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**Final Design Report**
**August 11, 1992**

(NASA-CR-192039) CONCEPTUAL DESIGN OF A MARS TRANSPORTATION SYSTEM Final Design Report (Minnesota Univ.) 165 p

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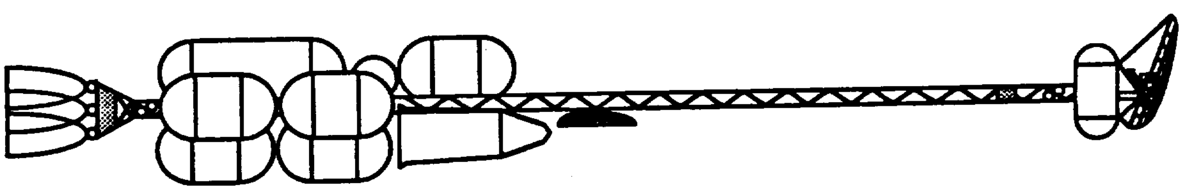
# Conceptual Design of a Mars Transportation System

***A One Year Design Project by the Senior Design Curricula in the Aerospace Engineering and Mechanics Department, University of Minnesota***



## NASA / USRA / University of Minnesota

**August 11, 1992**
**Final Design Report**

# FOREWARD

*"What is a man, if the chief good and market of his time be but to sleep and feed? A beast, no more."*

- William Shakespeare
Hamlet IV.iv.35-37

Challenge is perhaps the greatest stimulant known to Man. It has enabled peoples throughtout history all over the world to accomplish what seem impossible tasks. Whether that challenge is thrust upon them or taken up willingly, it is what keeps people alive; it gives them purpose. Without it, the human spirit would soon wither, atrophy, and die. It is, therefore, not in the interest of economics, nor politics, nor any other practical reason that we should reach for Mars. It is purely for the challenge. It is to satiate that very human need for challenge and adventure. Those who endeavor to reach those goals are alive in the truest sense of the word. This report is dedicated to them and to all who strive to reach their dreams, whatever they may be.

# Acknowledgements

This project would not have been possible without the dedication and hard work of many people. First and foremost, the students who were involved with the design. The names below include all of the students who participated in the class over the three quarter course sequence. Some students are listed more than once because they worked in more than one area. Asterisks indicate those individuals who served as lead engineers for their disciplines.

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Since most of the students did not have any practical design experience entering the class, the instructor, Andrew Vano, and teaching assistants David Rutherford and Chris Thyen, played a large role in getting the project off the ground and keeping it on track. Without their help and guidance the project would have been much less rewarding to all involved.

Finally, credit must be given to the engineers at NASA and other aerospace companies who helped the students throughout the course of the project. Whether through helping with literature searches, answering questions on particular points, or simply offering suggestions, these individuals helped the project immeasurably.

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This report was compiled from the work performed throughout the course of the project. All reports were written on Macintosh computers using Microsoft Word 4.00D software. Drawings were made with Claris MacDraw II 1.01 and ClarisCAD 2.0. Graphs were generated on Cricket Graph 1.3 and numerical tables were developed on Microsoft Excel 3.0.

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# List of Acronyms

AIMS .......... Airplane Information Management System
B-Al .......... Boron-Aluminum
BFO .......... Blood Forming Organ
deltaV or dV .......... change in velocity
DIPS .......... Dynamic Isotope Power System
ECLSS .......... Environmental Control and Life Support System
EOI .......... Earth Orbit Injection
ET .......... External Tank
ETO .......... Earth to Orbit
EVA .......... Extra Vehicular Activity
FEM .......... finite element model
G or g .......... acceleration of gravity (Earth standard)
GCR .......... Galactic Cosmic Radiation
HABP .......... Hypersonic Arbitrary Body Program
Hz .......... Hertz (cycles/second)
IMA .......... Integrated Modular Avionics
IMU .......... Inertial Measurement Unit
ITMG .......... Integrated Thermal Mircometeoroid Garment
LEO .......... Low Earth Orbit
LPNTR .......... Low Pressure Nuclear Thermal Rocket
MADM .......... Mars Ascent / Descent Module
MEV .......... Mars Excursion Vehicle
MHM .......... Mars Habitation Module
MMC .......... Metal Matrix Composite
MMH .......... Monomethylhydrazine
MOI .......... Mars-Orbit Injection
mT .......... metric ton
MTS .......... Mars Transportation System
MTV .......... Mars Transfer Vehicle
NERVA .......... Nuclear Engine for Rocket Vehicle Applications
NTO .......... Nitrogen Tetroxide
NTR .......... Nuclear Thermal Rocket
OMS .......... Orbital Maneuvering System
PLS .......... Personnel Launch System

# List of Acronyms
## (continued)

PLSS....................Personal Life Support System
RAS....................Robot assemly system
RCC....................Reinforced Carbon-Carbon
RCS....................Reaction Control System
rpm....................revolutions per minute
SG....................Supprt group
SPE....................Solar Particle Event
SSF....................Space Station Freedom
SWISTO....................SWIngby STopover Optimization
TEI....................Trans-Earth Injection
TMI....................Trans-Mars Injection
TPS....................Thermal Protection System

# 1.0 SYSTEMS INTERGRATION

## 1.1 Introduction

In conjunction with NASA Marshall Space Flight Center and several major aerospace corporations the University of Minnesota has developed a scenario to place humans on Mars by the year 2016. The project took the form of a year-long design course in the senior design curricula at the University's Aerospace Engineering and Mechanics Department. Students worked with the instructor, teaching assistants and engineers in industry to develop a vehicle and the associated mission profile to fulfill the requirements of the Mars Transportation System. This report is a summary of the final design and the process though which the final product was developed.

### 1.1.1 Goals of Project

The mechanism which made this effort possible is the Universities Space Research Association (USRA) Advanced Design Program (ADP). Because so much research goes on at both university and industrial centers USRA was created to facilitate productive interaction between these two spheres of information. Seasoned engineers receive fresh ideas from students while students have an opportunity to work on a real-world design problem. The hope is to produce benefits for all parties involved in the program.

In addition to the goals and objectives laid out by USRA, the University of Minnesota design curricula has four main goals for students involved with the program:

- To develop the student's concept of the design process and trade studies.
- To enhance the student's ability to communicate in written and oral presentations.
- To introduce organizational sturctures.
- To provide project ownership.

The combined focus of these goals is to prepare the engineering student for his or her introduction to real-world designing, where engineers typically work as large teams. The hope is that by the end of the program students will be better prepared for the professional environment and will have developed their creative sense as designers.

## 1.2 Project Statement

A set of mission requirements and assumptions were drawn up to help the students define the design problem and to force them to adhere to specific requirements.

### 1.2.1 Mission Requirements

1) Three missions to Mars will be planned starting in 2016.
2) The astronauts will remain on the surface for 30-100 days.
3) Landing sites will be selected for scientific interest and will include polar, volcanic, and canyon regions.
4) Artificial gravity will be provided for at least the outbound portion of the mission.
5) The above objectives will be met in a cost effective manner with acceptable levels of safety.

### 1.2.2 Mission Assumptions

1) The surface of Mars has been accurately mapped.
2) A satellite system for navigation and communication is in place around Mars.
3) Nuclear propulsion and aerobraking are mature technologies.
4) A Heavy Lift Launch Vehicle (HLLV) system is operational.
5) Space Station Freedom is operational.
6) A minimal lunar base is operational.

## 1.3 Course Structure

As stated earlier, one of the goals of the Aerospace Enginering and Mechanics department design program is to teach engineering design methodology. This means that such skills as trade analysis and group interaction are stressed rather than optimization of specfic subsystems.

Students are divided into subgroups to break the design problem down into reasonable projects. Figure 1.3.1 displays the discipline breakdown used in this design project. Each discipline had specific duties and was required to present findings on assigned action items to the class in both written and oral presentations. A lead engineer from each discipline was resposible for coordinating interaction between his or her discipline and other disciplines in the design team.

Class meetings were conducted twice each week for ten weeks each quarter. These meetings primarily served as an opportunity for the disciplines to interact through discussions and presentations. The results from each weeks' discussion and design work was submitted in a weekly status report. These reports were made available for all people in the design so that discussion on the material could be conducted. These reports were compiled at the end of each quarter and were presented in written and oral presentations by each discipline.

### 1.3.1 Three quarter sequence

Since most of the students had little background developing spacecraft, the course was designed into a three quarter sequence which would

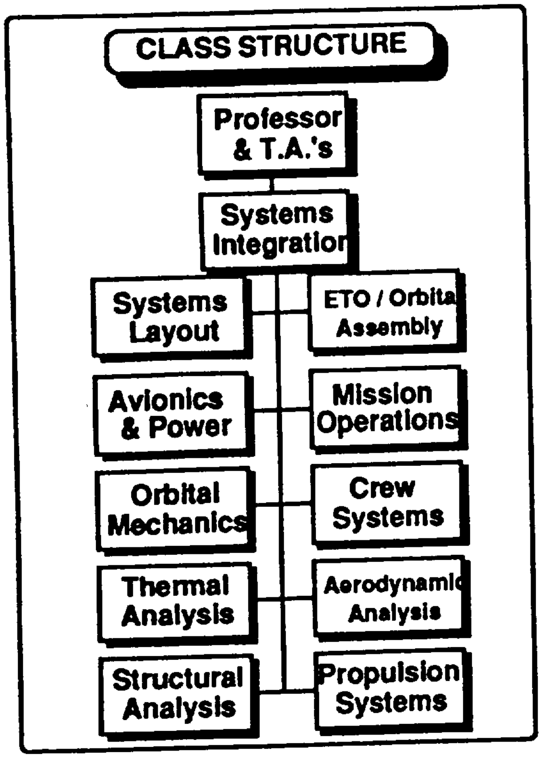



**Figure 1.3.1: Design team structure**

emphsize teaching design methodology. This called for introducing the new material first and then allowing for design work in the following quarters. Figure 1.3.2 displays a graphic representation of the design process taught in the course and how it was iplemented over the three quarter sequence.

The individual disciplines spent most of Fall quarter developing a knowledge basis on the material being studied and reporting it to the team. The end result of Fall quarter was an initial baseline configuration which was modified during the following two quarters into the final design.

The bulk of the design work was accomplished in Winter quarter. A similar format of action items and presentations was used along with special Configuration Control Board (CCB) meetings which did not entail discussion but only presentation of the current vcehicle and mission profile development status.

Spring quarter was used to optimize a few of the specialized subsystems on the design, perform numerical and physical tests of design elements, design and build scale models of the vehicle, and to prepare the final report.

### 1.3.2 Modeling activities

The final phase of the design course involved developing physical or numerical models of the design elements. A single scale model of the entire vehicle was constructed as well as models oseveral of the other subcomponents. In certain cases, physical models were impractical and were substituted by numerical analysis of the component.

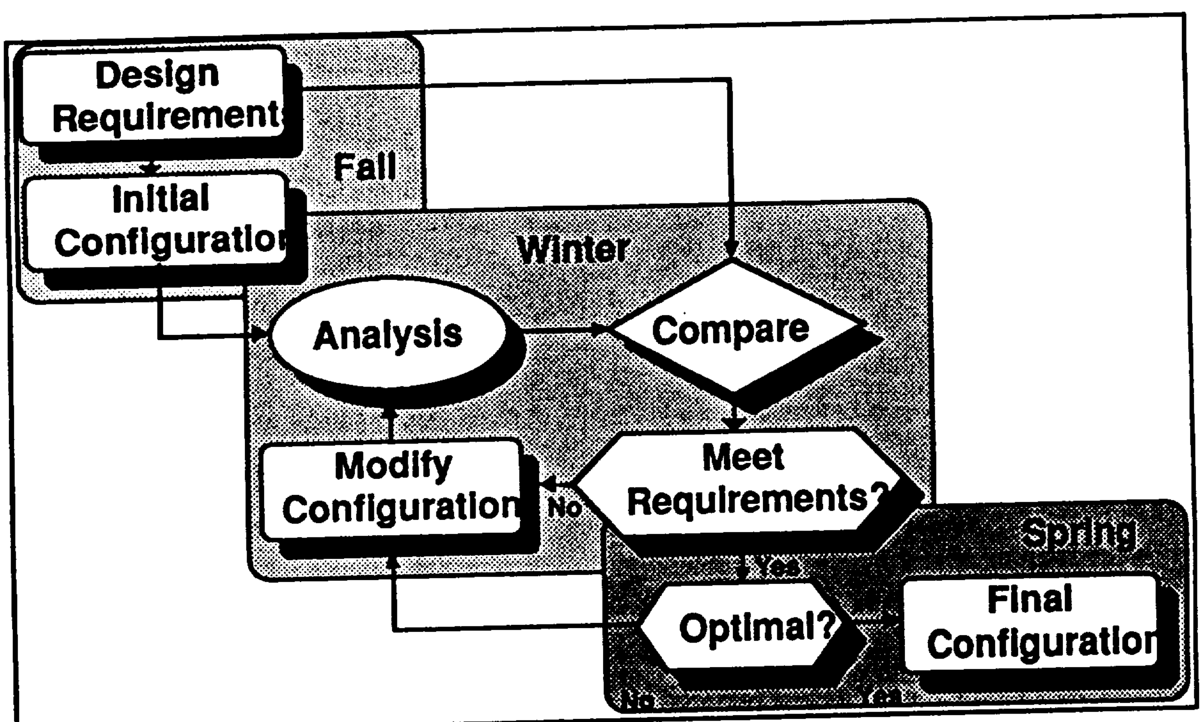



**Figure 1.3.2: Design process flowchart**

## 1.4 Vehicle Summary

The final cconfiguration of the Mars Transportation System (MTS) is illustrated in Figure 1.4.1 and consists of eight major components: the Mars Transfer Vehicle Habitation Module (MTV HAB), the propulsion system and fuel tanks, the biconic lander, the Earth return aeroshell, the Mars ascent/descent vehicle, Mars aeroshell, a communication and navigation array, and a central truss which serves as a common mounting bus for all of the other sub-components. Masses of the major components are presented in Table 1.4.1.

| Component | Mass(mT) |
|---|---|
| Truss | 15.7 |
| Biconic | 143.4 |
| Aeroshell(Earth) | 12.5 |
| MTV HAB | 63.9 |
| Ascent/descent vehicle | 26.9 |
| Engines, fuel tanks, and RCS | 57.0 |
| Aeroshell(Mars) | 39.7 |
| Total(dry) | 359.1 |
| Propellant | 705.2 |
| Total (with 15% contingency) | 1118.2 |

**Table 1.4.1:** MTS Vehicle mass breakdown

### 1.4.1 Mission profile

In addition to designing the vehicle for a mission to Mars the design team also developed a mission profile for such a mission. To fulfill the three mission requirement three separate missions were planned each utilizing a conjunctin class inbound Venus swingby type orbital configuratin and having an average trip time of 520 days. No effort was made for *in situ* resource utilization or reusing elements of the MTV from one mission on subsequent misions because of the uncertainty inheirent in these choices.

## 1.5 Conclusion

There is no doubt that this project achieved its primary goal of teaching design methodology and developing professional engineering skills in the students. Everyone involved with the project learned a great deal of information which will directly or indirectly change the way they approach problems in the future.

In addition to this primary goal, students also developed their abilities to work in team situations and other real-world engineering skills.

Finally, this report is respectfully submitted in hopes that it will be a useful contribution towards placement of humans on the surface of Mars.

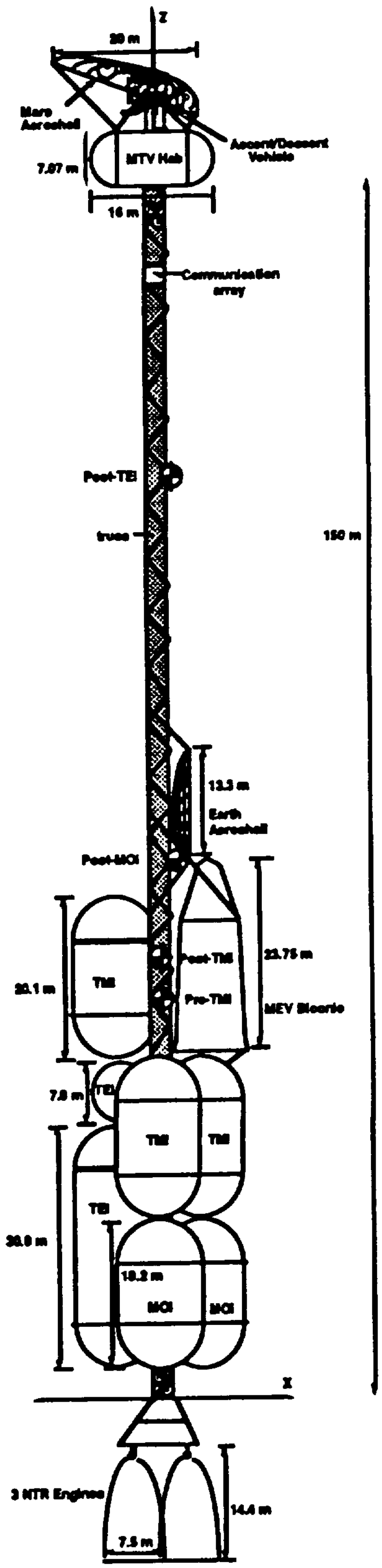



**Figure 1.4.1:** Mars Transportation System

# 2.0 SYSTEMS LAYOUT

## 2.1 Introduction

The Systems Layout discipline was responsible for 2-D drawings of the MTS configuration, the mass analysis, center of mass, and moment of inertia calculations. A specification sheet was established to be used as a brief overview of the current vehicle status. The sections that follow will first introduce the final MTS configuration, and then expand upon major and minor components of the design.

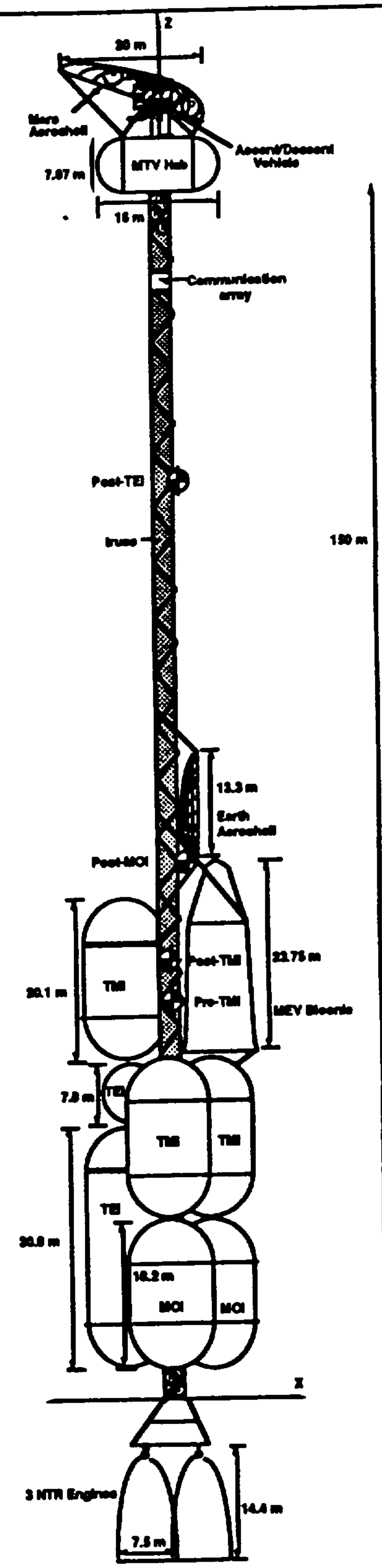
## 2.2 MTS Overview

The design requirements for the MTS involve transporting a six person crew to the Martian surface, providing for a sixty day surface stay and transportation back to Earth. The MTS design has seven major components necessary to complete the mission. These components can be seen in Figure 2.2.1 and are: 3 NTR engines, 8 fuel tanks, MEV Biconic, Earth aeroshell, MTV Habitat, ascent/descent vehicle, and Mars aeroshell.

The minor components on the MTS which are not emphasized in Figure 2.2.1 are: four RCS tanks, two located at the end of the truss near the engines, and two located near the MTV Habitat, the thrust structure which attaches the engines to the end of the truss, and the communication array.

The major and minor components are all assembled onto the 150m long equilateral triangular truss. Each side of the truss is 3m. The length of the completely assembled MTS is 185.07m. The dimensions of the major components can be seen in Figure 2.2.1. The major components are explained in detail in Section 2.3.

Shown in Figure 2.2.1 are the center of mass location of the MTS at Pre-TMI, Post TMI, Post MOI, and Post TEI. The fuel tanks are labeled in accordance to the tank that is supplying the fuel for the corresponding burn (ie. TMI,MOI, TEI). The center of mass and moment of inertia calculation which follow have been made with respect to the axis system illustrated in Figure 2.2.1; the X-axis is horizontal, Z-axis is vertical, and the Y-axis is coming out of the page. The center of mass and moment of inertia are the global values for the MTS.




Pre-TMI

| | | | |
|---|---|---|---|
| Mass : | | 1118.1 | mT |
| center of mass: | x = | 0.0 | m |
| | y = | 0.0 | m |
| | z = | 47.8 | m |

**Figure 2.2.1: MTS**

| | | | |
|---|---|---|---|
| moment of inertia: | Ix = | 2,161,367 | $m^2$-mT |
| | Iy = | 2,163,059 | $m^2$-mT |
| | Iz = | 59,642 | $m^2$-mT |

**Post-TMI**

| | | | |
|---|---|---|---|
| Mass : | | 782.1 | mT |
| center of mass: | x = | 0.0 | m |
| | y = | 0.0 | m |
| | z = | 52.2 | m |
| moment of inertia: | Ix = | 1,914,515 | $m^2$-mT |
| | Iy = | 1,916,207 | $m^2$-mT |
| | Iz = | 37,436 | $m^2$-mT |

After the TMI burn, the three TMI fuel tanks will be jettisoned from the MTS.

**Post -MOI**

| | | | |
|---|---|---|---|
| Mass : | | 585.7 | mT |
| center of mass: | x = | 2.5 | m |
| | y = | 0.0 | m |
| | z = | 67.2 | m |
| moment of inertia: | Ix = | 1,070,197 | $m^2$-mT |
| | Iy = | 1,071,895 | $m^2$-mT |
| | Iz = | 15,427 | $m^2$-mT |

After the MOI burn, the two MOI fuel tanks will be jettisoned from the MTS. The biconic and the descent vehicle inside of the Mars aeroshell will also be launched from the MTS once the proper position in the Martian orbit has been reached.

**Post-TEI**

| | | | |
|---|---|---|---|
| Mass : | | 194.4 | mT |
| center of mass: | x = | 1.8 | m |
| | y = | 0.0 | m |
| | z = | 117.5 | m |
| moment of inertia: | Ix = | 313,623 | $m^2$-mT |
| | Iy = | 314,721 | $m^2$-mT |
| | Iz = | 2,892 | $m^2$-mT |

After the TEI burn, the only components remaining on the MTS will be the two TEI tanks, the ascent vehicle upon its return form the Martian surface, and the MTV Habitat.

## 2.3 Component Summary

The seven major components previously described in Section 2.2 are outlined in further detail in the following summary. The components are shown here exactly as they are assembled to the truss with respect to the axis system established for the MTS. The center of mass calculations were made using the coordinates drawn on each component. The positive direction for the Z-axis is to the right of the origin, for the X-axis, positive is in the vertical upward direction from the origin. All distances are taken to start at the origin. The center of mass and moment of inertia are local calculations. The units for the local moment of inertia are $m^2$-mT.

### A. NTR Engine

| | | | |
|---|---|---|---|
| Mass of single NTR engine : | | 2.0 | mT |
| Center of mass: | x = | 0.0 | m |
| | y = | 0.0 | m |
| | z = | 8.8 | m |

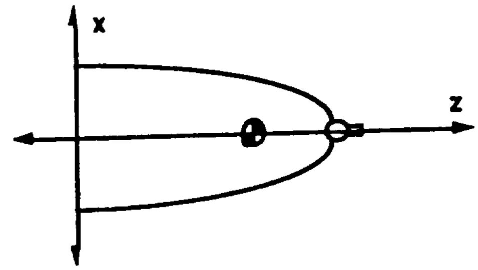



**Figure 2.3.1: NTR Engine**

| | | | |
|---|---|---|---|
| Moment of Inertia: | Ix = | 246 | $m^2$-mT |
| | Iy = | 246 | $m^2$-mT |
| | Iz = | 1 | $m^2$-mT |

Three NTR Engines are attached to the thrust structure at the end of the truss. Each NTR Engine is 14.4 m long, and the exit nozzle diameter is 7.5 m.

### B. Fuel tank

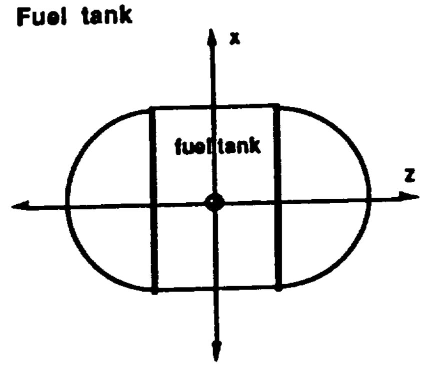



**Figure 2.3.2: Fuel tank**

Mass of fuel tanks vary depending on which burn the tank is used for (ie. TMI, MOI, TEI). Refer to the specification sheet for dimensions and masses of each tank.

| | | | |
|---|---|---|---|
| Center of mass: | x = | 0.0 | m |
| | y = | 0.0 | m |
| | z = | 0.0 | m |
| Moment of Inertia: | | | |
| TMI tanks: | Ix = | 4649 | $m^2$-mT |
| | Iy = | 4649 | $m^2$-mT |
| | Iz = | 1756 | $m^2$-mT |
| MOI tanks: | Ix = | 3473 | $m^2$-mT |
| | Iy = | 3473 | $m^2$-mT |
| | Iz = | 1540 | $m^2$-mT |
| TEI tank: | Ix = | 14,911 | $m^2$-mT |
| cylindrical | Iy = | 14,911 | $m^2$-mT |
| | Iz = | 2825 | $m^2$-mT |
| TEI tank: | Ix = | 106 | $m^2$-mT |
| spherical | Iy = | 106 | $m^2$-mT |
| | Iz = | 106 | $m^2$-mT |

There will be three TMI tanks which will be jettisoned from the MTS after the TMI burn has been completed. There are two MOI tanks which will also be jettisoned from the MTS when the burn is completed. There are two TEI tanks which are different sizes. One being a cylinder, the other is a sphere. These tanks will remain on the MTS to add mass to the end of the truss as an aid in creating gravity on the return trip to Earth.

### C. Earth Aeroshell

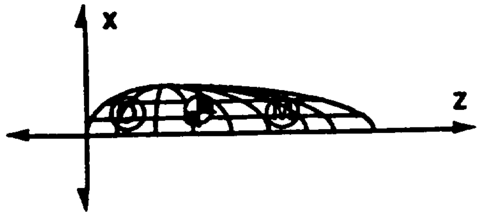



Figure 2.3.3: Earth Aeroshell

| | | | |
|---|---|---|---|
| Mass of Earth Aeroshell: | | 3.9 | mT |
| mass including TPS, methane and oxygen fuel tanks: | | 11.3 | mT |
| Center of mass: | x = | 0.9 | m |
| | y = | 0.0 | m |
| | z = | 4.8 | m |
| Moment of Inertia: | Ix = | 230 | $m^2$-mT |
| | Iy = | 166 | $m^2$-mT |
| | Iz = | 71 | $m^2$-mT |

The Earth aeroshell will be used for returning the 6 person crew to Earth's orbit. Prior to entry into Earth's orbit, the ascent vehicle will detach from the MTV Habitat, and dock inside of the Earth aeroshell. The remaining MTS will be placed on a trajectory to the Sun.

### D. MEV Biconic

Before departure from MTS:

| | | | |
|---|---|---|---|
| Mass of MEV Biconic: | | 143.4 | mT |
| Center of mass: | x = | 0.3 | m |
| | y = | 0.0 | m |
| | z = | 14.4 | m |
| Moment of Inertia: | Ix = | 32,979 | $m^2$-mT |
| | Iy = | 32,979 | $m^2$-mT |
| | Iz = | 1952 | $m^2$-mT |

After descent to the Martian surface:

| | | | |
|---|---|---|---|
| Mass of MEV Biconic: | | 98.2 | mT |
| Center of mass: | X = | 0.9 | m |
| | Y = | 0.0 | m |
| | Z = | 8.4 | m |

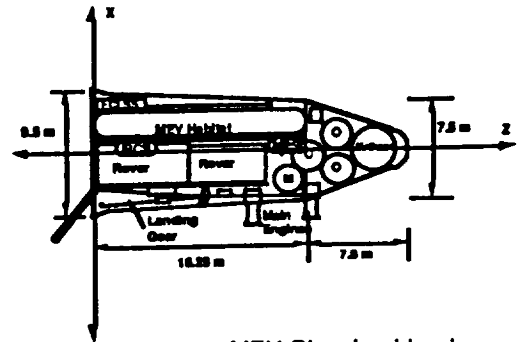



Figure 2.3.4: MEV Biconic side view

The MEV Biconic will contain the living quarters for the six person crew for the sixty days spent on the Martian surface. The living area can be seen in Figure 2.3.4 and Figure 2.3.6. The MEV biconic will be jettisoned from the MTS while it's orbiting Mars. The biconic has two main engines which extend from the bottom of the biconic and four RCS pods which will aid in the roll, yaw, and pitch maneuvers. Nine fuel tanks are needed for the biconic ascent to Mars: six liquid oxygen, and three liquid methane. The bottom level of the biconic is used for transporting the three rovers, nuclear reactor (power supply), and ground equipment to the Martian surface. The biconic and its contents will remain on the surface of Mars when the sixty day mission is complete. Side view and top view of the bottom layer of the biconic are shown here as Figures 2.3.4 and 2.3.5.

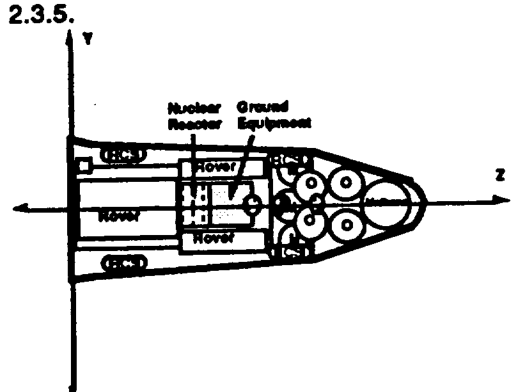



Figure 2.3.5: MEV Biconic top view

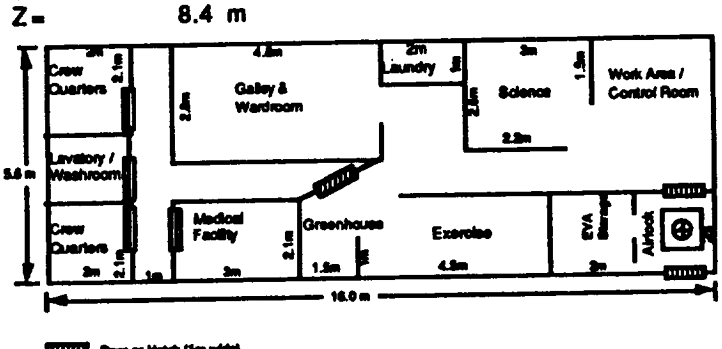



Figure 2.3.6: Floorplan of Biconic Habitat

### E. MTV Habitat

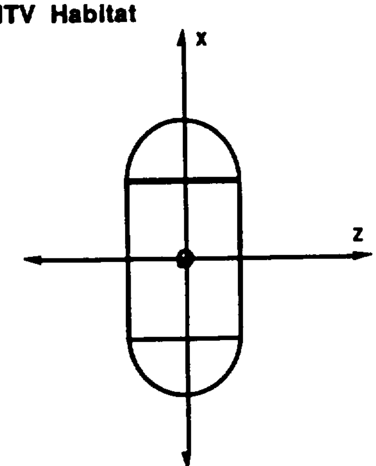



**Figure 2.3.7:** MTV Habitat

| | | | |
|---|---|---|---|
| Mass of MTV Habitat: | | 63.9 | mT |
| Center of mass: | x = | 0.9 | m |
| | y = | 0.0 | m |
| | z = | 0.0 | m |
| Moment of Inertia: | Ix = | 400 | $m^2$•mT |
| | Iy = | 156 | $m^2$•mT |
| | Iz = | 156 | $m^2$•mT |

The MTV Habitat is the environment in which the crew will be living in for the journey from Earth to the Martian orbit, and then from the Martian orbit back to Earth's orbit.

### F. Ascent/Descent Vehicle

| | | | |
|---|---|---|---|
| Mass of Ascent/Descent Vehicle : | | | |
| Descent vehicle | | | |
| before leaving MTS | | 26.9 | mT |
| after touchdown on Mars | | 30.1 | mT |
| Ascent vehicle | | | |
| before leaving Mars | | 30.5 | mT |
| after return to MTS | | 9.4 | mT |
| Center of mass: | x = | -2.9 | m |
| | y = | 0.0 | m |
| | z = | 0.0 | m |
| Moment of Inertia: | Ix = | 165 | $m^2$•mT |
| | Iy = | 95 | $m^2$•mT |
| | Iz = | 166 | $m^2$•mT |

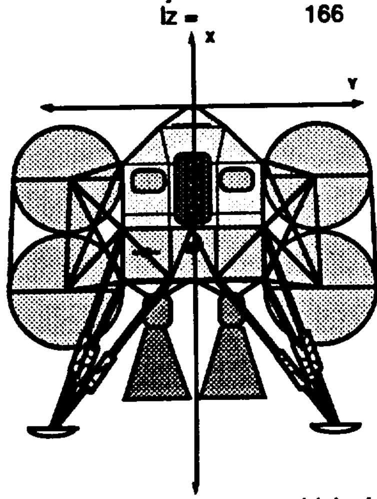



**Figure 2.3.8:** Ascent/descent vehicle front view

The ascent/descent vehicle is the transfer mechanism for the crew from the MTS to the Martian surface. The descent fuel is inside of the Mars aeroshell. The ascent vehicle is the descent vehicle minus the landing gear plus any samples taken from the Martian surface. The ascent vehicle, upon its return to the MTS, will dock with the MTV in the same position as before.

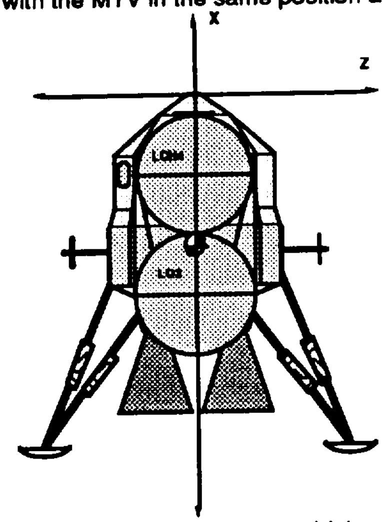



**Figure 2.3.9:** Ascent/descent vehicle side view

### G. Mars Aeroshell

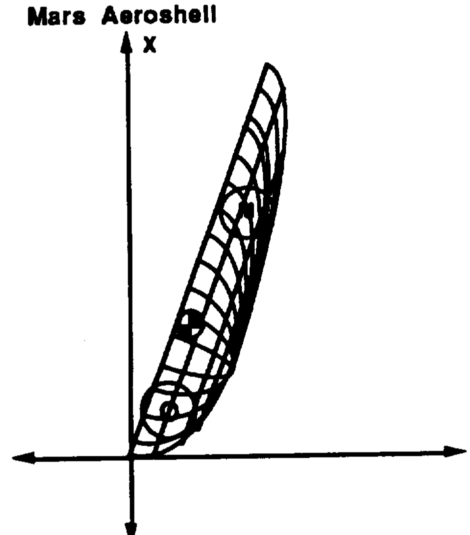



**Figure 2.3.10:** Mars Aeroshell side view

| | | | |
|---|---|---|---|
| Mass of Mars aeroshell: | | 7.9 mT | |
| mass including TPS, methane and oxygen fuel tanks | | 39.7 mT | |
| Center of mass: | x = | 6.3 m | |
| | y = | 0.0 m | |
| | z = | 2.9 m | |
| Moment of Inertia: | Ix = | 135 | $m^2$•mT |
| | Iy = | 199 | $m^2$•mT |
| | Iz = | 1803 | $m^2$•mT |

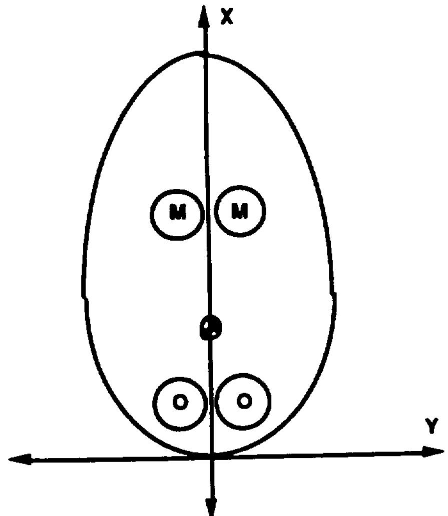



**Figure 2.3.11:** Mars aeroshell topview

The Mars aeroshell is designed to brake and protect the ascent/descent vehicle from entry into the Martian atmosphere. Once the descent vehicle has been braked to an adequate speed, the aeroshell will be jettisoned. The descent vehicle will then rely on its engines for touchdown.

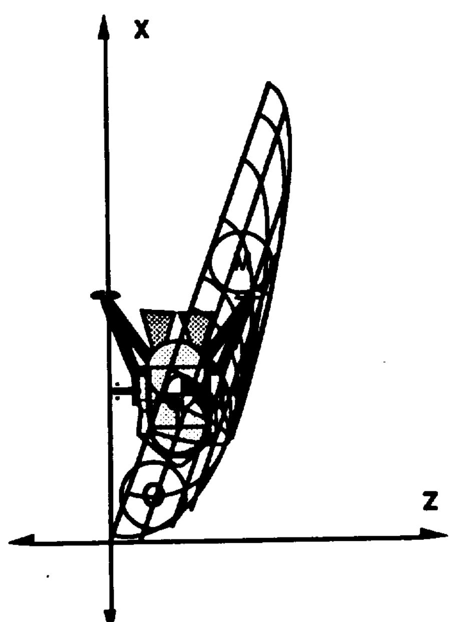



**Figure 2.3.12:** Lander and aeroshell package

| | | |
|---|---|---|
| Mass of aeroshell & lander: | 66.6 | mT |
| Center of mass: | | |
| x = | 6.0 | m |
| y = | 0.0 | m |
| z = | 2.8 | m |

The descent vehicle was placed inside of the aeroshell, calculating the center of mass for the combined vehicle.

A complete mass breakdown for the entire MTS can be found in Appendix 2.1.

## 2.4 Summary of Moment of Inertia and Center of Mass Calculations

Inertia may be defined as the change in force, or torque, necessary to make a unit change in acceleration, or angular acceleration, of a known body. In considering moment of inertia calculations, the assumption was made that the rotating body in question was perfectly rigid. Physically, the moment of inertia of a body is a measure of its resistance to angular acceleration. These are important aspects when considering a body which is free to rotate in all three dimensions of space.

It will be necessary to have the proper moments of inertia inputted into the onboard computer of the MTS so that any deviation from the norm can be corrected properly by reaction control systems. This is especially important when considering that the entire MTS will be subjected to a rotation during parts of the mission in order to maintain gravity for the astronauts.

The global moments of inertia are calculated by using the parallel axis theorem and the appropriate R value for each stage of the mission. Due to the changing center of mass for each stage, the R value changes each stage.

Center of mass calculations were a necessary first step in the moment of inertia calculations. Center of mass figures were also necessary for other disciplines in considering reaction and control of the vehicle.

## 2.5 Specification sheet

The specification sheet was created so that the MTS mission could be briefly summarized. As masses and dimensions were created and revised, the specification sheet was used to illustrate the changes as the MTS evolved. The final specification sheet can be found in Appendix 2.2.

## 2.6 Drawings

The role that drawings played in the System Layout group was vital in establishing the direction of the design process. Initially, 2-D drawings were used as visual aids for the design team during the brainstorming sessions. As the decisions were made in configuring the MTS, drawings of the MTS components were generated weekly and placed on the MTV. This was accomplished through the specification sheets.

Pro/Engineer, a Computer Aided Drawing (CAD) program was acquired by the design team. The 3-D drawings produced were used in the animation of MTS mission profile.

Pro-Engineer enables 3-D drawings to be created out of all the components which make up the MTS. The 3-D configuration aspect of Pro-Engineer has an appreciable advantage over Claris CAD and MacDraw II which was used in creating all of the drawings in the previous section. In creating drawings in 3-D, the program requires specific dimensions. Without dimensions, Pro-Engineer would not be able to regenerate any of the parts that are sketched; This program helps serve as a layout tool which requires that the components of the MTS be correctly specified to eliminate parts interference. This is a major contribution in making sure that the components securely fit onto the truss. Every component of the MTS has been drawn to a one-meter-to-one-inch scale in Pro-Engineer. The major components that have been generated using Pro-Engineer can be found in the Appendix 2.3 and are as follows: Figure A is the aeroshell, Figure B is a single NTR engine. Figure C, one of the fuel tanks, Figure D is the MEV biconic, and Figure E is the external view of the MTV habitation module with the lander and the aeroshell assembled.

The animation of the MTS mission profile required that the components drawn in Pro-Engineer be transferred into the animation program, Visualizer.

Pro-Engineer was also used to assist the actual modelling of the aerobrake because of its difficult geometry. Pro-Engineer allowed the aerobrake to be converted into an IGES surface file. The file was then fed into Master CAM to run toolpaths. Finally, the file is fed into a CNC milling machine where the mold of the aerobrake was cut.

## 2.7 Second and Third Missions

The only design modifications that will have to be made are those involving the size and placement of the fuel tanks. All the other major components are to remain the same in size and location. For a more detailed discussion on the fuel tank sizing, refer to the Structural Analysis portion of the report.

# 3.0 MISSION OPERATIONS

## 3.1 Introduction

This design of a Mars Transportation System includes 3 missions, each designed to carry a crew of 6 astronauts to Mars where they will stay on the surface for 60 days. While on Mars they have the use of extensive science equipment.

In planning the mission, the major consideration was minimizing IMLEO. Other factors which were taken into consideration were safety of the crew, mission complexity and best use of mission time to accomplish the goals of going to Mars.

## 3.2 Earth-To-Orbit

ETO operations entail every aspect of processing and launching the components of the MTS as well as the required orbital assembly. To determine the maximum size of the components and thereby calculate the amount of orbital assembly, a primary launch vehicle must be chosen.

### 3.2.1 Launch Vehicles

Several things must be taken into consideration when choosing a launch vehicle. These include total mission delta V ; propellant combination; thrust-to weight ratio; staging philosophy; expendable, recoverable, and/or reusable parts; mission architecture; technology risks; environmental issues; and launch location.

With these considerations taken into account a vehicle was chosen that fit the Mars mission needs and was the most likely to exist in time for the first mission. This vehicle is the Mars NLS derived HLLV with a core consisting of four STME engines burning LOX/LH2 and four boosters, each with two F-1A engines burning LOX/RP. The usable shroud diameter is 14.02 meters(46 feet) and the usable payload cylinder length is 30.48 meters (100 feet). The NLS derived HLLV is capable of carrying 220 metric tons into a 220 nautical mile, circular orbit, the same orbit that Space Station Freedom (SSF) will be in. This vehicle is shown in Figure 3.1 at the end of the Mission Operations section.

The crew will be carried in a Personnel Launch System (PLS) to the MTS. The PLS has a ten person capacity, two of whom are crew. Therefore, the entire crew for the Mars mission will be able to arrive together at the MTS. The vehicle to launch the PLS has not yet been chosen but one option is a space shuttle external tank (ET) derived one-and-a-half stage core.

### 3.2.2 Ground Facilities

New ground facilities will certainly be needed to construct and handle the new HLLV as well as the components of the MTS itself. These facilities need to be built and operating well before the scheduled first mission in 2016. The PLS is planned for the crews of SSF and therefore this system should already be in place for the Mars mission.

The NLS derived HLLV is a much larger launch vehicle than NASA has ever used. Therefore, many new ground facilities will need to be built and old ones modified. These include a booster processing facility, booster transporter, vertical payload transporter, mobile launch tower, and payload processing facility. New facilities will also have to be built in order to construct the aeroshells and biconic. The MTV habitat and the ascent/descent lander may be built in existing facilities or those built for SSF. Most of the new facilities will be built in the southeastern United States and the components then shipped to Kennedy Space Center where the launches will take place.

### 3.2.3 On-Orbit Assembly Facilities

Many options exist for the location and facility to assemble the MTS. These include 1)SSF, 2)a tethered facility off of SSF, 3)an assembly flyer platform, 3)a dedicated assembly node, 4)using low lunar orbit, 5)and using the MTV itself as a platform. Each of these options has its own advantages and disadvantages. The decision process put heavy emphasis on safety and the likelihood of the facility existing by 2014, the time when the launch will occur. This eliminated all the options except for the SSF, the tethered facility off SSF, and using the MTV itself. SSF was eliminated as an assembly point due to concerns of safety for SSF and the likely hood that SSF would be able to assemble the MTS. Therefore, it was decided the safest and most reliable choice for assembly would be to use the MTV itself as a platform and include a robotic assembly system (RAS) along in the first launch to assemble all the components of the vehicle. The RAS is shown in figure 3.2. The plan still requires localized debris shielding and additional reboost and control systems on the MTV itself.

### 3.2.4 Launch Strategy

The launch strategy was devised with the following considerations in mind: maximizing the usable payload volume and mass, minimizing the total number of launches, including the truss and RAS in the first launch for assembly purposes, sending the crew up last to minimize their total time in space, and launching the fuel as late as possible to minimize boiloff. The launch strategy that resulted is shown at the end of the Mission Operations section as Figures 3.3 - 3.9.

### 3.2.5 Orbital Assembly Operations

Since the total on-orbit assembly time is 1.75 years, debris protection is a primary requirement for orbital assembly. The assembly begins with the HLLV achieving a stable orbit adjacent to the assembly platform. A cargo transfer vehicle (CTV) then unloads the components of each HLLV and transfers them to the assembly facility. The components are then constructed and attached using the RAS that is to be in the first launch. Each system has interfaces which require verification after the assembly process. These interfaces are checked robotically if possible otherwise with system self-checks or by the on-orbit assembly crew. The estimated time for assembly of the components of each launch is well under the ninety days needed for launch processing, therefore, orbital assembly should not be a time constraint on ETO.

Some of the in-space operations that will take place include :

Receive / Rendezvous / Dock HLLV
Activate communications and power systems
Checkout on-board health monitoring systems
Inspect components
Verify health after launch
De-mate components from HLLV
Maneuver components to storage
Manipulate components into position
Attach components to truss / structure
Verify joint connections
Make connections between components
Verify connections / Check for leaks
Provide inspection data to Mission Control
Provide debris protection
Manage altitude

Orbital assembly for the Mars mission will be relatively easy as compared to that of SSF and the assemblers will be more experienced, learning from SSF. Therefore, orbital assembly should not have any problems sticking to the Mission schedule. Figures 3.10 - 3.16 at the end of the Mission Operations section are a graphic summary of the orbital construction phases including estimates of the time in hours required for some of the orbitall operations. The letters following the times below reflect the probable assembler with E meaning robotic control from Earth, S for robotic control from SSF, and H for Human EVA.

## 3.3 Three-Mission Profile

All three missions will have a 6-person crew with a 60-day stay on the Martian surface. This was decided because we felt that the expense of developing the flexibility necessary for different mission profiles outweighed any benefit of changing the profile extensively.

### 3.3.1 Launch Windows

The TMI launch window at Earth will be limited to 30 days, and perhaps less because of problems designing and accommodating fuel tanks for all possible $\Delta V$'s during a long launch window.

| Mission # | Window Start Date | Window End Date |
|---|---|---|
| 1 | 1/25/16 | 2/24/16 |
| 2 | 7/19/18 | 8/8/18 |
| 3 | 4/19/22 | 5/19/22 |

**Table 3.1:** Launch window dates

Ideally, missions should occur during a solar minimum when there is minimum danger of dangerous solar activity. Missions 1 and 2 happen during a solar minimum. Mission 3 happens during a solar maximum, which means there is a high risk of solar events which can seriously damage people and equipment. If it is decided that this is too risky, the next available launch window for our mission profile does not occur until around 2040.

## 3.4 Multi vs. Single Stages

An important first step in the design process was deciding what the mission profile should look like–that is, how many stages. This involved basically two decisions: a) whether to go to the Martian surface with one or two MEV landers and b) whether to go from Earth to Mars with one or two MTV's. If we went with two MEV landers, the first would carry the surface habitat and equipment while the second would bring the crew down. (If the MTV is in two stages, than so must the MEV.) The functions of the two MTV's would be essentially the same as 2 MEV's: the first bringing the surface habitat and supplies, and the second transporting the crew. Some advantages and disadvantages of these two choices follow:

2 MEV's
<u>Disadvantages</u>
—More mass to Mars.
—More expensive.
<u>Advantages</u>
—Larger habitat.
—More room for experiments, etc.
—Safer on descent, assuming single stage is a biconic.

2 MTV's
<u>Disadvantages</u>
—Habitat is subject to radiation and other wear.
—2 vehicles must be developed.
<u>Advantages</u>
—Less $H_2$ in LEO. (First MTV uses low energy Hohmann transfer.)

Although the choice was very close, the team finally decided to have one MTV and have the MEV land in two parts. The first stage of the MEV will be a biconic lander and the second stage will carry the crew only with an aeroshell. The decision to go with a two-part MEV was largely a result of the recommendation of Aerodynamic

Analysis. In an effort to save on MOI fuel, it was decided that all or part of the MEV be separated from the MTV before its propulsive brake at Mars. However, because of the high entry velocity and low density of the Martian atmosphere, this will not be done.

It was initially proposed that the crew come down with a rover to transport them to the habitat once they touched down. However, because it appears reasonable that the lander can be brought down accurately, there will not be a rover with the crew.

## 3.5 Mission Risks

### 3.5.1 Mission Risk Tree

Figure 3.17 is a risk tree for the mission. The primary goal of this figure is to identify the primary phases where decisions affecting the mission in a big way might occur. Following is a list of possible problems which might be reason to abort or change the mission profile. (Refer to Figure 3.17.)

