proposed for SS)
VLBI	Very Long Baseline Interferometry	
WM	Watste Management	
w.r.t.	with respect to	
WVE	Water Vapor Electrolyzer	(humidity control; hydrogen, oxygen production)

Not preferred:

ERV	Earth Return Vehicle	("return" implies haste, geocentric bias)
MEM	Mars Excursion Module	(use MDV; "MEM" implies short stay)
MM	Mission Modules	
mt, MT	metric ton	(use "t", tonne, 1000 kg, or 1 Mg)

MRSR element acronyms: EMTV,MOV,MES/MLM,RVR,MAV/RM,ERV/EOC,EOC/RS

Summary of Vehicle and Facility Acronyms

MSS	Mars Spaceship	(the spaceship that is assembled in LEO)
TMIS	Trans-Mars Injection System	(propulsion and guidance system for TMI)
MTV	Mars Transfer Vehicle	(configuration during flight to Mars)
IMM	Interplanetary Mission Modules	(Hab/Lab/Log modules for crew in space)
MOCS	Mars Orbital Capture System	(Mars aerobrake+retro-propulsion+ G&C)
MCV	Mars Cargo Vehicle	(logistics vehicle sent for cargo staging)
MDV	Mars Descent Vehicle	(the vehicle which de-orbits to land)
MAV	Mars Ascent Vehicle	(the vehicle which is launched to Mars orbit)
MELS	Mars Entry & Landing System + terminal propulsion + G & C)	(de-orbit propulsion + aerobrake + parachute
MLMM	Mars Landed Mission Module(s)	(Hab/Lab/Log modules)
MLOE	Mars Landed Operations Equipment	(Science, Transportation, Construction, Manufacturing equipment -- substitue S, T, C, M for O)
RVR	Rover	
MOV	Mars Orbiting Vehicle	(configuration in Mars orbit, not incl. the MDVs)
TEIS	Trans-Earth Injection System	(propulsion and guidance system for TEI)
ETV	Earth Transfer Vehicle	(config. of the MSS for Mars to Earth transfer)
MTM	Mars Transfer Modules	(Hab/Lab/Log modules for crew in space)
EOCS	Earth Orbital Capture System	(Earth aerobrake + retro-propulsion, if required)
ECCV	Earth Crew Capture Vehicle	(small vehicle for crew EOC and/or EELS)
EELS	Earth Entry & Landing System	(see MELS subsystems)

Acronyms in reserve:

ATAC	Advanced Technology Advisory Committee
BIT	Built-in Test
BITE	Built-in Test Equipment
BMA	Berthing Mechanism Assembly

BMR	Body-Mounted Radiator	
CFES	Continuous Flow Electrophoresis System	
CHCES	Crew Health Care and Exercise System	
CHX	Condensing Heat Exchanger	
CIL	Critical Items List	
C&M	Control and Monitor	
CMA	Contaminant Monitor Assembly	
COTS	Commercial Off the Shelf	
C&W	Caution and Warning	
CY	Calendar Year	
DAS	Data Acquisiton System	
DDT&E	Design, Develop, Test, and Evaluation	
DGC	Dry Goods Carrier	
DKC	Design Knowledge Capture	
D&PD	Definition and Preliminary Design	
EDC	Electrochemical Depolarized Concentrator	
EDP	Embedded Data Processor	
EGSE	Electrical Ground Support Equipment	
EMU	Extravehicular Mobility Unit	
FBCC	Full Body Cleansing Compartment	(see WBCC)
FDFI	Fault Detection/Fault Isolation	
FF	Free Flier	
FOC	First Operating Capability	
FY	Fiscal Year	
GC	Gas Chromatograph	
GDMS	Ground Data Management System	
GFE	Government-Furnished Equipment	
GFP	Government-Furnished Property	
GFS	Government-Furnished Services	
GN2	Gaseous Nitrogen	
GN&C	Guidance, Navigation, and Control	
GSE	Ground Support Equipment	
HEPA	High Efficiency Particulate Filter	
HPTA	High Pressure Tank Assembly	
HSE	Habitation Support Equipment	
H/W	Hardware	
HX	Heat Exchanger	
ID	Identification	
I/F	Interface	