Delay TMI:
- –System failure
- –Crew Health
- –Legal or Political
- –Delayed ETO
- –Solar storm

Elect Mars Fly-by:
- –Crew Health
- –Critical Damage to a Critical System
- –Excess $H_2$ boil-off or other unexpected loss of consumables

MTS does not enter parking orbit successfully:
- –Miscalculation
- –Engine failure

Delay or cancel surface expedition:
- –Bad weather
- –Irreparable system failure on MEV before separation from MTV

Habitat fails to function properly:
- –Landing system failure (crash)
- –Damage from impact (may, however, be repairable)
- –Bad weather

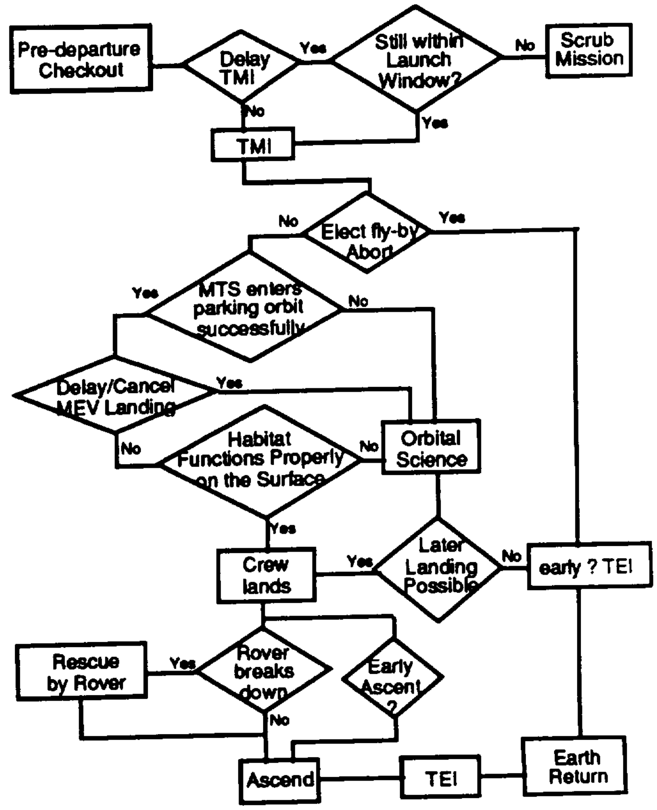



**Figure 3.17:** Mission Risk Tree

### 3.5.2 Critical Systems Analysis

Following are the results of a survey asking the different disciplines what problems may occur and what, if any, solutions could be implemented.

**Crew Systems**

Critical System: ECLSS

This system is composed of various subsystems, the failure of one resulting in the complete shutdown of the whole life support system. There is a complete back-up system onboard the MTV HAB Crew Systems estimates down-times to be less than 24 hours. If the secondary system should also fail, the crew would still be reasonably safe due to the only 'partially closed loop' nature of the system.

**Avionics and Power**

Critical System: Phased array antenna

High redundancy assures prevention of complete failure.

Critical System: Computer System

In case of failure, there are numerous redundancies in the system. Computers will be shielded against solar flares and GCRs.

Critical System: MTV Power Supply

3 separate Dynamic Isotope Power Systems will be utilized. The isotope heat source cannot be serviced by humans because of

radiation, but the chances of failure are extremely small.

Critical System: Crew Lander–Fuel Cell
While a minor hydrogen or oxygen leak is not repairable, it is not a serious problem because there are sufficient reserves in other tanks. If a fuel cell unit fails before descent, it may be replaced with a spare unit on the MTV. If failure occurs on the Martian surface, it is not likely repairable. A back-up fuel cell may be used with minimum power.

Critical System: Surface Habitat Power Supply
Power is supplied by a nuclear reactor with a turbine powered alternator. In case of any failure, there is a 7-day back-up of fuel cells. Three possible failures are foreseen:

1) Reactor failure is irreparable and will result in early return to the MTV.
2) In the turbine-compressor:
   a) Bearing failure may be repairable depending on what bearing.
   b) Turbine blade failure are not repairable.
   c) Fluid leaks are not repairable.
3) In case of transmission line failure, back-up power lines can be engaged inside of minutes.

Irreparable power failures will result in an early return to the MTV. Failure of these systems is unlikely, however, because they are already proven technologies.

**Aerodynamic Analysis**
Critical System: RCS control system
Critical System: Thermal Protection System
Problems are generally repairable, taking a matter of hours.

**Thermal Group**
Critical System: Solar Flare Early Warning System
The Solar Flare Early Warning System must be tested periodically and repairable in order to ensure sufficient advance warning.

If fuel boil-off exceeds predicted quantities, the mission may need to be aborted, depending on the seriousness of the problem.

**Propulsion**
Critical System: Main Engine
Critical System: RCS
In case of turbopump/feedline failure, there should be sufficient redundancy to provide back-up for engines. Any repairs would be extremely difficult (if not impossible) and time-consuming.

**Structures**
The following possible problems will be remedied through prevention and robust designs:
Crew lander landing gear failure
Thrust structure deformation
Radiation degradation
Dynamic instability of the truss
Crew habitat depressurization.

Critical System: Crew Habitat
Depressurization may occur due to a meteorite or airlock failure. There is currently no repair for this.

### 3.5.3 Radiation
During the trip to and from Mars the biggest danger is the possibility of a solar flare. During the first two missions, this danger is at a minimum. However, solar activity will be at a maximum during the third mission. A small storm shelter which is part of the MTV HAB will protect the astronauts and main computer from the majority of the radiation. A warning system will be provided to alert the crew a few minutes ahead of time in case of a solar flare. Upon receiving the warning the crew will enter the storm shelter and stay there for hours or even days until the danger has passed.

The crew will also receive constant exposure to galactic cosmic radiation from which there is no reasonable defence. This is an unavoidable danger to the crew and equipment. The only way to reduce the exposure is to reduce the trip time.

### 3.5.4 Rover Crew Rescue
If a manned rover breaks down far away from camp, how can the stranded crew be rescued? There appear to be two options:

1. The crew are rescued with another rover -or-
2. The crew are rescued by 'hopping' the crew lander.

The major reason rescue with another rover would not be feasible is the time that it would take. Therefore, if rescue is to be made by rover, crews must not travel more than 7 days journey by rover. This time would depend on the capacity of both the stranded rover and the rescue rover to support life of the crew. Another option would be to rescue the stranded crew with a remotely controlled rover.

If the crew are rescued by a sub-orbital lander hop, it would require additional fuel. If it is possible to manufacture fuel (LOX and $LCH_4$) on the surface, there may be enough fuel to allow a hop. However, this technology will not have been tested on Mars before the first mission, so it will not be possible to rely on it for the first mission.

As a result, the manned rover should stay within range of rescue by one of the unpressurized rovers on the first mission If the proceedure for producing fuel *insitu* proves practical on the first mission, a lander hop may be relied upon for rover crew rescue on subsequent missions.

### 3.5.5 Dust Storms

While on the Martian surface, there will inevitably be dust storms. The habitat and equipment must be designed to withstand this environment. However, because of the low density of the Martian atmosphere, the wind is not very forceful. The main problem will probably be low visibility, and by dust working its way into equipment.

### 3.5.6 Ascent Vehicle/MTV Rendezvous

In case the targeted rendezvous with the MTV is missed, sufficient contingency fuel exists to rendezvous a second time within hours.

## 3.6 Earth Orbit Injection and Return

### 3.6.1 EOI Trade Study

In deciding how to return to Earth, there are numerous options:

1. The MTV may brake into a low energy parking orbit.
2. The MTV may brake into a higher energy parking orbit.
3. The crew may use an earth return vehicle to propulsively brake into a low or high energy parking orbit, from which they will either dock with SSF under their own power or they will be "rescued" by another vehicle which would take them to SSF.
4. The crew may use an earth return vehicle and aerodynamically brake into a parking orbit.
5. The crew may use an earth return vehicle and aerodynamically brake to a splashdown in the ocean.

Figure 3.18 shows graphically the different options available for Earth Orbit Injection.

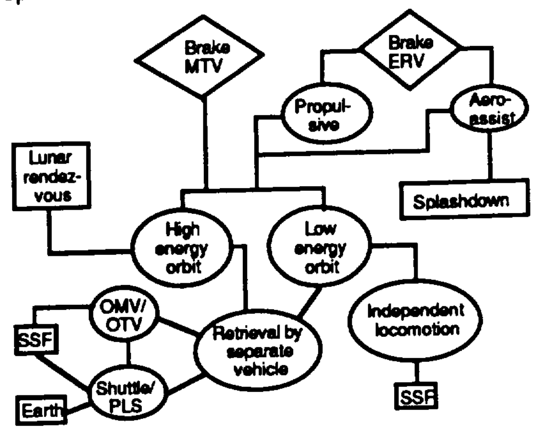



**Figure 3.18:** EOI Options

Figure 3.19 is a chart comparing IMLEO using different forms of EOI for the first mission. These masses were obtained utilizing Propulsion Systems's spreadsheet for calculating fuel masses. The "Total NTR burn" refers to insertion of the MTV into a 400 km parking orbit "Propulsive ERV" refers to propulsively braking the ascent pod into the same orbit. The "Limited NTR burn" refers to insertion of the MTV into a very high energy orbit. The "Total aerobrake" figures assume an additional mass for earth return of 20 tons and "Aerobrake ascent pod" assumes 13 tons for EOI.

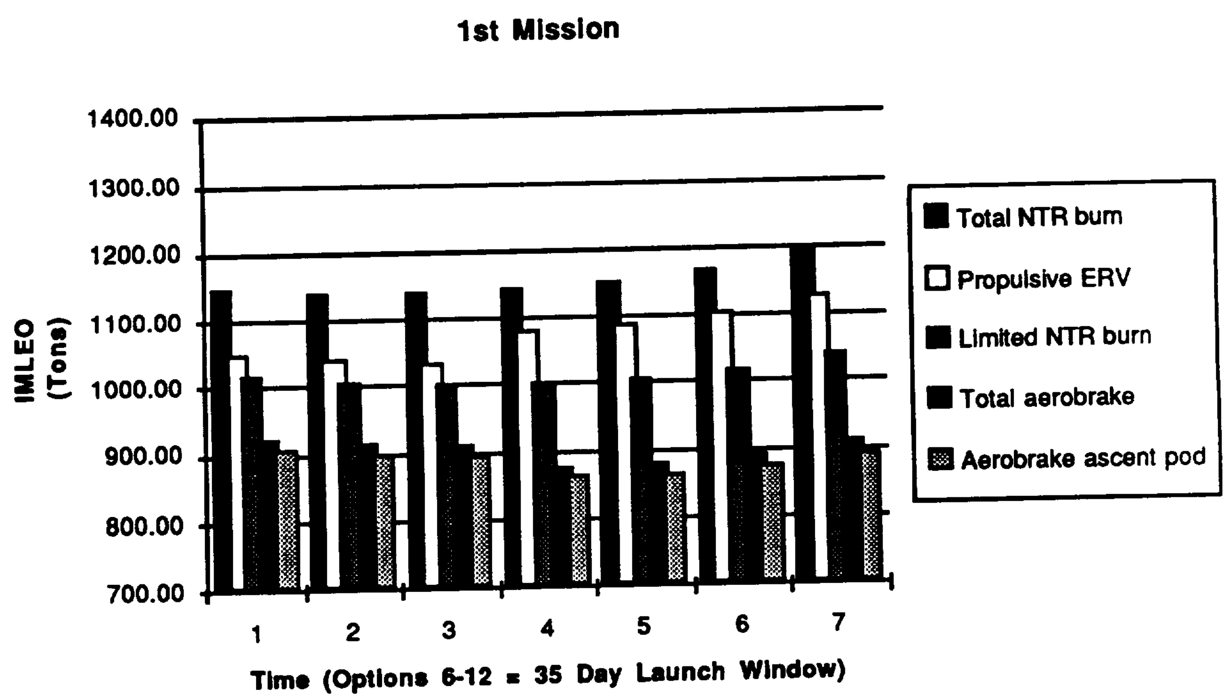



**Figure 3.19:** IMLEO for various Earth capture scenarios

#### 3.6.1.1 Braking the Entire MTV

As seen in Figure 3.19, braking the entire MTV at Earth requires about 300 tons of fuel (over the pure aerodynamic option) for the first mission and even more for some others. Since IMLEO is most expensive, this option is rejected.

#### 3.6.1.2 ERV Splashdown

It has been suggested that the crew return to Earth should always be accomplished through ballistic reentry and splashdown.[3.1] The reason given for this is a phasing orbit wait of weeks to reach SSF in LEO. However, because of the high entry velocity, this option is expected to require too much thermal protection and g's too high for the crew.

### 3.6.2 Final Earth Return Sequence

The crew will therefore return separately from the MTV using an aerobrake to capture the Earth Return Vehicle. Propulsion will be necessary to supplement the aerobrake. The ERV will not use a pure propulsive brake because the required fuel mass is in some cases much higher than the aerobrake/fuel combination mass.

The final earth return sequence follows:

1. Crew enters ascent module.
2. Ascent module moves to aerobrake at rear of vehicle by means of maneuvering thrusters.

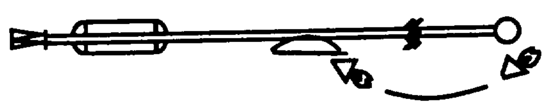


**Figure 3.20a:** MTV EOI configuration

3. ERV detaches from MTV.
4. ERV brakes in the atmosphere, perhaps requiring several passes.

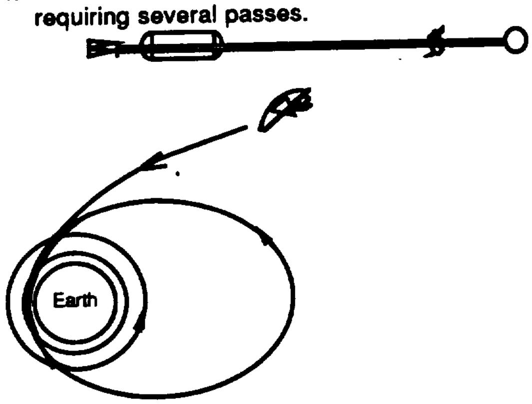



**Figure 3.20b:** EOI Orbital capture

5. ERV propulsively forms circular orbit of correct altitude and inclination. With an inclination change of 5° and circularization, total propulsion required for this is expected to be 1 km/s.
6. ERV rendezvous with SSF or Personnel Launch System.

If this orbit injection were done completely aerodynamically, it would take about 30 days. A balance must therefore be found between time of capture and required fuel mass for propulsion.

Transfer into an orbit around the moon would require an additional $\Delta V$ of about 0.9 km/s and at least 3 days.

If there is any fuel left over in the TEI tank, it may be used to capture the MTV as much as possible into a parking orbit before the crew departs.

## 3.7 Mission Timeline

A picture profile of the mission follows showing how the MTV configuration changes during the course of the mission. In Appendix 3.1 is a table which shows a more detailed timeline valid for all three missions.

### 3.7.2 Daily Crew Schedule

Even with years of training, it is important to remember that the crew is made up of living, human beings. It is therefore necessary to establish a work schedule which is compatible with mission requirements as well as human requirements such as privacy, leisure, and personal choice. With these considerations taken into account along with the specific minimum requirements of 8 hours/day of sleep, 1.5 hours/day of exercise, 3 meals/day and 40 hours/week of work, the following crew schedule was devised (the times are relative, but could be matched to that of Houston control):

| ACTIVITY | TIME | LENGTH |
|---|---|---|
| Post-sleep | 7:00 | 1.5 hrs. |
| Work | 8:30 | 5 hrs. |
| Lunch | 13:30 | 1 hr. |
| Work | 14:30 | 3 hrs. |
| Free time | 17:30 | 5.5 hrs. |
| Sleep | 23:00 | 8 hrs. |

**Table 3.2:** Daily crew schedule

The post-sleep period includes breakfast and personal hygiene. The work periods consist of specialized experiments, team projects, general maintenance, and further crew training. The free time period includes dinner, exercise, recreation, and the possibility to retire to bed early.

With 8 hours of work per day, each crew member will receive two days off each week. They all share Sunday off with perhaps one crew member "on call" each Sunday in case of an emergency. Each also has one day off per week that varies from person to person and possibly from week to week. The medical doctor can choose his second day off as situations can unexpectedly arise. Also, during important events such as the Mars approach or the Venus swing-by, it may be necessary for the crew to give up their variable day off for extended experiments.

With the proper training and with attention to the human factor when creating a daily schedule, the crew should be able to complete its mission requirements.

### 3.8 Crew Training

Training for the Mars mission is obviously important. Any adventure as complex as this requires the human components to function at least as well as the mechanical components. Much of the training will be done in established NASA facilities. Further training in such places as Antarctica, the lunar base, and Space Station Freedom is also essential to the mission's success.

The testing and training of the crew should take several considerations into account. The first of these is the actual technical training. The crew needs to learn how to apply their general skills to the specific mission requirements that will exist. The crew also needs to develop their secondary skills which could prove essential in a case of another crew member's illness or other unforeseen circumstances. Learning to function under space flight conditions is also required.

A second consideration for crew training involves human relations. The Mars mission provides a unique environment for the crew and the possibility of mental illness is very real. For this reason, it is important for the crew to build a sense of team unity as the training progresses. Also necessary is the ability to see the signs of hostility or withdrawal and the proper reactions to such signs.

A third consideration is stress regulation. The duration of this mission makes stress a large human factor. The crew is likely to go through some frightening situations which can lead to poor performance. Stress regulation can lessen the effects of fear through threat control (knowing how to counteract dangerous situations); fear control (being able to control the bodily and psychological states that define fear); and by alleviating chronic stress through biofeedback, relaxation, and meditation training.

Antarctica provides a unique location for testing the habitats, equipment, and supplies that will be used during the mission. The harsh environment will provide realistic conditions for the ruggedness of the materials, the personalities and abilities of the crew in confined conditions, and the testing and refining of scientific procedures. At the same time it is not as far away as the moon. The lunar base would provide a location for additional testing of surface procedures along with harsher conditions than Antarctica would provide and gravity more like that of Mars The Space Station allows for practice in micro gravity conditions and in the realities of space.

### 3.9 MEV Crew Operations

On the ground four astronauts will be in charge of running the unpressurized rover along with the large coring drill on daily missions that will all have to take place within 25 km of the landing area because the maximum range for the rover is 100 km in a day. This is based on an average speed of 10 km/hr and a 10 hour driving day. The ten hour figure is the maximum time that the astronauts can last outdoors with the oxygen available to them in their tanks. So, a 25 km drive will take approximately three hours, and with a four hour drilling period, and a three hour drive back to the habitation, a 10 hour day occurs. This crew will obtain core samples at radial directions around the base camp and take those samples back to the habitation module for testing. The drilling process with the large drill is a lengthy one and to obtain just a few samples will take a large part of the 60 days on the planet.

They will send the robotic rover out on its assigned surveying and mapping missions and will keep tabs on its location at all times. They will recall the rover when its sample return capacity (10 kg) is full.

These crew members will also be in charge of testing the samples brought back by the pressurized rover team. They will be responsible for setting up and taking down all equipment at the beginning and end of the missions. They will be responsible for setting up and monitoring the Mars environmental station on a daily basis. They will also be responsible for keeping in close communication with the pressurized rover on its missions, with Earth, via the MTV and with the robotic rover on its missions. In emergency situations these crew must summon the robotic rover and send it to the pressurized unit with extra oxygen tanks if the pressurized unit breaks down.

These four crew will work in two teams of 10 hours on, 10 hours off shifts.

#### 3.9.1 Pressurized Rover Sample Return Missions

Figures 3.22, 3.23, 3.24 are geological maps which detail the paths taken for the sample and return missions. The missions are numbered in order of succession. The small circle in the middle of each map is a 25 km range around the landing site where the habitat crew will use the large drill to obtain layers of rock samples. The crosses outline a 500 km range around the site.

The sample return missions have been based on a 10 hour day, with driving time averaging 10 km/hr. The two-man crew will spend half the mission driving and half the mission stopping in one-day spurts to take samples and drill. The pressurized rover's oxygen capacity can last about a dozen days maximum, so no missions will last over 10 days. In case the terrain is treacherous or the crew needs to spend more time at one site than another enough time has been left on each individual mission to allow for a worst-case scenario. Thus, unless the rover breaks down or a crew member is injured they will definitely have enough oxygen to make every sample return mission. The crew will rest and/or test samples in the habitat module for three days in between missions.

### 3.9.2 Five Pressurized Missions

On the first mission, since everything is new, all six crew members will help with the set up of the equipment, environmental station, laboratory, and telescopes. Two days will be allotted for this activity. After the setup phase has been completed the crew will begin their assigned missions. The sample missions will take 40 days total, with 12 days off, two days for set up, leaving six days of leeway if some contingencies arise.

On the second mission, if everything runs smoothly, the sample missions will take 46 days, with 12 days off and only two days of leeway.

On the third mission the sampling will take 42 days, with 12 days off with six days of leeway left.

Table 3.3 shows how long each sample return mission is and what will be sampled.

| Location | Mission | Range (km) | Length (days) | Geologic Samples Found |
|---|---|---|---|---|
| Olympus Mons | 1 | 374 | 8 | $Aoa_2$, $Aoa_1$, Aop |
| | 2 | 253 | 6 | $Aoa_2$, Ae, Aop, Aos, $Aoa_3$ |
| | 3 | 495 | 10 | $Aoa_2$, Aop, Amm, $Aoa_1$ |
| | 4 | 352 | 8 | $Aoa_2$, Aop, Aos |
| | 5 | 363 | 8 | $Aoa_2$, Aos, Aop |
| Valles Marineris | 1 | 434.5 | 10 | $Hpl_3$, HNu |
| | 2 | 451 | 10 | $Hpl_3$, As, Hf, HNu |
| | 3 | 495 | 10 | $Hpl_3$, Hvl, HNu, Hch |
| | 4 | 462 | 10 | $Hpl_3$, HNu, Avf |
| | 5 | 236.5 | 5 | $Hpl_3$, HNu |
| Mangala Valles | 1 | 335.5 | 8 | Nplr, Hchp, $Npl_1$ |
| | 2 | 341 | 8 | Nplr, $AHt_3$ |
| | 3 | 440 | 10 | Nplr, $Npl_1$, Hch |
| | 4 | 440 | 10 | Nplr, $AHt_3$ |
| | 5 | 242 | 6 | Nplr |

**Table 3.3**: Pressurized Rover Missions

## 3.10 Equipment for the Missions

### 3.10.1 Aboard the MTV

While aboard the MTV photographic equipment will acquire medium resolution pictures of stellar phenomena as the ship passes by, especially of Venus and the vicinity during the swingby. Fields and particles detectors will be used to test for the composition of the fields and the particles that exist in deep space. Remote sensing equipment will be used to attempt to detect waves from stellar bodies as well as capture the atmospheric waves that exist around Venus. An ultraviolet radiation monitor will be attached to the outside of the MTV to monitor the intensity and direction of the suns' radiation.

During the Venus swingby a Venus probe will be released down to the planet surface and will relay information to the MTV. Also dropped off on the Venus swingby and robotically emplaced in a heliocentric orbit around Venus will be solar flare monitors. The harmful radiation from gamma rays, x-rays, and energetic particles can then be predicted for the regions near future manned bases and interplanetary vehicles with warning times ranging from seconds (catastrophic blasts) to seconds (normal flares).

Aboard the MTV space environment and life sciences biological equipment will be used to test the effects of prolonged space travel on the human body. This will be especially important on the return journey after the astronauts have spent time on the Martian surface and are returning home with only three-eighths gravity.[3.1]

During the forty hours of work each week, the numerous experiments performed may include:

HUMAN PHYSIOLOGY
—bone demineralization, cardiovascular deconditioning, muscle atrophy, vestibular dysfunction, immune system, drug efficacy.

HUMAN PSYCHOLOGY/SOCIOLOGY
—isolated, confined and hazardous environment, stress assessment consequences, micro societal interactions.

ASTRONOMY
—astrophysics, planetary science, solar research including sunspots, flares, corona, and early radiation warning.

SPACE ENVIRONMENT EFFECTS
—micro gravity, variable-g, ultra-high vacuum, HZE particle irradiation.

LIFE SCIENCES
—ECLSS demonstrations, plant growth, microbiology, radiation environment.

MICRO METEOROID DETECTION
—effects on vehicle.

| | MASS (Kg) | VOLUME ($m^3$) | POWER (W) |
|---|---|---|---|
| Fields and Particle Analysis | 50 | 0.2 | 21.5 |
| Solar Flare Monitor[1] | 300 | 1.3 | 15 |
| Venus Probe[2] | 300 | 5 | 147 |
| Large Camera | 25 | 0.22 | 0 |
| Remote Sensing Equipment | 150 | 0.6 | 100 |
| 25 cm Aperture Telescope | 130 | 1.3 | 200 |
| UV Radiation Monitor | 15 | 0.11 | 50 |
| Imaging Impactors (12)[3] | 600 | 2.4 | 0 |
| Film Locker | 115 | 0.25 | 0 |
| Space Environment/Life Sciences Biological Equipment | 200 | 1 | 100 |
| TOTALS | 1885 | 12.38 | 633.5 |

**Table 3.4:** MTV Science

[1]Dropped off on Venus flyby, & robotically emplaced in a heliocentric orbit around Venus, to monitor the solar flare activity. The harmful radiation from gamma rays, x-rays, and energetic particles can then be predicted for the regions near manned bases and interplanetary vehicles with warning times ranging from seconds to days.

[2]The Venus Probe will be released upon swingby of Venus, and will relay information to the MTV

[3]Photographic equipment will get medium resolution pictures from the MTV, and these ranger-like impactors will be released from orbit and provide submeter images of small regions in selected sites (to provide for landing and roving traffic ability)

### 3.10.2 Ground Operations Equipment

When touchdown occurs for each mission and the crew has successfully unpacked the stowed equipment they will begin to ready the site for exploration. At each site a Mars environmental station will be put in place to monitor the seismicity, heat flow, atmosphere, and other parameters of interest around each site.

A geoscience laboratory will be set up inside the habitat to do basic studies of rocks and minerals in support of geologic field work. It allows on-going field studies to have laboratory support. This is especially useful for the crew that will be using the unpressurized rover on short EVA's around base camp. The laboratory elements consist of microscopes, spectrometers, diffractometers and thin sectioning equipment that will allow the astronauts to do basic radiometric carbon dating and mineralogical classification experiments.

Each astronaut will carry with him an astronaut field package which consists of tools (hammers, chisels, shovels) and cameras. Using this equipment, astronauts will perform field studies on the surface to determine the local and regional geology of the planet, and to provide ground truth for orbiter studies.

Mars drills will also be a basic necessity for the crew. The recovered Mars core samples can provide data on surface composition and history, atmosphere-surface interactions, surface volatile (most importantly water-laid) inventories, climate history, and the organic history of Mars regolith and near-surface rocks. A small drill can be carried as part of and operated by a robotic rover. Another small drill can be carried along with the pressurized rover to distances away from the habitat. The large drilling rig will require astronaut-assisted setup. The drilling scenario involves three stages: (1) A small machine similar to the Apollo and Luna drills deployed from the rover and/or operated by the robotic rover extracts as many cores as power will permit, with examination of the cores by the robotic rover equipment or in the pressurized rover laboratory with selected samples being transported back to the habitation module or selected for transport back to earth. (2) More extensive drilling with a small manned drill at regolith sections where the first layer of rock is the subject of interest. (3) A deeper drilling scheme to examine the underlying layers of rock. The automated drill can drill cores of 2 cm diameter to a depth of 1m in soil. The manned drill can drill cores of 2 cm diameter to a depth of 10m in soil and 10 cm in crystalline rock. The heavy drill can cut cores to 10 m in solid rock. The extent and rate of drilling will be limited by the power and total energy available. Power may have to be time-shared with other systems drawing from the same power source (particularly in the case of the deep drill).

Automated telescopes will be emplaced around the landing site to allow the crew to view large portions of the planet surface at all times. The automated scope will keep tabs on what is happening atmospherically at all times of day.

Mars balloons will be used to fly over lava flows, canyons and channels to bring automated equipment and surface cameras to places where the robotic rover cannot go.

Sample return containers will be packed full of samples for the return journey to earth. Each fully loaded container will weigh 200 kg and there will be a total of ten containers returning to the MTV, for a total weight of two metric tons.

| | MASS (Kg) | VOLUME ($m^3$) | POWER (W) |
|---|---|---|---|
| Geoscience Lab Equipment | 100 | 0.5 | 200 |
| Robotic Rover [1] | 800 | 16 | RTG |
| Pressurized Rover | 4000 | 70 | dips |
| Power Cart+Radiatior | | | |
| Pressurized Rover Lab | 100 | 0.5 | 200 |
| Unpressurized Rover | 550 | 14 | Lithium Battery |
| Mars Environmental Station | 100 | 1 | 250 |
| Mars Drills[2] | | | |
| Rover Drill | 30 | 0.1 | 500 |
| Manned | 50 | 0.1 | 500 |
| Large Drill | 500 | 10 | 5000 |
| Astronaut Field Package (Cameras, Tools) | 90 | 0.5 | 0 |
| Sample Return Containers | 200 | 0.3 | 0 |
| Automated Telescope[3] | 400 | 1.5 | 200 |
| Mars Balloons | 200 | 1 | 3.5 |
| Electric Hoist (23mT) | 28 | 0.1 | 190 |
| TOTALS | 7148 | 115.5 | 7043.5 |

**Table 3.5:** MEV and Ground Operations Equipment

[1] Rover can be controlled remotely and will explore the surface to obtain samples from diverse geological units and conduct a basic survey of the terrain.

[2] The large drill will be attached to the rover and carried by the rover power cart. It is capable of Mars regolith trenching (will be important in the channels and canyons), and has the capability to drill/core to 10m

[3] The automated telescope is self-leveling and self-aligning. It acquires targets, tracks them, and sends the information back to base camp. It will be put in place by astronauts on expedition.

Both the pressurized rover and the robotic rover will contain some basic geoscience laboratory equipment for the determination of radiometric age and mineral type so that analysis can be done in the field on short notice so that a return to the major laboratory is not necessary. Table 3.5 lists all science equipment and the rover specifications. Figure 3.22 shows the unpressurized rover. Figure 3.23 is the pressurized rover and dips system.

## 3.11 Features of the Landing Sites

### 3.11.1 Olympus Mons

This site is located at 13° N latitude, 139° W longitude and is at an altitude of 2 km, according to the Mars baseline pressure estimate.

The western volcanic assemblage consists of relatively young materials erupted from and around large volcanoes and fissure vents. The Olympus Mons formation consists of a broad range of geologic members. The oldest are aureole deposits of uncertain composition and origin. They are broad, flat, sheetlike, deposits whose surfaces are grooved, ridged, and faulted. Formation of the aureoles is considered to have been caused by either volcanic or gravity-assisted tectonics. Proposed volcanic origins include lava flows and ash-flow tufts of material. Postulated gravity-assisted processes include thrusting of layered material from beneath the Olympus Mons shield, and gravity sliding and spreading of the sides of the volcano. These tectonics might have occurred with the aid of ground ice that spread over the plain surrounding the volcano. The aureoles are overlapped by the steeply sloping shield member of the Olympus Mons formation that was extruded from the summit and sides of the volcano. This member and the aureoles are, in turn, buried in places by flows of the plains member that originated from fissures east of the shield. The lowermost aureole member underlies and overlaps fractured materials. A few older flows from Olympus Mons are sharply truncated at the steep slope or are degraded and exposed in windows above the slope. Where these older surfaces exhibit faults, they have been mapped as younger fractured material. Although crater counts of the aureole deposits have been attempted, their accuracy is questionable because mass wasting of the rough and apparently soft surfaces of the aureoles promotes rapid deterioration of the crater forms. The density of positively identifiable impact craters on the aureoles is lower than that of younger materials that surround the aureoles.

The mission to this area will allow the crew to come close to a semi-active volcanic assemblage. They will be able to explore the surrounding countryside to determine the age of these volcanoes, when they last erupted and what types of ashen substances they have spewed over the years.

### 3.11.2 Valles Marineris

This site is located at 6° S latitude, 78° W longitude and is at an altitude of 1 km.

It is hard to say exactly when the canyon system known as Valles Marineris was actually formed, but the canyons are composed of a sequence of thick plateau rocks, capped around their western part by ridged plains material. Significant canyon slope retreat appears to have followed the emplacement of these cap rocks, but it is not known if all of these units were breached by the developing canyon system. Possibly the ridged plains material was in part lava flows extruded from fault fissures that later formed Valles Marineris. Windblown and fluvial excavation of the central Valles Marineris region is assumed to be minor at best. The following sequence of formation of the canyons and associated outcropping channels is a good approximation. (1) Deep-seated heating from the planet core resulted in crustal expansion and rifting. (2) Possible concurrent volcanism from rift faults formed the ridged lavas that surround Valles Marineris. Withdrawal of this lava caused further canyon expansion. (3) Layered deposits were emplaced either in lakes that filled the canyons or as ash-fall deposits. (4) Outflow channels formed from catastrophic release of water from some canyon lakes. (5) The canyons continued to expand by faulting and land slides. (6) Eolian and fluvial erosion removed and transported most layered deposits.

At this site the crew will be able to explore the effects of volcanic tectonics on the formation of the canyons. They will also be able to determine when the water that carried the ashen sediments was present and whether it occurred because of migrating glaciers from the ice caps, or whether lakes and streams were present.

### 3.11.3 Mangala Valles

This site is located at 7° S latitude, 147° W longitude and is at an altitude of 2 km.

The formation of the channels has long been attributed to erosion by running water, as well as other suggested processes that include wind erosion, lava erosion, and glaciation. The key to determining the cause of formation is whether or not there was enough water present to actually erode these huge channels. The deeper channels probably originated by tectonics and by sublimation of ground ice, which produced incipient linear depressions that rivers later eroded into channels. Various processes have been proposed to account for the release of water. Some suggest that large volumes of water may have come from the melting of closely spaced ice in the subsurface because of volcanic heating. Small channels could have retreated headward occasionally intercepting subsurface fluid reservoirs, causing sudden increases in flow rates or there could have been underground lakes that protruded toward the surface because of volcanic activity.

The crew will determine the effects that volcanic tectonics played in the formation of the channels. It needs to be discovered whether or not the polar ice caps at one time stretched into the southern hemisphere and caused major glacial slides and left these underground wells of water that might have created these channels.[3.1]

## 3.12 Geologic Phenomena of the Landing Sites

Gray-scaled geologic maps of the area in the general vicinity of the landing site can be seen in Figures (3.24,3.25, and 3.26). The geologic units that are to be sampled are the different colors of the maps. The small circle in the middle is a 25 km area surrounding the lander that will be the primary heavy drill area. The dark crosses are markers of 500 km range around the site.

### 3.12.1 Olympus Mons

The actual landing site ($Aoa_2$) is a member consisting of an aureole lobed ridge on the southwest side of Olympus Mons. The site overlooks unit Aop which is a plains member made up of overlapping smooth lava flows ranging in shape from narrow tributaries to broad flows. Located within 500 km of the landing site are geologic finds characteristic of the western volcanic assemblage. The members are listed below.

Aos - A shield member composed of lava flows of channels and levees that extend down the sides of Olympus Mons. Collapse pits are common in this area and tend to cause large holes in the surface

$Aoa_3$ - An aureole lobe similar to the landing site.

Ae - Broad level plains chiefly formed from the windblown deposits of Olympus Mons. The surfaces appear rough, deeply etched, and striated in the directions of the prevailing winds.

Amu - An ash-fallout area thick with the windblown accumulation from the volcano and from wind erosion.

$AHt_3$ - Makes up the central shield area around the volcano. It contains many faults and fractures.

As - Slide material of volcanic-debris avalanches resulting from either slope failure or explosive volcanism. They may also be the debris of rocks and gravel left by a melting glacier from the former local ice caps.

$Aoa_1$ - Overlaps younger and older fractured materials. An aureole member that resembles the landing site.

Hf - Younger fractured material. Forms raised surfaces of moderate relief with fractures and collapse depressions.

$Aa_3$ - Smooth plains west of Olympus Mons.

Amm - Rough surface, deeply eroded pyroclastic rocks or layers of relatively soft eolian deposits.

The dark lines in the picture represent faults in the surface. The small crosses around Olympus Mons are the crest of the basal scarp of Olympus Mons.

### 3.12.2 Valles Marineris and Canyon

The site ($Hpl_3$) is a smooth unit consisting of flat, featureless plains formed from eolian deposits that bury most of the underlying rocks. It is located on a ridge between Candor Chasma and Hebes Chasma, two deep canyons, and is within Valles Marineris. The deposits located within 500 km of the site are listed below.

As - Rotational slide deposits from canyon walls, surfaces are longitudinally striated and were probably formed by gas or water lubricated debris flows.

Np1 - Ridged unit due to normal faulting with intervening areas rougher and more densely cratered.
Hr - Ridged plains material characterized by parallel wrinkled ridges 30 to 70 km apart.

Hsu - Relatively smooth long narrow lava flows and sheet flows where pit craters are common.

HNu - Walls and interior mountains in Valles Marineris and associated canyons. Erosion remnants and exposures of plateau sequence, highly deformed terrain materials, and ancient crater rims.

Hvl - Volcanic or lacustrine material deposited during intermediate to late stages of canyon development.

Avf - Canyon floor material. A mixture of landslides and debris flows from canyon walls, eolian material, volcanic deposits and possible lacustrine deposits.

Hch - Generally a tiled area of highland blocks within the canyons formed by chaotic events.
Hsl - Similar to Hsu but more highly cratered and faulted.

Hf - Relatively smooth, raised surfaces of moderate relief with fractures and collapse depressions.

The dark lines with circles are faults in the surface. The dark lines with diamonds are wrinkled ridges.

### 3.12.3 Mangala Valles and Channel

The site is composed of ridged plains material and is located just south of the Amazonis Sulci, the transition zone between the highland and lowland physiographic sequences. The deposits located within 500 km of the site are listed below.

Hchp - Channel deposits longitudinally striated of flood-plain material occurring adjacent to channels and in lowland plains below channel mouths.

$Npl_1$ - Highly cratered, uneven surface of moderate relief with fractures, faults, and small channels common. Materials were probably formed during a period of high impact flux, and are most likely a mixture of lava flows, pyroclastic material, and impact brecchias.

Hch - Channel and flood-plain materials of alluvial origin with some surfaces sculptured by flood waters. The chaotic materials were probably formed by the disruption of the terrain by ground water release.

Nplh - Rough, hilly material. Ancient highland volcanic rocks and impact brecchias uplifted by tectonism.

Amm - Pyroclastic rocks or layers of relatively soft windblown deposits.

$AHt_3$ - Central shield region with faulted materials.

Mountain - Directly to the south of the landing site is an unnamed mountain that can be explored by air.

The dots signify the Amazonis Sulci region.[3.2]

## 3.13 Conclusion

At the end of the of the design study it was obvious to the Mission Operations group that a number of areas need intensive research and development if the envisioned mission is to succeed.

Much development is required for all of the MTS components. Perhaps the most important of these is the heavy launch capacity without which no large space effort is possible. The ground operations equipment for the Martian surface also needs to be developed. Time spent on the Martian surface will be exceedingly valuable and what the crew accomplishes will be directly affected by their equipment.

In addition to the hardware related studies, a number of operations topics need to be studied more in the future. For Earth Orbit Injection,in particular, all of the different alternatives need to be carefully evaluated to be sure of the most efficient and safest alternative. For the mode of EOI chosen for this mission(aerobrake into LEO), careful study needs to be done on the time and fuel required. Much more analysis also needs to be performed on critical systems and risk assessment.

Finally, one must always remember the human component. If people go to Mars, the risks must be weighed very carefully, for they are potentially great. It will be a very long time before space travel can be called even reasonably safe.

**FIGURE 3.1 NLS Derived HLLV with 4 LOx/RP Boosters**

PAYLOAD: 224 mT
Shroud Volume:14.02 x 30.48 m
Shroud Mass: 51 mT
Number of Boosters: 4, each with two F-1A engines

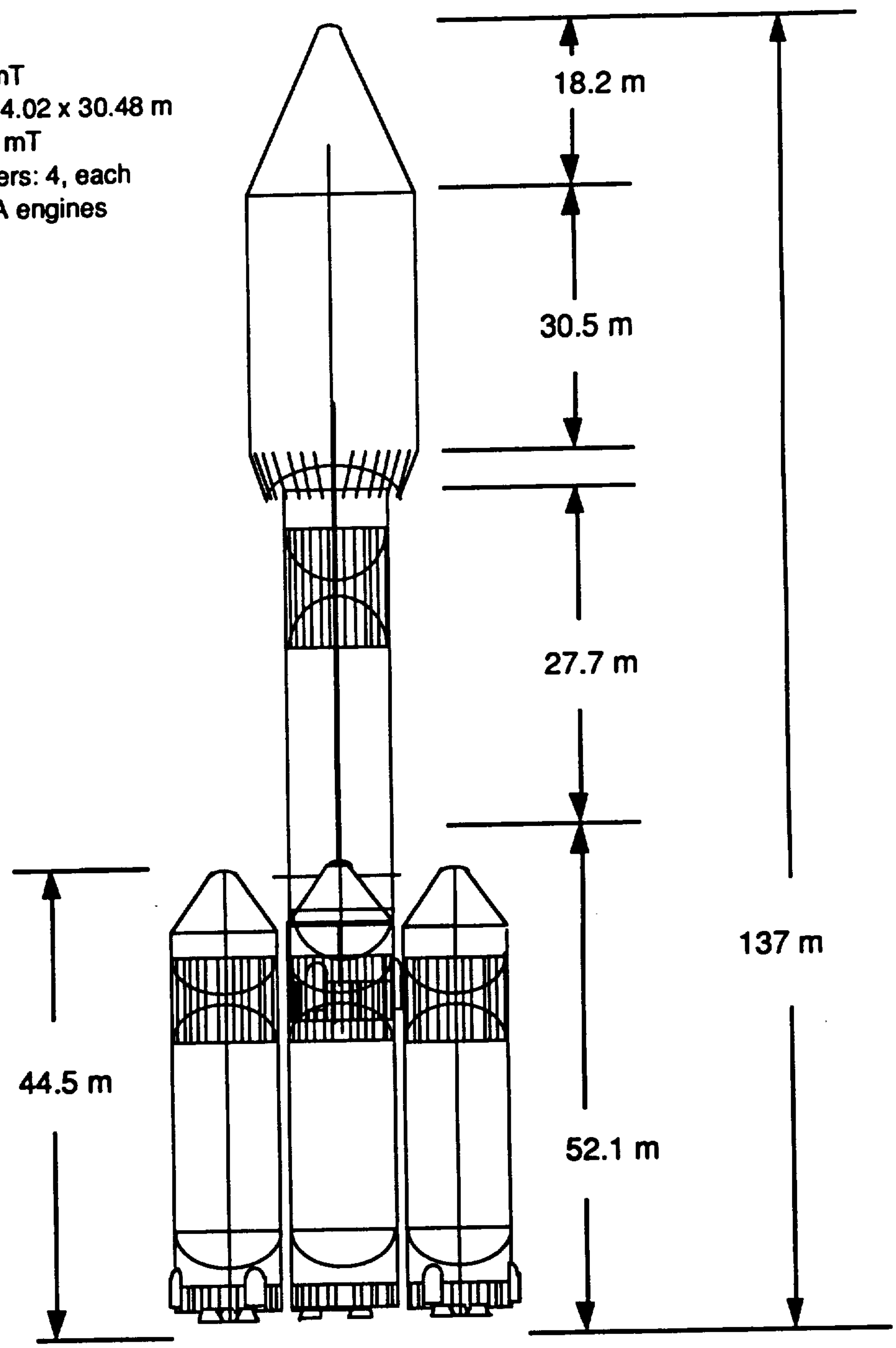
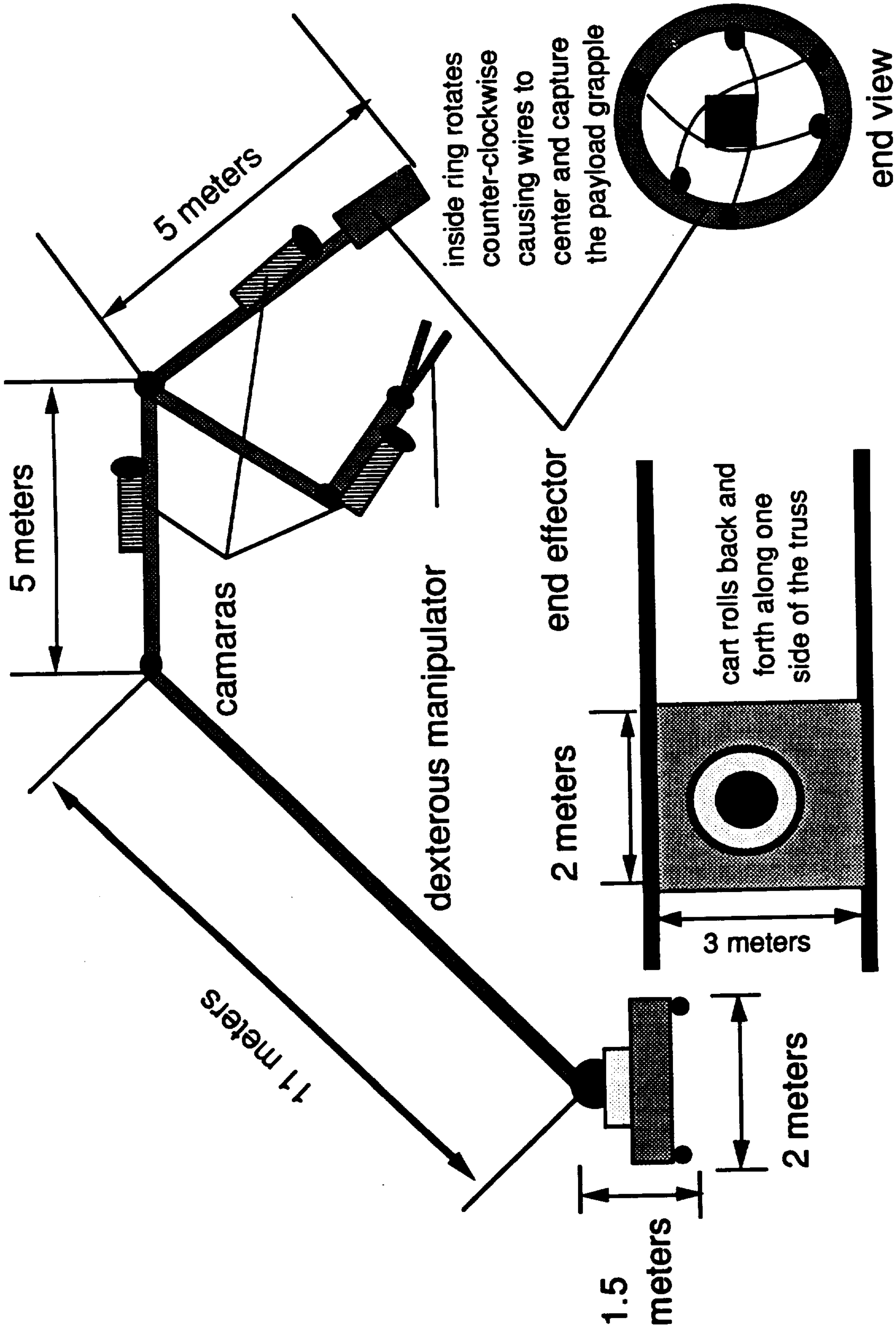


FIGURE 3.2 REMOTE ASSEMBLY SYSTEM



Launch Number 1
Mission Clock: T-727 days
Launch Vehicle Description: NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size: 14.02m x 30.48m
Mass to ETO: 174 mT
$M_F$: 79%
Payload Description:
- Biconic
- Remote Assembly System (RAS)
- Four 27 m sections of truss
- One 21 m section of truss
- One 16.5 m section of truss

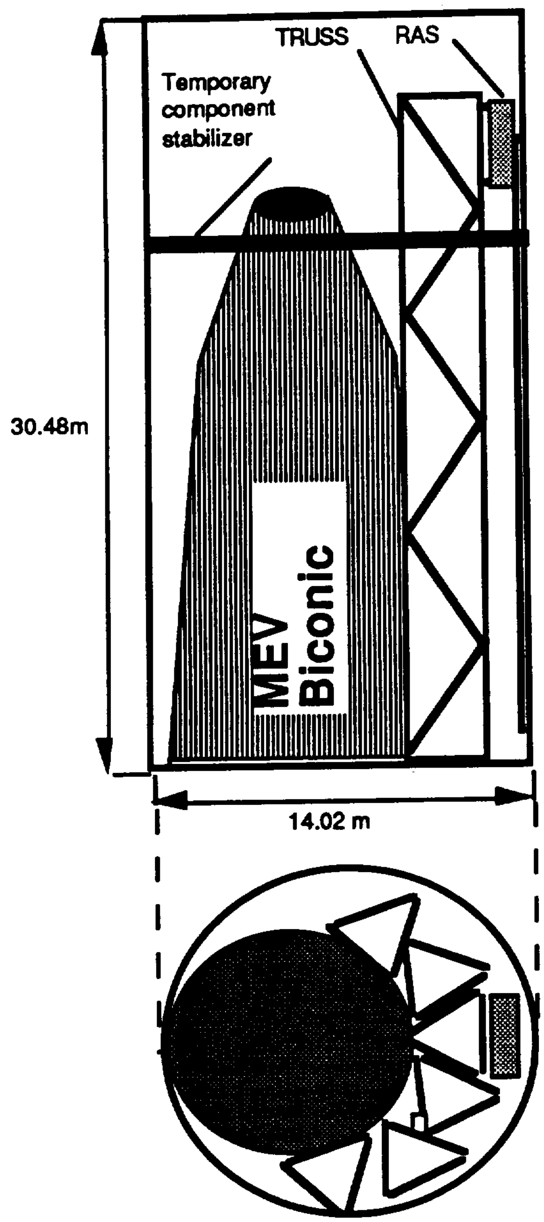



Figure 3.3: Launch 1 Payload

Launch Number 2
Mission Clock: T-637 days
Launch Vehicle Description: NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size: 14.02m x 30.48m
Mass to ETO: 148 mT
$M_F$: 67 %
Payload Description:
- MTV Habitat with 4.5 m Section of Truss Attached
- Earth Return Aeroshell
- Mars Aeroshell with Lander Attached

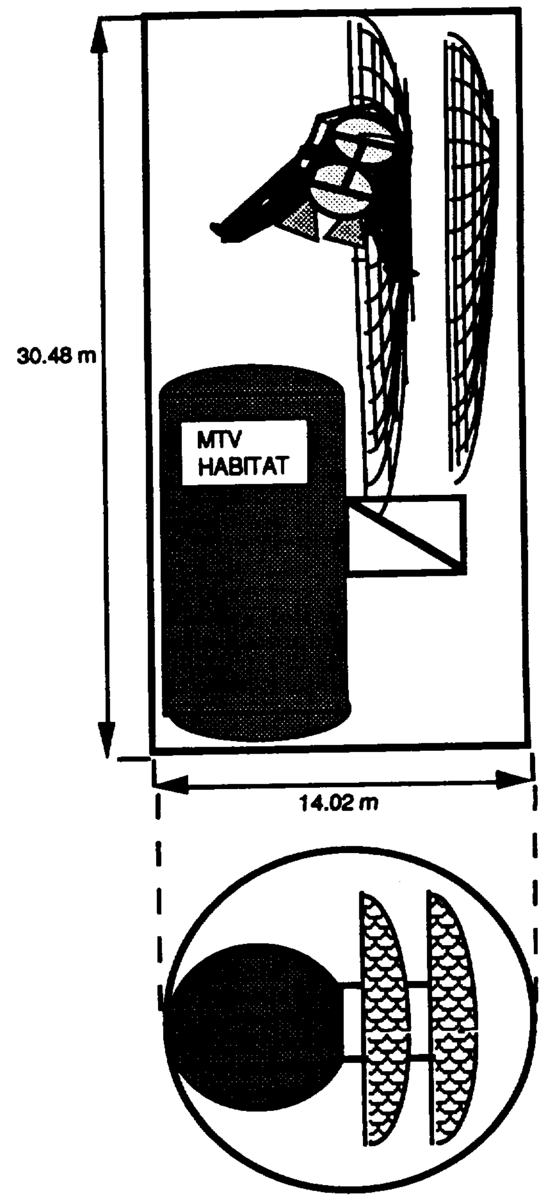



Figure 3.4: Launch 2 Payload

Launch Number 3
Mission Clock: T-547 days
Launch Vehicle Description: NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size: 14.02m x 30.48m
Mass to mT
$M_F$:47 %
Payload Description:
MOI Tank 1 with Fuel
NTR Truss Structure, Base, & Engines

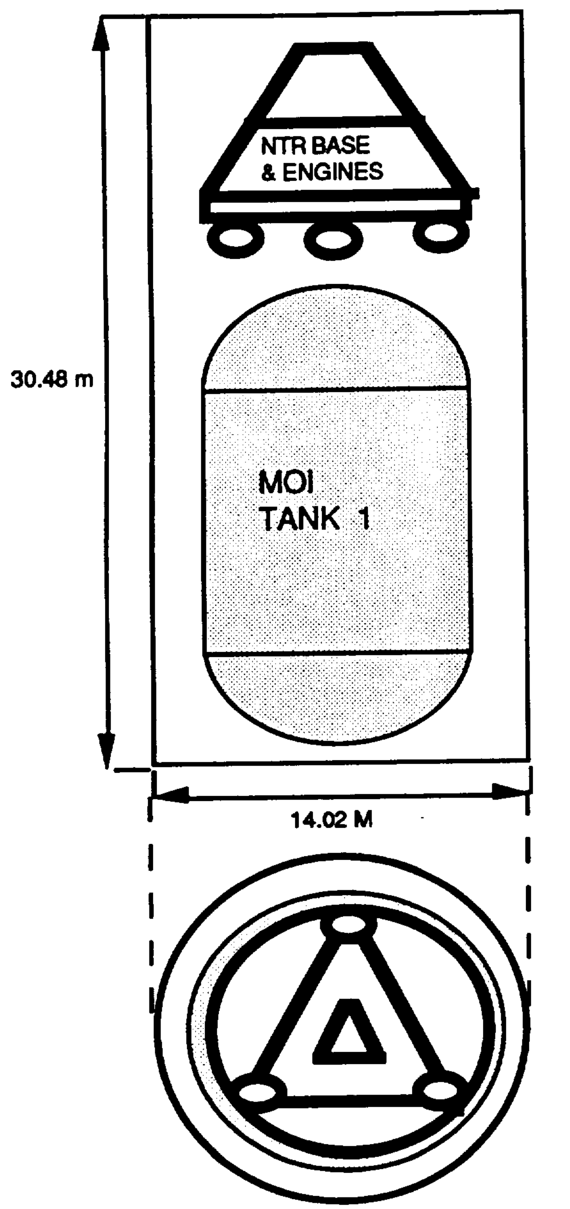



**Figure 3.5:** Launch 3 Payload

Launch Number 4
Mission Clock: T-457
Launch Vehicle Description: NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size: 14.02m x 30.48m
Mass to ETO:100 mT
$M_F$:46 %
Payload Description:
MOI Tank 2 and Fuel
NTR Nozzle

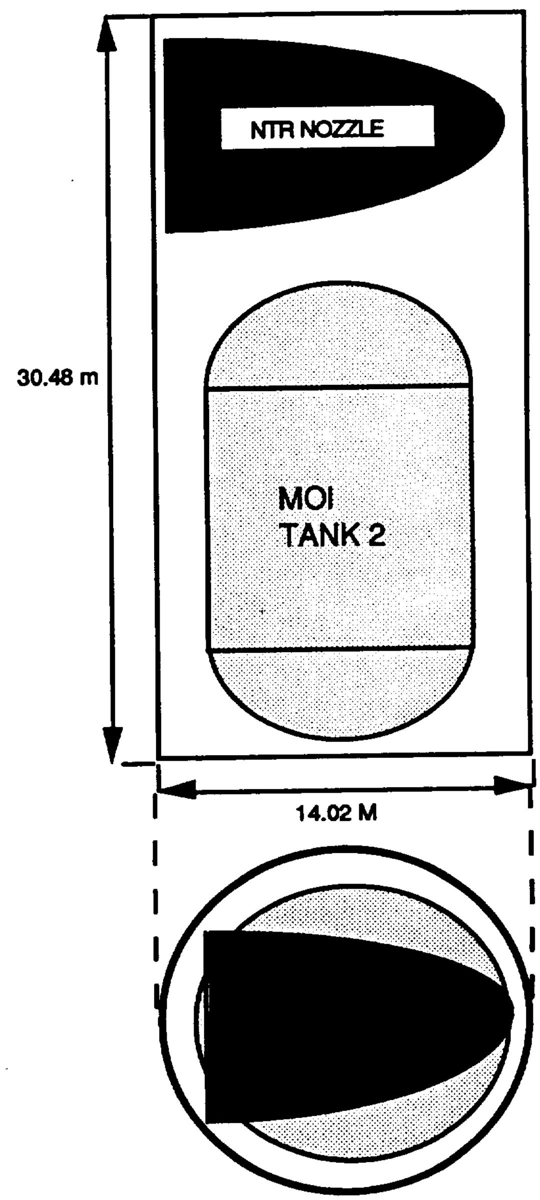



**Figure 3.6:** Launch 4 Payload

Launch Number 5 and 6
Mission Clock:T-367 days and T-277 days
Launch Vehicle Description:NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size:14.02m x 30.48m
Mass to ETO: 112 mT
$M_F$:51%
Payload Description:
TMI Tank and Fuel
NTR Nozzle

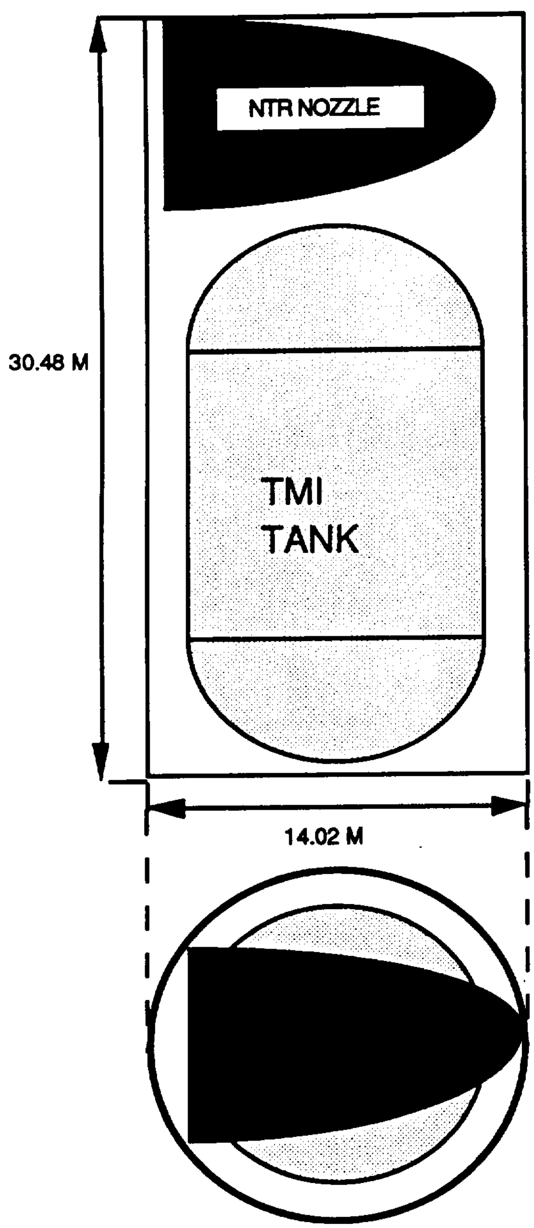



**Figure 3.7:** Launch 5 & 6 Payload

Launch Number 7
Mission Clock:T-187days
Launch Vehicle Description:NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size:14.02m x 30.48m
Mass to ETO:165 mT
$M_F$:75 %
Payload Description:
TEI Tank 1 and Fuel

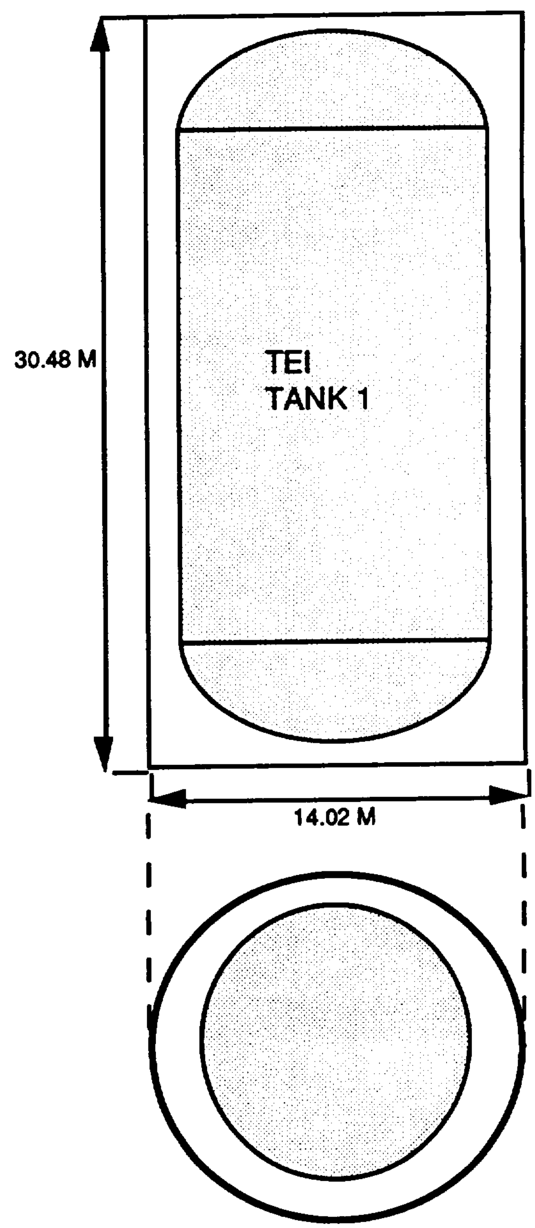



**Figure 3.8:** Launch 7 Payload

Launch Number 8
Mission Clock:T-187days
Launch Vehicle Description:NLS derived HLLV
Final Position: 220 circ. @ 28.5°
Shroud Size:14.02m x 30.48m
Mass to ETO:121 mT
$M_F$:55 %
Payload Description:
TEI Tank 2 and Fuel
TMI Tank 3 and Fuel

Launch Number 9
Mission Clock:T-7 days
Launch Vehicle Description:PLS
Final Position: 220 circ. @ 28.5°
Payload Description:
Crew

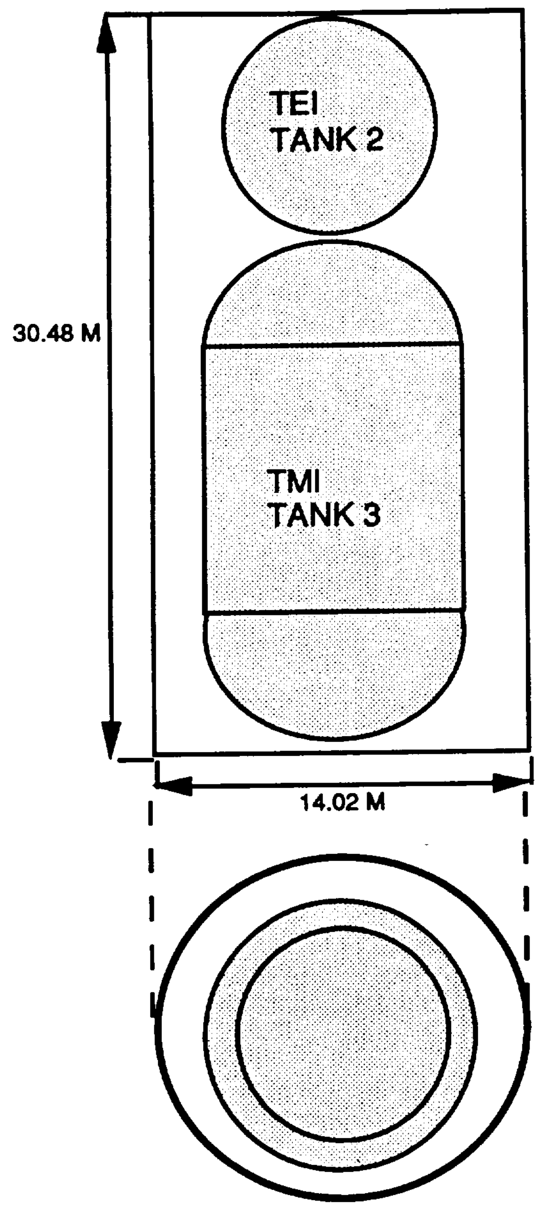



**Figure 3.9:** Launch 8 & 9 Payload

ASSEMBLY 1:

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move comonents with CRV | |
| Assemble 7 pieces of truss | |
| Check Navigation Equipment | |
| Check Communication Equipment | |
| Check main RCS System | |
| Check Structural, Electrical, & Fuel connections | |
| TOTAL FOR ASSEMBLY 1: | |

ASSEMBLY 2:

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach MTV habitat | |
| Attach Earth Aeroshell | |
| Attach Mars Lander / Aeroshell | |
| Check Structural connections | |
| Check MTV habitat systems | |
| TOTAL FOR ASSEMBLY 2: | |

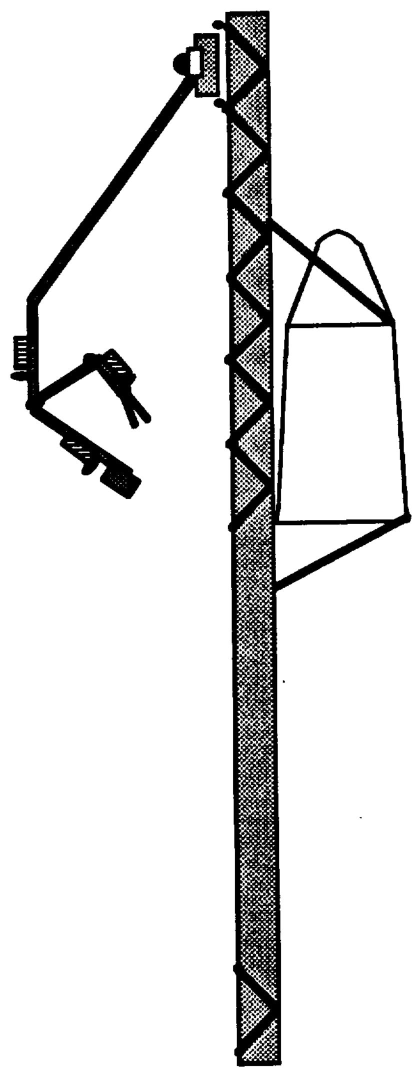


Figure 3.10: Assembly Phase 1

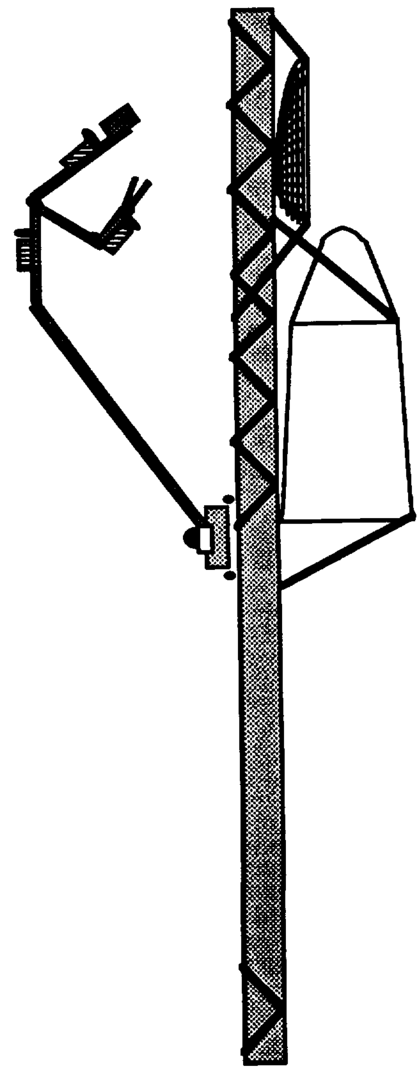


Figure 3.11: Assembly Phase 2

ASSEMBLY 3:

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach NTR Base with Engines | |
| Attach MOI Tank 1 | |
| Make fuel line connections | |
| Check structural, electrical and fuel line connections | |
| TOTAL FOR ASSEMBLY 3: | |

ASSEMBLY 4:

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach NTR Nozzle | |
| Attach MOI Tank 2 | |
| Make fuel line connections | |
| Check structural, electrical and fuel line connections | |
| TOTAL FOR ASSEMBLY 4: | |

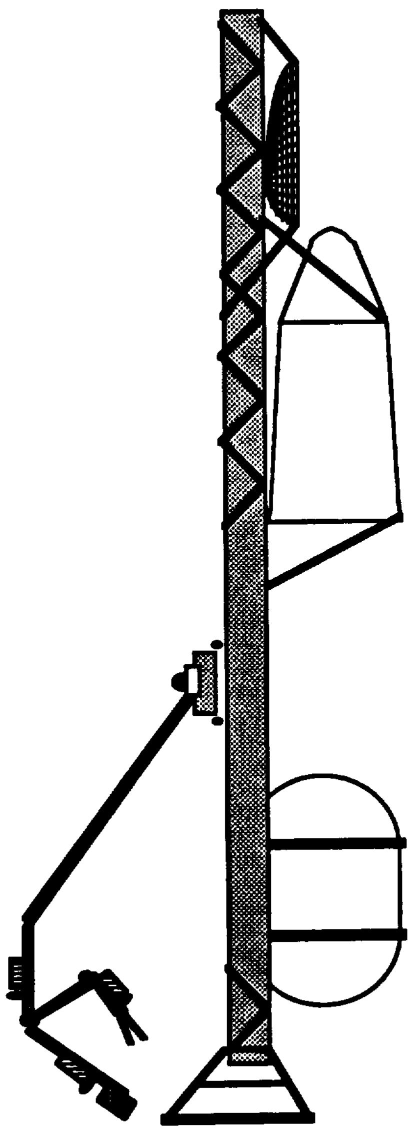


**Figure 3.12: Assembly Phase 3**

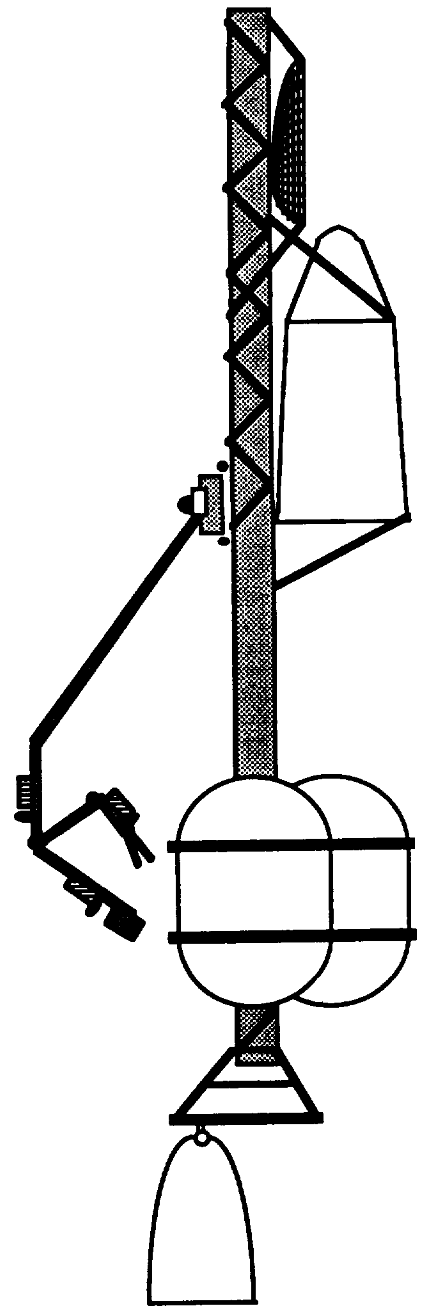


**Figure 3.13: Assembly Phase 4**

ASSEMBLY 5 & 6

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach NTR Nozzle | |
| Attach TMI Fuel Tank | |
| Make fuel line connections | |
| Check connections | |
| TOTAL FOR EACH: | |

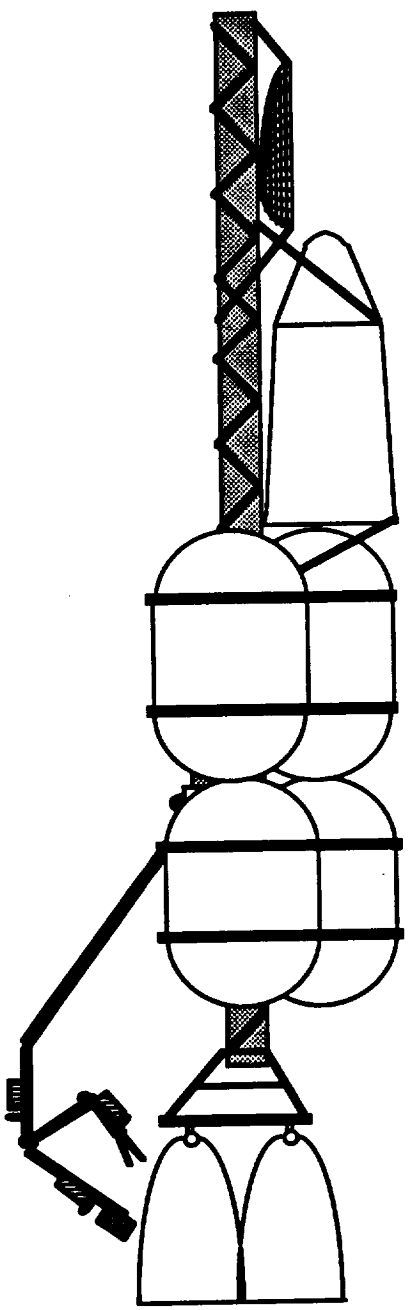


**Figure 3.14: Assembly Phases 5 & 6**

ASSEMBLY 7

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach MOI Tank 2 | |
| Attach TEI Tank 1 | |
| Make fuel line connections | |
| Check connections | |
| TOTAL FOR ASSEMBLY 7: | |

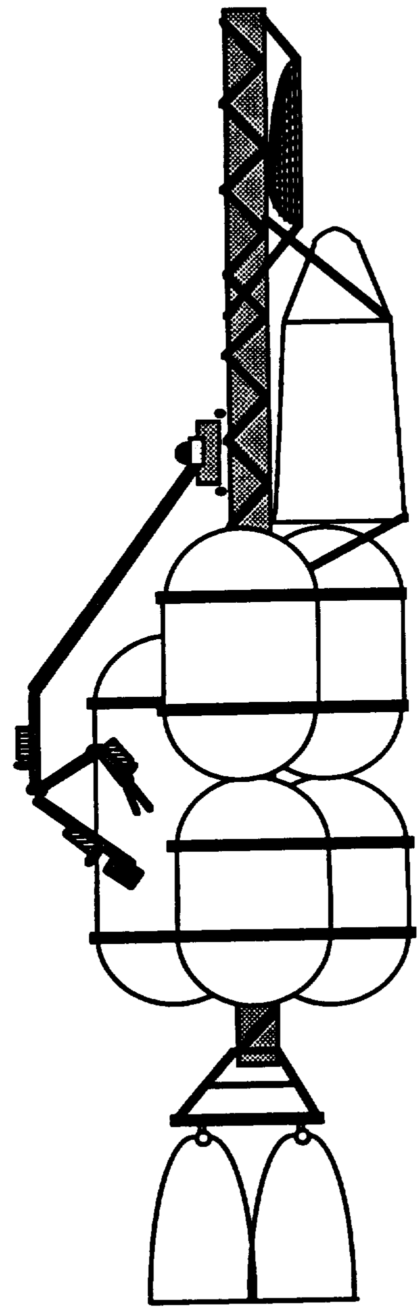


**Figure 3.15: Assembly Phase 7**

ASSEMBLY 8

| TASK | TIME |
|---|---|
| Detach components from HLLV | |
| Move components with CRV | |
| Attach TEI Tank 2 | |
| Make fuel line connections | |
| Check connections | |
| TOTAL FOR ASSEMBLY 8: | |

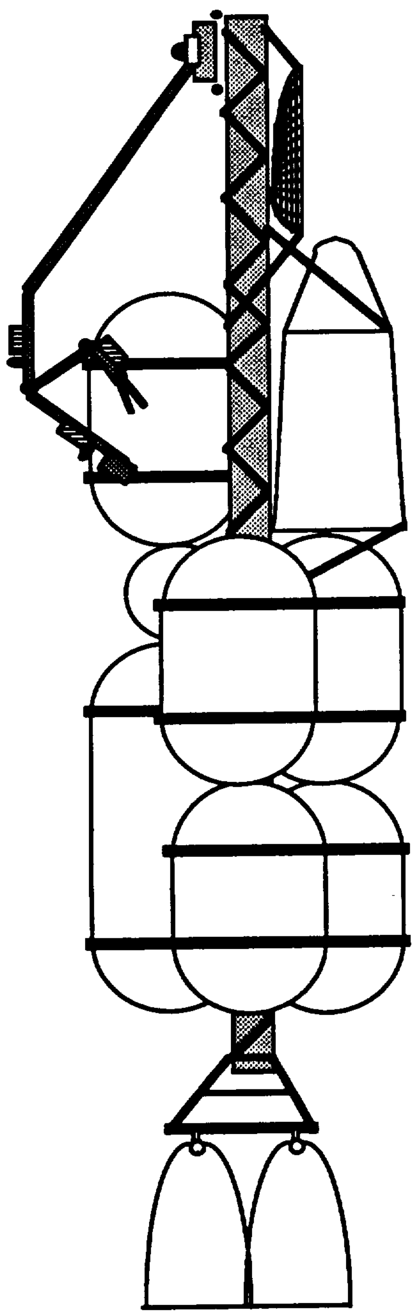


**Figure 3.16: Assembly Phase 8**

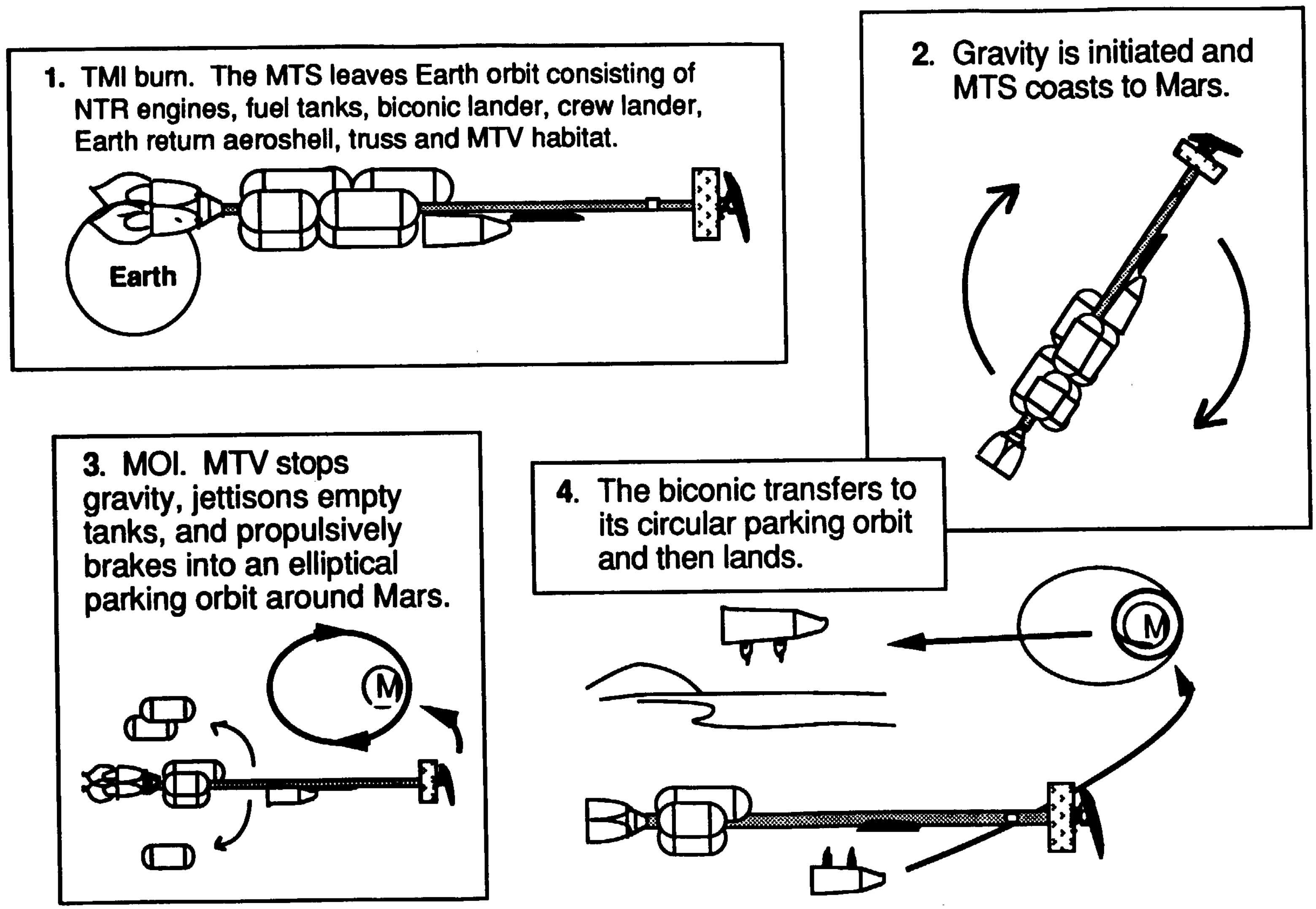
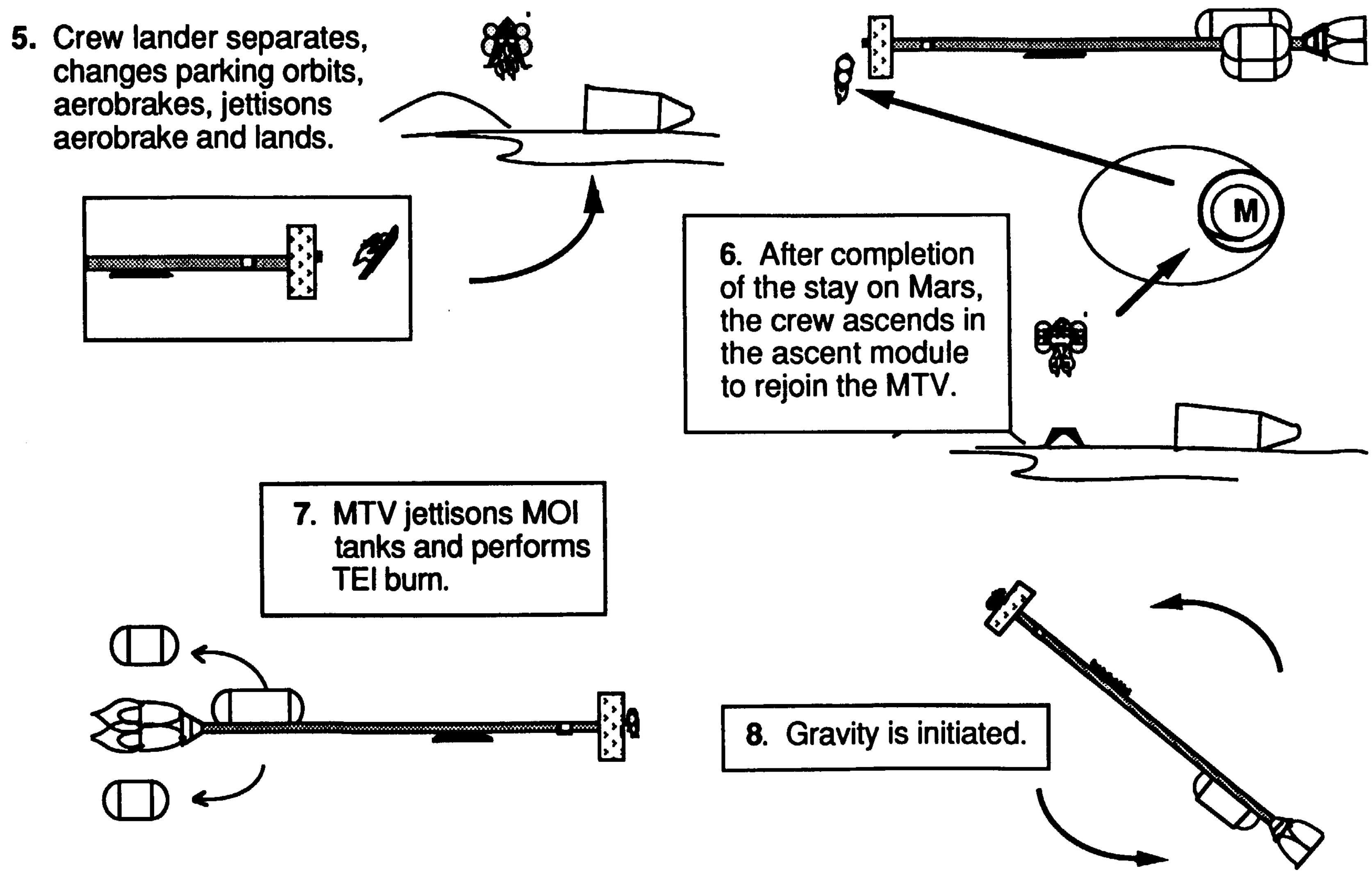
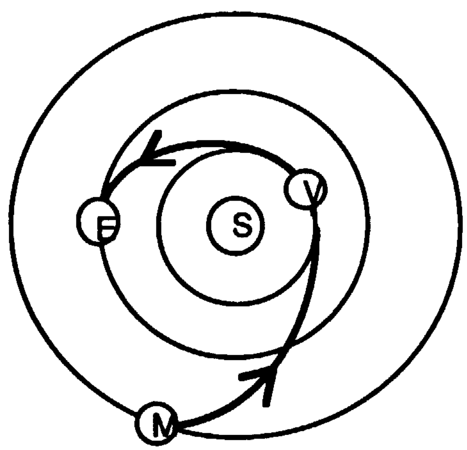
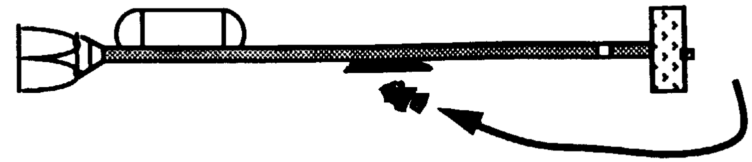



**Figure 3.21:** Mission Profile

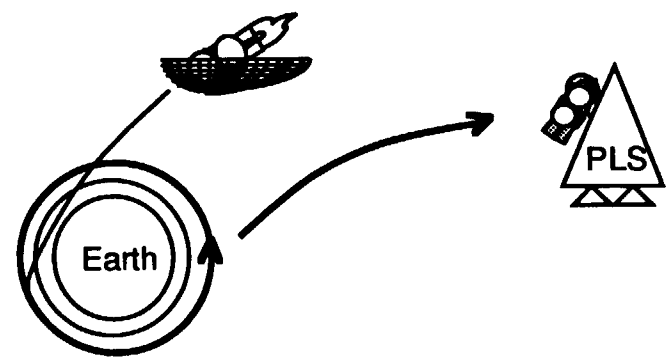

5. Crew lander separates, changes parking orbits, aerobrakes, jettisons aerobrake and lands.
6. After completion of the stay on Mars, the crew ascends in the ascent module to rejoin the MTV.
M
7. MTV jettisons MOI tanks and performs TEI burn.
8. Gravity is initiated.

9. On the way home, the MTV stops artificial g for a gravity boost from Venus. One week before arrival at Earth, gravity is finally stopped.



10. The crew enters the Mars ascent vehicle, which maneuvers to the aerobrake near the middle of the MTV.




11. With the help of the Earth's atmosphere, the ascent pod brakes into a Low Earth Orbit. There the crew will rendezvous with the Space Station or a Personnel Launch System vehicle. The MTV remains in orbit around the sun.

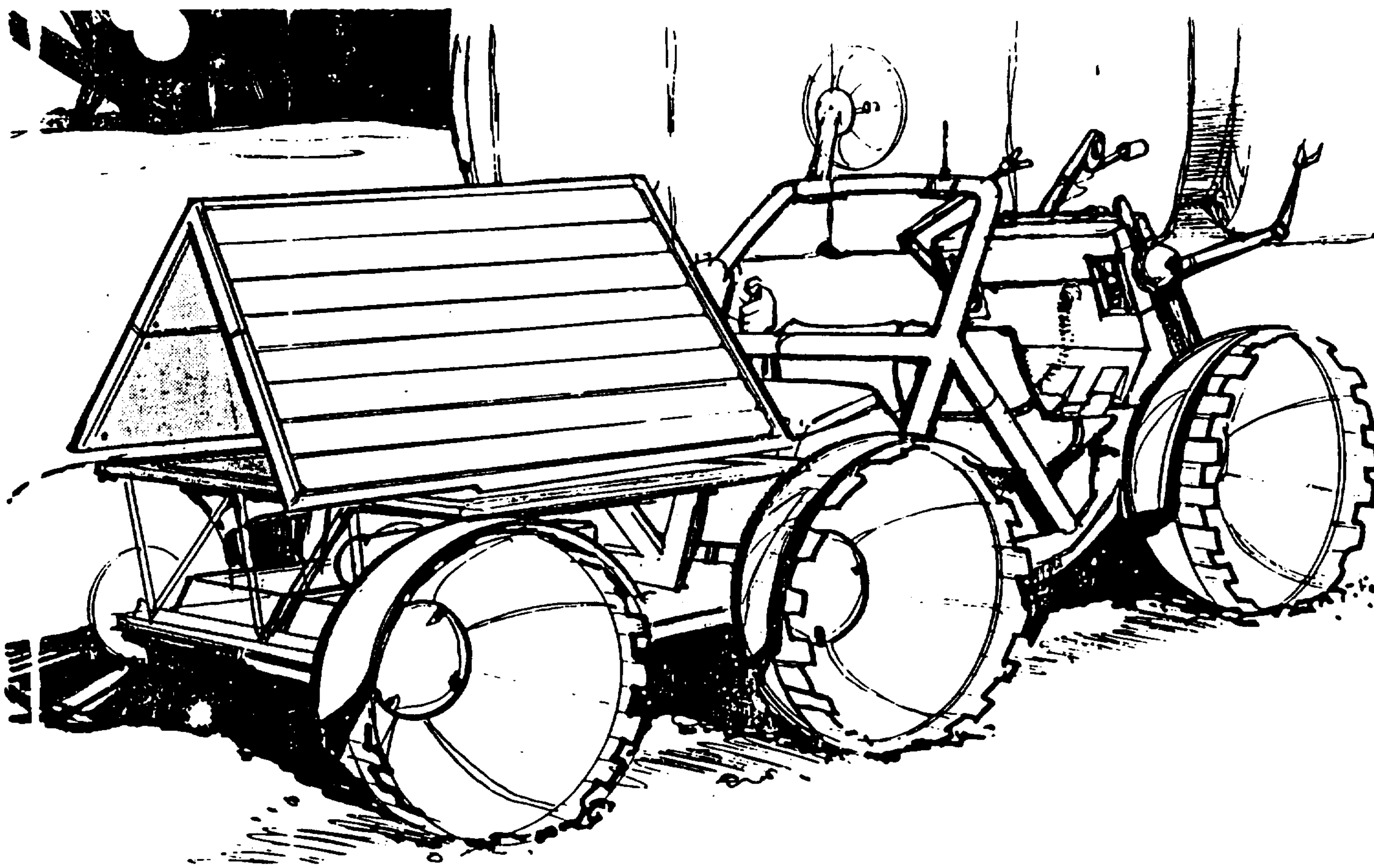


Figure 3.22: Unpressurized Mars rover

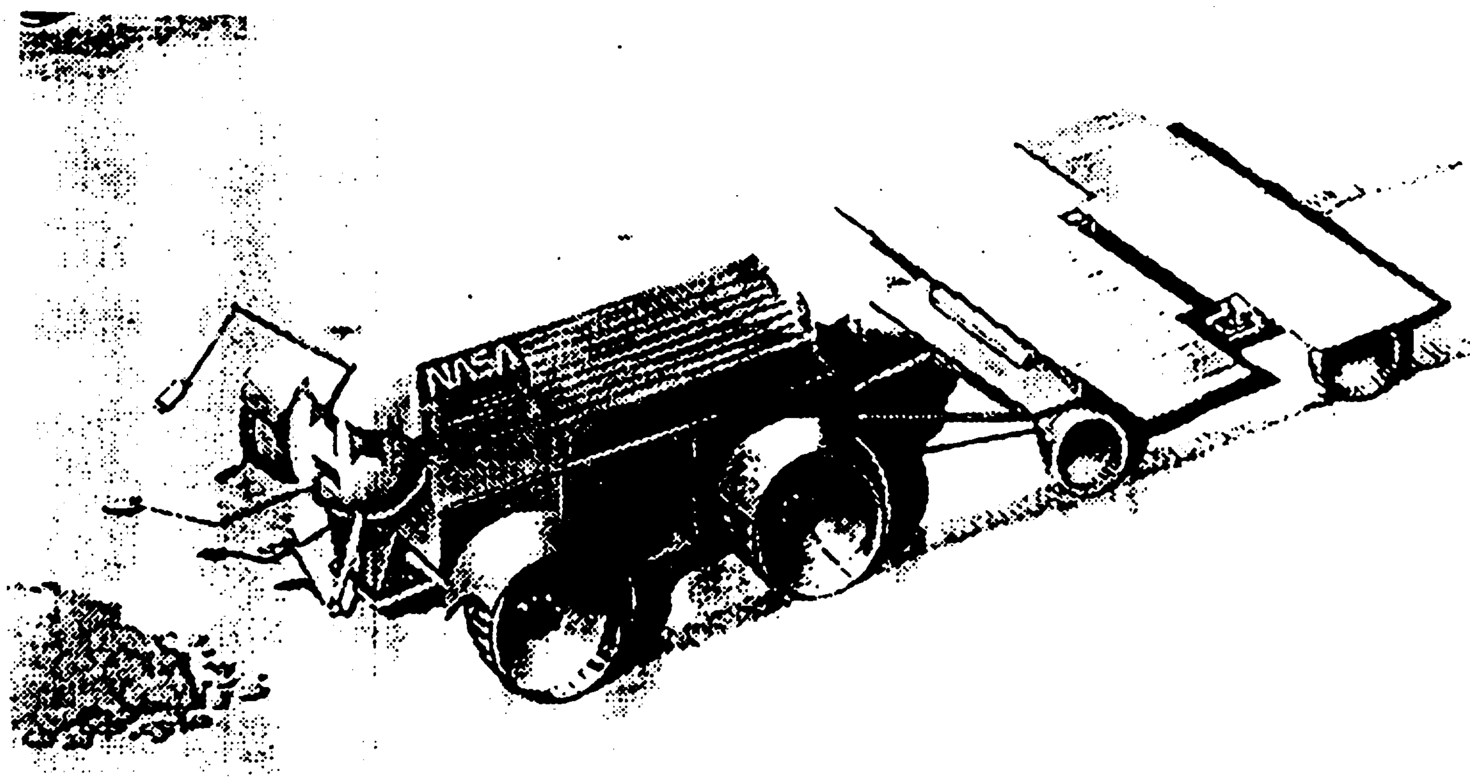



Figure 3.23: Pressurized rover and DIPS cart

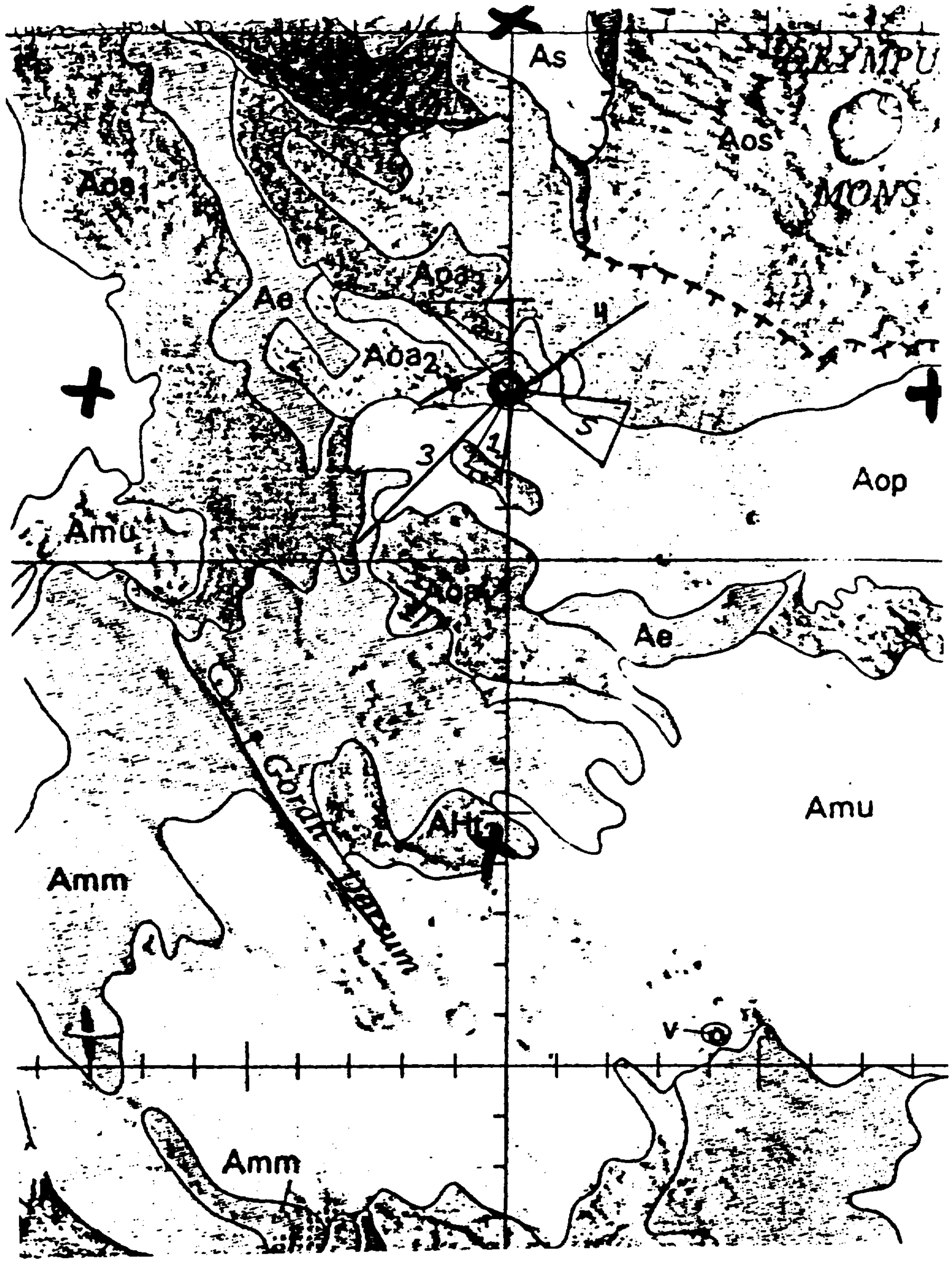



**Figure 3.24: Geologic map of Mission 1 landing site**


Figure 3.25: Geologic map of Mission 2 landing site

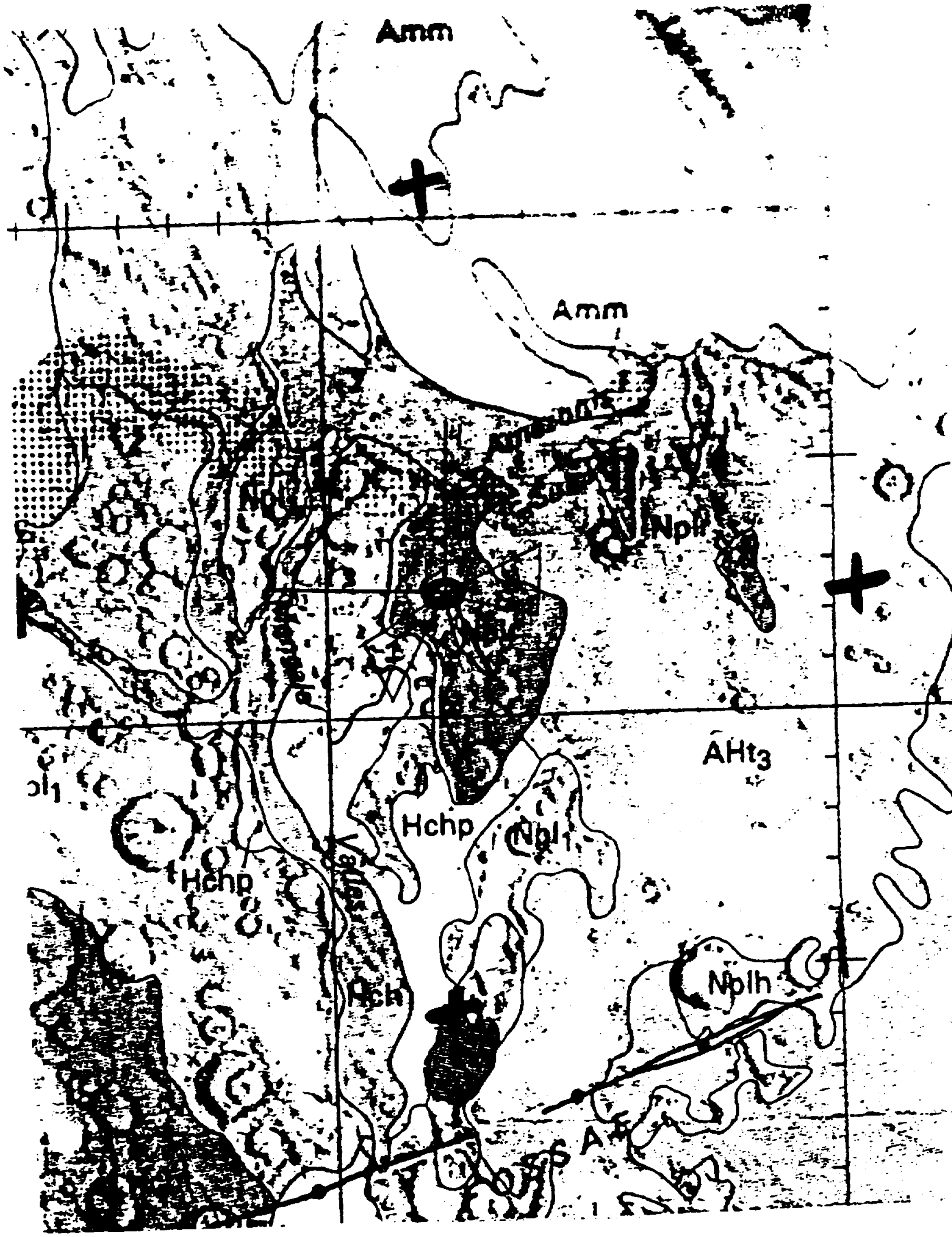



Figure 3.26: Geologic map of Mission 3 landing site

# 4.0 CREW SYSTEMS

## 4.1 Introduction

The crew systems discipline is responsible for anything directly related to the crew while taking part in these missions. For example, considerations include design of life support systems and layout of the crew modules to name a few.

## 4.2 Crew Selection and Size

As described in a paper by Tillman Ergonomics Company[4.5], there are two criteria for choosing the size of the crew for long-duration space missions; functional requirement analysis (FRA) and task requirement analysis (TRA). Functional requirement analysis evaluates crew size options for completing specified functions in a mission. Task requirement analysis evaluates crew size options for doing operating and maintenance tasks on a specific hardware design.