IFA	In-Flight Anomaly	
I/O	Input/Output	
IR&D	Independent Research and Development	
ITC	Internal Thermal Control	
IVA	Intravehicular Activity	
JEM	Japanese Experiment Module	
LLI	Limited Life Item	
LOC	Lines of Code	(see SLOC)
LS	Life Sciences	
LSE	Laboratory Support Equipment	
MACO	Manufacturing, Assembly, and Checkout	
MEL	Master Equipment List	
MTBF	Mean Time Between Failure	
PEP	Portable Emergency Provisions	
PGS	Power Generation Subsystem	
PHS	Personal Hygeine System	
RDT&E	Research, Development, Test, and Engineering	
R&MA	Restraints and Mobility Aids	
RO	Reverse Osmosis	
S/C	Subcontractor	
scc	standard cubic centimeter	
SEU	Single Event Upset	
SFSPE	Static Feed Solid Polymer Electrolysis	
S/H	Safe Haven	
SLOC	Software Lines of Code	(see LOC)
SR	Standard Rack	
TC	Thermal Control	
TCC	Trace Contaminant Control	
TD	Technology Demonstration	
TDRS	Tracking and Data Relay Satellite	
TDRSS	TDRS System	
TIMES	Thermoelectric Integrated Membrane Evaporator System	
T/L	Timeline	
TOC	Total Organic Carbon	
torr	pressure unit	
UBC	Universal Bar Code	
ULC	Unpressurized Logistics Carrier	
V&V	Validation and Verification	
WBCC	Whole Body Cleansing Compartment	(see FBCC)

WCA	Worst-Case Analysis
WQM	Water Quality Monitor
WR	Wardroom
WS	Work Station

Appendix F

Conversion Factors

APPENDIX F: METRIC SYSTEM CONVERSIONS

Conversions

From English	To SI units	Multiply by:
atm	N/m^2	1.013 E+5
AU	km	1.496 E+8
BTU	J	1055
BTU	W-h	.2931
BTU/ft^2s	W/m^2	11350
BTU/s	W	1054
cal	J	4.187
cal	W-h	1.163 E-3
eV	J	1.602 E-19
ft	m	.3048
ft^2	m^2	.09290
ft^3	m^3	.02832
gallon	m^3	3.785 E-3
HP	W	745.7
in	m	.02540
in of Hg @ 0°C	N/m^2	3386
knot	m/s	.5144
lbf	N	4.448
lbf-ft	N-m	1.356
lbf/in^2 (psi)	N/m^2	6895
lbf-sec/lbm (I_{sp})	N-sec/kg	9.81
lbm	kg	.4536
$lbm\text{-}ft^2$	$kg\text{-}m^2$	.04214
lbm/ft^2	kg/m^2	4.882
lbm/ft^3	kg/m^3	16.02
lbm/ft^3	g/cm^3	.01602
mb	N/m^2	100.0z
mile	m	1609
mile/hr	m/s	.4470
naut. mile	m	1852
rad	J/kg	.01000
slug	kg	14.59
$slug/ft^3$	kg/m^3	515.4
torr @ 0 °C	N/m^2	133.3

SI Prefixes

Factor by which unit is multiplied	Prefix	Symbol
10^{12}	tera	T
10^9	giga	G
10^6	mega	M
10^3	kilo	k
10^2	hecto	h
10^{-2}	centi	c
10^{-3}	milli	m
10^{-6}	micro	μ
10^{-9}	nano	n
10^{-12}	pico	p
10^{-15}	femto	f
10^{-18}	atto	a

Note:
1 N/m^2 = 1 Pascal (Pa)
1 rpm = .10472 rad/sec
1 W = 1 J/s
1 g/cm^3 = 1000 kg/m^3
1 N = 1 $kg\text{-}m/s^2$
1 gal (galileo) = 1 m/s^2
1 Earth-g = 9.81 m/s^2

Temperature

From	To	Use Formula:
°F	K	$t_k = (5/9)(t_f + 459.67)$
°F	°C	$t_c = (5/9)(t_f - 32)$
°C	K	$t_k = t_c + 273.15$
°R	K	$t_k = (5/9)\, t_r$

Non-SI, but recognized and defined in SI units:
t = metric ton = tonne
= 1000 kg = 1 Mg = 2204.6 lbm
l = liter = 10^{-3} m ^{3}
min = minute = 60 s
h = hour = 3600 s
d = day = 86400 s

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lbm	kg	.4536
$lbm\text{-}ft^2$	$kg\text{-}m^2$	.04214
lbm/ft^2	kg/m^2	4.882
lbm/ft^3	kg/m^3	16.02
lbm/ft^3	g/cm^3	.01602
mb	N/m^2	100.0z
mile	m	1609
mile/hr	m/s	.4470
naut. mile	m	1852
rad	J/kg	.01000
slug	kg	14.59
$slug/ft^3$	kg/m^3	515.4
torr @ 0 °C	N/m^2	133.3

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10^2	hecto	h
10^{-2}	centi	c
10^{-3}	milli	m
10^{-6}	micro	µ
10^{-9}	nano	n
10^{-12}	pico	p
10^{-15}	femto	f
10^{-18}	atto	a

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