To perform an FRA for this mission several things must be discussed:

- •Describe the mission specific functions (science, piloting, etc. )
- •Analyze the functions -- How will they be performed? How will personnel and equipment be used in the loop?
- •Tradeoff crew size options
    - Safety
    - Mission Success
    - Launch Weight
    - Automation

To perform a TRA for this mission the following items must be defined:

- Define Activities -- activities required to maintain and operate the vehicle.
- Categorize the activities
- Determine the automation level
- Allocate activities
- Determine best crew size based on least person-minutes

The Crew Systems group formulated a questionnaire for the other disciplines to better help analyze these requirements.

### 4.2.1 Crew Sizing

After the initial decision for a crew size of seven, there were concerns if this was the optimum size. Ideally, from a crew systems point of view, there would be as many people going as possible. The psychological benefits increase as the crew number increases. Another point brought up was the importance of an odd number of crew members. Molly Elrod[4.21] of NASA Masrshall Space Flight Center suggested that an even number of crew members would be better. Part of her reasoning was that with an odd number it would be possible that four crew members could "*gang* " up on three crew members for example. Crew Systems had decided on an odd number of crew members from surveying past space missions such as the Apollo and Space Shuttle. Ms. Elrod noted that these space trips were much shorter than the planned Mars expedition and that any conflicts in space should be solved by the expert of that discipline. For example, if there was a pressure leak in one of the MHM compartments the engineer would decide the best course of action to resolve the problem.

Intrying to reduce the crew to its optimal size it was decided that each crew member have more than one skill. While specializing in one areal, each crew member has secondary skills which allow him or her to fill in for another crew member. The following is a break down of a six member crew and the skills they must posses:

1. Pilot
    a. Piloting various crafts
    b. Mission commander
    c. Systems trouble-shooter
    d. Science assistant
    e. Monitoring and maintenance of MTV orbit while on Mars
2. Doctor
    a. Help maintain and monitor crew health
    b. Medical and Psychological knowledge
    c. Crisis management specialist
    d. Main consultant with SG on Earth
    e. Crew trouble-shooting
3. Engineer
    a. Structural
    b. Electrical
    c. Computer operations regarding overall systems integration
    d. Maintenance
        1. Habitats
        2. Rovers
        3. Lander
        4. Aerobrakes
    e. Co-pilot
    f. Science assistant
4. Scientist
    a. Astrophysicist
    b. Crew scheduling
    c. Computer operations
    d. Communications
5. Scientist
    a. Meteorology
    b. Geology
    c. Computer operations
6. Scientist
    a. Navigator
    b. Biology
    c. Agriculture
        1. Genoponics

2. Hydroponics
d. Computer operations
e. Geology

**4.2.2 Orbital Mechanics Concerns**
Early in the design project a concern was raised by the orbital mechanics group that since every orbit made by the MTV around Mars will incur a 5% error, it might be necessary to leave a crew member in the MTV for course corrections. Assuming that course corrections will be able to be made from the surface of Mars, it is preferrable to bring all crew members to the Martian surface. Not only is each crew member a vital part of the mission operations, but more experiments can be performed by a lrger crew.

## 4.3 Human Performance Factors
The following topics that will be discussed are from the psychology discipline called Human Factors. Engineering traditionally ignores the human in the man-machine loop. The Crew Systems group spent a substantial amount of time researching the needs for the human aspects of this mission.

**4.3.1 Effects of Noise on Performance**
The internal acoustics of the MEV and MTV are of great importance due to the effects of noise on the crew. Normal noise limits for operating conditions are from 30 to 40 dB. Over long periods of time noise greater than 40 dB is an annoyance to most humans. Prolonged exposure can lead to difficulty and loss of sleep, increased stress, and an increase in anxious emotions (fear, surprise, anger, etc.). In addition, with noise above this level, people have to raise their voices in order to be heard from more than two meters away.

In addition to noise in the audible range (20 - 20,000 Hz) concern exists for noise in the ultrasonic and infrasonic regions. Although not heard, these noises can have adverse physiological effects. Non-audible noise above 40 dB starts to interfere with communication and also disrupts other tasks. The lack of understanding between crewmates increases and with it the possibility of human error[4.1].

**4.3.1.1 Acoustic Requirements**
The main requirements for acoustics as stated in NASA's Man-Systems Integration Standards are:

1. Noise cannot cause harm to crew
2. Noise cannot interfere with communication of any (non)verbal kind
3. Noise cannot induce fatigue
4. Noise cannot contribute to the overall deterioration of the equipment.

To meet these requirements specialized equipment will be installed on the MTS. The instrumentation will include acoustic measuring devices for monitoring sound levels. The maximum allowable sound pressure level (SPL) in a 24 hour period should not exceed 115 dB and short term excessive noise (noise greater than 85 dB will require hearing protection regardless of duration) not to exceed five minutes. No noise is to exceed 120 dB in any octave band or 135 dB ever. Impulse sounds are not to exceed 140 dB at anytime.( An impulse sound is that which changes 10 dB in a second or less) Passive hearing protection is not an acceptable answer for long periods of excessive noise, therefore special precautions must be made in designing the MTV and MEV to minimize their nominal SPL's. Finally, background noise is not to exceed the 50-NC contour or dip below the 25-NC contour (see Fig. 4.1) for extended periods[4.1].

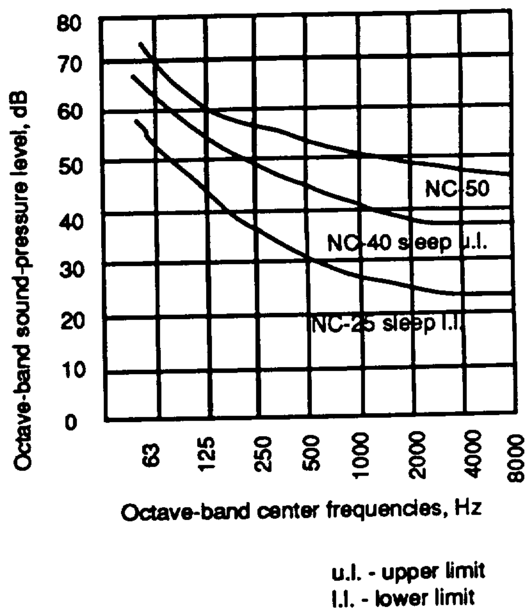



**Figure 4.1:** Indoor Noise Criteria (NC) curves

**4.3.2 Olfaction and Taste**
With a decrease in gravity there are accompanying physiological effects. Low gravity contributes to a slight redistribution to bodily fluids and sinus congestion is often associated with it. Although there is a lower level of olfactory sensation there is still an increase in the unpleasant odors present. Particulate matter in low gravity situations does not settle as fast -- if at all -- as earth's one-g case. Therefore, odor problems are likely to increase, making air filtering a must. If left unchecked, unpleasant odors can lead to nausea, sinus congestion, headaches, coughing, and overall general annoyance.

Low gravity's effect on taste is minimal and easily overcome. Low gravity leads to a decline in taste sensitivity. This is remedied by hightening the seasoning of foods. Considering the long duration and high stress level of this mission, this factor is not insignificant in contributing to overall crew morale[4.1].

### 4.3.3 Vestibular Effects

The two prevalent, adverse, vestibular results of low gravity are spatial disorientation and space adaptation syndrome (SAS). Spatial disorientation is important in the design of hardware and other controls. SAS is akin to motion sickness and can often be overcome with pharmaceuticals. Most adverse vestibular effects subside after two to four days and usually effect less than 50% of the crew. Hopefully, the crew will have adapted on the MTV trip and will have minimal side-effects in the MEV[4.1].

### 4.3.4 Kinesthesia

"Kinesthesia is the sense mediated by end organs located in muscles, tendons, and joints, and stimulated by body movements and tensions"[4.1]. Kinesthesia manifests itself in sensitivity degradation. For example, increments in mass must be greater, than in earth's 1-g situation, for discrimination. Also, the ability to estimate the mass of an object decreases.

### 4.3.5 Low Gravity Considerations

Additional physiological effects of reduced gravity loading are calcium loss (akin to osteoporsis), a decline in muscle mass, and cardiovascular deconditioning. Calcium loss leads to weakening of the bones. The rate of deterioration slows after four to five months, but the loss to this point is not insignificant. Currently, calcium, dietary supplements have only been marginally successful. Preventive measures such as certain exercises have been shown to decrease the muscular and cardiovascular decremental effects in 0-g environments. The adverse problems are lessened even more for any gravitational loading greater than 0-g. Primary tests to be conducted on the Space Station Freedom and on the lunar surface (1/6-g) will contribute useful data to help combat these problems[4.6].

### 4.3.6 Sleep Considerations

Sleep for crew members is an important consideration for design of crew work/recreation time. A suggested minimum sleep allotment for an individual, in a 24 hour period, is eight hours with at least six hours of uninterrupted rest. In addition, the sleep period should be preceded by one hour of nondemanding mental stress. On average, performance decreases with less than six hours of rest and negligible increases in performance occur after ten hours of sleep. On past space missions, sleep has not been hampered much by decreased gravity, some amount of insomnia can be counted on though. Sleep aids (drugs) should be available via a controlled access system to combat sleeplessness[4.1].

### 4.3.7 Nutrition

The nutritional needs (not so much specific nutritional content, i.e.; fats, carbohydrates, vitamins, etc. but overall considerations) for the crew is of paramount importance. Food will effect crew performance and morale. In this area it is important that a past history of crew members is available. Their preferences and habits will affect a pre-mission menu selection. The more earth-like the food, the more likely the crew will accept it on a regular basis. This includes desires for fresh fruit and vegetables. Frozen forms of the aforementioned are the most highly accepted. The importance of a well balanced diet is to add variety and provide each crew member with recommended daily allowances (RDA) of vitamins and minerals. A system of consistent meal scheduling is needed to avoid meal skipping. A variety of food will be a variable in crew morale, Snacks, with minimum preparation time, should be available to satiate in between meal cravings. On days where extravehicular activity (EVA) operations are to take place, high energy content food should be available. This includes snacks with high energy content. A system for monitoring the crews nutritional habits will be needed to make sure they are eating correctly. Excess particulate airborne matter, particularly odors, can affect appetite. This is important in air filtering considerations. The amount (weight) of food will depend on many factors; activity level (exercise, EVA operations, daily activity), crew size, individual preferences, etc.[4.1].

#### 4.3.7.1 Ingestible Water Needs

Daily needs per person for drinking, food rehydration, and water in food and drink is about 3 kg (6.6 lb). This does not include water for washing, hygiene, waste removal, etc. More water may be needed depending on activity levels (exercise and EVA operations in particular), cabin temperature and humidity, and as individual history would recommend. Also, additional water may be needed to counteract fluid redistribution due to low gravity effects. Fluid redistribution in the human system has not been observed well enough in long term reduced gravity to be able to suggest the amount needed[4.1].

### 4.3.8 Crew Exercise

Low gravity affects the body's muscular and cardiovascular/respiratory systems in adverse ways. The decreased gravity decreases the normal loads on bones and muscles associated with earth's gravity. Over long periods of time, this leads to degradation of muscular, skeletal, and respiratory systems. The degradation of the crew's bodies, if allowed to manifest, will greatly decrease the crew's ability to function. Prolonged exposure to low gravity without preventive measures will also decrease the body's immune system and strength levels will go way down, making even simple tasks difficult or impossible. "After the first three manned Skylab missions, which ranged in duration from 28 to 84 days it was observed that after a 28 day exposure to the microgravity of space, two of three crew members experienced a 23% loss in strength of the extensor muscles of the thigh and leg and about a 10% loss in flexor muscles"[4.10]. On Skylab flights, with exposure to a 0-g environment for 84-days, there was only a 10% loss of muscular strength[4.10]. The discrepancies in losses and days spent are due to the fact that on the longer fligh more exercise was utilized. In 0 -g the human system needs, on average, two to four hours of exercise daily. This can be a big problem as it takes away from an individual's work availability, it is viewed as inconvenient, crew morale generally goes down, and motivation becomes very difficult[4.1]. The two to four hour need for exercise is greatly reduced as the gravity increase towards earth's 1-g. On Mars (3/8-g) the minimum can be reduced to about one to two hours daily. Also, each individual's program should be personalized. Hemodynamic countermeasures have included MAST/anti-g force suits in the past but become disliked by most individuals as time increases[4.9]. The first couple of days in new g environments should include a minimum of exercise until SAS is overcome.

The basic needs for exercise are in the proper facilities. Exercise equipment should require minimum preparation and storage time. The room with the equipment should be closely monitored for temperature and humidity and may include facilities for personal hygiene. More than likely the later will be omitted as personal crew quarters will have hygiene maintenance facilities available. It is important for each crew member to realize the importance for exercise and in maintaining a regular exercise schedule. They can be made aware of the short and long term (even post mission) benefits. Individual histories will reveal what types of exercises the crew members prefer. The underlining fact is still: they must do it in order to successfully complete the mission. Different types of exercise include isometric, isotonic, tension, and aerobic exercises, the idea being to work as much of the body as possible to prevent "disuse atrophy." Different types of monitoring can be used, not only for mission's record but to record and chart personal progress. Short term monitoring can include heart rate, power output, blood pressure (BP), $VO_2$ max, etc. Long term monitoring can show progress or regression, goals, electrocardiograms (ECG), maximums, etc., and provide feedback on performance. Exercise doesn't have to be mundane work that an individual doesn't particularly look forward to, it can be fun. To add excitement to exercise there should be variety, extra sensory aids (music, video, social interaction, etc.), games, and other incentives (TBD)[4.1].

### 4.3.9 Psychological Factors For the Habitation Module Design

Psychological considerations for the design of habitation modules include[4.1]:

- •Compartment spaciousness
- •Perceived temperature
- •Psychological response to light
- •Stress reduction

Compartmental spaciousness is affected by room arrangement, room shape, and color. It has been shown that odd shaped rooms will appear larger than conventional rooms with the same internal volume. Brightness, saturation, and color choice will also affect human perception of the room.

Perceived temperature is the psychological perception of room temperature due colors, hues, brightness, and saturation. As an example, hot temperatures are red, orange, yellow, etc. and cool temperatures are blue, green, violet, etc..

Good lighting takes more into account than just illumination. Room purpose, task assignment, and emotional considerations all have to be dealt with when designing lighting for rooms.

Stress reduction can take on many forms. Pictures of natural landscapes and wall murals are a few examples of how stress can be reduced.

This is just a brief overview of the physical factors that can affect the psychological well being of the crew. These subjects will need to be studied more thoroughly in the future before the final design for the actual vehicle is made.

### 4.3.10 Windows

The need for windows can be seen by investigating reports from the South Pole Station[4.19]. It has been reported that residents of the South Pole Station stand in line to look out one of two windows available. Another situation involves astronauts and cosmonauts, they have been documented as spending numerous hours gazing out of the portholes on spacecraft. These two examples clearly indicate the need for windows on the habitation module.

Another reason to include windows in the habitats (primarily in the MHM) is to prevent the effects of sunlight deprivation. This effect has been observed in cave explorers[4.20]. After three days without sunlight, the explorers fell into an unusual pattern of 50-60 hours of work, eight hour meals, and 24 hours of rest. Clearly the omission of windows (or simulated windows or lighting) for the habitats would be detrimental to the mission.

Windows provide many functions[4.1]. It provides a psychological link with the outside environment, helps to alleviate the claustrophobic effects of confined and cramped quarters, provides a point of entry for light from a natural source which in turn will proved the crew with a sense of day and night cycles (again primarily in the MHM). They will also provide sensational, educational, and recreational views for the crew. Windows also have many practical functions such as monitoring and support of crew activity outside the habitation module as well as a communication link.

## 4.4 Psychosocial and Psychological Human Performance Factors

The whole point of a mission to Mars of this type is to make it a manned mission, to get people to the surface of Mars. Machines have done it, it is now time to make the next big step. For this reason it is of vital importance that considerations for the crew, other than tasks or functional requirements, are made. It is important how the crew as individuals and as small groups act and interact with each other and themselves.

### 4.4.1 Psychosocial Factors In Crew Considerations and Selection

In order for a successful mission as much about the mental aspects of the crew must be known. Important areas of interest are:

- Mental health of individual
- Past history
- Prone to psychiatric problems?
- Family history
- Mental tests/ evaluations
- Interpersonal considerations
    - Past history
    - Ability to conform to group
    - Ability to work/ relate to others
    - Cultural/gender differences
    - Personality conflicts
    - Need for privacy
    - Leadership,style and the ability to deal with imposed forementioned
- Ability to deal with isolation and separation
    - Family, friends, etc.
    - Home and work
    - Earth the planet
    - Familiar earth cues
    - Monotony
    - (Un)conscious conflicts

Tests and observations with similar conditions have been done on Earth. Expeditions in Antarctica, stays on submarines, and space capsule simulators have provided most of what is know about extended stays from civilization. These studies will be useful for future reference. The observations made have been consistent with what has been experienced with long-period stays in space stations. An interesting fact is that people born and raised in small town environments seem to deal with stressful small group situations better than people raised in big city environments[4.13]. One problem that will have to be considered is the "break-off" phenomenon experienced by pilots flying at high altitudes. The break-off phenomenon manifests itself with feelings of transcendence, religious conversions, and a sense of unreality. There is no way to predict when or if this will happen to any given individual. In the past this condition has resulted in serious problems in airplane flights and space shuttle missions. One can only speculate to what extent this might happen in the MTV trip and its effects as Earth slowly disappears from view.

These problems and similar others must be addressed and discussed with the crew and their families before the mission. The families must understand what the crew members might go through and how to deal with the reintegration into the family upon return from the long mission. An option (may not be feasible in early missions) would be in bringing spouses and/or families on the mission.

### 4.4.2 Psychological

In reviewing submarine and Antarctic expedition literature[4.14] there are three basic stages of psychological development in long duration crews:

1. Early part of trip
    - Stages of anxiousness and psychosis as the crew adjusts to trip
2. Middle of trip
    - Depression and monotony
    - As crew settles in for long trip
3. End of mission/trip
    - Aggressiveness and immature behavior
    - Crew's anticipation for end of trip

These stages can be broken up into further subcategories such as Earth-Mars trip, duration on Mars surface, and Mars-Earth trip. Similar obser-vations have been made on long duration Soviet space flights[4.13].

#### 4.4.2.1 Interpersonal

Interpersonal factors effect how people and groups will work and relate with each other. Over time tension increases with heterogeneneous crew members. "Cliques" or subgroups tend to form. Also, scapegoating becomes more prevalent with crew members who don't conform and to those who have problems with leadership. A need for privacy becomes more important for each individual and territorial behavior is evident in many cases. This is an important factor in architecture design for work and living quarters.

#### 4.4.2.2 Soviet Experience

The Soviets have compiled results from observations made of their cosmonauts' long duration stays in space. Soviet's experience is that the worst problem in space is that of asthenization[4.13]. Symptoms are:

- Irritability
- Emotional lability
- Hypodynamia
- Psychosomatic symptoms
- Depression

Some type of monitoring will be needed to detect early warning signs of this disorder (discussed in more detail in Sections 4.3.2.3-4). The Soviets have found that a Earth based support group (SG) has been very successful. The SG would consist of a psychologist, a psychiatrist, behavioral scientists, family, friends, celebrities, etc. In addition to the SG on Earth, additional countermeasures on the MTV and MEV should be available for psychological comfort. These may include:

- Music and videos
- Sounds of Earth (birds, rain, traffic, etc.)
- Exercise
- Video link with Earth for communication with SG
- Work and variety of schedules
- Crew physician

Also, in the past it has been suggested that the crew members develop potential on their free time. Past crew members have composed music and written stories for example. This will be a consideration in deciding what supplies to bring. In general the Soviets have been successful in maintaining the crew's sanity utilizing a SG and the methods described in Sections 4.3.2.3-4.

#### 4.3.2.3 Pre-Flight

Before the mission, in addition to task and functional evaluations, the prospective crew members should be given psychiatric evaluations and test such as the Fundamental Interpersonal Relations Orientation - Behavior (FIRO-B). The FIRO-B and others have shown promising results in evaluating people and testing compatibility. A minimum criteria for acceptability can be determined. After a crew has been decided upon they should spend as much time together as possible to realize conflicts and problems before the mission.

#### 4.3.2.4 In-Flight

The ideal situation would be to have a SG along on the mission but that is not at all feasible for cost and weight reasons. The crew will have one physician. The physician's clinical experience with a variety of cases will be of benefit to the mission. He/she can prescribe tranquilizers and medications, carry out research on the trip's long duration effects on the crew, and can take action upon warning signs or trouble situations (it should be noted that the physician is part of the crew and will have some bias and should therefore be in frequent contact with the SG). In addition to the physician's experience, the crew should be trained in crisis intervention and brief psychotherapy. The commander holds the right to at any time call an impromptu discussions in order to air grievances. Also, a private communication link for concerned/distressed crew members to the SG should be available.

### 4.4.3 Conclusion to Psychosocial and Psychological Considerations

The aforementioned problems have effect on many aspects of the mission that include crew mental health, architecture, communications, supplies nonessential to mission, crew selection, crew's performance abilities, and Earth based support. The importance of the crew's health is vital if the mission is to succeed.

## 4.5 Multi-g Effects Introduction

The crew systems team has analyzed the effects the harsh environment of space has on human beings. The group took an extensive look at the effects of mulit-g loadings on the crew and human system.

### 4.5.1 Multi-g Loading Effects On the Crew

Although high-G effects are important to the crew at all times, during the mission, it is especially important in the vehicle that the crew uses to ascend and descend to the surface of Mars and at Earth reentry. There are adverse effects to the human system caused by the different accelerations on the vehicle. There are two main considerations due to the loading.

The first consideration is that increased g-loading limits the mobility of the crew (see Table 4.1 for healthy, unconditioned individuals). This will effect to what extent the crew can manually control any given craft.

| Acceleration | Possible Reach Motion |
|---|---|
| 0 - 4 g | Arm |
| Up to 5 g (9 g if arm is counter balanced) | Forearm |
| Up to 8 g | Hand |
| Up to 10 g | Finger |

**Table 4.1:** Reach movements possible in a multi-g environment[4.2].

In addition to decreased mobility there are health effects on the human system[4.2] (see Fig.4.2 for sign convention). The following outline is for healthy, unconditioned individuals, it can be used to gain a starting point for which design considerations can be referenced.

A. Upward Acceleration Effects ($+G_Z$) (In Seated Posture)

| | |
|---|---|
| 1 $G_Z$ | Earth equivalent |
| 2 $G_Z$ | Increased weight; increased pressure in buttocks; drooping of body tissue |
| 2.5 $G_Z$ | Difficult to raise oneself. |
| 3-4 $G_Z$ | Impossible to raise oneself; difficulty inmobility (Table 4.4.1); progressive dimming of vision after 3-4seconds. |
| 4.5-6 $G_Z$ | Progressive **blackout** after about 5 seconds; mild to severe convulsions in 50% of the subjects during or following **unconsciousness**; **gustatory sensations**; loss of space and time perception for up to 15 seconds upon post acceleration. |

B. Downward Acceleration ($-G_Z$)

| | |
|---|---|
| -1 $G_Z$ | Unpleasant; but tolerable |
| -2 to -3 $G_Z$ | **Headache**; blurred vision **graying**; or reddening of vision after 5 seconds; may leave **petechial hemorrhages** |
| -5 $G_Z$ | Five seconds is maximum achieved by subjects |

C. Forward Acceleration ($+G_x$)

| | |
|---|---|
| 2 - 3 $G_x$ | Increased weight and abdominal pressure; slight spatial distortion; 2 $G_x$ tolerable up to 24 hours; 4 $G_x$ tolerable up to 60 minutes |
| 3 - 6 $G_x$ | Chest pain; loss of peripheral vision; difficulty in breathing and talking |
| 6 - 9 $G_x$ | Increased chest pain; shallow respiration; further reduction of vision; occasional **lacrimation**; body, legs, arms cannot be lifted above 8 $G_x$ and the head above 9 $G_x$ |
| 9 - 12 $G_x$ | Severe breathing difficulty; lacrimation; fatigue |
| 15 $G_x$ | Extreme difficulty in breathing and speaking; severe chest pain; loss of sense of **touch**; complete loss of **vision** |

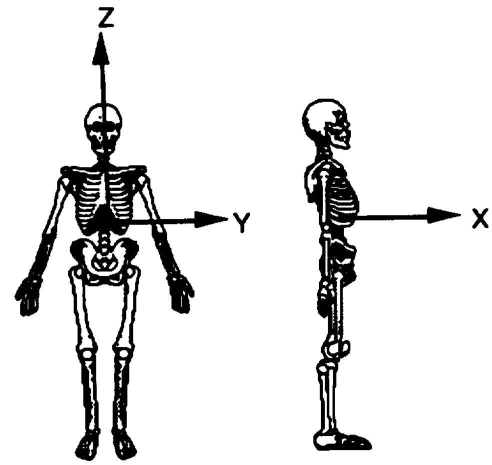



**Figure 4.2:** Body Reference Coordinates

D. Backward Acceleration ($-G_X$)

Akin to $+G_X$ effects except that the vector force is in the opposite direction. This results in easier breathing and forward head tilt. With increased g's the hemodynamic effects to the cerebrum are similar to those in the $+G_X$ situation.

E. Sideways Acceleration (+ / - $G_y$)

| | |
|---|---|
| + / - 3 $G_y$ | Discomfort after 10 seconds; pressure on body components on side corresponding to the acceleration |
| + / - 5 $G_y$ | 14.5 seconds leads to **external hemorrhage**; severe **headache** |

Acceleration limits that should be observed are shown in Figure 4.3. The graph is plotted on a logarithmic scale to show limits at very small time intervals. For example, the acceleration with a duration time of 0.01 minutes (0.6 sec) or less is called an impact acceleration or shock.

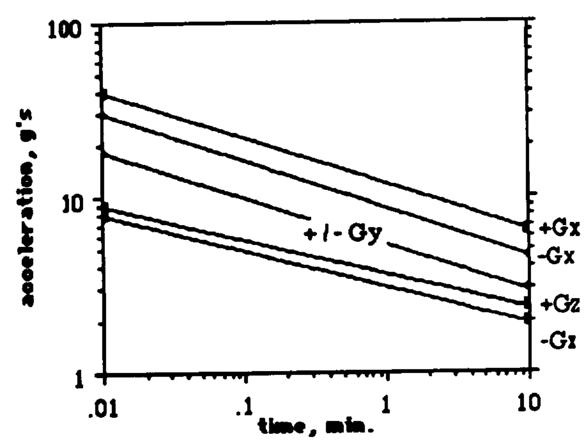



**Figure 4.3:** Linear acceleration limits for unconditioned and suitably restrained crew members[4.2]

### 4.5.2 Rotational Effects On The Crew

In addition to straight movement (Section 4.4.1), rotational motion is also a consideration in the crew's well-being. The overall effect on the human system is due to three factors; 1) rotational rate (usually expressed in revolutions per minute or rpm), 2) the placement of the axis of rotation, 3) center of rotation with respect to the body.

Most humans, without experience, cannot tolerate rotation rates in the area of 12-30 rpm. Rotation rates up 60 rpm for 3 or 4 minutes about the y-axis and z-axis have been described by some individuals as pleasurable. Rotation rates in excess of 80 rpm about the y-axis and 90-100 rpm around the z-axis are not tolerable.

In general, shooting for a design parameter of no more than approximately 6 rpm is the best choice. It should be noted that random "tumbles" are less tolerable than rotations in a consistent direction. Adverse effects on the human system include; spatial disorientation, headaches, nausea, and mobility degradation. These all lead to an overall performance degradation.

### 4.5.3 Effects of Long Duration Exposure to Various Multi-g Environments

The above discussions only consider multi-g effects on healthy, unconditioned individuals. An analysis involving prolonged exposure to multi-g environments must be mentioned. The TMI trip will be under 1-g and its effects on the crew are outlined above. After spending 60 days on Mars with 3/8 g the crew will be in a semi-weakened state and when the crew is ready to reenter Earth they will also be in a much greater weakened state. The weakened state of the crew will lower their tolerance level to g-loading. The amount to which this will be a problem is a problem in itself. Meaning, that because there is a lack of available examples, people on extended space missions, much known about the possible outcomes is pure speculation. The planned mission scenario would involve 60 days in 3/8 g on Mars and additional 250 days at 0.5 g on the TEI MTV trip. The following data is a projected tolerable g-loading the crew will be able to handle:

1) Trip to Earth Orbit From Earth's Surface
   - Full g-loading as outlined in Section 4.5.1 - 4.5.2
2) MTV TMI
   - Full g-loading as outlined in Section 4.5.1 -4.5.2
3) Lander Trip To Martian Surface
   - Full g-loading as outlined in Section 4.5.1 - 4.5.2
4) Lander Trip From Martian Surface to MTV
   - Muscular strength loss of 5-10%
   - Other Physiological Losses 5-10%
   - Resulting g tolerance levels shown in Figure 4.4(a)
5) MTV TEI
   - Same as #4
6) MTV Upon Reaching Earth Orbit and Lander Trip From MTV to Earth's Surface
   - Cumulative Physiological losses On the order of 40%
   - Resulting g tolerance levels shown in Figure 4.4(b)

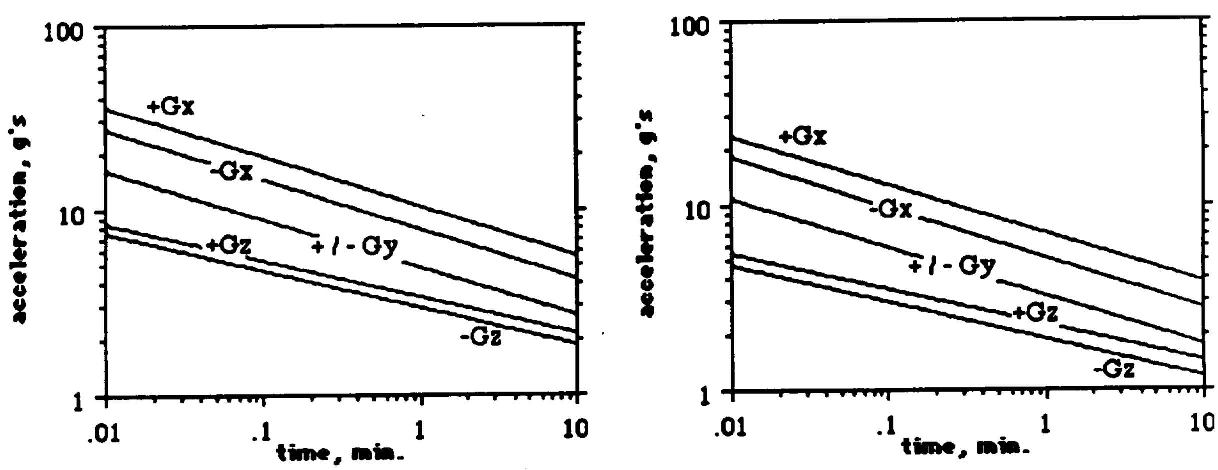



**Figure 4.4(a),(b):** Linear acceleration limits for crew personnel after (3) 60 day stay on Mars, and (4) after 60 day stay on Mars and 250 day TEI.

#### 4.5.4 0-G Effects In Worst Case Scenarios

The increased g-loadings on the crew upon different delta V's can very well have severe adverse effects on the crew. Two additional points were brought up in further discussion that are important in design considerations. One, the g-loadings in Figure 4.4(b) were patterned after a 250 day TEI trip at 0.5-g. It has been suggested that numbers for a maximum of a 300 day TEI at 0-g should be looked at (also a case of 200 days at 0-g plus 60 days at 3/8-g plus 300 days at 0-g). This analysis examines what might possibly happen if for some reason the artificial gravity system fails at different stages of the mission. Two, true the astronauts will be in a weakened state after a 300 day TEI at 0-g and will not be able to tolerate very high g-loadings, but what is the maximum amount crew members can sustain and survive. This is important because the difficulty in decreasing the Earth lander's g-loading upon transition from the MTV to the Earth's surface increases substantially as the allowable g-loading is decreased.

##### 4.5.4.1 0-g Environment For 300 Day TEI

The maximum g-loading the crew will be able to experience,upon reaching Earth, will be in the +x direction (forward acceleration). After the total mission time, including

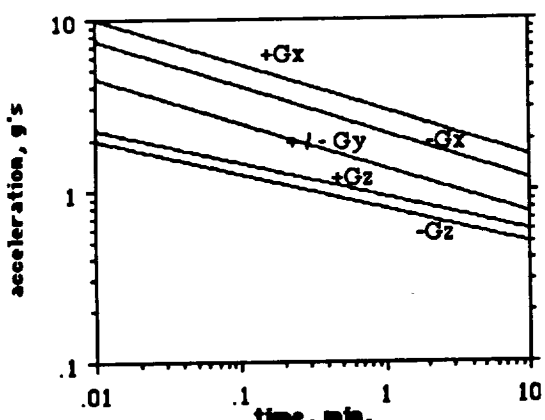



Figure 4.5: Maximum tolerable g-loading for the crew at EOI (after 200 days TMI at 1-g, 60 days Mars Surface at 3/8-g, 300 days TEI at 0-g).

TMI, time on Mars, and TEI, the maximum acceleration in the +x direction the crew will be able to survive is outlined in Figure 4.5.

Exceeding the limits shown in Figure 4.5 might result in severe physiological damage that could jeopardize the lives of the crew. Notice that the crew will be in very poor condition upon reaching Earth. Individuals will have problems even just living in Earth's 1-g environment and will require much rehabilitation and medical attention upon reaching the surface. The aerodynamics group suggested a g-loading on the order of 5g's for approximately 10-20 seconds during reentry. By orientating the crew in the correct position they should be able to handle the aforementioned loadings with some difficulty. Avionics may want to devise a system where the lander can be controlled by some external means as the crew may be incapable of moving in increased g conditions. Also, the crew will more than likely experience lacrimation, severe breathing difficulty, and possibly complete loss of touch and vision. This will effect the crew for more than the 10-20 seconds of increased g-loading during reentry as they will have to be constantly acclimating to the difficult stresses put on their bodies.

##### 4.5.4.2 0-g Environment For Entire TMI and TEI

If for some reason the artificial gravity system should fail immediately, the crew will have to tolerate a micro-gravity environment during the 200 day TMI and the 250-300 day TEI. Since the mission cannot be aborted after initial burn towards Mars the crew will have to be prepared for the worst. Upon reaching Mars the crew should have little problems with g-loadings and will be able to complete scientific studies on Martian surface. The down side is that they will probably not make it back to Earth safely if at all. These extended periods of exposure to micro-gravity has irreversible effects on the human system that will prevent the crew from surviving. The crew can maintain some relative health on the TMI by increasing daily exercise sessions to 2-5 hours. On the TEI detrimental human system failures will compound. Bone decalcification, fluid redistribution, and cellular breakdown will surmount to a point where individuals will cease to function properly and will finally succumb to death. This risk must be explained to the crew before the mission.

#### 4.5.5 Multi-g Summary

If all goes as planned the crew should be able to survive all predicted g-loading situations and also survive the worst case situation described in Section 4.5.2.4.1.

### 4.6 Space Suits

Space suits are going to be needed for different parts of the mission and for different reasons. Two designs are depicted in Figures 4.6 and 4.7 and are explained briefly below.

#### 4.6.1 Liquid Cooling Garment (LCG)

The LCG is worn during all EVA activities and can be worn in emergency situations. This unit keeps the human system at a correct temperature by absorbing heat and transferring unneeded heat to the sublimator in the personal life support system (PLSS) or to a vehicles main ECLSS system. The garment consists of a stocking type undergarment with liquid filled tubes running along the exterior.

### 4.6.2 Pressurized Garment Assembly (PGA)

Over the LCG another suit can be worn to work in conjunction with the LCG. This suit is the PGA. The PGA provides a portable life-support chamber in which the astronauts were during EVA maneuvers or in case of a vehicle breach. A diagram of a PGA can be seen in Figure 4.6. This suit is designed to be worn for 115 hours in a worst case scenario. Urine can be disposed of via a urine collector transfer (UCT) connector without ever having to remove the suit. As visible in the picture there are also connectors for additional life-support on the chest.

The helmet is attached separately. It does not turn with a persons head but remains stationary. The helmet is a clear Lexan bubble-type visor with heads-up-display (HUD). There are connectors for communications, water, and food (for extended EVA or extended stays in the lander) attached to the suit and fit inside the helmet near the astronaut's mouth.

### 4.6.3 Integrated Thermal Micrometeoroid Garment (ITMG)

Together with the LCG and the PGA the ITMG forms the total extravehicular mobility unit(EMU) for EVA operations. The PLSS in worn on the back like a backpack (see Figure 4.7). The PLSS can provide life-support for up to four hours at one time. Crew systems is assuming that this time will be improved upon by the start of the mission and that a PLSS system which can provide a minimum of ten hours of life-support will be available.

## 4.6 MEV Lander

The lander will be the vehicle that takes the crew to the surface of Mars from the MTV. It will also be used as an ascent vehicle for the crew back to the MTV in orbit around Mars. The lander will be positioned behind the aerobrake for the descent stage and will ascend by itself on the return trip. Estimates of volumes and masses for crew systems only are given as follows:

| | |
|---|---|
| • Habitable Volume | 13.58 $m^3$ |
| • Stowage | |
| • EVA Suits | 1.40 $m^3$ |
| • 72 Hour Contingency Provisions | |
| • Food | 0.20 $m^3$ |
| • Water | 0.20 $m^3$ |
| • ECLSS | 5.39 $m^3$ |
| Total Volume | 20.77 $m^3$ |

The habitable volume is the space for the six person crew not occupied by structures (example: this volume does not include the seats the the crew will occupy during ascent/descent but the space immediately above the seats).

The food and water needed for contingency will have a mass of 6.0 kg. The only life support system the crew will have in the lander will be a minimal ECLSS (1100 kg total) system that is accessed by the crew through umbilicals in their PGA suits. The mass for the crew and their suits will be approximately 1010 kg. The total approximate mass for the crew systems is 2116 kg.

### 4.7.1 Lander Design Considerations

There are a few design considerations from the crew systems point of view that must be taken into account before just assigning space for different applications:

One, there is a need for two windows in the lander structure. These serve to aid guidance in the landing on the Martian surface and the redocking of the lander to the MTV.

Two, there is to be an escape hatch 0.81 m square accessible to the crew. This hatch will provide the crew means for leaving and entering the lander at six different times:

1) Entering lander from MTV
2) Leaving lander on Mars surface
3) Entering lander on Mars surface for ascent phase
4) Departing lander to gain access to MTV
5) Enter lander to depart MTV and return to earth, 6) Departing lander upon completion of total mission.

Three, there must be enough free space for the crew to accomplish certain tasks. This includes gaining access to there EVA suits and being able to don and doff these garments. Also, there must be enough room for the astronauts, fully suited, to gain access to and use the hatch to the outside Martian environment with out interfering with vital, internal lander equipment.

Four, the six chairs that the crew occupies should be situated as to position the +x-axis of the human body in approximately the direction of the lander flight as much as possible.

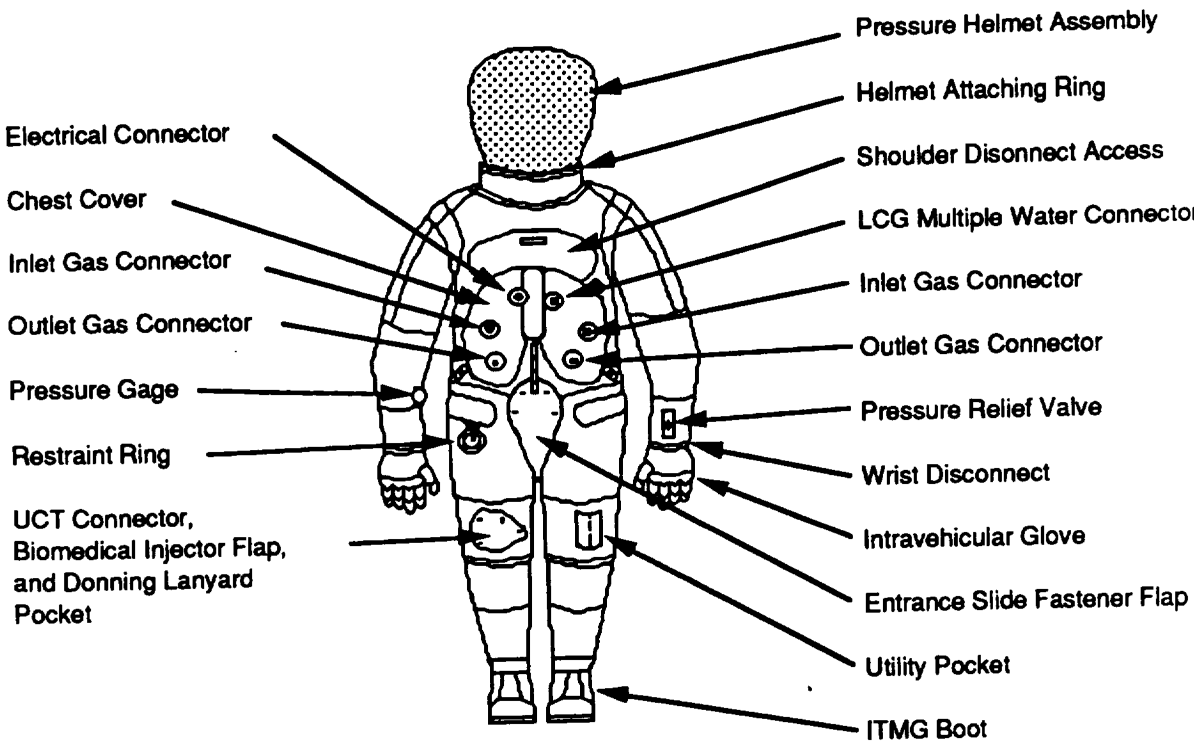



**Figure 4.6:** Pressure Garment Assembly

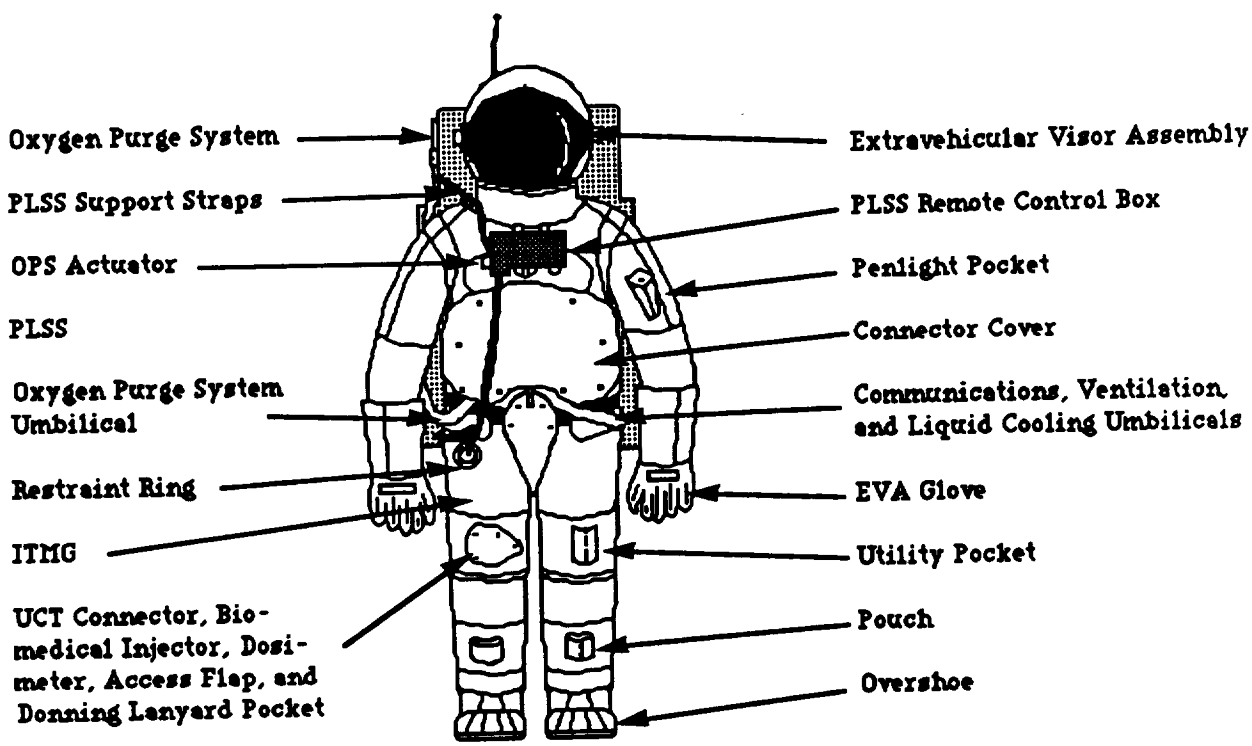



**Figure 4.7:** Integral Thermal Micrometeoroid Garment

## 4.8 MTV Habitation Module

The MTV habitation module will house the astronauts both to and from Mars. They will spend the majority of their trip here. It is estimated that they will spend approximately 450-470 days in the habitation module. Careful consideration has gone into its design to insure the safety of the crew as well as to make them as comfortable as possible.

### 4.8.1 Storm Shelter (MTV)

The storm shelter on the MTV will be located roughly in the center of the module and will contain both floors. It's general dimensions will be 2m x 5m x 5m. It's walls will be composed of 7cm of solid aluminum. This will give it an overall mass of 16.64 mT. The Thermal Group has determined that this level of shielding will be adequate.

The Structures Group has requested that the storm shelter be used as structural support to help support the loads of the ascent/descent module which will be hanging off the end. Since the storm shelter is a square, a truss system will have to be constructed between it and the pressure vessel. A tether system cannot be used since it must be able to withstand loads both in tension as well as compression.

The storm shelter will contain two storage rooms and the airlock. All drinking water will be stored in here to prevent its exposure to radiation. It has also been suggested that the main CPU be stored in here.

### 4.8.2 Pressure Vessel (MTV)

The pressure vessel for the MTV will serve several purposes. It must adequately protect the crew, and provide them with a pressurized environment in which to live. The pressure vessel will not, in any way, protect the crew from any type of radiation. This would induce a great mass penalty and is not necessary.

The pressure vessel will be constructed of 0.5 cm of aluminum, as recommended by the Structures Group. To do the mass analysis, it is assumed that it is a cylinder 16m long, 7.07m dia., with flat end caps. Using these assumptions, we obtain a mass of 5.86mT. It is understood that flat end caps would not be possible due to the large stresses induced at the corners. This, however, will not greatly affect its overall mass.

### 4.8.3 Floors

The floors inside the MTV habitation module will be similar to those used in Skylab. They will be an Aluminum "waffle grid" with beam supports. They will have a density of approximately 13.3 kg/m$^2$. Since we have 160 square meters of floor space, this gives us a total floor mass of 2.13 mT.

### 4.8.4 Partitions

The partitions (walls) inside the habitat will be of a laminated type construction. They will be composed of KEVLAR and NOMEX, and will be 10cm thick. Their exact construc.ion is detailed in Fig. 4.8.

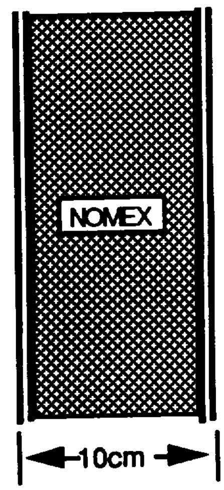



**Figure 4.8:** Schematic of Partition

The actual number of pieces needed along with their exact dimensions have also been computed and are given in the following table:

Pieces:
4 x 6.465m
4 x 4.800m
4 x 7.465m
8 x 2.000m
2 x 1.000m
1 x 0.700m
2 x 0.800m
2 x 0.900m
1 x 0.230m
2 x 1.200m
1 x 3.100m
2 x 1.100m
2 x 2.615m
---------
Total = 110.18m

If a ceiling height of 2.3m is used, the total wall area is 253.4 square meters. This will give a total wall mass of 0.9mT.

## 4.9 Layout of the MTV Hab

The layout of the MTV habitation module is shown in Fig. 4.9, and Fig. 4.10 As can be seen, it is a two level configuration inside a single pressure vessel. The actual living quarters contain 160 square meters of floor space with an internal volume of over 375 cubic meters.

The present configuration has a mass of just over 72mT. Since mass is a very important factor in the overall design, a continuing effort is being made to further reduce this mass.

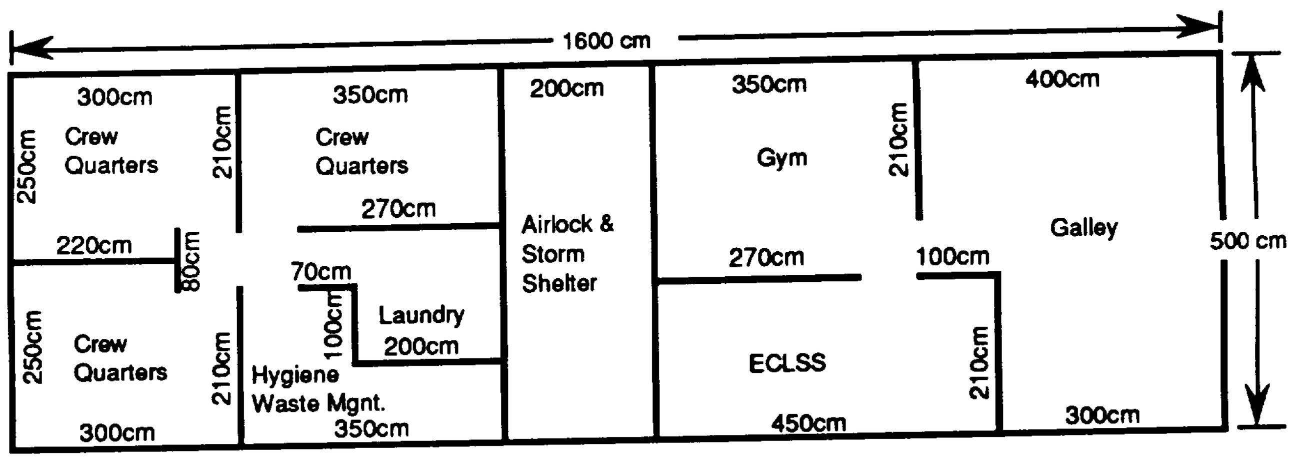
### 4.9.1 Crew Quarters
The crew quarters are grouped together just left of the storm shelter in the lower level. Each room will be occupied by two of the six crew members. An attempt has been made to keep this area of the habitat as quiet as possible by isolating it from the major activity areas. This is done to provide sleeping comfort for the crew members.

### 4.9.2 Hygiene and Waste Management Facility
The hygiene and waste management facility (bathroom) is located near the crew quarters for convenience. It will contain a toilet, shower, and wash basin. It is adjacent to the laundry to minimize the amount of plumbing required.

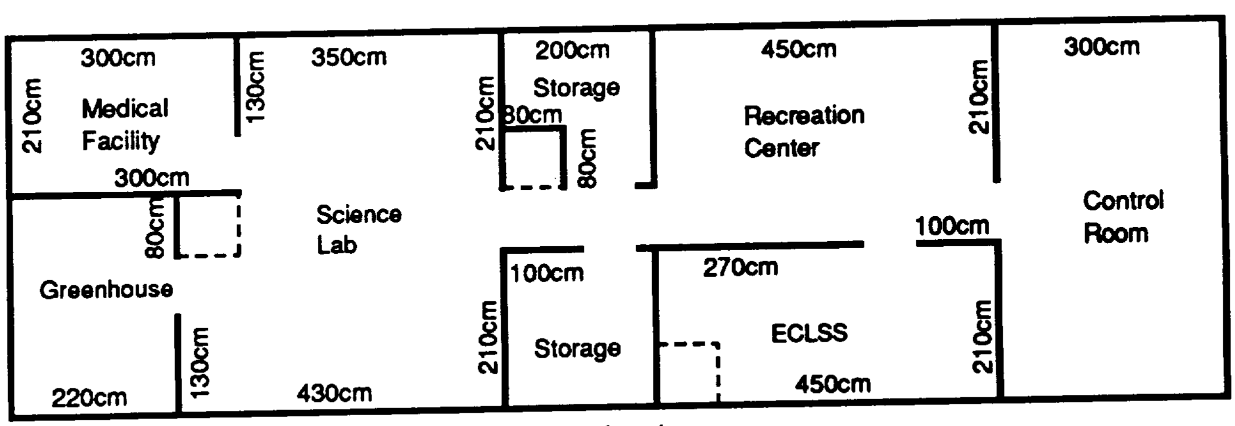
### 4.9.3 Laundry
The laundry will contain two washers and two dryers stacked on top of each other. Note that the laundry is not a separate room, but is rather simply cut into the hall which greatly reduces the amount of space needed and helps in the reduction of mass.

### 4.9.4 Gym
To help maintain the physical well being of the crew, a gym has been provided on the lower level. This will be especially important during the portions of the trip when the crew is subjected to a zero-g environment. Exercise equipment will include devices to improve muscular as well as aerobic development.

### 4.9.5 Galley
The galley is where the crew will eat all their meals. It may also be used as a general meeting room. Any equipment requiring the use of water (sink, dishwasher, etc...) will be located in the the lower left corner to reduce the amount of plumbing required.




Lower Level




Upper Level

**Figure 4.9:** MTV Habitat (Top View)

#### 4.9.6 Airlock and Storm Shelter

The airlock and storm shelter have been incorporated into a single structure. The crew will have to be in this area during solar flares. There is some concern having an airlock this large. It is very difficult to totally evacuate this area of all air when leaving or entering the vehicle. This could make the loss of atmosphere considerable.

#### 4.9.7 Medical Facility

In the event that a crew member becomes ill, he/she will be attended to in the medical facility. A window will be placed between the medical facility and the greenhouse. This serves to improve the psychology of the ill member and increase the healing process.

#### 4.9.8 Greenhouse

The greenhouse will be used to aid in the experimentation of plant growth in space. It is possible that this could provide the crew with fresh food, if it is determined to be safe. It should be pointed out, however, that the crew will, in no way, depend on this as a primary source of nourishment. Its primary purpose is purely scientific.

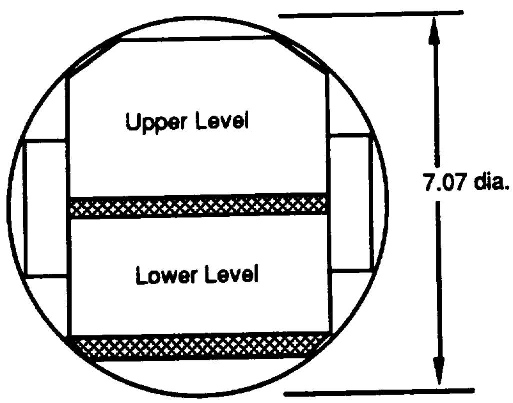



**Figure 4.10:** MTV Habitat end view

#### 4.9.9 Science Lab

During the duration of the trip the crew will perform many scientific experiments in the science lab. To reduce plumbing, as before, all equipment needing water will be located such that it is directly above the hygiene and waste management facility. The Mission Operations Group will determine the exact experiments to be performed and the equipment necessary.

#### 4.9.10 Recreation Center

The psychology of the crew is as important as there physical well being. It is important that the crew have a chance to relax. The recreation center will contain audio/visual materials, books, etc. A large couch will be placed along the far wall.

#### 4.9.11 Control Room

The control room will function as the main command center for the ship. All of the navigational and control systems will be located here. Control of all the ships subsystems will also be done from here. The ship's main computer, however, will not be here. It will be contained in the storm shelter to protect it from the damaging effects of radiation.

#### 4.9.12 ECLSS Rooms

Two rooms are provided to contain the ECLSS equipment. They are centrally located and stacked one on top of the other to once again reduce plumbing. These rooms will contain the ECLSS systems described in Section 4.10 as well as the thermal control equipment.

#### 4.9.13 Storage

Storage will be provided in several ways. First, there are the major storage rooms on the upper level. Second, a storage compartment will totally surround the module and utilize space that otherwise would be wasted. Personal belongings will be stored in the individual crew quarters. No storage is provided in the floor since it is very difficult to cut holes in the floor without greatly reducing its strength

### 4.10 ECLSS

A possible ECLSS system recommend for use is a partially closed system outlined in Fig. 4.11. With this type of system, none of the drinking water will be recycled. Only nonpottable water should be recycled. This greatly increases the safety factor of the mission.

On the MEV a totally closed loop system may be used. This would further decrease the logistics requirement. If a system failure should occur, the crew could simply abort the remainder of their surface stay. The MTV will be provided with a secondary system identical to the first. This system will serve as a backup and may also be used during times of excess activity.

The MEV system is actually composed of four subsystems as shown in Fig. 4.12. As is evident from the diagram hydrogen and methane are produced. The hydrogen produced could possibly be used to compensate for boil-off . The methane could be used as fuel for the reaction control systems. The actual amounts of these substances being produced has yet to be determined.

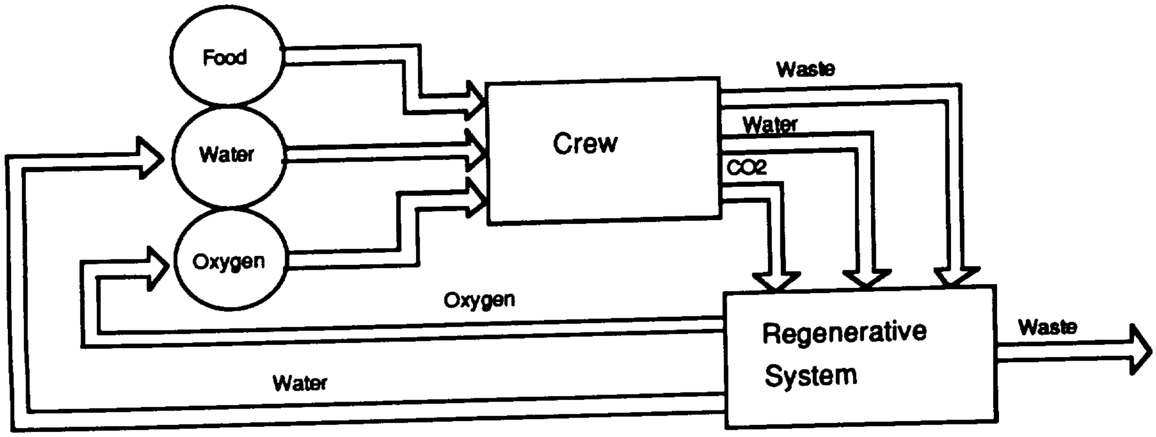



**Figure 4.11:** Partially Closed ECLSS System

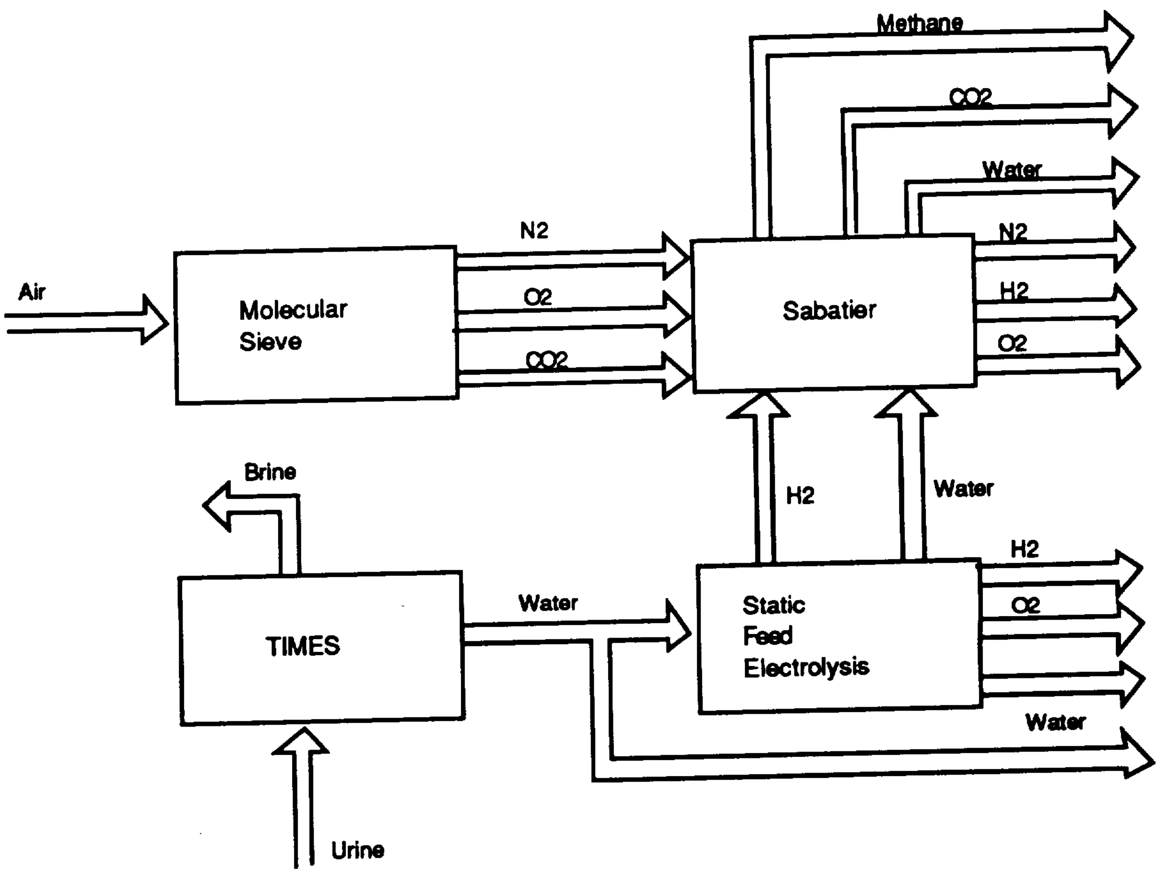



**Figure 4.12:** MEV ECLSS

## 4.11 Mars Habitation Module (MHM)

While the MHM was designed with many of the same considrations of the MTV hab, it contains a much smaller volume. The much smaller enclosed volume is due to several reasons. One, the crew will be on the surface of Mars for only 60 days and will be able to tolerate smaller living conditions. The crew will also be participating more EVA which will augment the semi-cramped living conditions. Two, much of the scientific equipment will be stored on the rovers thus minimizing the room needed on the MHM. A layout of the MHM can be seen in Figure 4.13. The left end of Figure 4.13 is the forward end while the right end is aft. A more detailed layout of the MHM crew quarters can be seen in Figure 4.14(a) and 4.14(b).

In addition, a hatch was added in the floor for access reasons. The crew will be able to obtain and don their EVA suits in the airlock and use the hatch in the floor to leave the MHM. While descending from the MHM to the lower level of the biconic, the astronauts will be aided in mobility by use of a type of small elevator (single passenger). The hatch in the side will be for emergency use only.

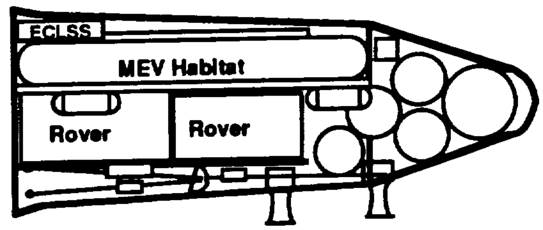



Figure 4.15: MHM Placement in the biconic

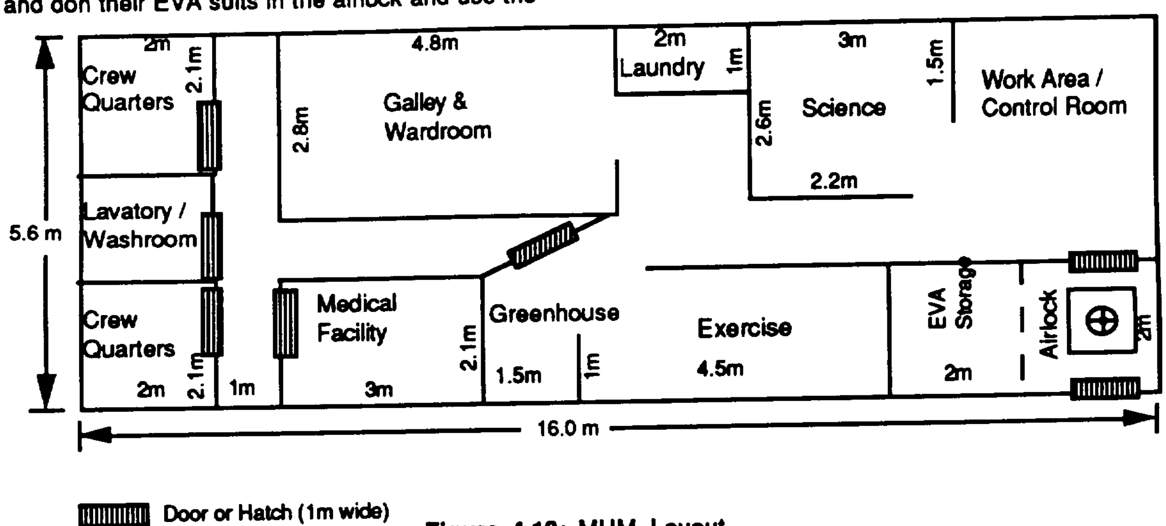



Figure 4.13: MHM Layout

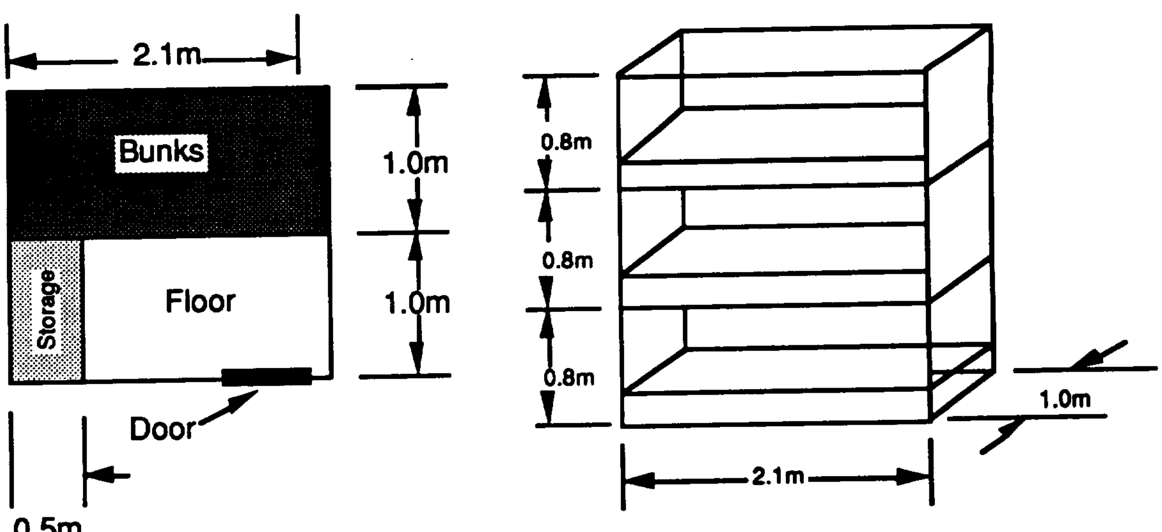



Figure 4.14(a): MHM Crew Quarters (top view), (b): Bunk Bed Assembly

The total volume and mass necessitated by crew systems is listed in Appendix 4.1. This table does not include mass and volumes for needed for other groups such as structures or avionics for example. For a 60 day stay, the allotments outlined in Appendix 4.1 seem to be about as minimal as it is going to get. The MHM total, approximate volume is about 250m$^3$.

Since the MHM will be positioned in the biconic, which lands horizontally, the MHM will consist of a horizontal cylinder positioned within the biconic roughly as shown in Figure 4.15.

## 4.12 Crew Scheduling

Although a definite hourly schedule for the crew is not given in this week's report, there has been research into what different aspects will/will not require crew time. There are two main categories of crew time. First, support time is the time the crew spends actually participating in mission specific activities. Second, overhead time is the time spent to keep the crew alive and well. These two categories are further broken up as follows:

- Support time
  - EVA operations
    - Ship/equipment maintenance
    - Mission specific operations
    - Ground operations
  - IVA operations
    - Ship/equipment maintenance
    - Science experiments
    - "Flight" required operations
    - Additional work
- Overhead time
  - Sleep
  - Eating
  - Cleaning
    - Hygiene
    - Clothes
  - Exercise
  - Recreation
  - Personal time
  - Sickness (not scheduled)
    - Medical treatment
    - Recovery

A rough outline of the crew's daily routine is as follows:

| | |
|---|---|
| • work | 10 hr. |
| • go to sleep ans wake up | 2 hr. |
| • sleep | 8 hr. |
| • Misc. | 4 hr. |

Work will include exercise and any other type of maintenance work that has to be done. Also, the crew will have a few days off periodically as the trip will be demanding mentally and physically.

## 4.13 Radiation Introduction

Crew Members on Mars will be exposed to basically two different types of radiation: Ionizing, and Non-Ionizing. Ionizing radiation (which is the primary concern for crew members on the surface of mars) affects biological systems by breaking chemical bonds. This can cause acute damage to tissue, latent genetic and cancer effects, or the possibly sudden illness, incapacitation, or death. Non-ionizing radiation is not strong enough to break the chemical bonds in a biological system, but over a prolonged period of time can cause irreversible damage to the biological system.

### 4.13.1 Atmosphere and Elevation Effects on Radiation

The Martian atmosphere will provide an acceptable layer of protection from radiation. This protection will depend upon the surface pressure, atmospheric density, and the thickness of the layer of atmosphere above the point of interest. Each factor depends upon elevation.

These three factors are inversely related to the elevation. That is, as elevation is increased, the surface pressure, density, and the thickness of the layer of atmosphere above the point of interest will decrease. Since radiation is inversely related to these three factors, radiation levels will increase as elevation is increased. Therefore, maximum atmospheric radiation protection will occur at the lowest elevation possible.

### 4.13.2 Sources of IR

Ionizing radiation is emitted from two sources, either natural or man made. Natural sources are the primary source for ionizing radiation.

There are several types of natural radiation found in the space environment. The major sources of ionizing radiation that are of interest are:

- Solar Cosmic Rays
- Galactic Cosmic Rays

The other natural sources of radiation either have a sufficiently low enough particle energy or a low enough flux density to be considered negligible. A possible exception would be trapped radiation belts due to Mars magnetic field.

Man made sources of ionizing radiation would include:

- Electric Power Sources
- Small Radiation Sources
- Induced Radiation

The primary man made source of concern would be the electric power source. This source could possibly be either a nuclear reactor or a radioisotropic source. Small radiation sources are considered to be any machine or materials which uses or produces radiation. Induced radiation is produced from the interaction of spacecraft materials with proton from GCR and solar flares. An example is the conversion of non-radioactive aluminum to radioactive sodium[4.1]. The last two sources, small radiation sources and induced radiation sources, can be easily shielded and therefore neglected in comparison to electric power sources.

An additional comment is needed for induced radiation. Even though it has been stated that this type of radiation can be neglected in comparison to electric power sources, this only holds true during 'normal' radiation incidents. 'Normal' being defined as radiation received daily rather than during solar flares. During a solar flare, large quantities of protons are produced which interact with the shielding materials to produce massive amounts of secondary radiation. Secondary radiation is hard to shield against. The material which under 'normal' conditions protects the crew now becomes a hazard. This problem needs to be addressed in more depth in future studies.

**4.13.2.1 Solar Cosmic Rays**
Solar cosmic rays (SCR) are the product of giant eruptions on the suns surface. These eruptions are periodic with a cycle of approximately 11 years. The eruptions usually last from half an hour to an hour, but the bombardment of solar particles emitted from these eruptions can last for days. A solar flare event is electromagnetic radiation emitted by the sun during a solar flare. The radiation received from solar flares has several names[4.1]:

- Solar Cosmic Radiation
- Solar Proton Event Radiation
- Solar Energetic Particles (SEP)

**4.12.2.2 Galactic Cosmic Rays**
Galactic cosmic rays (GCR) are radiation which is produced from a source outside our immediate solar system and is assumed to relatively constant through out our solar system. The amount of radiation due to GCR is a function of both altitude and latitudinal position. Using earth as an example[4.1], it can be illustrated that lower doses of radiation are received when the altitude and orbital inclination (position with respect to earth, 0 is the geomagnetic latitude zero and 90 is polar) are at their lowest.

**4.12.2.3 Mars Magnetic Field**
It was mentioned that the Mars magnetic field could be neglected as a source of trapped radiation belts. This assumption has been proven valid for the following reasons.

| | | Integrated Dose Equivalent, rem, at an altitude of- | | | |
|---|---|---|---|---|---|
| | | 0 km | 4 km | 8 km | 12 km |
| Galatic Cosmic Rays | Skin | 13.2 | 15.9 | 18.9 | 22.4 |
| | BFO | 11.9 | 13.8 | 15.8 | 18.0 |
| Solar Flare Event Aug-72 | Skin | 9.0 | 21.9 | 46.2 | 82.6 |
| | BFO | 4.6 | 9.9 | 18.5 | 30.3 |
| Solar Flare Event Nov-60 | Skin | 9.7 | 15.1 | 21.9 | 29.6 |
| | BFO | 7.3 | 10.8 | 14.8 | 19.1 |
| Solar Flare Event Feb-56 | Skin | 11.0 | 13.4 | 16.2 | 19.1 |
| | BFO | 9.9 | 11.8 | 13.6 | 15.3 |

**Table 4.2:** Integrated Dose Equivalent for the Mars Low-Density Atmosphere Model

| | | Integrated Dose Equivalent, rem, at an altitude of- | | | |
|---|---|---|---|---|---|
| | | 0 km | 4 km | 8 km | 12 km |
| Galatic Cosmic Rays | Skin | 11.3 | 13.4 | 15.8 | 18.6 |
| | BFO | 10.5 | 12.0 | 13.7 | 15.6 |
| Solar Flare Event Aug-72 | Skin | 3.9 | 9.5 | 21.1 | 42.8 |
| | BFO | 2.2 | 4.8 | 9.5 | 17.4 |
| Solar Flare Event Nov-60 | Skin | 6.4 | 10.0 | 14.8 | 21.1 |
| | BFO | 5.0 | 7.5 | 10.6 | 14.4 |
| Solar Flare Event Feb-56 | Skin | 9.2 | 11.1 | 13.3 | 15.9 |
| | BFO | 8.5 | 10.1 | 11.7 | 13.4 |

**Table 4.3:** Integrated Dose Equivalent for the Mars High-Density Atmosphere Model

There are two key elements that are essential for a planet to have radiation belts similar to earth[4.17]. These are:

1) Strong magnetization of its body,
2) Sufficient exposure to the solar winds.

By using data from Mariner IV, it has been shown that Mars magnetic field is negligible. It has also been estimated that the Martian surface magnetic field strength is less than 1000 gammes ($1*10^{-3}$ gause). By comparison, the Earth's magnetic field strength is 50,000 gammes (0.5 gause).

Mars does have exposure to the solar winds, but with an inadequate magnetic field, radiation belts are not present in sufficient strengths to pose a problem. This analysis has been confirmed by the Mariner IV fly by of Mars. Therefore, avoidance of radiation belts for the MEV and MTV will not be a concern.

#### 4.13.4 Acceptable Doses

The National Council on Radiation Protection and Measurements (NCRP) has proposed guidelines, instead of limits, to the amount of radiation levels acceptable. Limits are not possible because there are outside circumstances which can not be controlled. Table 4.4[4.8], gives these proposed radiation exposure guidelines for body forming organs at a depth of 5 cm (rem=roentgen equivalent man).

| Period | BFO (5cm) |
|---|---|
| 30 days | 25 |
| Annual | 50 |
| Career | 200+7.5(age-38) females<br>200+7.5(age-30) males |

**Table 4.4:** Acceptable Doses (rem)

#### 4.13.5 Model For the Prediction of Radiation On Mars

GCR flux intensities are believed not to vary significantly within the solar system and therefore a currently used model for GCR distribution (derived for earth) can be applied to Mars. The GCR distribution model was derived (at 1AU) from spectra received from the three largest solar flares (occuring in August 1972, November 1960, and February 1956). By using 1 AU for Mars instead of its 1.5 AU, a conservative model can be applied for the GCR distribution on Mars.

For solar flares, a baryon transport code (BRYNTRN) is used. Integrating BRYNTRN with an existing transport code for GCR, the distribution of radiation can be computed. This numerical method has be calculated to have an accuracy of 1%[4.3].

##### 4.13.5.1 Predicted Doses Near the Martian Surface

Table 4.3 gives the predicted dose equivalents for the Mars Low-Density Atmospheric Model and table 4.4 gives the predicted dose equivalents for the Mars High-Density Atmospheric Model[4.1].

A conservative estimate for the predicted radiation doses on the Martian surface can be calculated by summing up the radiation received yearly from GCR and add one giant solar flare to that value. Using this approach, the annual dose for BFO is given in Table 4.5.

| Altitude | 0 km | 12 km |
|---|---|---|
| High Density Model | 19 rem/yr | 33 rem/yr |
| Low Density Model | 22 rem/yr | 48 rem/yr |

**Table 4.5:** Predicted radiation doses on the Martian surface

#### 4.13.6 Predicted Radiat6on Levels

As illustrated in Table 4.2, the radiation received during the worst case scenario (low-density, high altitude) will be 48 rem/yr. This is below the allowed 50 rem/yr limit. Therefore the Martian atmosphere will provide adequate protection from space radiation if men and women could be instantaneously transported to Mars.

However, this is not the case. Future astronauts will travel through space and during the course of their journey they will be constantly bombarded by space radiation. This dictates that future astronauts will require additional radiation protection while on the martian surface to keep their overall radiation doses to a minimum, preferably keeping to the ALARA principle (As Low as Reasonably Achievable).

For a complete analysis of the amount of radiation protection required, radiation doses received in transit to and from Mars will be needed.

An estimate for the amount of radiation received by the crew for the first mission (a 60 day stay on the Martian surface) is given in Table 4.6.

| | Altitude | |
|---|---|---|
| | 0 km | 12 km |
| High Density Model | 3 rem | 6 rem |
| Low Density Model | 4 rem | 8 rem |

**Table 4.6:** Estimates of radiation doses received during a 60 day stay

Estimates of the radiation received during the second and third mission will be calculated in the future.

As can be seen from Table 4.6, not much shielding will be required to adequately protect the crew while on the martian surface. In fact, that amount of radiation that the crew will receive while on Mars is important, but small compared to the amount they will received in space. Crew Systems feels that if the crew can be protected against 10 rem of radiation (the worst case scenario including a factor of safety and excluding massive solar storms), for a 60 day stay, the crew will be adequately protected.

### 4.13.7 Protection Against Ionizing Radiation

The most damaging space radiation is the solar particle event, which may or may not accompany some solar flares. Radiation doses received from these events can easily exceed any and all acceptable standards resulting in rapid death.

To aid in the protection of astronauts from the damaging events, two ideas are presented[4.6]. The first is a solar shelter in which the crew could safely weather the solar particle events (which can last from a few hours to a few days). The storm shelters would be stocked with enough supplies to last the estimated duration of an event and would situated in the most heavily shielded area of the Mars Habitation Module.

The second idea is some kind of early warning system. This could be a solar monitor system which would continuously monitor the sun for solar activity and would send out warnings of solar particle events.

Some other preventive measures would include personal radiation detectors to monitor each crew members radiation levels, a system of keeping track of each crew members radiation history, and radiation detectors situated in the habitation module

### 4.13.8 Non-Ionizing Radiation

The major effect of non-ionizing radiation (NIR) is the production of heat in the tissue. Other effects caused by ultraviolet radiation and other sources are chemical changes within the cell as well as electron excitation.

### 4.13.9 Sources of Non-Ionizing Radiation

The primary source of NIR is from man-made sources, even though a significant amount of NIR is from naturals occuring sources. The natural sources of NIR are[4.1]:

- Continuous Solar Emissions
- Solar Flare Events
- Galactic Cosmic Rays
- Magnetic fields from planets

Continuous solar emissions are the primary source of concern for NIR. This radiation originates from the sun which continuously emits a broad band of electromagnetic radiation. This radiation has a frequency between 3 Hz to around $3*10^{15}$ Hz (the actual dividing line between electromagnetic ionizing radiation and non-ionizing radiation is foggy at best, the accepted division occurs at a frequency of $3*10^{15}$, below this frequency is the non-ionizing radiation and about is ionizing radiation). NIR from solar flare events and GCR are at such low doses that they do not appear to present a problem. Magnetic field strengths of planets varies from planet to planet. The magnetic field strength of Mars is virtually nonexistent and will not present a problem.

Non-ionizing radiation can be categorized into several types of radiation:

- Optical Radiation
- Radiofrequency Radiation
- Extremely Low Frequencies (ELF)
- Pressure Waves

### 4.13.10 Limits of Non-Ionizing Radiation

The limits radiofrequency radiation are the limits found in American National Standards Institute (ANSI) Radio Frequency Protection Guides (RFPG)[1]. These limits are shown in Table 4.7.a and Table 4.7.b.

| Frequency Range (MHz) | $E^2$ ($V^2/m^2$) | $H^2$ ($A^2/m^2$) |
|---|---|---|
| 0.3 - 3 | 400000 | 2.5 |
| 3 - 30 | 4000 | 0.025 |
| 30 - 300 | 4000 | 0.025 |
| 300 - 1500 | 4000 | 0.025 |
| 1500-100000 | 20000 | 0.125 |

**Table 4.7:** Radiofrequency exposure limits

$E^2$ is the electric field strength, $H^2$ is the magnetic field strength, and the power density is the equivalent plane wave power density. These are the recommended values which should not be exceeded.

| Frequency Range (MHz) | Power Density (mW/cm$^2$) |
|---|---|
| 0.3 - 3 | 100 |
| 3 - 30 | 900/f$^2$ |
| 30 - 300 | 1 |
| 300 - 1500 | f/300 |
| 1500-100000 | 5 |

**Table 4.7(b):** Radiofrequency exposure limits

| Duration of Exposure Per Day | Effective Irradiance E(W/cm$^2$) X *10$^{-6}$ |
|---|---|
| 8 hrs | 0.1 |
| 4 hrs | 0.2 |
| 2 hrs | 0.4 |
| 1 hr | 0.8 |
| 30 min | 1.7 |
| 15 min | 3.3 |
| 10 min | 5 |
| 5 min | 10 |
| 1 min | 50 |
| 30 sec | 100 |
| 10 sec | 300 |
| 1 sec | 3000 |
| 0.5 sec | 6000 |
| 0.1 sec | 30000 |

**Table 4.8.b** Permissible UV Exposures

| Wavelength (nm) | TLV (mJ/cm$^2$) | Relative Spectoral Effectiveness (S) |
|---|---|---|
| 200 | 100 | 0.03 |
| 210 | 40 | 0.075 |
| 220 | 25 | 0.12 |
| 230 | 16 | 0.19 |
| 240 | 10 | 0.3 |
| 250 | 7 | 0.43 |
| 254 | 6 | 0.5 |
| 260 | 4.6 | 0.65 |
| 270 | 3 | 1 |
| 280 | 3.4 | 0.88 |
| 290 | 4.7 | 0.64 |
| 300 | 10 | 0.3 |
| 305 | 50 | 0.06 |
| 310 | 200 | 0.015 |
| 315 | 1000 | 0.003 |

**Table 4.8.a** Radiation Exposure-Actinic UV (8 hr period)

For incoherent Ultraviolet (UV) Radiation, several different limits have been set depending upon the type of UV radiation. For the UV-A (315nm - 400nm) Spectrum, the unprotected skin can received no more than 10 W/m$^2$ for periods of exposure greater than 1000 seconds and no more than 1 J/cm$^2$ for periods of exposure less than 1000 seconds.

For the actinic (UV Spectrum 200nm - 315nm, it is also the UV which causes chemical change), the limits are shown in Table 4.8.a. and Table 4.8.b. TLV is the Threshold Limit Value.

For Broad Band Frequencies (400nm - 1400nm), the limits are shown in Table 4.9.

| Wavelength (nm) | Blue Light Hazard-Function (B) | Burn Hazard Function (R) |
|---|---|---|
| 400 | 0.1 | 1 |
| 405 | 0.2 | 2 |
| 410 | 0.4 | 4 |
| 415 | 0.8 | 8 |
| 420 | 0.9 | 9 |
| 425 | 0.95 | 9.5 |
| 430 | 0.98 | 9.8 |
| 435 | 1 | 10 |
| 440 | 1 | 10 |
| 445 | 0.97 | 9.7 |
| 450 | 0.94 | 9.4 |
| 455 | 0.9 | 9 |
| 460 | 0.8 | 8 |
| 465 | 0.7 | 7 |
| 470 | 0.62 | 6.2 |
| 475 | 0.55 | 5.5 |
| 480 | 0.45 | 4.5 |
| 485 | 0.4 | 4 |
| 490 | 0.22 | 2.2 |
| 495 | 0.16 | 1.6 |
| 500-600 | (a) see below | 1 |
| 600-700 | 0.001 | 1 |
| 700-1050 | NA | (b) see below |
| 1050-1400 | NA | 0.2 |

(a) 10 Exp[(450-lambda)/50]
(b) 10 Exp[(700-lambda)/505]

**Table 4.9**: Retinal hazards from broad band optical sources

## 4.14 Crew Systems Related Emergencies

The following is a list of crew related emergencies and the rough procedures/consequences that follow:

- Fire
  - For small fires, fire extinguishers are readily available
  - For large fires, the ships automatic fire extinguishing system is enabled. This system is equipped with an additional manual controls.
- Food storage failure
  - This would include refrigeration failure in which case there should be enough dry food to supplement the shortage.
- ECLSS failure
  - If the ECLSS completely fails, the crew will be able to survive a minimal time in their space suits or transfer to/from the lander for more time. The ECLSS has enough backup and warning systems that this case is highly unlikely.
- Crew member(s) become mentally unstable
  - In this case the medical officer (if the medical officer becomes unstable then the highest in command) shall make the ultimate decision what is done with the individual.
- Crew health problems
  - Again the medical officer takes responsibility.
- Artificial gravity fails
  - This was covered in winter quarter's final report.
- If any major systems fail while on Mars
  - The ground mission is cut short, and the crew returns to meet up with with the MTV as soon as possible.
- In most emergency situations, the crew member in charge of the particular subsystem makes the ultimate decision as to what course of action to take (i.e., A leak in the pressure vessel - the engineer decides the course of action).

## 4.15 Conclusions and Recommendations

The crew systems group's report has given many of the considerations neccesay for design of the this type of mission. With more time, more detailed descriptions of certain systems and designs could be made. These might include; the exact designs of the ECLSS systems, control layouts (i.e. buttons, switches, etc.), and space suit designs. Also, more research is needed in the fields of multi and low gravity environments for extended periods of time on the human system.

# 5.0 ORBITAL MECHANICS

## 5.1 Introduction

What follows is a basic summary of trajectories and launch dates that could be used to complete a manned Mars mission in approximately 500 days. Some of the main focal points within the design requirements are launch windows, fuel mass minimization, trip time considerations, surface stay time, and accessibility to the three landing sites as specified by the mission operations group.

Given these design requirements the main objectives of the orbital mechanics group are to minimize fuel and trip time requirements. This task is made difficult since minimizing fuel requirements generally translates into an increased trip time through space.

However, by choosing an appropriate interplanetary trajectory, Martian parking orbits, and carefully selecting launch windows with relatively low velocity requirements, a compromise between fuel and time constraints can be achieved.

## 5.2 Choosing an Interplanetary Transfer Trajectory

For a mission to Mars the type of interplanetary trajectory that proves to require the least time in transit is one that involves a Venus swingby. The swingby can be used for either the outbound leg (Earth to Mars) or the inbound leg (Mars to Earth). Of these two choices a swingby during the inbound leg is the most attractive since velocity requirements for departure from Mars and arrival at Earth can be reduced. It is important to note that a swingby only changes the direction of a spacecraft's velocity vector and not its magnitude.

The only major disaΔVantage in using a Venus swingby has to do with timing. That is to say, Earth, Mars, and Venus must be in a certain position in order for the swingby trajectory to be feasible. The amount of time between inbound swingby opportunities with Venus can be anywhere from two to seven years apart.

Figure 5.2.1 illustrates an inbound Venus swingby and the relative positions of Earth, Mars, and Venus at each of the four major velocity changes. These ΔVs will be refered to as: TMI - trans Mars injection, MOI - Mars orbit injection, TEI - trans Earth injection, and EOI - Earth orbit injection.

## 5.3 Launch Window Selection and Analysis

### 5.3.1 Window Requirements for a Manned Mars Mission

An opportunity for a TMI launch from an Earth orbit, or window can be defined as the time period in which a launch can be successfully achieved based on given mission requirements.

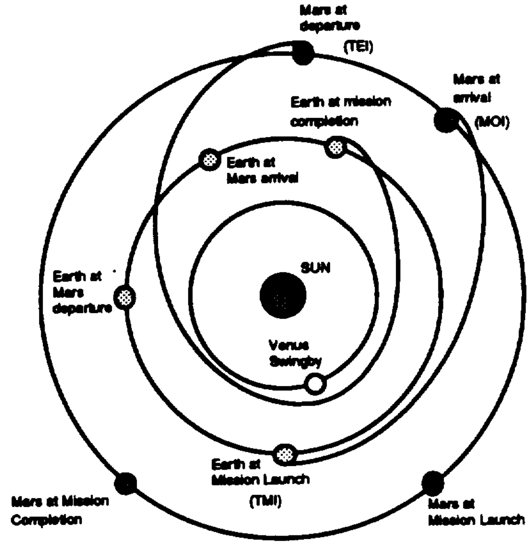



**Figure 5.2.1**: Inbound Venus swingby

For a manned mission to Mars it was determined that three basic requirements must be satisfied for a given launch window. These requirements are:

1. Utilization of an inbound Venus swingby.
2. Reasonable ΔVs for TMI, MOI, TEI, and EOI.
3. Radiation effects on the crew due to solar maximums.

### 5.3.2 Using SWISTO in Determining Launch Windows

SWISTO is an acronym for SWIngby STopover Optimization. It is a computer program that was written for NASA to aid in feasibility assessments of interplanetary trajectories. Given various trajectory and spacecraft performance specifications the program will compute velocity and date requirements that are based on minimizing propellant mass.

SWISTO utilizes the method of patched conics in its analysis. The patched-conic method provides good and relatively quick estimates of ΔV requirements for use at the preliminary design level. It is important to note that actual mission design and execution must employ the most accurate, and thus more time consuming, numerical integration techniques.

Table 5.3.1 lists the major input parameters that were used in our study of a 500 day class, manned, Mars mission.

In addition to these input parameters the user enters a range of launch dates from Earth, the corresponding arrival dates at Mars, and a range of swingby dates at Venus.

| | | |
|---|---|---|
| *Trajectory type:* | Inbound Venus Swingby | |
| *Trip times:* | Outbound leg | = 200 days |
| | Inbound leg | = 250 - 270 days |
| | Stay at Mars | = 60 days |
| | Total trip time | = 510 - 530 days |
| *Orbital parameters:* | • 400 km circular Earth parking orbit | |
| | • 0.50 elliptical Mars parking orbit with periapsis radius of 300 km | |
| *Vehicle performance:* | Engine Isp = 1100 sec. | |

**Table 5.3.1:** SWISTO input parameters

Given the input parameters and dates SWISTO will compute and provide as output the velocity requirements and their corresponding dates. Table 5.3.2 summarizes the output data generated by SWISTO.

| | | | |
|---|---|---|---|
| *Dates:* | Launch (TMI) | *Velocities:* | $\Delta V1$ (TMI) |
| | Arrival (MOI) | | $\Delta V2$ (MOI) |
| | Restart (TEI) | | $\Delta V3$ (TEI) |
| | Swingby at Venus | | $\Delta V4$ (EOI) |
| | Return (EOI) | | |

**Table 5.3.2:** SWISTO output

**5.3.3 Determination of Launch Windows**
At this point the output data provided by SWISTO is analyzed and launch dates that satisfy the first two requirements described in Section 5.3.1 are defined as possible launch windows. For the first requirement (feasibility of an inbound Venus swingby) SWISTO will provide a value of 0 for the passage date if a swingby is not possible. Satisfying the second requirement (reasonable $\Delta V$ values for TMI, MOI, TEI, and EOI) is done by establishing some sort of maximum allowable $\Delta V$. This can be done in one of two ways. The first is to set a maximum allowable $\Delta V$ that cannot be exceeded by each injection maneuver separately. The other is to set a maximum allowable total $\Delta V$ that can not be exceeded by the sum of all four injection velocities.

The following graphs illustrate three launch windows that were found using SWISTO. For each window there are two graphs. One graph shows an individual breakdown of the four required $\Delta V$s vs the corresponding launch date from Earth, and the other graph plots the sum of all $\Delta V$s vs the Earth launch date.

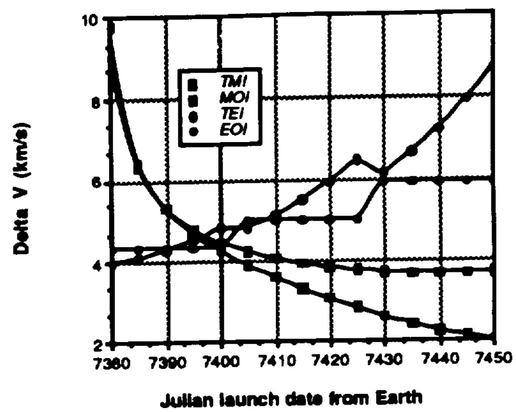



**Figure 5.3.2:** Individual $\Delta V$s for window#1

Setting a maximum allowable $\Delta V$ of 5 km/s in Figure 5.3.2 yields a launch window with Julian dates of 7395 to 7425. These dates translate to a 30 day time period from January 7, 2016 to February 7, 2016. During this time period all four $\Delta V$s do not exceed 5 km/s.

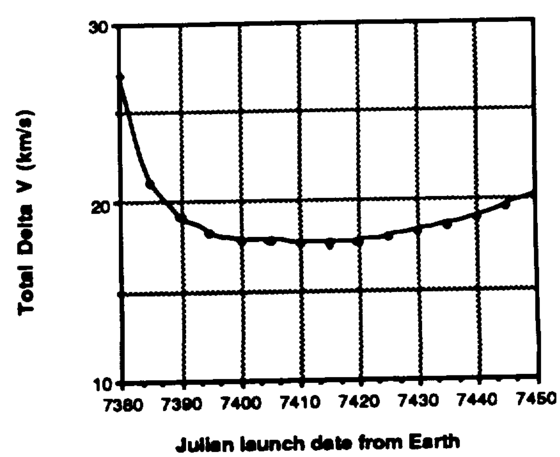



**Figure 5.3.3:** Total $\Delta V$ for window #1

From Figure 5.3.3 a larger launch window is obtained by setting a maximum allowable total $\Delta V$ at 20 km/s. The launch window now corresponds to a 63 day time period from December 29, 2015 to March 1, 2016.

The governing factor in deciding which window to use depends mainly on the TMI $\Delta V$. The TMI will generally require the most propellant mass since the mass of the MTV will be greatest in LEO prior to TMI. A consistently lower TMI curve in figure 5.3.2 makes for a larger launch window, but the individual ΔVs become too high. Therefore the first set of boundary dates will be used.

Figure 5.3.4 displays the individual velocity breakdown for the second launch window. Setting the maximum allowable velocity at 8 km/s yields a 20 day launch window from July 19, 2018 to August 8, 2018.

LAUNCH WINDOW #2

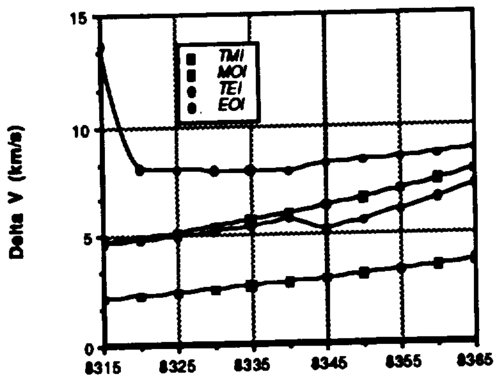



**Figure 5.3.4:** Individual ΔVs for wndow #2

LAUNCH WINDOW #2

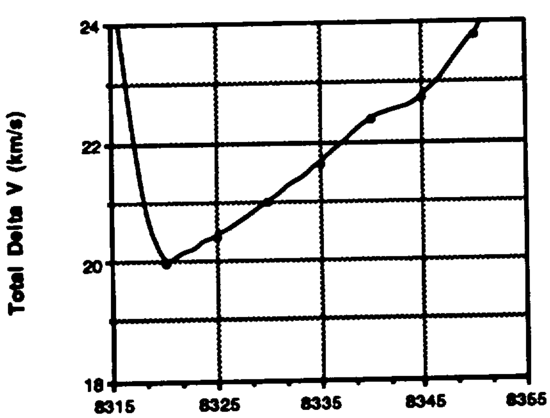



**Figure 5.3.5:** Total ΔV for window #2

Figure 5.3.5 displays the total ΔV for the second launch window. However, since the individual TMI curve is "high" it is better to determine the launch window from Figure 5.3.4.

The ΔV breakdown for the third window is as follows:

LAUNCH WINDOW #3

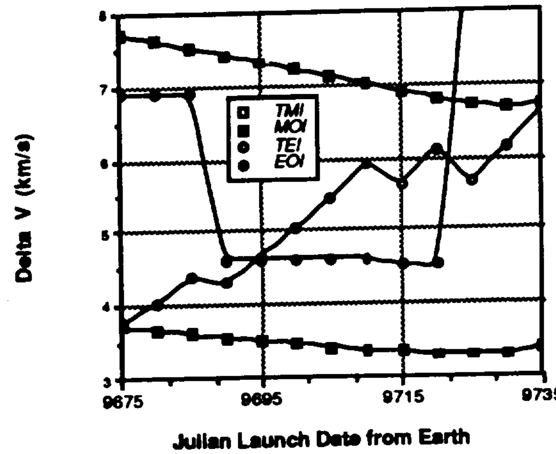



**Figure 5.3.6:** Individual ΔVs : Window #3

The third launch window is similar to the second in that the individual TMI curve is "high" again. Therefore, Figure 5.3.6 is used in determining the third launch window. This window occurs over a 30 day period with a maximum allowable ΔV of 7.5 km/s. The dates corresponding to this window are April 19, 2022 and May 19, 2022. This can be seen in the following figures. Once again both figures were used in determining the final window boundaries.

LAUNCH WINDOW #3

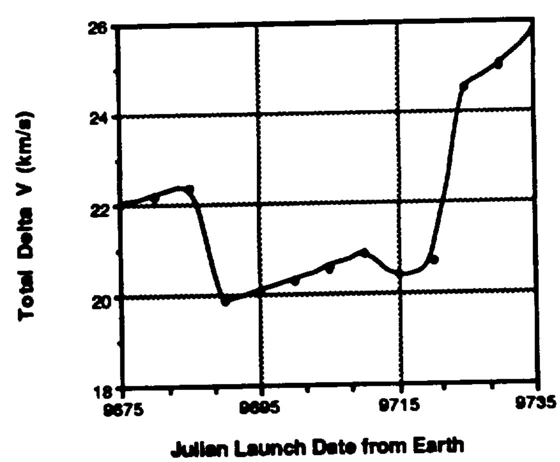



**Figure 5.3.7:** Total ΔV for window #3

### 5.3.4 Summary of Launch Window Opportunities

Table 5.3.3 summarizes the launch windows for an inbound Venus swingby. In looking at the table it is evident that the third launch window does not occur at a favorable radiation period with respect to the solar cycle.

| Launch Date(s) | Reasonable ΔV | Radiation Level |
|---|---|---|
| Jan 7, 2016 - Feb 7, 2016 | Yes | Low |
| Jul 19, 2018 - Aug 8, 2018 | Yes | Low |
| Apr 19, 2022 - May 19, 2022 | Yes | High |

**Table 5.3.3:** Launch Window Summary

## 5.4 Radiation Effects in Transit

Necessity for radiation protection stems from the fact that during transit to and from Mars the crew is vulnerable to solar radiation from the sun. Generally, within close proximity of a planet, protection from solar radiation is provided by the planet's magnetic field. However, the mass of material required to shield the crew in transit during a solar maximum (a period of intense solar activity) exceeds the maximum mass for a practical mission (i.e. the spacecraft simply becomes too "heavy" due to the radiation shielding). Therefore, it is necessary to launch at times that send the crew in transit during solar minimum periods.

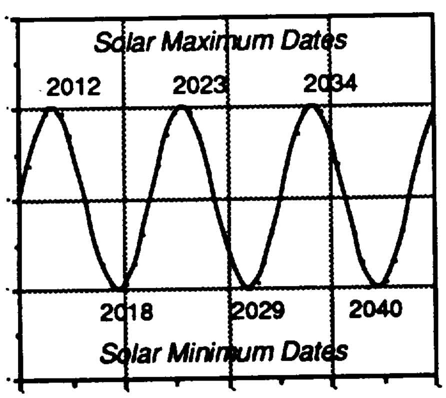



**Figure 5.4.1:** Solar cycle

The solar cycle is sinusoidal and repeats itself every 10.9 years. Figure 5.3.1 illustrates the solar cycle with the corresponding dates of maximums and minimums. From this figure desirable launch windows lie along points on the curve that are in the vicinity of the minimum point, generally in the lower half of each cycle.

The next figure is a plot of the launch dates with respect to eachother and the solar cycle.

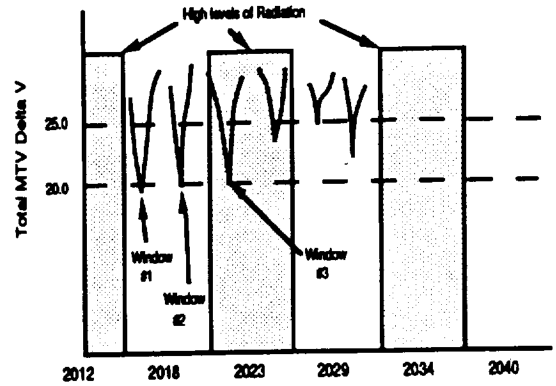



**Figure 5.4.2:** TMI date vs total ΔV & radiation

From this figure, the first three launch windows can be seen. The graph shows the total ΔV for a mission with respect to each possible TMI launch date. Some dates have significantly higher ΔVs than others due to the position of the planets. In order for our mission to be feasible, the total ΔV must fall below 20 km/s, which is shown on the plot. The three windows that meet this design limit are labeled on the figure.

Unfortunately, when comparing the launch dates with the solar cycle, it becomes clear that the third launch date falls within a high risk period. One possibility for avoiding this problem would be to launch at a later date when the risk factor is at a lower level.

The next local total ΔV minimum, meaning the lowest point on each segment of the total plot, occurs in the year 2029, but does not fall below the maximum 20 km/s limit. An identical situation occurs for the following local total ΔV minimum in 2031.

There might be a ΔV at or below 20 km/s in the next few years, 2032 - 2037, but these years correlate with another high risk radiation period. In other words there would be no difference between launching during this period versus the first high risk period, 2020 - 2025.

The only other alternative would be to attempt a launch between 2038 and 2044. This will mean, at the minimum, a 20 year waiting period between the second and third mission. By this time, the new technology available will have surpassed the technology of a more current design and prove to be a more useful project.

Given these considerations, our recommendation is to stay with the three dates proposed, hoping that by the year 2022 new radioactive shielding technology will be available to make the third mission possible.

## 5.5 Establishing Parking Orbits at Mars

### 5.5.1 MTV Parking Orbit at Mars

Parking orbit selection can greatly assist in minimizing the total fuel requirements for the MTV. The size and shape of the parking orbit at Mars also plays a significant role in determining the necessary fuel requirements. The variables used in determining the orbital configuration are orbit altitude, eccentricity, period, and velocities at the departure and redocking location. All these characteristics are related.

After analyzing the trade-offs the shape of the MTV parking orbit was chosen to be elliptical. This allows for smaller ΔVs upon arrival and departure from the Martian parking orbit, MOI and TEI. Fuel requirements and total mass of the vehicle are also reduced for the entire mission by this type of orbit. By changing the eccentricity or 'shape' of the ellipse, the ΔVs can be adjusted to find an optimal magnitude. As the eccentricity increases the ΔV for the MOI and TEI decrease. On the other hand, as the eccentricity increases so does the period and the ΔV required of the MEV to initiate its landing sequence and when attempting to redock with the MTV.

### 5.5.2 MEV Parking Orbit at Mars

A circular parking orbit was chosen for the MEV. This type of orbit has more consistent access to landing sites within the latitude limits designated by its orbital inclination. The period will be less than that of an elliptical orbit with the same periapsis altitude, allowing for a shorter descent sequence. Finally, the windows of opportunity to initiate the landing procedure along with redocking with the MTV, are significantly greater for the circular orbit. This increase can be attributed to the constant altitude throughout the orbit.

An elliptical MEV orbit does not prove to be a feasible orbit. Although it would require a lower separation ΔV, the orbit will have fewer windows for landing and redocking. Velocities and altitudes at various points in the orbit will not be constant, making landing and redocking extremely complicated. Higher eccentricities of elliptical orbits result in even fewer windows or opportunity.

An altitude of 300 km will be used for the MEV's circular orbit. At this altitude the MEV is close enough to stay within it's propulsive and maximum trip time limits. The MTV is very capable of establishing an elliptical orbit with a 300 km periapsis altitude. This also eliminates any possible need for an altitude change by either vehicle in preparation for landing the MEV. In a circular orbit at this altitude the MEV has a period of 1.89 hours and a constant velocity of 3.14 km/s. Figure 5.5.1 shows these characteristics in a graphical form. Please note that this figure is not to scale.

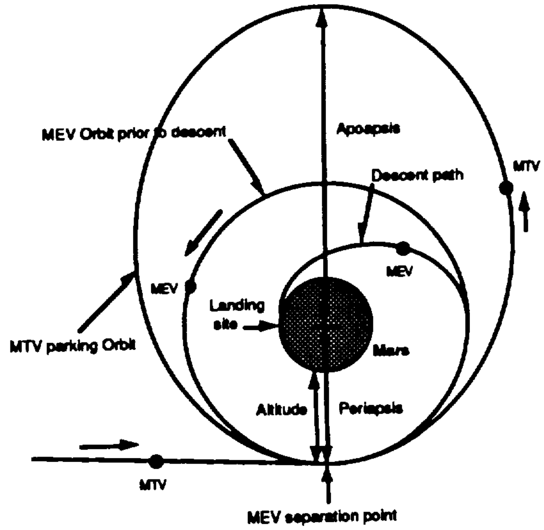



Figure 5.5.1 Orbital Configuration at Mars

### 5.5.3 Orbit Optimization

A fairly involved trade-off occurs when attempting to minimize the fuel mass requirements of the MTV and MEV. The ΔVs involved are: MOI and TEI burns for the MTV, and a change in velocity by the MEV from the periapsis of the MTV elliptical orbit to a circular orbit. Determining the MTV velocity at the separation point sets the MTV's period and eccentricity. The best design would minimize these ΔVs as much as possible while maintaining the mission objectives.

The MTV periapsis velocity is the key variable used in determining the ΔV and fuel mass requirements of both the MTV and MEV. To minimize the MTV's overall mission fuel requirements the periapsis velocity should be as high as possible. This minimizes both the MOI and TEI ΔVs, but increases the MEV circularization ΔV.

Minimization of MEV fuel mass would occur if the MTV enters a 300 km circular orbit. This would eliminate the MEV circularization ΔV. The MEV would be able to initiate its landing sequence directly from the MTV. The major drawback of this design would be the extremely large MOI and TEI ΔVs required.

When determining the optimal periapsis velocity of the MTV the fuel mass of each vehicle must stay within design limits and requirements. An optimal MTV periapsis velocity will occur somewhere between the two individual minimums mentioned above, based on fuel mass limits of both vehicles.

After working with the propulsion discipline, the optimal trade-off for the MEV and MTV $\Delta V$s were obtained. The MEV's optimal $\Delta V$ is 1.05 km/s, which establishes a circular orbit of a 300 km altitude. Therefore the MOI was determined to range from 2.0 to 3.5 km/s. The range of $\Delta V$s for the MOI result from the corresponding different launch dates that may be used within each launch window. Similarly, the MTV departure $\Delta V$, TEI, will also vary day to day based on the exact date of the TEI.

Working backwards from this MEV $\Delta V$, the goal was to establish a MTV periapsis velocity of 4.18 km/s. This resulted in an elliptical parking orbit with an eccentricity of 0.50, and a period of 5.34 hours. For a complete graphical representation see the following figure.

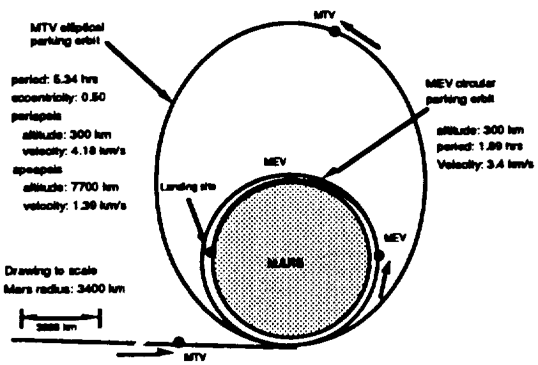



**Figure 5.5.2** Scaled Configuration at Mars

## 5.6 Summary of Mission Sequence of Events

Table 5.6.1 provides a sequential list of the $\Delta V$s executed within each mission. The magnitudes listed for the TMI, MOI, TEI, and EOI are given as ranges based on their variations with respect to each specific launch date.

| | | Vehicle (s) | $\Delta V$ |
|---|---|---|---|
| 1. | TMI | MTV | 4-7 km/s |
| 2. | 2 degree interplanetary plane change | MTV | 0.15 km/s |
| 3. | MOI : enter into an elliptical parking orbit | MTV | 2 - 3.5 km/s |
| 4. | MEV/hab separation (enter circular orbit) | MEV/hab | 1.05 km/s |
| 5. | Initiate descent | MEV/hab | 1.6 km/s |
| 6. | Landing | MEV/hab | aero |
| 7. | Take-off | MEV | aero |
| 8. | Reestablish a circular parking orbit | MEV | 1.6 km/s |
| 9. | Redock with the MTV in its elliptical orbit | MEV | 1.05 km/s |
| 10. | TEI | MTV | 4.5 - 7 km/s |
| 11. | * 2 degree interplanetary plane change | MTV | 0.15 km/s |
| 12. | * Venus Swingby | MTV | 0.0 km/s |
| 13. | EOI | MTV | 4.5 - 8 km/s |

* The Venus swingby might occur before the interplanetary plane change depending on the phase angle between Earth and Mars at the time of the initiation of the TEI.

**Table 5.6.1:** Summary of Overall Mission

The values used in the $\Delta V$ column of Table 5.6.1 were obtained assuming the following variable inputs:

1. An elliptical MTV parking orbit of:

   periapsis altitude : 300 km
   eccentricity : 0.50

2. A circular MEV parking orbit of:

   altitude : 300 km

## 5.7 3-D Relationship to Landing Sites for the MEV

### 5.7.1 Accessible Latitude Ranges

The coordinates for the landing sites are:

| | | |
|---|---|---|
| Mission #1: | 13 North | 139 West |
| Mission #2: | 06 South | 78 West |
| Mission #3: | 07 South | 147 West |

The MTV will arrive at Mars on the Martian ecliptic plane which is tilted 23 degrees with respect to the Martian equator. An orbiting spacecraft has access to any landing site at a latitude equal to or less than the angle of inclination. For our case, the MEV has access to any landing site below latitude 23 degrees north, and above 23 degrees south. Since this angle of inclination is greater than the latitude of each landing site, a plane change by the MTV or MEV to reach a particular landing site will not be required. The following figure illustrates the MEV's range of achievable landing sites at Mars.

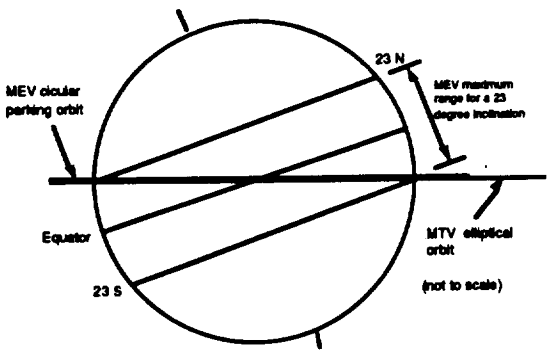



**Figure 5.7.1:** MEV latitude range at Mars

### 5.7.2 Ground tracks

One complete orbit will create one 'sine-like' wave line on a flat projection of the Martian surface which spans all 360 degrees of longitude. Since the planet rotates under the orbiting vehicle the next wave will be shifted to the west. Using the characteristics of the MEV's circular orbit and the Martian length of day, this converts to 27.78 degrees per orbit. When the MEV makes its descent it will follow one of these lines down to the surface.

### 5.7.3 Landing Accuracy

Calculations by the MEV aerodynamic analysis group show that the landing range from an altitude of 300 km is about 600 km perpendicular to the orbital plane. This means that a specific ground track can be used to initiate the landing sequence if it passes within 600 km of the landing site. This is very useful because the ground tracks do not overlap each other or the landing site exactly, thus allowing accessibility to the landing sites which are not exactly beneath any one specific orbit.

After the MEV has established its circular orbit, the control system being used will determine the exact time and location to initiate the descent sequence. This determination will be based on the following trade-off. The magnitude of the required MEV deviation from the orbital plane or ground track, versus how many additional circular orbits the MEV would have to performed before an ideal or better track can be achieved.

### 5.7.4 Orbital Repetition

The MEV circular orbit repeats approximately every 13 orbits whereas the MTV elliptical orbit repeats approximately every 9 orbits. Therefore the time spent in the MEV will be no longer than 24.57 hours. After the MTV has established its parking orbit, determining when the MEV and habitation module will separate in order to reach the landing site will be determined through the following manner.

In order to minimize the time spent in the MEV by the crew, a combination of revolutions will be performed by the MTV and MEV. This will be done in such a manner that the MEV will have to perform a minimum number of revolutions before initiating its descent sequence.

### 5.7.5 Perturbations

The previous data presented did not take into account the effects of perturbations. They are defined as a deviation from normal or expected motions. These orbits that have been designed will be subjected to several different perturbations with different magnitudes.

During the interplanetary orbits the MTV will be subjected to solar radiation and solar wind. This will create a need for continuous monitoring of position and direction during transit. Once the vehicles have established their parking orbits, these orbits will be effected by the oblateness of Mars. This uneven shape creates gravitational characteristics which are not even and cause parking orbits to be slightly altered after a period of time.

Several techniques are available to solve and design around the effects caused by these perturbations. The Cowell, Encke, and variation of parameters methods are ways of solving for these effects. Unfortunately, these calculations require large amounts of computation to solve for accurately and are not included in this report.

## 5.8 Redocking

A very intricate redocking sequence is necessary for these missions to be successful. For both the MTV and MEV, the instances of repetition are not exact, but are within the range of the propulsion systems to obtain redocking. The following data provides a numerical perception of the difficulty in redocking. It takes approximately 117 MTV and 392 MEV rotations for both vehicles to meet exactly at the periapsis of the MTV's elliptical orbit. Some of the combinations within this range can be met by the use of extra propulsion, but should only be used in case of an emergency. Since the MEV will spend most of the time on the surface, it is very essential that the launch is timed with the location of the MTV so that the vehicles will meet as quickly as possible. The MEV should not required more than two revolutions before it meets the MTV at the periapsis.

This procedure will be performed as follows. The MEV will wait on the surface for an optimal launch window based on the MTV's location within its elliptical orbit. These windows will not be very abundant due to the elliptical shape which yields only one redocking location, the periapsis. After the MEV has reached its parking orbit altitude of 300 km, it will maintain this orbital altitude and path until this orbit and the MTV's elliptical orbit meet at the periapsis of the elliptical orbit.

In the case of an emergency if it becomes necessary to leave the surface as quickly as possible, the MEV will launch into its circular orbit anyway. After establishing a circular orbit with an altitude of 300 km it performs several orbital rotations until the MTV is met at its periapsis. The major obstacle would be if the orbit do not 'line up' favorably, requiring numerous revolutions for alignment. This also would require a significant amount of fuel would be required to perform the redocking procedure.

## 5.9 Conclusion

### 5.9.1 Summary

This investigation was performed in order to study characteristics of transfer and parking orbits for a manned mission to Mars. Based on the analysis performed, three missions have been planned for the years 2016 - 2024. When designing the orbital characteristics for these three missions several general statements can be made.

First, a significant trade-off occurs in the MTV design between trip time and fuel mass. The shorter the trip time the larger the ΔVs and fuel mass. The MOI and TEI ΔVs can be decreased by choosing specific types and sizes of MTV and MEV parking orbits.

A final optimal orbital configuration for both the MTV and MEV entails a detailed trade-off involving landing site accessibility, ΔVs and fuel mass considerations for both vehicles, and surface to MTV transfer times.

One of the most important points to begin this type of analysis is from the coordinates of the landing site. The three sites used in this design do not require orbital plane changes at Mars. If this were necessary extra ΔVs would be necessary.

In summary, a trade-off between landing site accessibility and mission performance must be made when selecting transfer and parking orbits for a manned mission to Mars.

### 5.9.2 Recommendations

Although the research outlined in this investigation is complete with respect to satisfying the objectives of this study, a few topics still need to be addressed.

Other possible methods of reducing ΔVs, such as different mission scenarios. For example, two vs three dimensional burns and tangential burns at Mars. Or maybe multiple impulse maneuvers, and a more accurate calculation of the solution for these orbital selections based on numerical integration techniques.

# 6.0 AVIONICS & POWER SYSTEMS

## 6.1 Introduction

The functions defined for the MTS avionics system are: spacecraft guidance, navigation, and control; sequencing; systems monitoring; environmental control and life support system; landing and docking of the MEV and MHM on the surface of Mars; thermal control; data management.

### 6.1.1 Computer System

The computing needs to perform these functions as accurately and simply as possible will be met using Integrated Modular Avionics (IMA). The system is supported by a powerful computer with an operating system that allows independent processing of application software. Physical partitioning between software modules will be maintained through the use of line replaceable units. The IMA forms a subsystem with a common design, shared fault tolerant processing, and multiple vehicle interfaces. The computers are housed in a cabinet of hardware modules as seen in Figure 6.1.1.[6.1]

This computer system model, developed by Honeywell, is based on the working model Airplane Information Management System (AIMS). AIMS integrates the following in two housing cabinets: electronic flight instrument system/engine indication and crew alerting system display generators; flight management; on-board maintenance; communication management; and data conversion gateways. The AIMS operating system and hardware provide total function separation and isolation under a combination of partition errors. It prevents application software from controlling shared resources to the exclusion of other application software (time partitioning), prevents application software from contaminating memory areas of other applications or operating system software (space partitioning), and prevents failure of a hardware element, unique to an application, from affecting another application.[6.1]

An objective of this design is a capability to upgrade systems and to add new functions through on-board software loading of either revised application programs or new ones.

This is an important capability for our three mission goal; problems encountered in the first mission can be corrected easily and mission variations can be accomplished through revised application programs.

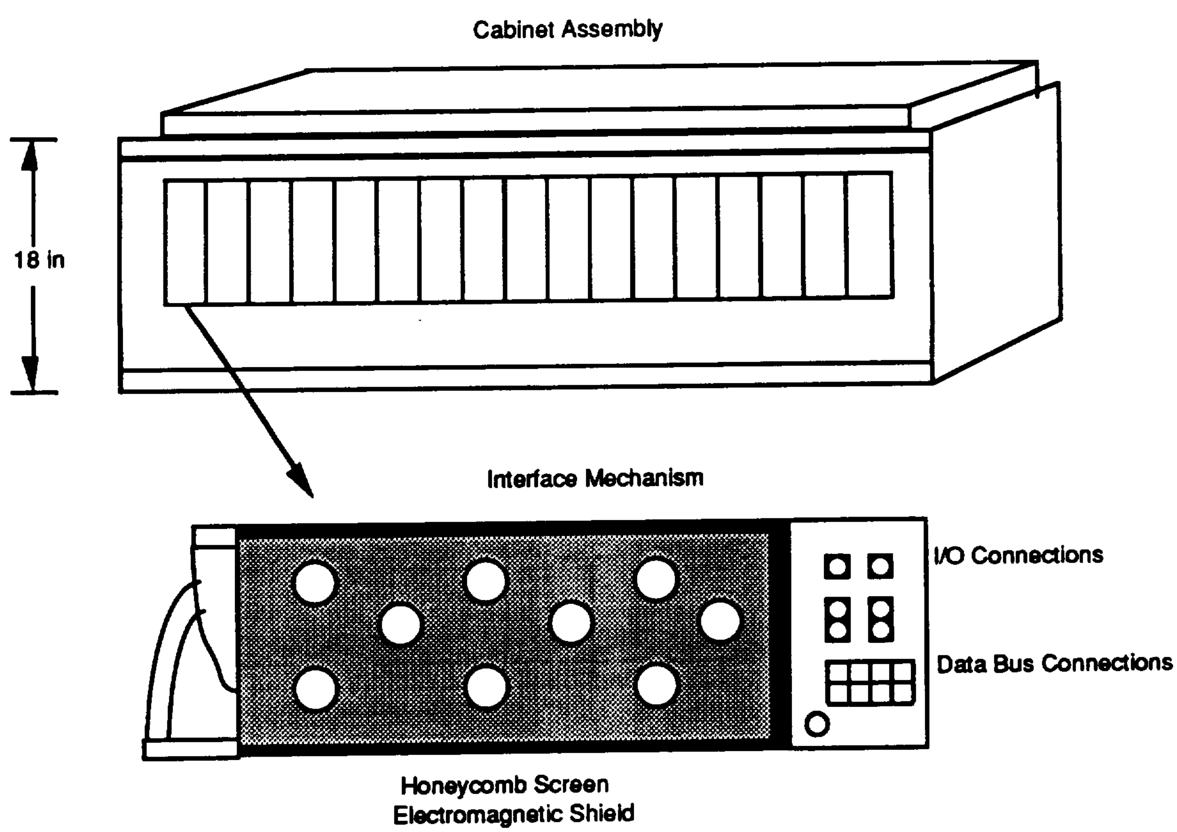



**Figure 6.1.1**: Integrated Modular Avionics package

#### 6.1.1.1 Interfaces

The avionics cabinets are interfaced to the avionics hardware by a data bus. The requirements on the bus, met by current technology (AIRINC 629 Data Bus) are a bi-directional data system with capabilities of transfering data at 2 Mbps. The bus is intended for use in all data functions, from the most important to the trivial. This connection forms an integrated system for performing all avionics functions of the vehicle. The hardware acts as an electrical interface, while the spacecraft functions and reactions are implemented in the software modules.[6.1]

A minimum of two backplane buses are used to transfer data between modules internal to the avionics cabinet. SAFEbus™ backplane bus is an evolving technology created by Honeywell. SAFEbus™ is being developed because no existing protocol adequately meets all key requirements of a bus in the IMA system (i.e.: time determinism, memory production, high performance, high efficiency, low pin count and fault tolerance). Each bus must be independently monitored on the basis of self test and status of data received over the bus.[6.1]

#### 6.1.1.2 Location

Avionics equipment may be distributed over a variety of locations. Location should be based on radiation protection (using spot shielding could de-emphasize this requirement), functional and operational performance, environment concerns, access for maintenance, integration with other systems, length and number of wire runs and number of interconnections within the source systems. Copies of the software modules will be stored in the storm shelter in the event of radiation contamination through the cabinet shielding.

#### 6.1.1.3 Avionics Maintenance

For a trip of this duration (520 days), centralized maintenance functions are necessary for all spacecraft avionics. The function shall provide fault detection, fault isolation, and fault containment. Faults should be identified before they are allowed to propagate to serviceable components. A central maintenance computer (CMC), is required to perform the maintenance functions. One function of the avionics components shall be to report their status, performance details, and faults during the operation of the spacecraft. All fault data will be saved and then down-linked to earth to clear accumulated data for the duration of the trip. If a critical failure does occur, the information will be immediately communicated to earth and any necessary action will be taken upon consultation. There will also be displays on the flight deck that can be used by the flight crew to determine the operational status of selected avionics equipment. A database of maintenance information, Electronic Library System (ELS), will be used with the IMA. This database includes troubleshooting tools, maintenance records, and schematic diagrams and should extend the ability of the spacecraft to be serviceable throughout the trip.

#### 6.1.1.4 Control Panel Displays

Liquid Crystal Displays (LCD) are the current technology with application potential to the MTS. The LCD display medium has significant advantages over cathode ray tubes (CRT) currently used in cockpits. Some of the advantages that apply to the MTS are: smaller volume, lower mass, increased reliability and readability. Immediate applications to the MEV and MTV exist for multi-functional control display units; electronic instrument displays, ELS displays for avionic and flight navigation management systems.

#### 6.1.1.5 Data Storage

The MTV main computer system will function with the aid of a powerful memory subprocessor dedicated to data storage and retrival. This subprocessor will utilize three types of data storage: random access memory (RAM), read only memory (ROM), and write once/read many times (WORM) formats. All data collected on the mission, ranging from ship diagnostics to physical records of the crew to experimental data, will be recorded on the WORM memory and down-linked to Earth via the Deep Space Network (DSN) for backup and analysis. The data storage on either end provides a backup in case of either one's failure. RAM will be provided by standard memory chips but ROM and WORM data will be stored and recalled with an optical disk system. Although this technology has not been qualified for space applications, it is currently used in the military jet aircraft and tanks. It is very rugged and can withstand large vibrational amplitudes and radiation doses. The memory subprocessor is located in the storm shelter to maximize its protection form GCRs and solar radiation. Using a bank of ten read/write units, each containing ten disks of 2Gb capacity each, the subprocessor should be able to record all of the mission data for later analysis. This amount may increase in the future as the technology improves. It is imperitive that all necessary mission information is contained on the ship in case of communications failure with the Earth. This system should provide that assurance.

### 6.1.2 Vehicle Attitude Control

The space environment for the MTS is divided into three regions where different forces dominate. Close to the surface of the planets Earth and Mars, aerodynamic torque will be the predominant consideration. As the vehicles leave the atmosphere of these planets, the aerodynamic forces will fall off and the magnetic and gravimetric torques will take over as the primary perturbation forces. Finally, the torque due to solar radiation caused by not only the induced radiation pressures, but also by the differential heating of the spacecraft, will dominate during the transit phases between the planets. Inertial torques also will affect the the attitude of the MTS. Inertial torques can result from the redistribution of fuel and crew activity. Torques due to micrometeorites are very small, even relative to crew activity, and are therefore considered negligible. Because torques exist throughout the spacecraft environment, there is the need for attitude stabilization and control. The MTS will be stabilized in two ways throughout the mission: spin stabilization and an active Reaction Control System (RCS)

#### 6.1.2.1 Reaction Control System

The most common form of passive control techniques is spin stabilization, in which the entire spacecraft is rotated so that its angular momentum vector remains approximately fixed in inertial space. With the exception of gravity-gradient stabilization, which is the form of stabilization for the MTV in Mars orbit, passive control normally requires the use of "active" control systems, such as mass expulsion or magnetic coils, to periodically adjust the spacecraft attitude and spin rate to counteract disturbance torques and to maintain constant artificial gravity effects for the crew. The MTS also requires a form of nutation damping to eliminate nutation caused by an unbalanced spacecraft or the elasticity of the spacecraft structure.

The form of active attitude control for the MTV is an active RCS based on a mass expulsion control system used for attitude maneuvering. Propulsive systems such as this are efficient in the execution of maneuvers, are simple to operate, and are not limited to a specific environment. However, they are expensive, require complex hardware and plumbing, and are limited in lifetime by the amount of fuel on-board. Propulsive attitude control systems can also be used to make orbit changes during a reorientation maneuver. Consequentially, the thrusters are usually fired in pairs to minimize translational motion. Gas thruster systems are commonly used with spin-stabilized spacecraft for attitude maneuvering and spin rate control, as will be the case for the MTS. For this type of spacecraft, a minimum of two reorientation thrusters and two spin rate control thrusters are required. For a three-axis stabilized system, six possible directions are available for maneuvering the spacecraft and a minimum of six thrusters are required. The other vehicles will use an RCS along these lines but sized appropriately for individual control needs.

#### 6.1.2.2 Attitude Sensors

The reaction control system aboard the MTV requires sensors to determine the vehicle's orientation relative to some fixed body. The attitude sensors that will be used on this mission will be an Inertial Measurement Unit (IMU) , and a Star Mapper.

The IMU consists of six accelerometers to measure velocity and six fiber optic gyroscopes to measure the vehicle heading. These accelerometers and gyros will be arranged in pairs along each axis as a check against a system failure as well as to insure fault tolerance.

The Star Mapper is a device with an optical system to focus the star field image on a solid-state detector array arranged in a slit pattern. The optics of the Star Mapper will be mounted to the MTS body whose rotational motion causes the star field image to sweep across the focal plane. As the individual stars cross the detectors, they generate electrical pulses that are processed to obtain the attitude information that is desired. This information is compared to the star field image contained in the guidance software of the IMA computer.

The inertial and stellar attitude sensors complement each other as well. The main drawback of the inertial guidance system is that it depends on integrating small changes in the attitude of the body to propagate the orientation in inertial space from some known initial value. Therefore small errors will accumulate and periodic updates based on an external source are required to maintain system accuracy. Conversely, the effective use of the Star Mapper will require an accurate knowledge of the initial attitude of the spacecraft on which to base its derived movements. As the MTS undergoes a course correction, the gyroscopes provide an accurate measure of the change in orientation and thus previous attitude information is refined using the new data to determine body orientation. The refined information is then used to update the gyroscopes to minimize the accumulated errors.

### 6.1.3 Guidance and Navigation

#### 6.1.3.1 MTV Navigation

MTV navigation will be accomplished with data supplied by the avionics navigation software. The software will be loaded into the IMA system shortly before the launch of the MTV so last minute delays in the launch time and date can be corrected or accounted for. The IMA will take input from the Inertial Measurement Unit and Star Mapper to record course deviations. At approximately the midpoint of the journey and again before aerobraking, the RCS will perform a course correction on the MTV. This maneuver will be executed on the consultation of the IMA with verification by both the inertial and stellar attitude determination system. The TEI portion of the trip will be navigated by the IMA based on the approximate date of departure from Mars. At least two course corrections will be executed as well for this portion of the trip.

#### 6.1.3.2 Landing on Mars

The landing of the MHM and the MEV on the surface poses a difficult challenge for the guidance, navigation, and control subsystems of the MTS avionics. Even on Earth, Global Positioning Satellites have measurement errors on the range of 5-20 meters (this is with the benefit of gravitational field mapping). The most accurate and simple landing scenario would involve three surface beacons to triangulate the approximate landing site. Landing the ascent /descent vehicle and MHM via triangulation will be very accurate (even to the extreme of pin point accuracy given unlimited fuel). The challenge now becomes landing the beacons properly around the projected landing site. This can be done using satellite triangulation. The advantage of using satellite triangulation to land the beacons rather than the ascent/descent vehicle and MHM is that the beacons are much lighter and therefore react more quickly and require less propellant than the 30 mT ascent/descent vehicle and the 70 mT MHM. If there are any errors in the landing of the beacons, their exact location will be known by the satellites and any triangulation corrections can be accounted for through the avionics computers on board the vehicles.

## 6.2 Power Systems

There are two items of importance that need be recognized before the power systems for this mission are defined. The first is that no power requirements, specific to our MTS vehicles, were developed by the other disciplines. Therefore, system capacity is based on the figures given by the various disciplines involved with the MTV and MEV vehicles. In addition to this fact, the design of the systems for each vehicle was dependent upon several parameters. These include variable power output, system redundancy, the safety inherent in the systems from both a crew as well as an electronics standpoint, efficiency of operation, and the maximum capacity versus mass of the systems in question.

To supply each vehicle with the necessary power, three main power production plants were considered. The systems chosen were due to the fact that they met at least four of the above five parameters that are set in the preceding paragraph. The first of these is a nuclear power plant, the second is the Dynamic Isotope Power System (DIPS), and the third is a Hydrogen-Oxygen fuel cell. Each of these will be detailed in the following section.

### 6.2.1 Nuclear Reactor Power

Nuclear production of electricity is based on the principles of nuclear fission in which high speed neutron particles collide with the relatively heavy hydrogen atoms. This collision splits the atom and releases large quantities of energy. This energy is then converted to electricity by employing Sterling-cycle convertors. This type of power source has several outstanding qualities. Among these is the ability to produce very large quantities of energy. The nuclear power source is rated at producing 30 kW per metric ton. This, coupled with the fact that it is a long term means of energy production shows that nuclear power has much to offer a mission of this type. Another benefit to employing a nuclear power source deals with the fact that this reaction can be halted, and restarted at any time. The ramifications of this are apparent if the need arises for repairs, or if the system needs to be put in storage for any portion of the mission. The major drawback to this system is that once the reaction is initiated, substantial radiation is produced. Since this will be detrimental to both the astronauts and to the computer components, dealing with a nuclear power source will necessitate radiation protection in some form.

### 6.2.2 Dynamic Isotope Power System

DIPS is based on utilizing the heat generated by the decay of radioactive materials. This heat is transferred to a working fluid that flows through a turbine which powers an alternator and a compressor to create electricity. This system is self contained, and is a very simple means of power production. This power source is capable of producing energy on the order of years and is very efficient. One drawback to this system is that it produces large amounts of radiation that can be hazardous to the crew and to the computer systems. Also, unlike the nuclear reactor, this radiation cannot be halted. The decay of the isotopes that produce energy will continue for the duration of the mission and could cause handling problems and storage problems as well. Also, the power production for these units are not extremely high at only 8.57 kW per metric ton.

### 6.2.3 Hydrogen-oxygen Fuel Cells

This power production system is based on the chemical reaction between hydrogen and oxygen. In such a reaction, energy is given off which can be harnessed for the powering of a mechanical unit. The by-products of this chemical reaction are pure water and heat. This system of power production is very appealing to the mission to Mars due to the fact that its levels of energy production are very high, producing 30 kW/mT. Additonally, there are no harmful emissions such as in the case of both the nuclear power source and the dynamic isotope power source. Thus, the safety of the crew and the proper functioning of the computer components would not be an issue if this source of energy were employed. There is, however, one significant drawback. The life of this system is relatively short, on the order of days. With this mission profile, it is easy to see that fuel cell power will play only a small role in the production of energy. The demand for a long lasting source of power dictates this reasoning.

**6.2.4 MTV power system**

The first vehicle system design considered was the MTV. A specific problem encountered with this particular system is the incredibly long life that the system must possess. In order for fuel cells to achieve this length of power supply, the amount of fuel required would become phenomenal. In fact, the mass of the fuel alone would approach 108 mT. This consideration alone dictated that this system was not feasible for the power source aboard the MTV.

The second proposal was based on using the NTR's nuclear reactor as a heat source, with a back up power plant consisting of the above described fuel cells. Three NTRs would need to be utilized, so the idea of using the already-in-place reactors seemed to be a feasible choice. In addition to the simplicity of the design brought about by the utilization of existing units aboard the MTS, the system was practical in terms of its high power output as well as its reasonable mass. Upon further inspection, however, it was determined that the design of the cooling system became very complicated with the use of the NTR's, For this reason research into simpler design was initiated.

The solution to the question of MTV power generation was the Dynamic Isotope Power System (DIPS).

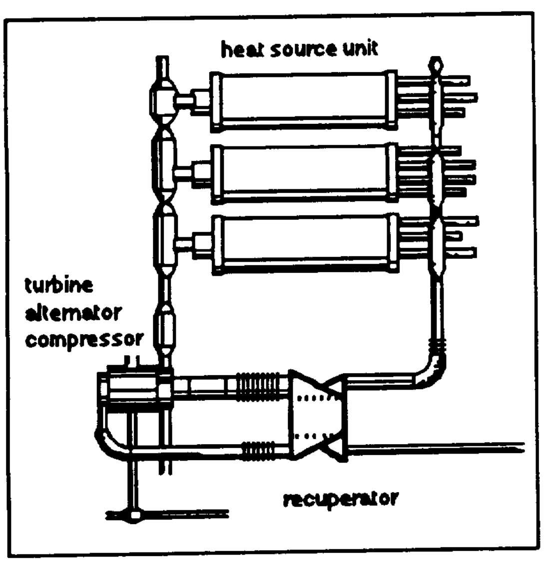



**Figure 6.2.1:** DIPS configuration

The MTV power system will utilize three separate DIPS systems to provide an adequate level of safety. Two of the DIPS units producing energy will comprise the normal system of power supply to the MTV. Each of these systems has a cylindrical volume 2 meters high by 1 meter in diameter. With two of these units on line, the power consumption of 19.86 kW that is the MTv's allotment will be provided. This allows the third DIPS unit to remain untouched for contingency power or as a redundancy factor should one of the on-line units fail. In addition to the normal mode of operation, these three units will provide more than enough safety due to the fact that one DIPS unit operating alone will provide adequate power to sustain the life of the crew as well as the functions of the necessary computer systems in case of an emergency.

The down-side to this seemingly ideal system of power production is that it does produce a significant amount of radiation. Therefore, the DIPS heat source will need to be shielded from the rest of the system as well as the crew. The easiest, and lightest way to accomplish this is to place them next to the NTRs to take advantage of the same shielding. The rest of the system should be mounted as close as practical to prevent heat loss before the fluid enters the turbine. With the output of each DIPS unit at 10 Kw, the total power output for all three of these units operating simultaneously is 30 Kw. The estimated mass for the MTV power system is 3.5 mT (7700 lbs.).

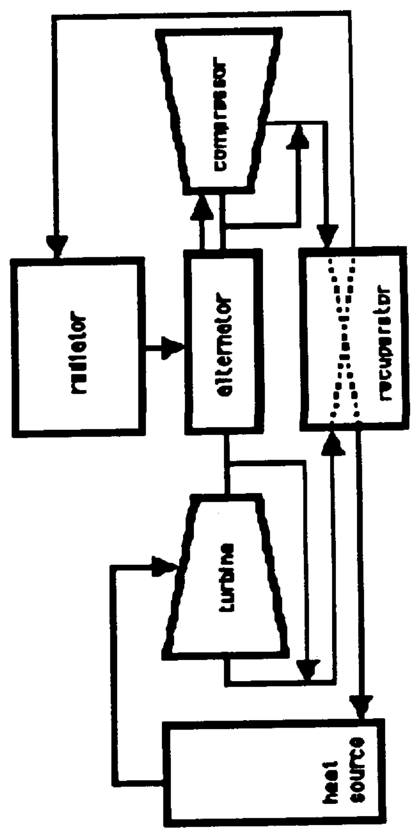



**Figure 6.2.2:** Schematic of DIPS cycle

### 6.2.5 Mars Ascent/Descent Vehicle

The Mars Ascent/Descent vehicle was the next topic in this discipline for the design of power systems. The main consideration in this case was system mass. In order to achieve a safe landing at the outset of the mission, and also a safe return from the planet's surface upon mission completion, a light-weight vehicle is of paramount importance. The main benefit of a light vehicle is in its response to attitude controls. If a light vehicle is being maneuvered, the reaction to thrusts is much more precise, and allows for much quicker reaction time in the event of an emergency situation. Hydrogen-oxygen fuel cells not only provide the ascent/descent vehicle with a high level of controllability, but it also allows for vehicle compactness. Because the only by-products of these fuel cells are heat and water, the placement of its units is of little consequence and does not require massive shielding to protect computers and crew. At an estimated power level of 15 Kw the mass of the system is 0.5 mT (1100 lbs). A breakdown of the mass is in the following table.

| | |
|---|---|
| Fuel Cell Unit | 125 kg |
| $LO_2$ | 124 kg |
| $LH_2$ | 16 kg |
| Fuel Tanks | 75 kg |
| Power Conditioning | 100 kg |
| Wire,Switches,etc | 60 kg |
| Total | 500 kg |

**Table 6.2.1:** Ascent/ descent vehicle power system mass

### 6.2.6 Earth Return Vehicle

The Earth Return Vehicle has the exact same consideration as the above ascent-descent vehicle. The only difference from the power standpoint is that the Earth Return Vehicle will need power sustained for a longer period of time. Therefore, this same design as the ascent descent will be implemented with the only modifications in the quantity of fuel carried.

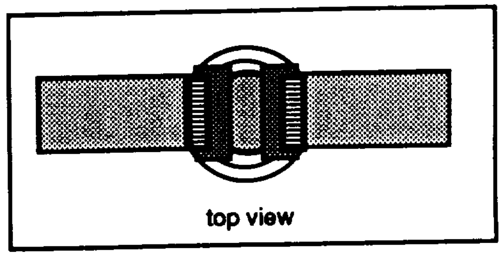
### 6.2.7 Mars Habitation Module

The Mars Habitation Module (MHM) was the last system requiring a power supply. Once again, the amount of time that this system needs to remain in operation creates a problem for the Hydrogen-Oxygen fuel cell system. Therefore, this particular system was discarded as a possibility. Also, the DIPS system was decidedly impractical due to the fact that the radiation it will be producing cannot be shut down while in transit to Mars and would thus add significantly to our radiation problem. The only remaining system capable of powering the MHM was a nuclear reactor. The output of this system is a maximum of 100 kW with a system mass of approximately 3 mT (6600 lbs.).

This system will utilize two Sterling-cycle converters to process the heat from the reactor into useable electric power. The dimensions of this system are as given in Table 6.2.2.

| | Stored | Deployed |
|---|---|---|
| Height | 3.0 meters | 3.0 meters |
| Width | 2.0 meters | 7.0 meters |
| Depth | 3.0 meters | 3.0 meters |

**Table 6.2.2:** Dimensions of nuclear reactor



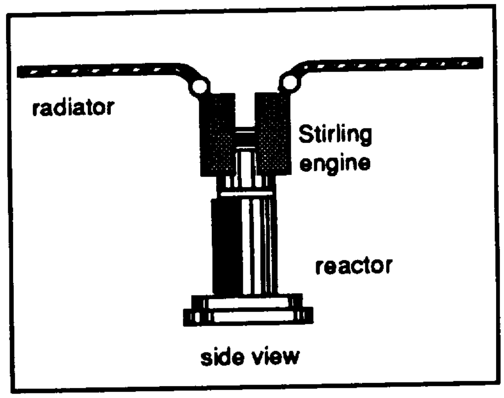




**Figure 6.2.3:** Nuclear power plant configuration

This system will be inoperable during the transit to the planet's surface, and will be stowed with its radiating fins folded to its sides which will allow for much simpler packing and much more efficient use of space in the biconic lander. Once it is on the surface of the planet Mars, the reactor will be towed away from the site and will be put on line. This will effectively reduce the astronaut's exposure to the deadly radiation, but will necessitate an alternate power source that will operate until the reactor is set up and producing power.

To supply power while the nuclear reactor is being assembled, and for powering the avionics during entry into the Martian atmosphere, hydrogen - oxygen fuel cells will be used. The fuel cell will also serve as a backup system in case of main system failure. The fuel cell will have a preliminary power output of 20 kW and a life of 120 hours without regeneration. The preliminary mass estimate for the fuel cells is approximately 1 mT (2200 lbs.). Together the estimated mass of the MHM power system is estimated at 4.5 mT (9900 lbs.).

## 6.3 Communications

### 6.3.1 Antenna Configuration

In the initial stages of planning for the required mission's communications, there were several topics of concern. The first task in the analysis of communications was to establish the parameters of the communications antennas that would be required to transmit data not only from the surface of Mars, but also from any point along the astronauts' transit from Earth to Mars and back again. Some of these factors include the size of the antenna, the power needed to achieve communications, the maximum data rate figures, the directionality of the communications links, and the desired gain of a deep space antenna system. It must be realized that all of these antenna characteristics are dependant upon one another. (Figure 6.3.1)

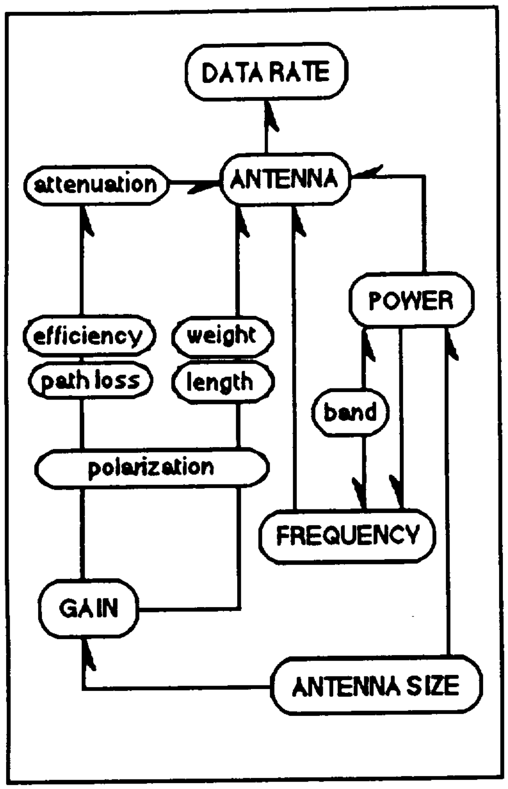



**Figure 6.3.1:** Data rate factors

As can easily be seen, some of these factors are directly dependant, while others vary indirectly. In any case, the alteration of one of these will ultimately change the requirements for all others. Not only does this make it difficult to analyze each of the parameters separately, but it also creates a very complicated engineering problem when the maximization of efficiency and the minimization of mass enters into the picture.

### 6.3.2 Data Requirements

As can be seen from Table 6.3.1, the data rates differ significantly. For example, the data rate required to send high rate video (comparable to normal television picture) is equal to 100 Mbps. To contrast this, only 0.02 Mbps are required in order to send a voice signal. The choice for determining which of these data would be necessary to send during the mission to Mars was among the first decisions to be made.

| DATA TYPE | DATA RATES |
|---|---|
| high rate video | 100.00 Mbps |
| low rate video | 1.00 Mbps |
| voice links | 0.02 Mbps |
| science | 200.00 Mbps |
| command links | 1.00 Mbps |

**Table 6.3.1:** Data rates

It was decided that 100 Mbps would be the target range for the MTS communication rate. From Table 6.3.1, it can be seen that this rate will enable high rate video to be sent in real time to the viewers on Earth. This does not mean, however, that scientific data will not be capable of transmission on the communications link. This information will only need to be compressed and deciphered at a later time which will slow the entire process considerably.

### 6.3.3 Signal Frequency

The signal of electromagnetic radiation, and its characteristics are directly dependant on the frequency that is chosen to carry the information. There are two relationships that must be balanced in this particular area when concerned with long range transmission. The first is the way in which the frequency affects the data rate, and the second is the way in which the frequency affects the attenuation of the radiation beam. In this case, as with most of the other signal characteristics, there is a trade off. This arises due to the fact that if the frequency is very small, there is very little attenuation of the signal. However, the lower the frequency, the less information can be sent. The decisive factor in this choice is that although attenuation is a problem in a heavy atmosphere, it can be neglected when operating in the vacuum of space. Therefore, the most beneficial solution is to employ a high frequency to take advantage of the large capacity for data transmission. The chosen frequency for the mission to Mars is 32.2 GHz, which is within the Ka-band of the electromagnetic spectrum. The utilization of this particular frequency will allow for approximately 5 times the data rates that have been achieved previously in NASA missions and will subsequently bring the goal of 100 Mbps easily within the reach of the present day technology.

### 6.3.4 Antenna Size

The antenna size is perhaps the most obvious of the parameters affecting the performance of the communications system. In order to determine this particular variable, the information learned from the data rate requirements, as well as the benefits of the chosen signal frequency needed to be taken into account. The distances that the signal must span also become a very significant factor in the determination of antenna sizing. In order to transmit an intelligible signal over a distance on the order of 200 billion meters (approximate distance from Mars to Earth), the smallest possible antenna able to be employed will be seven square meters. This size, however, will not satisfy the safety factor that has been set which dictates a minimal amount of transmission error. With the seven square meter system, error could be disrupted easily by slight solar activity, or by non-ideal atmospheric conditions at Earth and would in effect cripple the Mars-Earth communications link. Therefore, an antenna of sixteen square meters was chosen to be the primary link from the MTS to ensure intelligible communications. While this may seem to be an enormous jump in the size of the antenna, it would only require the enlarging of a three meter circular dish to a diameter of 4.5 meters. Other benefits that are associated with this increase in dish size are a reduction in the amount of power required, a decrease in required directional accuracy, increased gain of the system, and decreased difficulty in attaining the data rate goal.

### 6.3.5 Antenna Gain

The gain of any directional antenna gives a method of comparing the boresight axis power to that of an ideal isotropic radiator in which power is distributed evenly in all directions. Due to the relatively small distance, and therefore the small gain requirement, most satellites in Earth orbit are able to employ omni-directional antennas. However, in deep space this changes dramatically. Due to the long distances, highly directional antenna systems are needed to concentrate the energy along the chosen path. This becomes more and more important as the transmission distances increase since the signal will dissipate with the square of the increase in distance. To achieve a suitably understandable signal at Earth, this gain must be very large indeed.

#### 6.3.5.1 Paraboloid of Revolution

In the past, the most common directional antenna was the paraboloid of revolution. This type of antenna will generate a signal at the focal point of the parabola, thus creating parallel waves of energy concentrated along the boresight axis.

As can be easily seen , this method will definitely produce a certain concentration of energy in the predetermined direction. In the case of this type of system, the gain is directly dependant on the size of the antenna. It is reasonable to generalize the mass of this type of system as well. In fact, as the diameter of the antenna increases linearly, its mass increases quadratically. In addition to the mass of the antenna we must include the mass of the mounts and the driver, which will be well over twice the mass of the associated antenna.

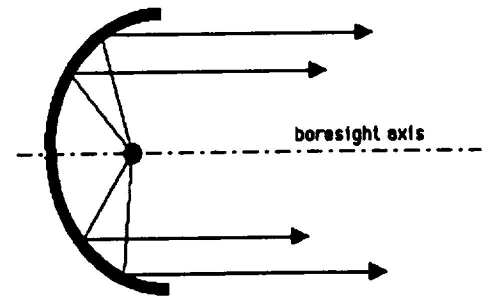



**Figure 6.3.2:** Paraboloid of revolution

#### 6.3.5.2 Phased Array Gain

The next obvious step was to compare the gains of the two available antenna systems. The equations for the measurement of each of their gains was determined and graphed versus antenna size. It was soon realized that the gain for a broadside array has at least the projected gain of an equivalently sized parabolic dish (see Figure 6.3.3). In addition, this comparison dramatically favors the broadside array configuration as the size of the antenna increases. Simply stated, the directivity of signal energy for the broadside array will be greater than that of the dish antenna. It will therefore be capable of sending more data with greater accuracy than the conventional system. With the requirements for high data transfer rates and with the need for transmission over large distances this increase in system gain would greatly enhance this crew's ability to achieve the given requirements.

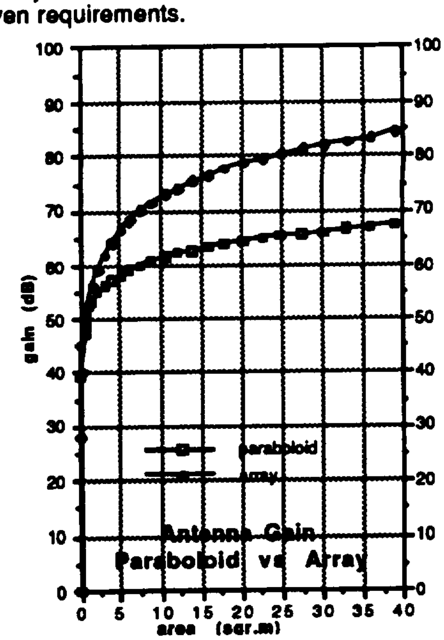



**Figure 6.3.3:** Gain comparison

### 6.3.6 Antenna Pointing

In addition to the increased gain of the broadside array, there is another benefit of this antenna configuration that can be exploited. The fact that the array antenna's communication beam is directed electronically, and not mechanically becomes a definite advantage. One of the requirements of the MTS is that it must employ artificial gravity for at least the out-bound leg of the journey. To simulate gravity, the MTS must be spinning in the plane of its velocity vector which will cause many difficulties in the pointing of the communications beam. The cumbersome mechanical units would not only be required to continually spin the antenna counter to the gravity-producing motion, but they would need to constantly alter the angle of the antenna to ensure its tracking of the planet Earth. With the phased array, the mechanical failures cannot occur and the directional alterations of the communications signal will be instantaneously produced.

### 6.3.7 Phased Array Antenna

The principle of the electronic beam steering is based on the electromagnetic radiation theory of constructive and destructive interference. With electromagnetic elements aligned and radiating in phase the constructive interference mandates the maximum energy to be situated along the array boresight axis while the destructive interference eliminates any energy transmission in other directions. The result of these two phenomena is an incredibly powerful wave that is aligned along the boresight of the array and has its wave front aligned perpendicular to the axis of the element array.

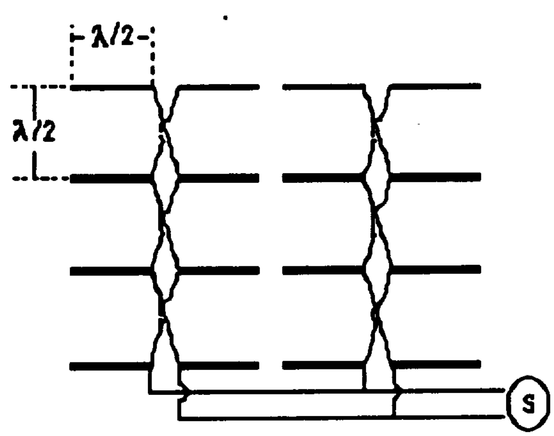



**Figure 6.3.4:** Broadside array elements

To achieve a signal transmission off the boresight axis, the elements need only be radiated slightly out of phase with one another. The result would be to force the radiating energy at an angle off the boresight. This angle, measured relative to the boresight axis is directly related to the phase shift between each of the adjacent elements.

### 6.3.8 Array Specifications

To put the finishing touches on the proposed communications array, there are several items that need mentioning. These deal with the specifics of the system, its redundancy, and several other considerations.

In further study into phased-array criteria, it has been determined that a beamwidth of one degree is easily possible with an array consisting of a grid of 10,000 radiating elements. The arrangement of such an array will be a square of 100 elements on a side with each side measuring just under 4.0 meters. Although this may seem like an extraordinary amount of elements, there are several points that will justify this number. The first reason is that, while a lesser amount of elements could be employed without jeopardizing the beamwidth of the directed energy, the gain of the system would diminish and thus the benefit of the phased-array system would be decreased. Subsequently, the resulting side lobes would be significantly larger. Another point is that this proposed system has much inherent redundancy.

It is extremely likely that during the mission to Mars, the vehicle will experience the failure of many of these radiating elements. The trouble may be caused by micrometeors, radiation, or simply by the fact that these elements have a projected life of 10,000 hours of continuous use. In any case, this array can lose a substantial number of elements before falling below the set performance criteria. As stated above, the beamwidth will remain unaffected, it is the gain that will be sacrificed. Also, the directional error will be affected by the number of elements; beam instability will increase as the number of elements is decreased. With the proposed 10,000 elements, the directional error should be less than 0.001 degrees.

The final argument for the proposed configuration deals with the ability to steer the broadside array electromagnetic beam. The array principle of beam steering is directly dependant upon the number of radiating elements such that the more elements present, the greater the possible angular deflection of the array. The proposed design will enable communication at 45 degrees of latitude off the boresight axis in each direction while maintaining a very minimal amount of directional error and power loss.

### 6.3.9 Array Simplification

One of the larger mass additions that will be associated with the phased array system deals with the controllers of the individual elements. In a three-dimensional coverage scenario, there will need to be a controller for each element present to ensure the maximization of the array capabilities. Considering the proposed system, this would dictate the need for 10,000 controlling systems for the operation of the array. The

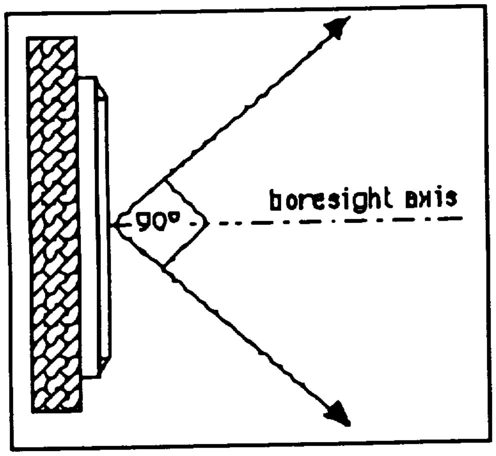



**Figure 6.3.5:** Llimits of directional steering

chance for mass savings would be very great if two-dimensional coverage would be sufficient for constant communication during the mission. If this is the case, the number of controllers dramatically decreases to 198. Thus, the simplification of the MTS communications array deals with the fact that the mission to Mars may only require the two- dimensional coverage noted above. The reason for this is that during the transit from Earth to Mars and back to Earth from Mars, the MTS will always be in the plane of one of the planet's orbits. As can be seen, not only does this promote the technical simplification of the array, but it also provides a significant mass savings over the three- dimensional array. While this mass savings would be desirable, the mass is not unreasonable at all even given the three dimensional coverage. The mass of the array has been calculated at 1.0 mT and the computer controls system accompanying the hardware comes to a total of 0.1 mT.

**6.3.10 Array Placement**

The location of the array is really very arbitrary except for one requirement. The maximum possible directional change that the communications link can be steered through is 50.0 degrees to each side of the boresight axis. The only requirement, therefore, is to ensure that the energy beam is not obstructed in any way by such MTS components as the aerobrake, the MTS habitation module, or the biconic. Dictated by this limitation, the closest that the communications array should be to these items will be set at 10.0 meters. The only other consideration that has been discussed is that the the array should be close to the crew to allow for easy access in the event that repairs or element replacement becomes necessary.

**6.3.11 Contingency Planning**

In addition to the array, it has been decided that an emergency parabolic antenna will be stowed on board the MTS in the unlikely event that the broadside array system fails during the mission. This emergency communications dish will measure 3.0 meters in diameter and be capable of transversing clearly one half the furthest distance from Earth to Mars. This may not seem adequate, however, it must be remembered that there will be relay satellites at both Earth and Mars that will be capable of a much delayed, but discernible round trip signal transmission. The stowing of this system will again be arbitrary, however, it must be accessible during the transit stages of the mission.

**6.3.12 Mars Satellite Relay System**

One of the other concerns that was introduced to the Avionics and Power discipline was the necessity for full-time coverage when the astronauts were on the Martian surface. With only the MHM as a transmitting source, there would be a twelve to fourteen hour "hole" in the communications each day. This lack of transmission time was not sufficient for the communications needs of the mission. Therefore, the use of a satellite relay station(s) on or around Mars became necessary. It was determined that the use of a three satellite communications support network would increase the communications availability at Mars to the 90% level. The Martian Satellite Relay System (MSRS) will consist of two units orbiting Mars aero-synchronously. These two "birds", in addition to the orbiting MTV will be sufficient to provide the coverage described above. Because communication satellite technology has been rapidly improving over the past few years, the task of designing the MSRS very simple indeed. The technology needed for the MSRS should be able to be purchased on a commercial level. This point will be adequately proven in the following two sections.

**6.3.12.1 GSTAR**

The GSTAR is a Ku-band satellite developed by the GTE Corporation. As of 1987, technology provided 60 megabits per second per transponder, of which it has sixteen. To give an idea of the data capacity this achieves, an operational GSTAR could handle 30,000 simultaneous phone conversations, 300 two way video conferences, or a combination of both. The satellite weighs upwards of 2000 kg. The limiting factor seems to be the station-keeping fuel supply which maintains its 0.1 to 0.05 degree synchronous orbit. The design requirements that have been introduced into the satellite system orbiting the planet Mars are far below the numbers that are seen in the GSTAR. Therefore, a similar satellite could be employed with several alterations. First, the number of transponders will need to be decreased to fit the needs of the communications network. Four transponders would be more than capable of handling the data load that has been foreseen during this mission. Secondly, the size of the antenna would need to be modified in order to achieve the high gains that are necessary to penetrate the large distances form Mars to Earth with a suitably error free transmission.

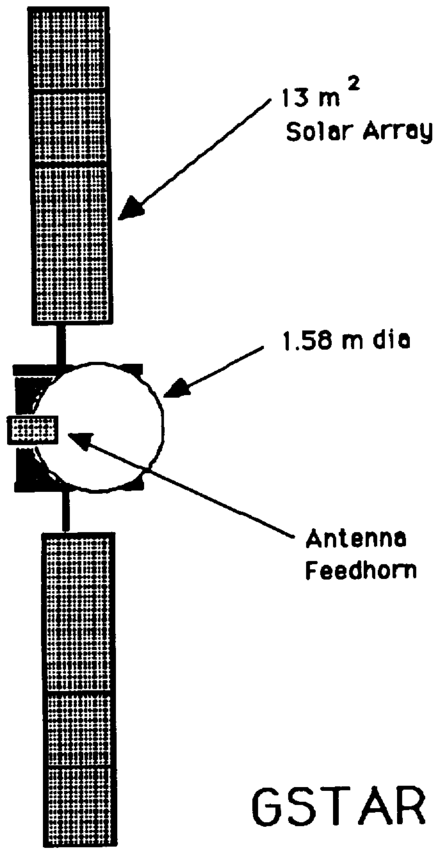



**Figure 6.3.6:** GSTAR satellite

#### 6.3.12.2 TELSTAR III

The TELSTAR III communications satellites now used by AT&T operate in the C-band range and would need to be modified to fit our design requirements, although the design is very compact and efficient. Each satellite posses 24 transponders that may be switched by ground command for various regional coverage. Each also has a capacity equivalent to 21,600 voice conversions in addition to high speed data services. Dimensions of this system are 2 m diameter by 2.5 m length when stowed for launch and 2 m diameter by 7 m when fully deployed. The TELSTAR III produces in upwards of 915 w and retains excellent station keeping performance despite its larger weight.

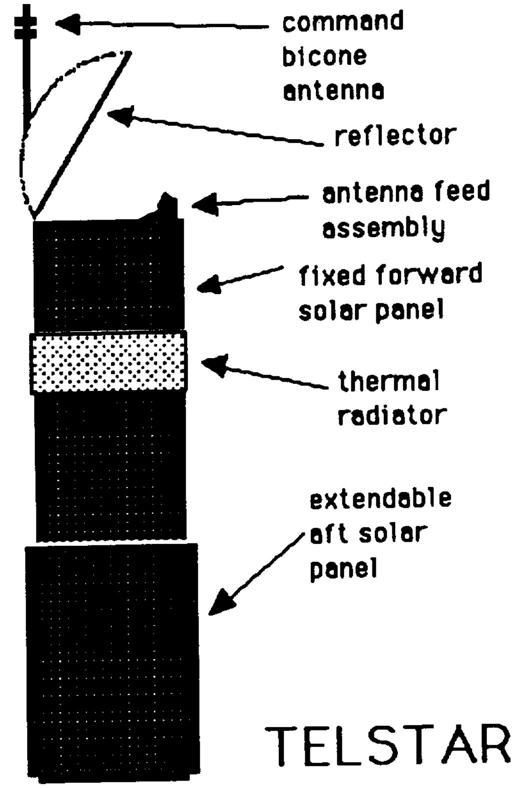



**Figure 6.3.7:** TELSTAR III satellite

These two examples show that the satellites required at Mars are already technologically available. Further study and research into more up-to-date technologies is needed, but at the time of this report none has been found that deal with the higher frequencies (Ka-band) that will be employed in Mars mission scenario. However, the actual model that will orbit Mars at the time of the mission will simply be an alteration from either of the two outlined above. The Mars aero-synchronous system will need about one-one-hundredth of the channel capacity mentioned above. But this serves only to increase the life of the satellites due to the savings in power requirement. Also, consideration must be given to the addition of alternate satellite capabilities such as micrometeorite detection capabilities, tracking systems, solar flare monitoring sensors, and terrain mapping functions. Satellite technology for the Mars Satellite Network should be easily available within the time frame of the first Mars launch.

### 6.3.13 Mars Surface Links

The final aspect communications dealt with in this mission analysis was the Mars surface links. One of the benefits of maintaining a communications satellite relay system, it was found, is that it can have many uses outside of those originally intended. In this case, no matter which stage of the mission is involved (or which landing site is being occupied) the two orbiting "birds" in addition to the orbiting MTV will provide full coverage for the surface-to-surface links without the addition of large linking disks or other ponderous systems. In fact, the habitation sites need only employ a 0.5 meter dish to ensure an absolutely clear signal to the orbiting units which in turn can communicate undisturbed with the rovers possessing the same type dish at any distance on the planets surface. For closer explorations, a simple omni-directional antenna will suffice just as it would here on Earth. Even in the case of emergencies, the omni antennas will be able to communicate adequately to ensure satellite-linkage no matter what the situation. As a result, the number of available communications lines are dramatically increased which serves not only to bolster the amount of data that can be sent simultaneously, but at the same time a large amount of redundancy is built into the system to prevent crippling due to one malfunction in the communication lines.

# 7.0 AERODYNAMIC ANALYSIS

## 7.1 Introduction

Perhaps,as with all aspects of space flight, our interest in re-entry aetrodynamics began in 1958 when the Soviet Union launched Sputnik. Hypersonic aerodynamics research groups were formed in NASA, the DoD and in academia. The research done by these groiups forms the backbone of re-enty aerodynamics.

The problem of getting back down from orbit is easily as difficult as the problem of gettin up. The issues of aerodynamic heating, stability and contriol, and the highly nonlinear equations which govern hypersonic speeds lead to serious research problems which must be solved before atmospheric re-entry is attempted.

## 7.2 Entry Dynamics/ Methods

Successful atmospheric entry requires aerodynamic shapes and high strength materials capable of surviving the intense heat of entry, up to 2800° C (5000° F), and loads of the magnitude of 100g's. Furthermore, a braking system, either propulsive or aeroassisted, must be included to ensure acceptable planetary impact.

It is the two vehicle parameters of lift to drag ratio, L/D, and ballistic coefficient, $C_B=m/SC_D$, which determine the type entry path followed. It should be noted the other two important parameters for these equations are the planetary radius, R, and the atmospheric density, r. These of course are constant when we consider Mars, though they will place constraints on the types of MEV's possible.

### 7.2.1 Ballistic Entry

Ballistic entry is the simplest method of entering a planet's atmosphere. By definition, ballistic craft generate no lift force, and are thus easily controlled. Ballistic craft can be delivered with meters of their target, although this assumes a very steep and fast entry path that may produce aerodynamic loads on the order of 100g and hypervelocity impacts with the ground. The large loads are not a problem if the vehicle is unmanned, however, some sort of braking system must be used to ensure the craft will survive impact. Possible braking systems include a reverse propulsion system or parachutes. Ballistic missiles also have very small landing footprint. That is, once they deorbit they are very limited in their cross range maneuvers and thus must be dropped from a very specific orbit to be able to reach their target.

### 7.2.2 Aeroassisted Entry

The basic idea of aerobraking would be to use the Mars atmosphere to slow the motion of the MEV descent to the Mars surface. There are a variety of aerobraking concepts that have been studied. While all the above entry methods use aerobraking to some degree, our definition of aeroassisted entry will be a vehicle similar to the lunar lander that uses a large aerobrake to slow its descent with no other aerodynamic control surfaces.

The MEV Aerodynamics group concentrated on two of these concepts initially: flexible/deployable aerobraking and rigid aerobraking. Of the two concepts, the design of rigid aerobraking configurations represents a more risk free design compared to the alternative. However, the assembly of the rigid aerobrake in-orbit has the potential to be a greater task than the alternative. The main advantage of a flexible/deployable aerobrake is the elimination of on-orbit assembly. However, flexible/deployable aerobrakes are more complicated in design and therefore can be more problematic.

### 7.2.3 Martian atmosphere

For aerodynamic analysis, the Mars atmosphere has an effective altitude of 100 km( 328,000 ft) above the surface. The majority of the information available on the Martian atmosphere is from the Viking missions. Many computer models of Mars have been built. Unfortunately, these predictions are based on "Earth type" assumptions and may not be valid.

Atmospheric mass varies seasonally by up to 25%. This is due to the fact that the Martian atmosphere consists mostly of $CO_2$ that condenses and sublimates. Other major constituents of the atmosphere are CO, $O_2$, $H_2O$, Ar, and $N_2$. In the summer on Mars large amounts of water vapor are present in the northern hemisphere and there is virtually no water vapor in the winter.

| CONSTITUENT | ABUNDANCE(cm atm) |
|---|---|
| $CO_2$ | 780 |
| CO | 5.6 |
| $O_2$ | 10.4 |
| $H_20$ | ~3 (variable) |
| $H_2$ | ~0.4 |
| $O_3$ | ~$10^{-4}$ |
| $N_2$ | < 400 |
| A + inert gas | < 1560 |
| $SO_2$ | < $3 \times 10^{-3}$ |
| $N_2O$ | < 200 |
| $CH_4$ | < 10 |
| $C_2H_4$ | <2 |
| $C_2H_6$ | <1 |
| $NH_3$ | <2 |
| $NO_2$ | < $8 \times 10^{-4}$ |

**Table 7.2.1:** Atmospheric constituents of Mars

Two key factors that must be examined in regard to the Martian atmosphere are the atmospheric waves and the seasonal dust storms. During dust free times, the waves are caused by vertical propagation of energy from the heating of the ground and near surface. This can cause a variation in the temperature and subsequently the density of 10%. During the dust storms, particles are carried aloft in excess of 50 km. Not only do these particles change the mass of the air, but they are heated by solar radiation causing an increase in the air temperature. During a dust storm, the air density can change by 20%.

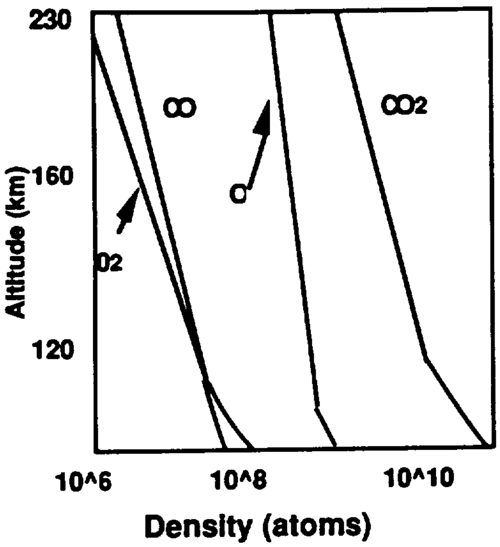



**Figure** 7.2.2: Mars atmospheric density chart

Unfortunately, global data is lacked on the dust storms. If at all possible, they should be avoided during the flight phases of the mission. Although the storms pack winds in excess of 200 km/hr, surface exploration can continue. The winds are of little concern, since the atmospheric pressure on Mars is around 7.5 millibar, as compared with an atmospheric pressure of 1000 millibar on Earth. It is known that regional storms generally occur when Mars is closest to the sun (southern spring and summer). Local storms, carrying particles to altitudes of 30 km and covering an area less than $10^6$ km$^2$occur more often.

The mean temperature throughout the atmosphere is shown in Fig. 7.2.3. Fluctuations in the temperature occur due to seasonal changes and a weather system similar to that on Earth. The temperature gradients that occur between the equator and the poles form jet streams. The jet streams are larger in magnitude than those on Earth, but due to the lower atmospheric densities are not very significant to the vehicle design.In the higher latitudes, the temperature ranges from 145°K during the polar night to over 210°K in the summer. In lower latitudes, the variations are less extreme.

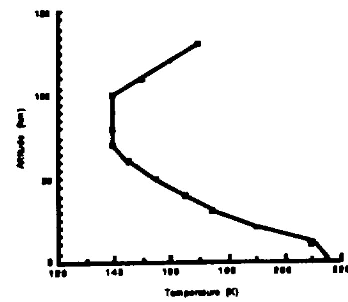


**Figure** 7.2.3:Temperature vs. Altitude for Mars.

### 7.2.4 Earth Atmosphere

The Earth atmosphere is composed predominantly of $O_2$ and $N_2$. Diatomic nitrogen comprises almost 78% of the Earth's atmosphere. Diatomic oxygen is another 21%. Argon composes another 0.9%. The remaining 1% is composed of different trace gases such as carbon dioxide and water vapor.

The atmosphere is divided into 4 basic regions: troposphere, stratosphere, mesosphere, and thermosphere. These divisions are broken down according to temperature gradients. Temperatures decrease in the troposphere, stratosphere with an increase in altitude. While they increase in the mesosphere and thermosphere with an increase in altitude. These layers are caused by selective absorption of solar radiation. At an altitude of 100 km, extreme ultraviolet radiation is absorbed by oxygen atoms. This process keeps the thermosphere temperature elevated. It also produces the ions that composed the ionosphere. These ions often cause transmission problems for spacecraft.

Atmospheric pressure is 1,013.25 millibar. Since the atmosphere is compressible, pressure does not decrease uniformly with an increase in altitude.

Most atmospheric temperature changes from compression and expansion are adiabatic. Thus convection currents, carrying warm air aloft have little overall effect on the atmosphere. Therefore the Earth's atmosphere is considered to be statically stable.

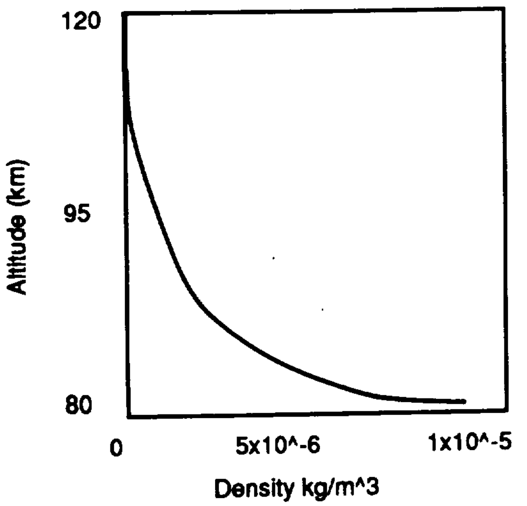



**Figure 7.2.4:** Atmospheric density vs. altitude for Earth

## 7.3 Aerobraking

The basic equations of atmospheric entry which govern all entry vehicles are as follows :

$$\frac{dV}{dt} = \frac{T \cos a - D}{m} - g \sin g$$

$$V \frac{dg}{dt} = \frac{T \sin a + L}{m} - \left(g - \frac{V^2}{r}\right) \cos g$$

$$\frac{ds}{dt} = \frac{R}{r} V \cos g$$

$$\frac{dr}{dt} = \frac{dh}{dt} - V \sin g$$

$$\frac{dm}{dt} = - \dot{m}(t)$$

$$L = \frac{1}{2} r V^2 S C_L$$

$$D = \frac{1}{2} r V^2 S C_D$$

$$g = g_o [R/(R+h)]^2$$

$$a = a(t)$$

**Table 7.3.1:** Aerodynamic entry equations

These equations, subject to the initial values of r, V, and g at the time of de-orbit; the vehicle's ballistic coefficient, $C_B$, and lift to drag ratio, L/D; and a model for the variation of atmospheric density, r, with altitude, completely prescribe the state of vehicle during entry, including velocity, acceleration, and trajectory.

One of the characteristics considered in the aerobrake design is called the flight control corridor. This corridor represents the maximum flight path that the entry vehicle can follow in order to maintain control without skipping off the atmosphere or reentering the atmosphere. Within this region there is a limiting corridor called the load relief corridor. The design of this region is to reduce the g force and heating of the reentry vehicle. This is obtained by flying at a higher altitude. However, the higher L/D ratio which results in more control capability of the aerobrake and larger control corridor is reduced by a load relief corridor, but the result is a decrease in g force and heating on the reentry vehicle.

A first order approximation to the size of the footprint can be found if the following approximations are made:

1) L/D remains constant during the descent
2) The bank angle, j, remains constant
3) The original MEV orbit is circular.

Fig. 7.3.1 shows the landing footprint. The exterior box represents lateral and longitudinal ranges while the interior "footprint" represents the actual achievable landing area. This area is smaller than the rectangular region for two reasons. One, a very steep descent trajectory implies very high heating and aerodynamic loads, and two, when flying for maximum crossrange ($D_y$), the craft's forward momentum is necessarily reduced.

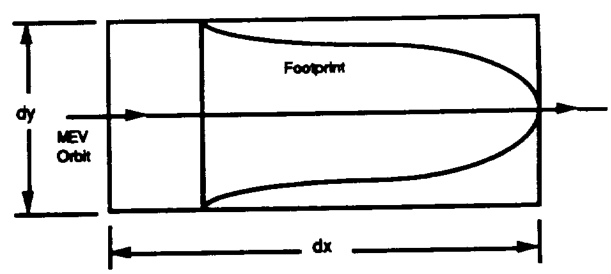



**Figure 7.3.1:** Landing Footprint

It should be noted that the downrange is not as important as descent can begin from any point in orbit. It is large crossrange which is desirable to reduce fuel costly orbit plane changes and to increase the margin of error in reaching the desired landing site.

## 7.4 Aerobraking Vehicles

The vehicles used in descent through the atmosphere need aerodynamic braking to slow to a velocity where the vehicle can land safely on the planet surface. The devices used are commonly known as aerobrakes. There are a variety of aerobrakes that can be used depending on the requirements of the mission. They are divided into categories of low, moderate and high lift to drag aerobrakes. The best aerobrakes, that meet the mission requirements are the low and moderate aerobrakes. There are two configurations to the descent vehicles one is the Biconic MEV and the other is the Aeroshell / Crew lander. The aerobrake used for the descent of the crew is a low lift to drag aeroshell.

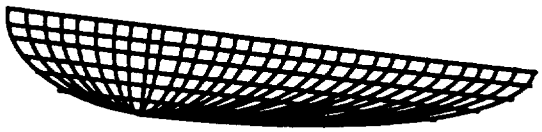


**Figure 7.4.1:** Low L/D Aerobrake

The basic idea used in biconic entry is to enclose the payload in a non-symmetrical container comprised of circular, conic, cylindrical, and spherical shapes. This unusual design provides a lift to drag ratio that lies between ballistic and gliding entry providing a vehicle with some cross range maneuverability. Biconic technology is new and expected to be a developed science by the end of the century.

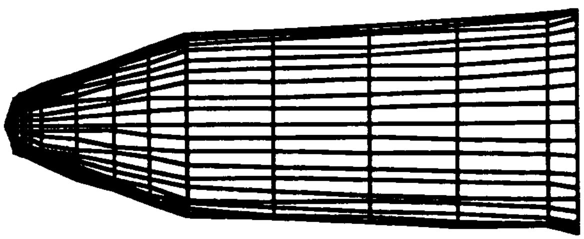


**Figure 7.4.2:** Moderate L/D Aerobrake

High L/D aerobrakes include glider type vehicles similar to the space shuttle. These aerobrakes tend to have less cargo room and are too complex to fit the needs of the Mars mission. The vehicle would have a lift to drag ratios of greater than 1.5 which would allow a fair amount of maneuverability while descending and thus a larger landing footprint. A glider would need a reversible propulsion system to allow landing at a safe impact speed. Due to the roughness of the Martian terrain, the vehicle would have to be a vertical take-off and landing (VTOL) craft.

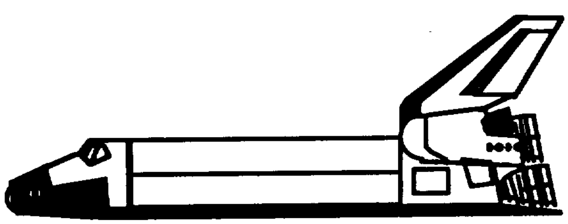


**Figure 7.4.3:** High L/D Aerobrake

## 7.5 Orbital Maneuvers

According to Orbital Mechanics the descent vehicles are both required to achieve a 300 km orbit before descending to the planet and that a plane change is not required. The 300 km orbit is achieved in order to be able to descend at any point on the orbital trajectory. The plane change is no longer needed because the plane of the MTV orbit already passes over the landing site.

There are two scenarios Aerodynamics is considering for achieving the 300 km orbit. The first scenario is that the entire orbital adjustments are to be done propulsively. The second option is to use the atmosphere to achieve the 300 km orbit.

<u>Option 1</u>

1) Propulsively inject the vehicle into a 300 km orbit.
2) Propulsively deorbit from 300 km parking orbit.

<u>Option 2</u>

1) Propulsively deorbit from MTV elliptical orbit.
2) Dissipate energy aerodynamically and propulsively burn to establish 300 km orbit.
3) Deorbit from the 300 km parking orbit.

Both of the vehicles have the ability to perform this maneuver, however it is suggested that it only be done with the biconic aerobrake. The large mass of the biconic is mainly caused by the amount of fuel needed to perform the orbital maneuvers required. "The propulsive requirements can be greatly reduced by utilizing the atmosphere of the planet."[7.2] Two major considerations that still need to be addressed is the time it takes to perform this orbital maneuver and the comparison between the amount of propulsion that would be used in either scenario.

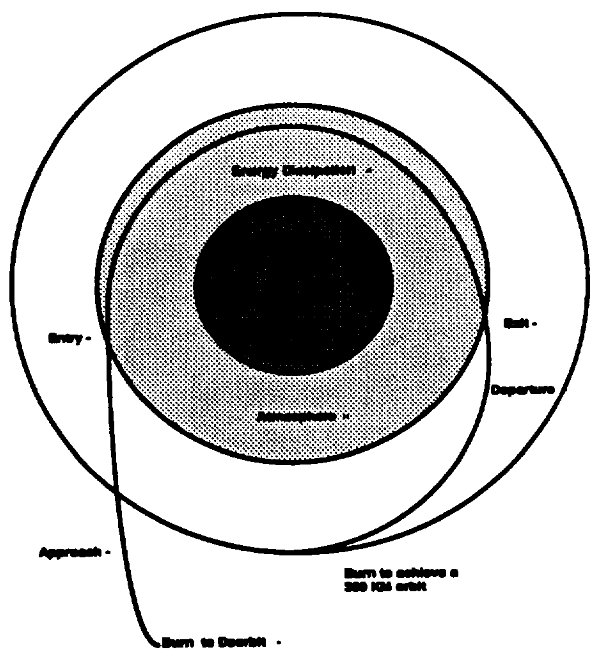



**Figure 7.5.1:** Orbital Maneuvers

The vehicles are also able to perform an orbital plane change during this same maneuver. In order to perform the plane change the vehicle needs to change the direction of the lift vector, which is done by banking the vehicle. The higher the L/D of the aerobrake, the larger the plane change possible. For the 1 sol orbit, a 12 degree plane change can be accomplished with a L/D of 0.8.[7.1] The biconic currently has a L/D of 1.2 and a maximum cross-range of 880km (14 plane change). The aeroshell currently has a L/D of 0.3 and a maximum cross-range of 330 km (5.89 degree plane change).

### 7.5.1 Aeroshell Maneuver

A generic entry into the atmosphere for an aeroshell has three phases in the descent. The first phase is the initial entry where maximum drag is maintained until the desired altitude is reached. The second phase of the maneuver occurs when the desired altitude is reached. At this point the aeroshell maintains a constant altitude and dissipates the excess energy. The final phase of the aeroshell maneuver is achieving a desired glide. The process for a preliminary analysis of the trajectory will be done is as follows:

1) Set the maximum g-loading and the maximum heat rate which the craft can maintain.
2) Pick a desired altitude for the dissipation maneuver.
3) For different ranges of ballistic coefficients the velocity at the dissipation altitude can be determined.

In other designs a maximum g-loading of 5 g's and maximum heat rate of 300 W /cm$^2$ has been standard values.Using the data found and the restraints of the vehicle, plots can be made to determine the envelope in which the vehicle can descend. Preliminary analysis has begun on the descent of the aeroshell.

The aerobrake maneuver involves an atmospheric entry known as the skip entry. The skip entry is accomplished by entering into the atmosphere from a hyperbolic transfer orbit. The vehicle enters the atmosphere at a very high speed. For example, at Earth, $V_E = 12.6$ km/s. The vehicle uses the atmosphere to lower its energy and the literally skips back out into space. Since the vehicle has a lower speed at this point, a much smaller $\Delta v$ is required to park the vehicle in orbit. This method can also be used to take a vehicle from a high energy orbit to one with a much lower energy. To accomplish this, a small propulsive burn is required to send the vehicle into the atmosphere. Then, the vehicle produces drag while in the atmosphere to lower its overall energy. Finally, the vehicle changes its flight path angle and enters into a lower energy orbit.

The key factor in this maneuver is the flight path angle of entry. There is a range or corridor of flight path angles for which aerobraking is safe. The minimum angle limit is the smallest entry angle, usually on the order of 8 to 10 degrees, that the vehicle can have without skipping off the atmosphere and back into a hyperbolic orbit. The maximum angle limit, usually between 12 and 15 degrees, is the angle at which the vehicle can enter the atmosphere without experiencing the maximum g-loading. This maximum g-loading is usually between 3 and 5 g's, where 1 g = 9.81 m/s.

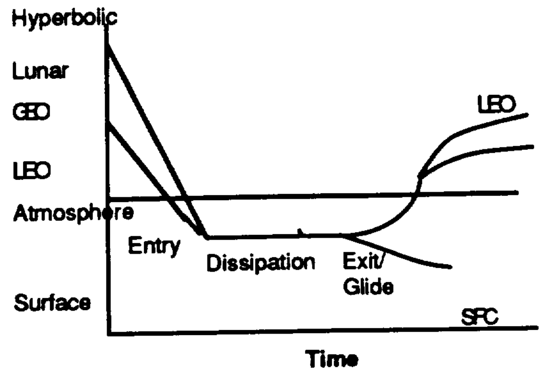



**Figure 7.5.2:** Generic aeroshell maneuver

### 7.5.2 Descent Trajectories

The parking orbit that will be established by the MEV vehicles prior to descent will be a 300 km circular orbit. The MEV will need to make an orbital plane change so that the vehicle is able to land at the desired site. This will have to be accounted for when making fuel estimates. From the new orbit the MEV vehicle will have to initiate a burn that will send the craft on a trajectory that intersects the planet. The smallest burn that will accomplish a suitable trajectory is the largest energy orbit that intersects the planet. Shown in Figure 7.5.3 is the trajectory that will have the smallest deorbital burn. The velocity at atmospheric entry can be found using the following equations:

(7-1) $E = u / 2a$

(7-2) $E = V^2 / 2 - u / r$

E energy of the trajectory
V velocity
u gravitational parameter
2a distance of the semi-major axis
r distance the craft is from the center of the planet

The velocity entering the atmosphere was found to be 3.5 km/s (11500 ft/s). This means that the vehicle will be approximately be traveling at Mach 13 when entering the atmosphere. This number was found so when using the program HABP a Mach number can be specified. The calculations of the stability derivatives and lift to drag ratio made by the program are dependent on the Mach number. This will give a more accurate estimate of what the vehicle will be experiencing during descent.

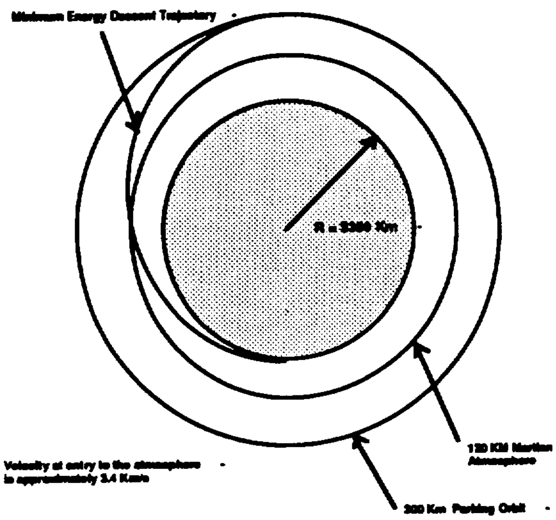



**Figure 7.5.3:** Descent trajectory

## 7.6 MEV Configuration Trade Study

While there are many different combinations of vehicles that can be used on the mission to Mars, but the vehicles that the Aerodynamics group considered are the following:

1) Configurations that were considered for the MEV.
   a. Single aeroshell
   b. Single biconic
   c. Aeroshell / biconic

2) Design constraints for the MEV.
   a. Minimal in orbit assembly
   b. Habitat is 450 $m^3$
   c. Enough supplies for a meaningful 60 day stay on Mars.
   d. Maximum weight 100+ mt
   e. Safety factors

Using these configurations and constraints as the ranges for the trade study, Aerodynamic Analysis analyzed the various vehicle configurations to determine which one would be optimal for the mission. Outlined below are the results from that trade study.

### 7.6.1 Single Aeroshell / Lander MEV

<u>Advantages</u>
1.Low fuel mass
2.Single integrated package
3.Abort option is possible
4.Lander has already been tested

<u>Disadvantages</u>
1.Small cargo volume
2.Untested wake flow
3.Earth orbit assembly required
4.Large lander size

### 7.6.2 Single Biconic MEV

<u>Advantages</u>
1. Single integrsted package
2. One descent pod saves on overall mass
3. Large cargo volume

<u>Disadvantages</u>
1. Ascent pod mechanics are complicated
2. No escape option for emergency jettison
3. Large size and mass of a single vehicle
4. Untested biconic configuration

### 7.6.3 Aeroshell / Biconic Multi-Vehicle MEV

<u>Advantages</u>
1. Crew would descend in a proven craft
2. Abort option for the crew
3.There can be a systems check of the habitat before descent of the crew.
4.A limited mission could be preformed if the biconic failed
5.More cargo room in the biconic
6.The lander is still able to asses landing area for the crew
7. Leaves an option for two MTV vehicles
8.Two smaller vehicles for descent
9.Ascent pod design is uncomplicated
10. No Earth orbit assembly required

<u>Disadvantage</u>
1. More mass to surface
2. Lander is away from the biconic
3. Untested wake flow for the aeroshell
4. Untested biconic configuration

This as the configuration that the Aerodynamics group decided upon. Having a multiple MEV receives the benefits of both the biconic and the aeroshell, while not being limited by the disadvantages of a vehicle being used exclusively. An advantage the *multi-vehicle* system has is the crew will descend in a proven craft. The lander that will be used is modeled after the Eagle lander used in previous missions. This vehicle will also have the option to abort the descent if problems occur. The safety and success of the crew during the first mission is most important, since it is our goal to complete three missions to Mars. Another safety advantage is the habitat can have a systems check before the crew descends; if problems do occur with the habitat a smaller mission can still be achieved with the crew lander. By having two smaller vehicles the space on the biconic will be used solely for cargo and the lander will be responsible for the ascent and descent of the crew. It will also allow for the deceleration of the vehicles to be accomplished easily. Another great thing about the configuration is that there would be no Earth orbit assembly required and leaves the option to have a multiple MTV.

The multi-vehicle configuration is the best design for our missions purposes. It optimizes the advantages of both the biconic and the aeroshell vehicles. The configuration provides the mission with more options, more experimental facilities, and a larger safety margin.

## 7.7 Biconic Design

The biconic is a moderate lift to drag aerobrake. The main function of a biconic is to carry the cargo to the planet. The biconic MEV that will be used in the mission is an unmanned vehicle which takes the habitation module and the supplies for the 60 day stay down to the surface of Mars. The MEV Hab is the living quarters that the crew will have for the duration of their stay on the surface of Mars.

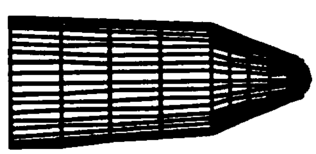


Figure 7.7.1: Biconic MEV

### 7.7.1 Landing Configuration for the Biconic

The first consideration with the biconic is if it is possible for the vehicle detach from the MTV and enter from an interplanetary orbit or if the vehicle needs to brake with the MTV and achieve an orbit around Mars before detaching from the MTV. To make a decision about where the biconic will detach from the MTV a further analysis of interplanetary entry is to be done, however at this time the aerodynamics group would want the biconic vehicle to remain attached to the MTV until a orbit around Mars can be achieved. One reason for this is that the longer the vehicle remains with the MTV, there will be less orbital corrections that would have to be made by the biconic. One correction that has to be made is an orbital plane change, this enables the craft to reach the landing site. Another reason is the longer the vehicle remains with the MTV the vehicle can have system checks done just before it detaches to descend to the planet.

There are different stages that the biconic vehicle will go through when entering the Martian atmosphere. The stages of descent are:

1. Change from an the orbit to the descent trajectory.
2. Initial entry into the atmosphere.
3. Peak heating of the vehicle.
4. Peak loading of the vehicle.
5. RCS for trajectory correction.
6. Deployment of the parachutes for stability.
7. Propulsive system for final braking.
8. Landing of the vehicle.

In stage one of the vehicles decent there is a need for propulsive impulse to put the vehicle on a descent trajectory and propulsive impulses for orbital corrections. From stage one an initial velocity for entry must be established. This will be used in calculation of the descent trajectory. In the second stage of descent the vehicle will enter the Martian atmosphere. The second stage has little affect from the upper atmosphere, but still needs to be considered in the entry of the vehicle. The third stage will begin when the vehicle is further into the atmosphere, the vehicles shell is important in this stage, because at peak heating rates, the material used will have to withstand extreme temperatures. The structure of the vehicle is important in the fourth stage when the vehicle would be experiencing the peak loading. During this stage the vehicle will have to with stand the highest loads. After the vehicle has gone through the first stages an RCS system will be needed to make trajectory corrections. Most of these corrections will be done during this stage. Near the final stages of descent parachutes will be deployed for stability purposes and the propulsive system will be used for the final braking.

### 7.7.2 Biconic Analysis

The original design is shown in Figure 7.7.2 . This design had a maximum L/D of about 0.7. Although it met all of the volume constraints, it did not provide an adequate crossrange.

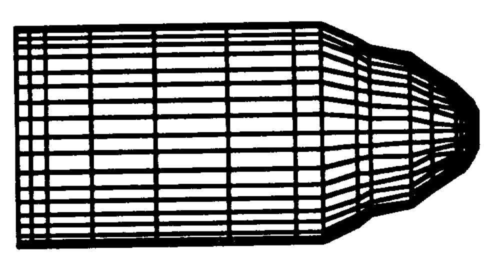


Figure 7.7.2: Basic Biconic

Two other alterations were made to the original biconic. First, the nose cone was replotted with more cross sections onto the input card for HABP. Although it increased computing time, it ensured a more aerodynamic and real nose cone shape. Second, the aft section of the biconic was sloped outward at 2.86°. This increased the diameter of the base, and also increased the original volume, which was to the our benefit. This was done after investigating several reports on biconics and how an increase in the lift could be accomplished by this method[7.1]. The altered shape can be seen in Figure 7.7.2

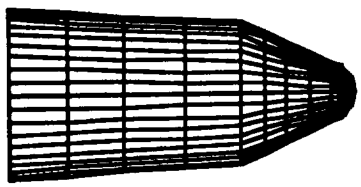


**Figure 7.7.3:** Flared aft section on the biconic

The design was the analyzed. It had already met the volume constraint. Next, the L/D was plotted. It had a L/D greater than 1.00 between the AOAs of 15 and 25 degrees with a maximum of 1.2 at 20 degrees AOA. It met and exceeded the L/D requirement. This design could effectively be flown at 20 degrees AOA with a safety factor of +/- 5 degrees before it fell below the required range. However, at low AOAs, the slope of the Moment Coefficient was positive. The design would either have to be altered to eliminate this, or small AOAs would have to be avoided in order to meet the slope of the Moment Coefficient requirement. It was decided after investigation into previous biconic designs that only a small alteration in the design could be done to eliminate this problem.

The Moment Coefficient can be changed by flaring the tail of the biconic. The sloped biconic design was then altered in such a manner. Initially, a 45 degree flare that extended 3% of the base diameter outward was tried. The result of the CM vs AOA was adequate, but the L/D dropped dramatically and did not attain the desired L/D of 1.00 at any AOA. This option was discarded. The sloped biconic was altered again Fig (7.7.3). This time, a 11.3 degree flare that extended 3% of the base diameter outward was tried. The Cm vs the AOA (Fig 7.7.4) was again adequate, and so was the L/D vs AOA (Fig 7.7.5) which was essentially the same as the sloped biconic's.

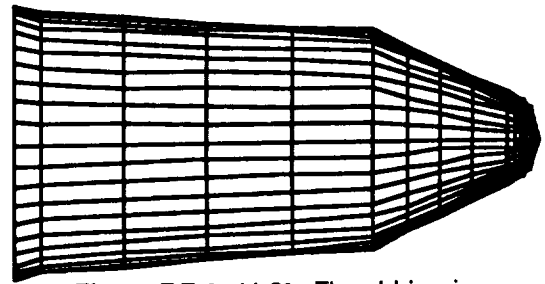


**Figure 7.7.4:** 11.3° Flared biconic

**7.7.3 Biconic stability**

Flaring the back end of the biconic is also beneficial for the maneuvers that need to be performed. Sections of the flare will have the ability to deflect which will change the flow across the aft section. This enables the craft to pitch, roll, and yaw for necessary flight corrections. The control panels will be more effective at lower altitudes and at slower velocities.

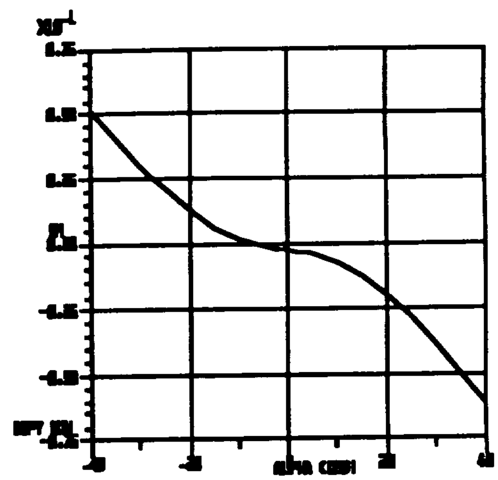


**Fig 7.7.5:** Cm vs. AOA: 11.3° flared biconic

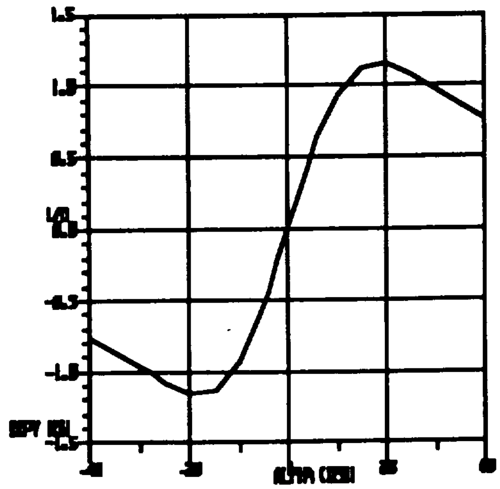


**Figure 7.7.6:** L/D vs AOA: 11.3° flared biconic

For a portion of the descent the vehicle will need to achieve level flight to dissipate some of its energy. The vehicle must also be able to have the trim angle near the maximum lift to drag. This will enable the vehicle to achieve a larger cross range during descent. The center of gravity has to be positioned appropriately for level flight at a specified AOA.To ensure the stability of the aerobrake at a particular angle of attack we must locate the center of gravity at a set location.

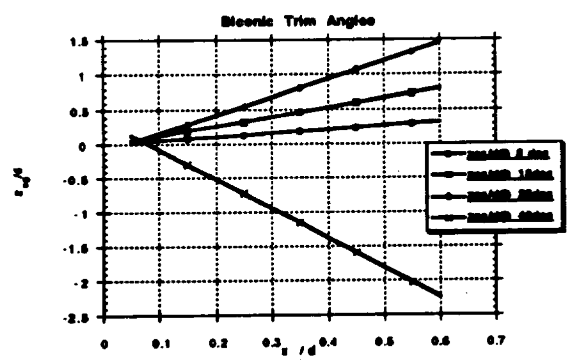



**Figure 7.7.7:** Biconic trim angles

It was seen in the Fig 7.7.6 describes the trim angle of attack for a specific location of the center of gravity. The reader should note the dramatic change in stability requirements for the biconic at a 40° angle of attack. This is because the biconic is a lifting body which in this case has become stalled.

### 7.5.5 Biconic Heating

The aerodynamics group has also done a preliminary design estimate of the heating rates associated with the biconic.

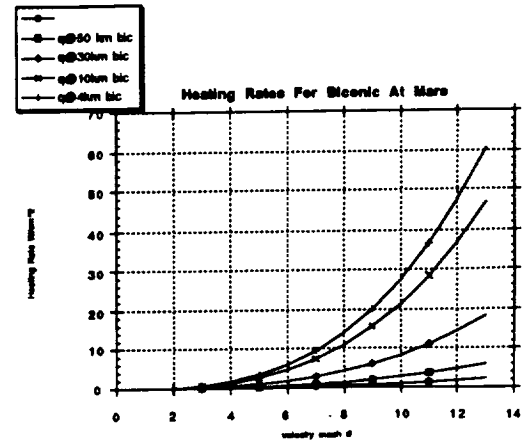



**Figure 7.7.8:** Biconic heating rate ($W/cm^2$ vs. Mach number)

It was seen that the shape of the graphs is similar to that of the aeroshell but the heating rates are much higher due to the much smaller radius of the nose of the biconic.

The final vehicle configuration that will transport the cargo down to the planet will be the 11.3° flared biconic. It meets our primary requirements for a Mars lander in the aspects of L/D, stability, heating, and volume.

## 7.8 Aeroshell Design

The vehicles used in descent through the atmosphere need aerodynamic braking to slow to a velocity where the vehicle can land safely on the planet surface. The devices used are commonly known as aerobrakes. There are a variety of aerobrakes that can be used depending on the requirements of the mission. They are divided into categories of low, moderate and high lift to drag aerobrakes. The best aerobrakes, that meet the mission requirements are the low and moderate aerobrakes. There are two configurations to the descent vehicles one is the Biconic MEV and the other is the Aeroshell / Crew lander. The aerobrake used for the descent of the crew is a low lift to drag aeroshell.

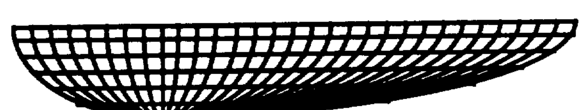


**Figure 7.8.1:** Mars Descent Aeroshell

The aeroshell is the means by which the crew lander decelerates through the atmosphere until final approach. The crew lander is the vehicle that the astronauts descend to the surface in. It also has the ascent pod on the vehicle which the astronauts will use to rendezvous with the MTV in orbit. The crew lander is modeled after the Eagle Lander that landed on the moon.

### 7.8.1 Descent Configuration for the Aeroshell

The aeroshell and crew lander will not detach until the MTV has propulsively braked and has maintained an orbit around Mars. The biconic MEV will have already landed and a system check made on the vehicles before the crew descends to the planet. The stages of descent for the crew are as follows:

1. Detachment from the MTV.
2. Achieving the parking orbit.
3. Change from the parking orbit to the descent trajectory.
4. Initial entry into the atmosphere.
5. Peak heating
6. Peak loading
7. Aerobrake jettisoned
8. RCS for trajectory correction
9. Deployment of parachutes for stability
10. Propulsive system for final braking
11. Landing

The detachment and initial descent of the aeroshell will be done with a propulsive system. This will involve achievement of a 300 km parking orbit and a propulsive burn to deorbit. Initial entry into the atmosphere is basic, but could involve the use of parachutes for an initial deceleration. Further into the atmosphere, the peak heating and loading will occur. The aeroshell has to be structurally sound to with stand the heating and the loading that will occur. The fluctuations in the Martian atmosphere may cause perturbations in the initially calculated descent trajectory. The perturbations can be accounted for by sending down a small probe prior to the descent of the vehicle. The next phase of the descent is to jettison the aeroshell after the vehicle has dissipated much of its energy and has slowed to a velocity at which detachment is possible. After jettison of the aeroshell, an abort option is available, but it must be remembered that the escape pod must have enough thrust to overcome the descent velocity and escape into a suitable orbit. When detaching from the aeroshell a parachute could be deployed to initially stabilize the craft before use of the RCS. Once the crew lander has detached from the aeroshell, the RCS system will be used to handle any course corrections that may be necessary to land the craft at its desired target. Parachutes will be used again to insure stability on the final approach. The final propulsive brake will have to be enough to decelerate the craft to a velocity that the landing gear can handle and enough to account for any variances in altitude that were not foreseen.

### 7.8.2 Aeroshell Analysis

The different shapes of a low lift to drag aeroshell were analyzed as part of the Aerodynamic Analysis research. The first thing done was a comparison of L/D's for the spherical aeroshell and the raked aeroshell. At low angles of attack the lift to drag for the raked aeroshell was 0.3, which was slightly higher than the spherical aeroshell. The change in shape was done by moving the location of the center point towards one of the edges.

Further analysis on the raked aeroshell showed that at higher angles of attack the lift to drag was as high as 0.6. However, when the vehicle is flying at such an angle of attack the wake flow becomes a major problem for the crew module. The super-heated gases bend towards the inside of the aeroshell. The amount that the flow of the fluid bends is called the Turning Angle. The total deflection of the fluid can be found by adding the angle of attack to the turning angle; this angle is known as the impingement angle. As the angle of attack increases so does the impingement angle, therefore staying at lower angles of attack is desirable.

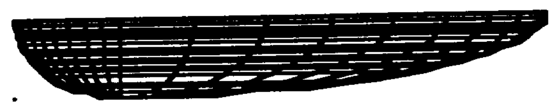


Figure 7.8.2: Raked Aeroshell

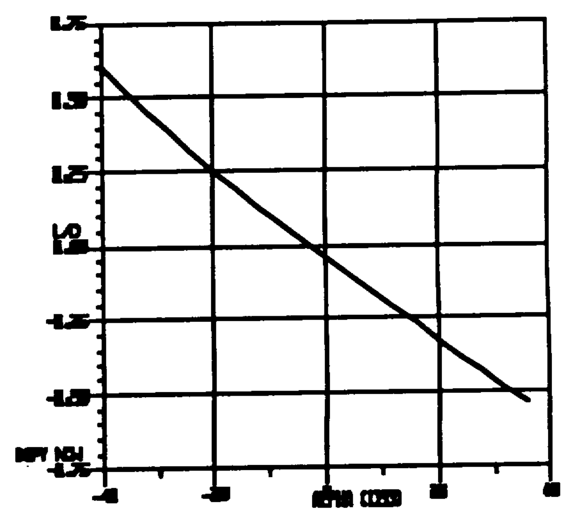


Figure 7.8.3: L/D vs. AOA

### 7.8.3 Aeroshell Stability

Another design factor that needs to be addressed is the stability of the aeroshell. The aeroshell stability will done by limiting necessary conditions for vehicle equilibrium and longitudinal static stability and proceeding with more complex conditions once these conditions are satisfied.

To achieve longitudinal static stability the slope on the Coefficient of Moment vs Angle of attack plot must be negative.

Cm/alpha < 0

The results that were obtained from HABP show that Aeroshell is statically stable over the angles that the vehicle will be descending at. The maneuver of the aerobrake will be done by sliding the vehicle along the tracks inside the aeroshell. This will change the location of the center of gravity and will pitch the aeroshell. This maneuver is necessary in order to change from the maximum drag to the trim angle of attack. During the initial descent the vehicle needs to decelerate as much as possible, therefore the vehicle will benefit most by entering at a zero angle of attack. However for the vehicle to be able to dissipate as much energy as it can level flight through the atmosphere should be achieved. Sliding the vehicle in side the shell on some tracks will change of the vehicle to the desired the orientation.

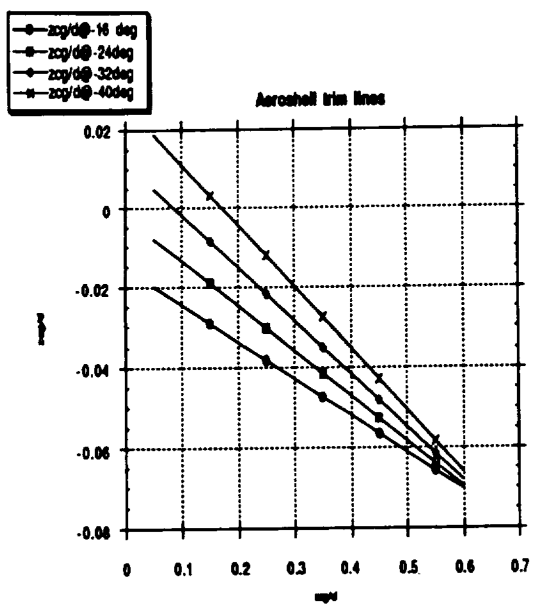



**Figure 7.8.4:** $Z_{cg}/d$ vs. $X_{cg}/d$

This graph represents the relationship between the location of the center of gravity and the angle of attack of the aeroshell. For each x and z coordinate of the aeroshell there is a corresponding trim angle of attack line. By using this graph the trim angle of attack can be designed just by locating our center of gravity at a specific point.

### 7.8.4 Heating Rates

Another design issue to take into account is aerodynamic heating. The Aerodynamics group used a basic "design" equation to give an order of magnitude estimate of the heating rates upon Mars entry.

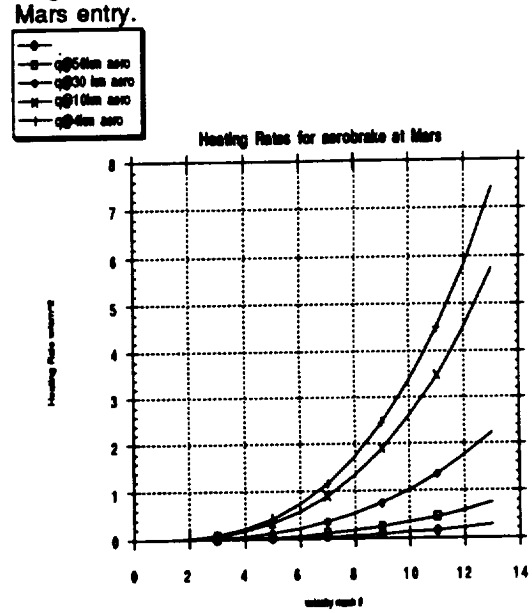



**Figure 7.8.5:** Heating rates for aeroshell

# 8.0 Thermal Analysis

## 8.1 Introduction

Thermal and radiation protection of the Mars Transportation System required some innovative answers to some tough problems. The tradeoffs between adequate radiation shielding and mass were tried to their limits. Since the main design driver was mass savings, sufficient means of protection were provided while chopping mass at every opportunity.

It was deemed that the cost in mass of reducing the radiation exposure was too high to be of much value. Therefore, other than a storm shelter, no radiation shielding will be placed around the MTV hab. Also, no radiation shielding will be provided for the MEV, owing to the Mars atmosphere and the short stay time.

In the case of thermal protection, the lightest, most heat resistant materials were placed on the vehicle, and masses were kept as low as possible. A new ablative substance, Zirconium Diboride, will be used along with a new RSI (Reusable Surface Insulation) tile system to dissipate the tremendous heat energy the aerobrakes will encounter.

A great deal of work went into deriving a suitable boiloff rate formula that could be used with the MTS design. This formula was incorporated into the spreadsheet developed and used by the MTS program, and the results are summarized in Figure 10.4.

## 8.2 Radiation Protection

Most of the attention in regard to radiation shielding was given to mass savings. The NASA and NCRP (National Council on Radiation Protection) proposed guideline is 50 REM per year BFO exposure. BFO stands for Blood Forming Organs, equivalent to a tissue depth of 5 cm, effectively the bone marrow. See Table 8.1 for more specific information on radiation limits. Realizing this value is only a guideline, and that no one knows for sure how much radiation exposure is really "safe", decisions were made to stay as close as possible to this limit, lopping off mass at every opportunity. Many new and different schema were proposed and looked at throughout the quarter, including a regolith bagging system to protect the astronauts on the surface and a liquid hydrogen shield around the circumference of the transfer habitation module. In the end, it was determined that a trip taken at solar minimum without outer hab shielding would best suit the needs of the mission.

| Timeframe | Skin Dose (REM) | BFO (REM) |
|---|---|---|
| 30 days | 150 | 25 |
| 1 year | 300 | 50 |
| Career | 600 | 100-400[a] |

[a] Dependent upon age, gender, and age of first exposure.

**Table 8.1:** Radiation Limits[1]

### 8.2.1 MTV Shielding

#### 8.2.1.1 Liquid Hydrogen

One of the first proposals for radiation shielding on the MTV was a liquid hydrogen system. Liquid hydrogen absorbs high energy particles very well for its mass. Dr. Cecil Waddington of the Physics department at the University of Minnesota[8.2] suggested looking at a system utilizing heavy water (deuterium bonded with oxygen - $D_2O$). Since the ECLSS system on the MTV hab was expected to produce hydrogen, liquid hydrogen appeared to be the more promising system.

The design of the shielding system utilized space between the two concentric outer pressure vessel walls to hold liquid hydrogen (see Figure 8.1). A series of aluminum mesh screens or drilled plates would provide uniform fluid flow and coverage, and possibly structural rigidity. Hydrogen would flow around the mesh by means of a pump/valve system employed by the ECLSS.

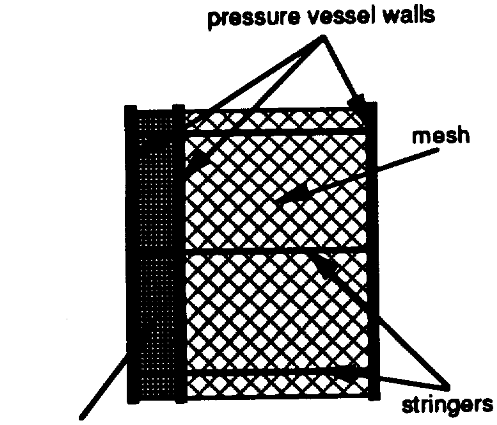



**Figure 8.1:** Cross section of pressure vessel walls

Problems surfaced along the way. One big trade study involved in such a system was thickness - thickness of the pressure vessel walls, the mesh, and the space allowed for hydrogen flow. Flow direction and stability was also a major problem. The artificial gravity system may have had detrimental effects on the proposed plan as well. The burden of having to overcome such hurdles prevented much study of this system. Its complexity without considerable mass savings led to its demise.

#### 8.2.1.2 Aluminum

Many of the same problems surfaced with the deuterium system, and so emphasis was placed on investigating aluminum. Since aluminum is relatively heavy, ways to reduce the amount required for shielding had to be investigated. The solar cycle provided clues.

#### 8.2.1.3 SPE and GCR

The two types of cosmic radiation are GCR (Galactic Cosmic Radiation) and SPE (Solar Particle Events). There is an opposing relationship between the two, wherein the effect of GCR is at a minimum when SPE is at a maximum (at solar max) and vice versa (at solar min). See Figure 8.2.

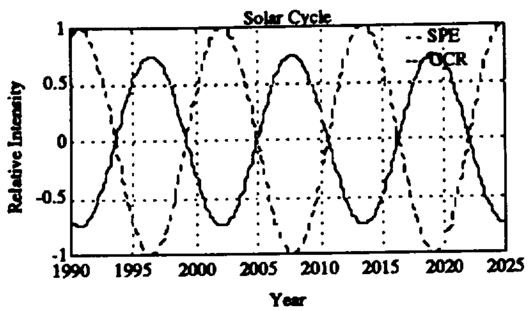



**Figure 8.2:** The Solar Cycle

Table 8.2 summarizes findings for various shielding levels at solar max and solar min[1].

| Hab Shield Level | GCR (REM/yr) | SPE (REM/hr[a]) | Total Dose (REM/yr) | Mass (mT[b]) |
|---|---|---|---|---|
| **None** | | | | |
| Solar max | 18.4 | 2.6 | 58.7 | 0 |
| Solar min | 49.1 | 0 | 49.1 | 0 |
| **7.5 cm** | | | | |
| Solar max | 13.9 | 0.4 | 19.9 | 89.5 |
| Solar min | 31.0 | 0 | 31.0 | 89.5 |

a Assuming August 1972 flare lasting 15.5 hours during solar max, no flares during solar min. 7.5 cm thick storm shelter assumed.

b Not including storm shelter; hab dimensions of 7.07m diameter and 16m length.

**Table 8.2:** Shielding levels and masses

The most important columns in the table above are the rightmost two. First notice the Total Dose column. With no shielding on the outside of the hab and a 7.5 cm thick storm shelter, the total dose received at solar max per year is near 60 REM. Adding 7.5 cm of shielding to the outside of the hab drops this value by a factor of three, but at the expense of 90 metric tons. Leaving at solar min with no shielding on the outside of the hab puts the astronauts right at their limit if no solar flares are experienced during that time. Adding 7.5 cm of aluminum shielding drops the value only slightly for the addition of 90 metric tons. Clearly, there is no value there.

If the mission commenced during solar minimum, the mass savings more than pay for the radiation protection lost. The NCRP annual limit will almost certainly be broken, but career dosage limits will not, taking into account the length of the mission.

The recommendation to the final design is to launch at solar min with no shielding on the outside of the habitation module. There will be a storm shelter within the hab with a shield thickness of 7.5 cm. Additionally, a solar x-ray detection system will be added to the total radiation protection package. When the detector measures a rate of energy flux deemed unsafe, an alarm will signal the crew to enter the shelter. When levels are once again safe, an all-clear signal will alert the crew that it is safe to exit the shelter. Inside the shelter, the crew will put on partial body protection suits for extra radiation protection.

### 8.2.2 MEV Shielding

Mass considerations and relative safety necessitated cutting out MEV radiation shielding. Table 8.3 shows BFO exposures for a 60 day stay on Mars[3].

| Atmospheric Model | 0km | 12km |
|---|---|---|
| High Density | 3 REM | 6 REM |
| Low Density | 4 REM | 8 REM |

**Table 8.3:** BFO Exposures

The mass of the shielding necessary to reduce these values was too high to seriously consider, although a regolith bagging system was studied, which proved effective for longer duration stay times.

#### 8.2.2.1: Regolith Bagging System

The system chosen for putting Martian regolith into place was based on a lunar regolith bagging system developed in a USRA design program at the Georgia Institute of Technology[8.4]. Mass savings was the biggest advantage of this system. Since the shielding material (regolith -- dirt) was already on the surface, it didn't have to be transported anywhere. The machinery needed to put the regolith where it belonged weighs only a few metric tons. The machinery fills fiberglass composite weave bags with Martian regolith. The bags are then moved to necessary locations by the astronauts and robotic rovers and lifts.

The bagging system designed for the Moon has the characteristics described in Table 8.4.

| | |
|---|---|
| Mass | 900 kg |
| Size | 1.6m X 2m X 4m |
| Max Operating Time Per Charge | 8 hrs |
| Recharge Time | 16 hrs |
| Work Rate | 120 bags/hr |
| | 960 bags/day |
| Energy Consumption | 5.7 kW |
| Max Bags Per Roll | 975 |
| Bag Size | 0.0283 $m^3$ (1 $ft^3$) |

**Table 8.4:** Regolith Bagging System

A bagging machine can produce 120 bags an hour, or 960 bags a day. That means that one machine will produce the necessary number of bags in about three days. There is still the question of how and when the crew can handle moving nearly 2500 bags of dirt, and also how long it will take them to do it. They will have to be mechanically assisted, no doubt.

The regolith bagging system was cut from the program due to the complexity involved, but for future missions longer in duration, it holds great promise.

### 8.2.3 Electrical Circuits

Today's electrical circuits are at micro-miniature sizes with very low power requirements. The 'critical charge' to change a '1' state to a '0' state is about 0.01pC. At this level if the low powered cosmic ray ($10^7$ eV) was to hit a silicon device it would produce a charge of 0.05pC which would flip the bit. This kind of arbitrary bit upsets could prove devastating if the wrong bit were to have its state changed. Table 8.5[8.10] summarizes the maximum radiation doses for various types of circuit technology.

| Technology | Rads |
|---|---|
| CMOS (soft) | $10^3$ - $10^4$ |
| CMOS (hardened) | $5*10^4$ - $10^6$ |
| TTL/STTL | > $10^6$ |

**Table 8.5:** Circuit radiation limits

## 8.3 Boiloff

The formula below was used to evaluate the heat transfer of the fuel tanks.

$$Q = -(T_1-T_2) * 2*p*k*\ln(r_2/r_1)*L$$

The values of $T_1$ and $T_2$ are the temperatures of the liquid hydrogen (33°K) and space (250°K) respectively. The constant k is the thermal conductivity of the SuperInsulation, which is $9.0\times10^{-6}$ W/(m*K). The radius of the tank is $r_2$, the tank radius plus the thickness of the insulation is $r_1$, and the length is L. The heat transfer is also equal to the boiloff rate multiplied by the enthalpy of evaporization. Setting the boiloff rate equal to 1% per month and finding $h_{g(LH2)}(T_1)$ to be 464 kJ/kg[8.5], and then setting the equation equal to the Q above allows solving for $r_1$. Subtracting $r_2$ from $r_1$ gives the thickness of the insulation and thus its mass. These calculations were absorbed into the boiloff spreadsheet. The thickness and mass of the SuperInsulation along with the boiloff calculations are best related in Figure 10.4.

## 8.4 Thermal Protection

### 8.4.1 MTV and MEV Habs

A heat dissipation system was to be designed by crew systems to operate as efficiently as possible with the ECLSS system.

### 8.4.2 Aerobrakes

The recommended TPS shielding for both of the aeroshells and the biconic is an advanced tile system composed of an advanced reusable surface insulator (FRCI-12 and FRCI-12--20) coated with a diboride ($ZrB_2$) and bonded to a composite skin.

#### 8.4.2.1 Problems

The thermal environment into which the aeroshells will be flying is only now beginning to be understood. Two different phenomena may present problems. The first is catalysis on the surface of the aerobrake, and the second is shock layer radiation[8.6].

#### 8.4.2.1.1 Catalysis

The high temperatures near the surface of the aerobrake cause molecules to dissociate. Due to catalysis, these molecules may recombine, leaving the energy of the dissociation on the surface of the aerobrake. The way to defeat this phenomenon is to use materials that do not readily recombine with molecules present (nitrogen at Earth, carbon dioxide at Mars).

#### 8.4.2.1.2 Shock Layer Radiation

No one knows to what extent this phenomena will change the heating environment. If this radiation occurs, it could raise the peak heating rate anywhere from 10 to 100%. The general problem is shown in Figure 8.4

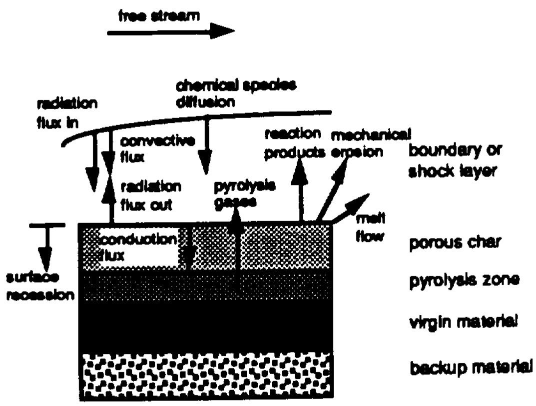



**Figure 8.3**: General Thermochemistry Problem[8.7]

Studies are being undertaken to model these problems in detail, and may add a great deal to the general understanding of the thermal environment.

#### 8.4.2.2 TPS Materials

Although many materials were researched as possibilities for thermal protection, not very many met the requirements for aerobraking in either Earth's or Mars' atmosphere. Those that seemed close are at the limit of today's technology. Tomorrow's technology should bring better solutions to the forefront. Since the design of the MTS incorporates so many different types of vehicles needing thermal protection, it made the job that much harder.

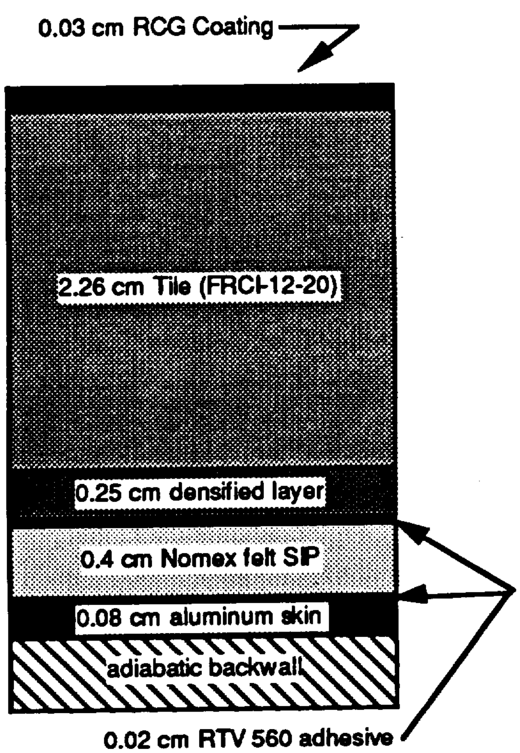



**Figure 8.4**: Tile TPS System[8.8]

There are basically two ways to shield vehicles from the massive energy flux experienced as it transcends a planet's atmosphere. The first is a metallic standoff scheme, which basically entails placing great amounts of metal on the front of the vehicle on top of a layer of insulation. As the vehicle slams into the atmosphere, most of the metal ablates (burns off), leaving the important parts of the vehicle intact. Whatever heat makes its way through the ablative material is absorbed in the layer of insulation. This is the type of heat shield used on the Apollo missions. The second type of shielding is with reusable surface insulation (tiles), such as are used on the Space Shuttle (see Figure 8.4).

New technology on the scene advances the tile system. In the new system, a toughened, high temperature thermal control surface would replace the RCG (Reaction Cured Glass) layer[8.9]. One of the latest developments in thermal control surfaces concerns the high density diboride composites of Zirconium and Hafnium ($ZrB_2$ and $HfB_2$). These substances are capable of withstanding the kinds of temperatures expected during aerobraking. In arc-jet testing done so far, the diborides have outperformed RCC (Reinforced Carbon Carbon) and have very little mass loss at high temperatures. Also, if the structure's skin were made from composite materials instead of aluminum, the SIP (Strain Isolator Pad) would no longer be necessary to handle the strain of expanding and contracting tiles. See Figure 8.5.

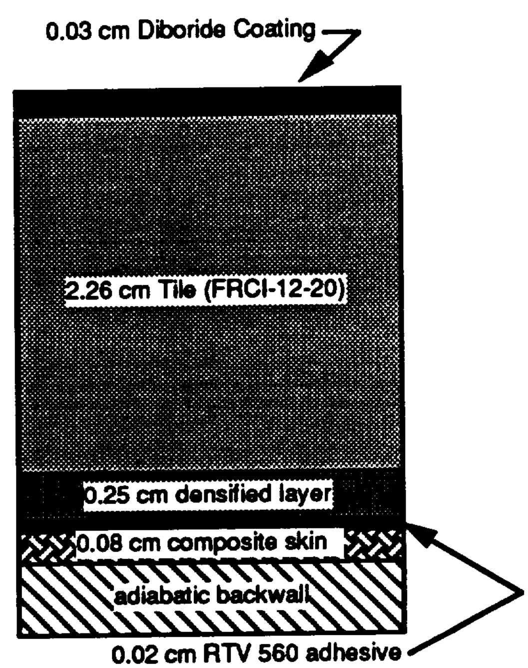



**Figure 8.5**: Advanced Tile TPS System

The aeroshells, weighing in at around 10 metric tons, had a ballistic coefficient of 300-400 kg/m$^2$.

The ballistic coefficient is a measure of the capability of a drag force to change the velocity of a vehicle. The higher the ballistic coefficient, the higher drag force needed to slow the vehicle. A higher drag force is achieved by dipping lower into the atmosphere. This raises the heating loads on the vehicle itself. The peak heating rate for the aeroshell was given as 450 W/cm$^2$ (396 Btu/ft$^2$-sec), and the stagnation temperature was given as 2700°C (5000°F). These values are at the limit of current technology, but the new tiles should be able to handle the load.

### 8.4.2.3 TPS Mass

Mass figures have been estimated for both the aeroshells and the biconic. Both structures have unorthodox shapes, but simple approximations gave good results without much error. Numerical methods appeared to be a waste of time. Approximating the aeroshell as a flat ellipse, the surface area comes out to be around 188m$^2$. The surface area of the biconic is estimated at 550m$^2$.

Figure 8.6 below shows the regions of the aeroshells where most of the heat will be encountered. Figure 8.7 displays the same information for the biconic. FRCI-12-20 (the black areas) will be placed over the hottest areas, and FRCI-12 (white) will cover the rest. The diboride coating will be used with either material, however. On the aeroshells, about 20% of the structure is covered with the heavier tile material; on the biconic, FRCI-12-20 covers about 35% of the area.

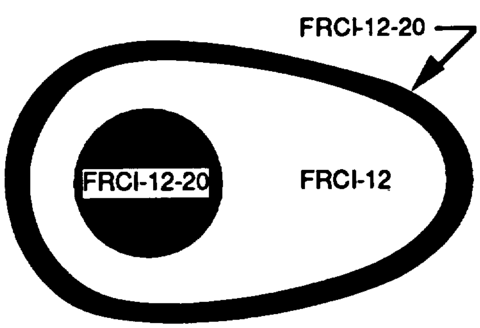



**Figure 8.6:** Heating Protection on the Aeroshells

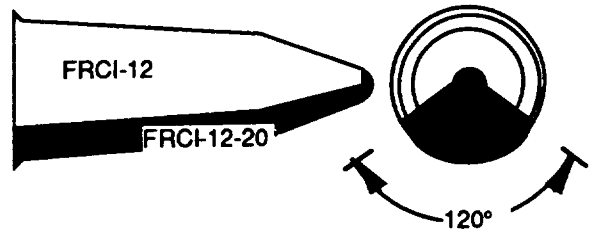



**Figure 8.7:** Heating protection on the biconic

| Material | Density (kg/m$^3$) | Thickness (m) |
|---|---|---|
| Diboride | 100 | 0.0003 |
| FRCI-12-20 | 320 | 0.0226 |
| FRCI-12 | 192 | 0.0226 |
| Densified Layer | 480 | 0.0025 |
| RTV Adhesive | 1410 | 0.0002 |

**Table 8.6:** TPS material characteristics[8.8]

Using the above figures, the mass of the TPS for the aeroshell is estimated at 1.2 metric tons. The mass of the TPS for the biconic is estimated at 4.0 metric tons.

## 8.6 Conclusion

In conclusion, the baseline specifications for themal and radiation protection on the MTS design are as follows. The MTS habitation module will have no exterior shielding due to secondary radiation effects encountered. There will be a storm shelter shielded with 7.5 cm of aluminum for protection from a catastrophic solar flare.

Thermal protection for entry into the atmospheres of Earth and Mars will utilize an advanced tile system incorporating new high temperature control surfaces. A thin layer of ablative Zirconium Diboride will be attached to the surface of the next generation ceramic tile to allow for the higher heating experienced at high entry velocities.

Boiloff rates are summed up in Figure 10.4.

# 9.0 PROPULSION SYSTEMS

## 9.1 Introduction

This section of the report will summarize the development of the Mars Transportation System propulsion subsystem. In addition, analysis of important propulsion related subjects such as propellant mass and volume estimates will also be included.

## 9.2 MTV Main Engines

Early in the quarter research was conducted into the subject of Nuclear Thermal Rockets (NTRs) for use as the MTV's main engines. A decision was made to narrow this research to Solid Core Nuclear Thermal Rockets since this particular type is most likely to be operational by the anticipated launch window.

### 9.2.1 The NERVA NTR

The Nuclear Engine for Rocket Vehicle Application (NERVA) program began almost 20 years ago with a goal to produce a high specific impulse engine for use in manned interplanetary missions. The NERVA rocket utilizes a solid core nuclear reactor to heat the propellant (liquid hydrogen) to temperatures over 2000°K. The hydrogen is circulated about the core as a coolant, and is then pumped through the core to be heated. The reactor core is a cluster of hexagonally shaped rods with holes bored along their axes. The propellant passes through the holes receiving heat from the nuclear reaction. The heated hydrogen then flows out of the core and expands into the nozzle, creating thrust. The fuel rods in the reactor are made of Uranium 235 dispersed in a graphite matrix. This unfortunately limits the reactor temperature to the melting point of graphite. More advanced NERVA designs have utilized fissionable materials dispersed in more heat resistant matrices to obtain higher reactor temperatures and hence improved engine performance. The improved NERVA designs boast specific impulses as high as 1050s and thrust to weight ratios of 3:1[9.2].

### 9.2.2 The Particle Bed NTR

In this type of system, the fissionable materials are contained in beads (0.5 mm diameter) of Zirconium Carbide (ZrC). This method of fuel containment vastly increases the surface area to propellant ratio and hence the heat transfer to the propellant. These fuel "kernels" are contained in a cylinder with a hollow flow channel in the center. The hydrogen enters the cylinder radially and is heated to 3000°K. It then flows out the hollow flow channels and out through the nozzle producing thrust. Particle bed rockets achieve specific impulses of 1000seconds (s) with a thrust to weight ratio of 30:1[9.6].

### 9.2.3 The Low Pressure NTR

The low pressure NTR is a concept which yields a high specific impulse from a simple engine. This engine utilizes a spherically shaped reactor core in which the liquid hydrogen receives heat as it flows radially outward from the center of the core. The hydrogen then flows into the spherical pressure vessel which contains the core and then outward through the nozzle. What makes this concept unique is that because the center of the core is unheated (and hence a region of low pressure) a turbopump is unnecessary. This results in an engine which is much less sophisticated than the two previously mentioned engines while maintaining a specific impulse of 1075 s and a thrust to weight ratio of 6:1[9.3,9.4].

### 9.2.4 NTR Comparisons

At the beginning of the NERVA program the Space Nuclear Propulsion Office set up the following guidelines for the design of an NTR:

1. Reliability
2. Specific Impulse
3. Thrust to Weight Ratio

First, let us address the subject of reliability. Solid core NTR technology has the distinct advantage of being developed and tested (operational prototypes from the NERVA program existed as early as 1970). However, solid core engines require complex turbopumping systems to provide adequate propellant to their engines. These turbopumps proved to be responsible for more than 50% of all NERVA test failures. Particle bed NTRs, although having never been tested, also require complex turbomachinery similar to that found in solid core NTRs. It would be reasonable to conclude that the particle bed NTR would be prone to similar failures. The low pressure NTR, although never tested, has a simplistic design which does not require turbopumps or complex feed systems.

All of the engines provide a specific impulse of 1000 s of higher with the low pressure NTR being the highest at 1075 s.

The particle bed NTR has the highest thrust to weight ratio at 30:1, followed by the low pressure NTR and solid core NTR at 6:1 and 3:1 respectively.

From these facts the following conclusions may be drawn. The solid core NTR is a proven system which would benefit from significant advances in pump technology since the end of the NERVA program in the early '70s. These facts coupled with a specific impulse of over 1000s makes the solid core NTR quite attractive. However, despite it's lack of testing, a strong case could be made for the low pressure NTR because of its simpler design, higher thrust to weight ratio than the solid core, and the highest specific impulse of the three engines. The particle bed seems to be the weakest of the three since it has never been tested and it has the lowest specific impulse. In defense pf the particle bed, it does have an extremely high thrust to weight ratio. However, current designs have the burn times of this engine at only a few seconds. In interplanetary travel, longer burns on the order of one hour are more favorable.

After careful consideration, the propulsion group elected to choose the Low Pressure Nuclear Thermal Rocket despite its lack of testing. The LPNTR's high specific impulse and low engine complexity were the deciding factors in this matter. After making this important decision, the group turned its attention to the LPNTR's specifications in order to optimize the engine for use on the MTV.

#### 9.2.5 Important Design Criteria

The LPNTR chosen for the MTV mission consists of a spherical nuclear reactor contained in a low thrust chamber and with a throat extending into a contour or bell shape nozzle.

#### 9.2.5.1 Spherical Reactor

The spherical reactor (Figure 9.2.1) has a cavity in the center containing 120 evenly spaced holes. Fuel enters the cavity from a pressurized tank and then expands radially through the holes where the nuclear reaction takes place. The nuclear reactor core is constructed from uranium$^{235}$ and zirconium with a density of 0.5 grams $U^{235}/cm^3$ in the fuel bed. The exterior structure of the reactor is constructed from beryllium. The fuel bed outer diameter is 70 cm and the inner diameter is 35 cm. The diameter of the entire reactor core is 1.2 m. The hydrogen gas will expand through the hole gaining heat and increasing in temperature up to about 3500°K.

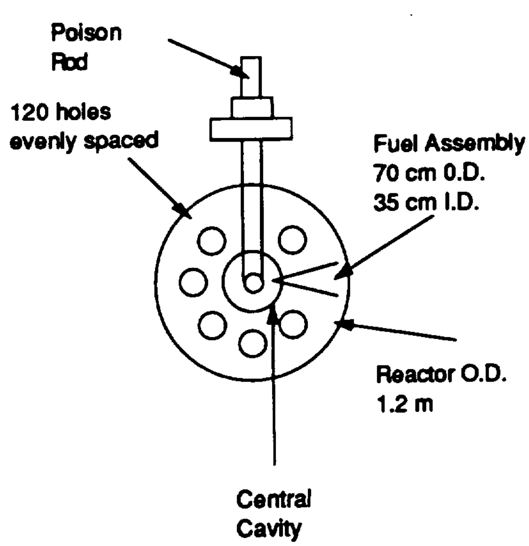



**Figure 9.2.1:** Spherical nuclear reactor core

#### 9.2.5.2 Thrust Chamber

As the heated hydrogen ($H_2$) gas exits the holes to the thrust chamber, the pressure in the chamber doesn't change much since there is no combustion in the process. As a result, the chamber pressure will be low (close to 1 atm in this case). Since the gas is hot and the chamber pressure is higher than the ambient pressure (vacuum) the gas exits the throat of the combustion chamber at transonic velocities and expands supersonically through the nozzle.

#### 9.2.5.3 The Contour or Bell Nozzle

After the hydrogen gas exits the throat a nozzle has to be extended in order for the gas to expand supersonically. Contour or bell nozzles (Figure 9.2.2) are highly recommended for this type of application because they diverge quickly after the throat (up to a 60° angle). This steep inclination allows the flow to expand rapidly after exiting from the throat and avoids flow separation. After the steep inclination, the angle converges slowly to minimize shock waves in the nozzle. Unfortunately, nozzles of this type have to be very large since they usually converge to an angle close to zero (5-8°). The use of such a nozzle for the LPNTR will improve the performance of the engine significantly, however (see Figure 9.2.3 below).

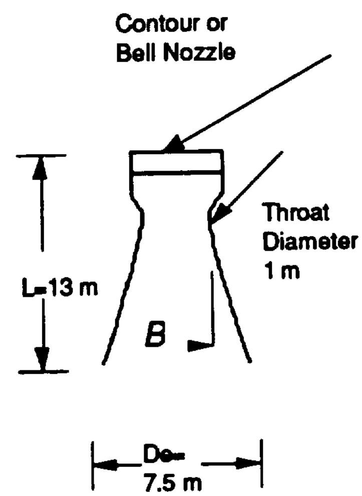



**Figure 9.2.2:** An example contour or bell nozzle

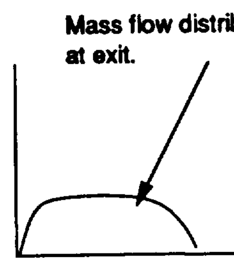



**Figure 9.2.3:** Exit mass flow distribution

#### 9.2.5.4 The Importance of Hydrogen

Due to the extremely high temperatures in the thrust chamber (3500° K), some of the diatomic hydrogen molecules ($H_2$) will dissociate into monotonic hydrogen (H) while maintaining the same enthalpy level. As the hot gas expands through the nozzle the temperature will decrease and hydrogen atoms will recombine, decreasing the enthalpy dramatically. Since the exit velocity depends on the change of enthalpy between the exit and chamber the exit velocity will increase due to hydrogen recombination in the nozzle. This will result in a higher level of thrust for the same mass of propellant. Hence, this characteristic of hydrogen plays an important role in the design and performance of a nuclear thermal rocket.

#### 9.2.5.4 List of LPNTR Parameters

The following is a list of the MTV LPNTR configuration characteristics:

a- Number of engines:......................3
b- Thrust per engine:........................113 kN
c- Nozzle throat diameter:................1 m
d- Nozzle length:.............................12.9 m
e- Nozzle area ratio:........................60
f- Flow rate:.....................................10.5 kg/s
g- Chamber pressure:......................1 atm
h- Chamber temperature:.................3200°K
I - Specific impulse:..........................1100 sec
j- Mass per engine:..........................1977 kg
k- Thrust / Weight ratio:...................6

**Reactor Data**

Thermal power:......................525 MW
Fuel region power density:.......4.6 MW/liter
Reactor geometry:.................spherical shell
Reactor core ID/OD:..............50cm/100cm
Reactor moderator:................Be + Zr
Reactor reflector:...................Be+ Graphite
Reflector thickness:...............10cm
Flow direction:........................Radial Flow
Fuel material:.........................UC-ZrC
Fuel form:..............................1mm beads
Fuel matrix melting point:........3700°K
Fuel density in fuel bed:..........0.5g $U^{235}/cm^3$
Fuel loading:...........................70 kg $U^{235}$
Fuel assembly type:...............Axial/Radial/Axial
Number of fuel assemblies:.....120

### 9.2.6 Operation of LPNTRs

#### 9.2.6.1 Propellant Delivery

Since there is very little back pressure from the combustion chamber, the very high tank pressure required to store liquid hydrogen will be used as the driver for the propellant delivery system. All that is required is a regulator to connect the piping system between the fuel tank and the reactor core. The fuel will enter the regulator at high pressure and will exit at the pressure set by the reulator. This exit pressure will be controlled electronically by the main computer in order to deliver constant exit pressure as the inlet pressure decreases with fuel depletion

#### 9.2.6.2 Fuel Tank Internal Structure

It is very important to keep the fuel inside the tank evenly distributed in the whole volume in order to maintain stable fuel delivery throughout the mission. In order to overcome this problem, an internal structure is included in the fuel tank design so all the requirements may be met.

This system consisits of a diaphragm that looks exactly like the fuel tank when it is fully expanded (see Figure 9.4.1). This diaphragm is placed inside the fuel tank and then filled with fuel. As the propellant is delivered for engine operation, high pressure helium is injected inside the fuel tank to compensate to the loss of fuel volume inside the tank.

The diaphragm system will maintain tank pressure and will also separate the helium from the liquid hydrogen in order to maintain the quality of the main propellant.

#### 9.2.6.3 Engine Cooling

Because no known structural material suitable for engine and nozzle construction can withstand the temperatures generated by the engine, these areas must be cooled. This is accomplished by an active cooling system which is similar in concept to the one found in car engines and the one currently used on the space shuttle main engines (SSMEs). Liquid hydrogen is delivered to these high temperature areas through a series of complex channels and tubes inside of the components. As the hydrogen absorbs the heat, it cools the part and is then routed into the combution chamber and expelled form the engine Unfortunately, to deliver the fuel into these small channels and back into engine chamber requires high pressure. This is due to the small diameter of the tube where viscous shear forces and friction will drop the pressure significantly in the direction of the flow.

This problem has been analyzed qualitatively and research has been conducted in order to find a definite answer to this problem. So far, two answers are available. First, the pressure drop, while significant, will be small in comparison to the tank pressure. In a LPNTR high pressure of the fuel tanks could be delivered to the propellant used for cooling with the use of a regulator. The second solution, used in most rockets, is to use a turbopump to drive the fuel through the cooling channels and into the combustion chamber.

#### 9.2.6.3 LPNTR Shutdown Proceedures

Shutting down the engine is one of the most difficult tasks a nuclear rocket designer is faced with. In order to understand the importance of engine shutdown, a fundamental understanding of nuclear core behavior is needed. The following is a brief description of how a neutronic Nuclear Thermal Rocket similar to one proposed for the MTV behaves.

Basic physics or chemistry states that atoms consist of proton that have positive charges and neutrons that are neutral with no charge. When the distance between two proton, two neutron, or a neutron and a proton become very, very small ($E^{-10}$ m), a nuclear binding force is developed among protons and neutrons to form a nucleus. Enriched uranium metal contained 235 protons and neutrons inside it's nucleus and the nuclear forces which hold the atom together are much larger than the electrical forces inside the nucleus. One way of splitting enriched uranium is by inserting one neutron into a nucleus. When the neutron takes it's place inside the nucleus the internal distribution of protons and neutrons will change. This will lead to a large imbalance in the electrical force among the protons, causing one of them to leave the nucleus along with several neutrons, gamma, and alpha particles. These ejected neutrons will in turn enter other nuclei, resulting in a 'chain reaction' which, if unchecked with result in a meltdown.

In order to control the rate at which fission occurs control rods made of neutron-absobing materials are inserted into the holes in the reactor core. The more of this material in the core, the fwer fissions that occur and the lower the output of the reactor. The role of the liquid hydrogen is to cool the reactor. As liquid hydrogen passes over the nuclear core, energy will be transferred into the hydrogen, increasing its temperature and decreasing the temperature of the core. If the flow of hydrogen were to be halted while the reactor was in operation, there would be no cooling agent in the core and the temperature would rise very rapidly to uncontrollable levels. In order to deal with such problem liquid hydrogen has to be dumped to the nuclear core at high rate to slow down the reactivity level of the nuclear core.

Standard shutdown proceedures, then, would involve idling the engine by inserting all of the control rods to minimize the fissions and then purging the engine with liquid hydrogen to remove all residual heat from the core. Addionally, boiloff hydrogen could be routed through the core during idle periods in order to keep it cool.

## 9.3 MEV Main Engines

After considering a variety of chemical propulsion units, the decision was made to utilize a $LO_2$-$LCH_4$ type rocket engine for the MEV habitation module (biconic) and the ascent/descent vehicle (aeroshell).

### 9.3.1 Engine Specification

Since the Propulsion Systems group was unable to find any documentation on the use of $LO_2$-$LCH_4$ engines for space missions, the group decided to design an engine based on some performance parameters available for $LO_2$-$LCH_4$.

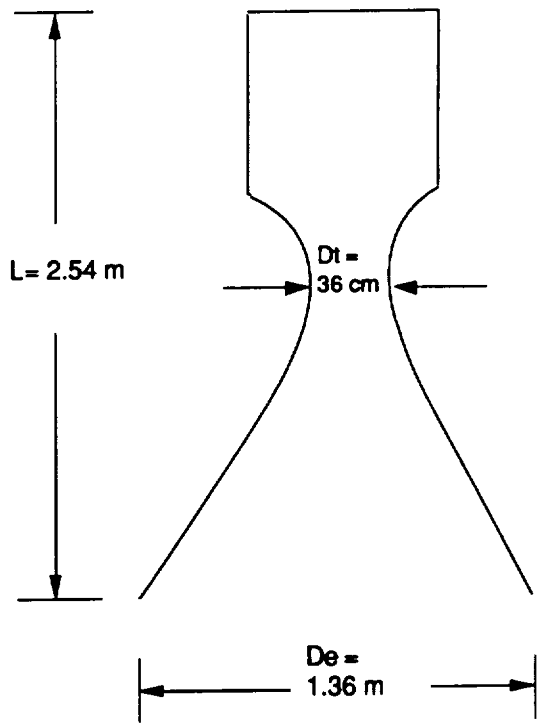



**Figure 9.3.1:** MEV Engine Schematic

The engine specification is as follows:

a- Thrust in vacuum: ...................... 133 kN
b- Specific impulse: ...................... 380 sec
c- Main propellant: ...................... $LO_2$-$LCH_4$
d- Mixture ratio: ...................... 3.5
e- Chamber pressure: ...................... 10.3 MPa
f- Mass: ...................... 256
g- Length: ...................... 2.54 m
h- Exit diameter: ...................... 1.32 m
i- Expansion ratio: ...................... 225
j- Exit area: ...................... 1.36 $m^3$
k- Throat area: ...................... 0.006 $m^3$
l- Flow rate: ...................... 35.7 kg/s

### 9.3.2 Number of Engines

The Propulsion Group is proposing to use two engines for the biconic and four engines for the aeroshell. In addition, the engines for the aeroshell descent are to be used for the crew pod during the ascent stage. It is important to note that the main engines have to be connected to the crew pod only so they can be used for the ascent stage.

The main reason for using multiple engines is safety. This consideration will help in overcoming the problem of engine failure as well as giving the flexibility of changing the thrust vector through gimballing, allowing for greater vehicle control (these engines can be gimballed ± 7°).

#### 9.3.2.1 Acent/DescentVehicle Main Engine Configuration

Upon consideration of the maneuvering that the ascent/descent vehicle has to perform, the Propulsion Group decided to use four engines with two nozzles of each size. Each phase of the vehicle's mission would have one engine out capability for optimal performance and two engine out capability possible.

The reason for the two different nozzle sizes results from the different environment the ascent/descent vehicle will operate in. It is well known in rocket engine design that the most efficient rocket performance is obtained if the exit pressure matches the ambient pressure. If exit pressure is higher than the ambient pressure, the nozzle is said to be underexpanded and results in energy loss. In this case performance could be improved by expanding the nozzle. If the exit pressure is lower than the ambient pressure shock waves are formed at the nozzle exit and there would be also a possibility to move toward the nozzle depending on the pressure difference, and that might cause deterioration and destruction of the nozzle due to the high pressure and temperature formation. If all engines chosen to meet the Martian surface requirement, the efficiency of such engines would drop significantly since the nozzle would be underexpanded during its orbital maneuvers at Earth and Mars. As a result, the Propulsion Group decided to have two different nozzle sizes to optimize the performance.

## 9.4 Tank and Feedline Systems

The following section provides an overview of the systems available for the storage and feed of propellants to the main engines for both the MEV and MTV.

### 9.4.1 Propellant Feed System

Two types of propellant feed systems are mainly available for the MTS mission: gas-pressure systems and turbopump systems.

#### 9.4.1.1 Gas-pressure systems

One of the most common means of pressurizing propellants is to force them out of their tanks by displacing them with high pressure gas. A separate gas supply tank provides the gas (usually Helium) to the top of the main propellant tank to displace the propellant and force it out. This feed system is being used in many applications especially for small attitude control rockets. The reason that this system has been used for attitude control rockets is because of its simplicity and its low mass in comparison with using a pump feed system.

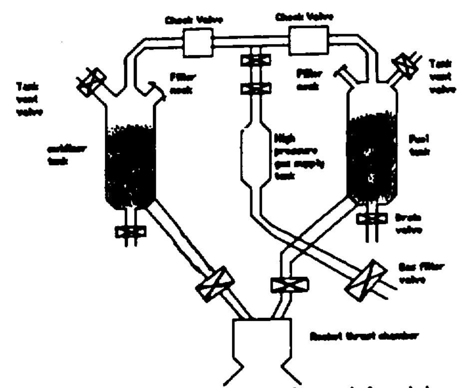


Figure 9.4.1:Typical tank and feed layout

#### 9.4.1.2 Turbopump System

The turbopump feed system pressurizes the propellants by means of pumps, which in turn are driven by turbines. Turbopump systems are usually used on high thrust and long duration rocket systems. This is part of the reason that this system is more likely to be used in the main propulsion system for the MEV.

### 9.4.2 Propellant Tanks

The first obstacle that would face the designer of a propellant tank, especially if the fuel and oxidizer are both cryogenic, is the boiloff effect. Careful design and construction in order to insulate these tanks from the surroundings will require thick walls with insulators. However, even with heavy insulation, it is impossible to prevent the evaporation of the cryogenic fluid, and therefore the cryogenic tank must include vents or other pressure relief provision to prevent self-over-pressurization.

After taking into consideration the above two constraints, the designer has to choose the optimum shape of the propellant tank in order to increase volume and decrease mass. The optimum shape for a propellant tank is a sphere. However, it won't always be feasible to design spherical tanks. This is due to the high drag that might be generated from such configuration especially in the earth atmosphere. The MEV could contain spherical propellant tanks due to the minimal effect of drag in the Martian atmosphere.

## 9.5 Reaction Control Systems

After investigation of different methods and systems of attitude control the propulsion group decided to model the MTS RCS after the reaction control system (RCS) currently used on the space shuttle.

### 9.5.1 Space Shuttle RCS

The space shuttle RCS uses the hypergolic propellants, nitrogen tetroxide (NTO) and monomethylhydrazine (MMH). The complete RCS consists of primary, vernier, and orbital maneuvering system (OMS) thrusters. The primary thrusters (model R-40B) are designed to produce perigee and orbital adjust forces. Each engine is capable of producing 4000N of thrust with a specific impulse of 300 seconds. These engines are also capable of "pulse" operation (ie for stationkeeping). The R-40B is able to produce pulses as small as 40 milliseconds. The vernier thrusters (model R-1E) are intended to be used for attitude control, orbit adjustments and stationkeeping. They are each capable of 110N of thrust at a specific impulse of 290 seconds. The OMS thrusters are intended to be used for orbital insertion, orbit transfer, rendezvous, and deorbit burns. Each engine is capable of 16700 N of thrust. Unlike the other rockets of the shuttle RCS, the OMS engines are capable of thrust vector control (TVC). Both OMS engines are controlled by a closed loop servo system. These servos provide pitch and yaw for the engine nozzles.

The propellants are fed to these engines from small spherical tanks made from titanium. These systems are also all pressure fed.

#### 9.5.1.1 Observations on Shuttle RCS

A cluster of primary and vernier thrusters comprise a module or "pod". There are a total of 3 pods. One in the nose, and two in the tail just below the OMS engines. There are two major advantages to this kind of configuration. First, Since each pod is a complete package unto itself (tanks and feeds are inside the hull just underneath each pod), it may be utilized almost at any location on the structure. Second, there is no need for additional feedline layout from a central location.

#### 9.5.1.2 Application of Shuttle Concepts to MTS

It seems that a system of pods as described above may be suitable for stationkeeping and orbit maneuvers as well as artificial gravity control. Note that these engines use NTO-MMH. Although it has a lower specific impulse (about 300 seconds) than other propellants such as $LO_2$-$LH_2$ or $LO_2$-$LCH_4$, it is very low in tank volume. This makes the pods smaller and more modular. Taking the above facts into consideration along with the extensive flight testing of these systems, the shuttle RCS is an attractive blueprint to use for the design of the MTS RCS.

### 9.5.2 The RCS Pod

Each thruster pod will consist of a number of primary and vernier type thrusters oriented radially and tangentially with respect to the center of the hull (see Figure 9.5.1,2). Each thruster pod is fed by two propellant tanks, one for the fuel (monomethylhydrazine) and one for the oxidizer (nitrogen tertroxide). These small thrusters are pressure fed requiring a tank pressure of approximately 2.5 MPa. This pressure is provided by two small helium tanks.

The layout of the pressurization and feed system is shown in Figure 9.5.3. Note the parallel feeds and multiple valves for sufficient system redundancy and increased reliability. This system is a simplified version of the space shuttle's forward RCS feedline layout[9.1].

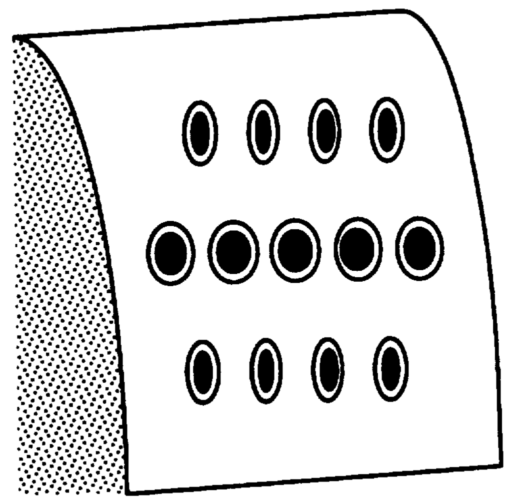


**Figure 9.5.1:** External appearance of RCS pod

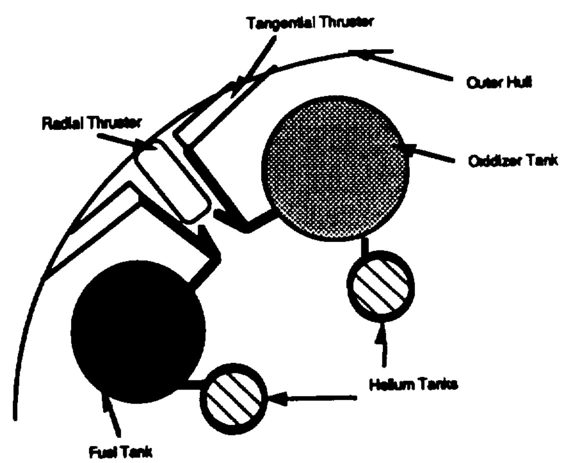



**Figure 9.5.2:** Cutaway schematic of RCS pod

### 9.5.3 MEV Habitation RCS

The Propulsion Group decided that the habitation biconic utilize four RCS pods. Each pod will consist of a number of thrusters, one bank radially oriented and two banks tangentially oriented (see Figure 9.5.1 above). Note that the thruster nozzles are below the surface of the hull, thus protecting them from the extremely high temperatures encountered during descent. Not shown are the fuel tanks and feed lines, which are also beneath the outer hull (see Figure 9.5.2).

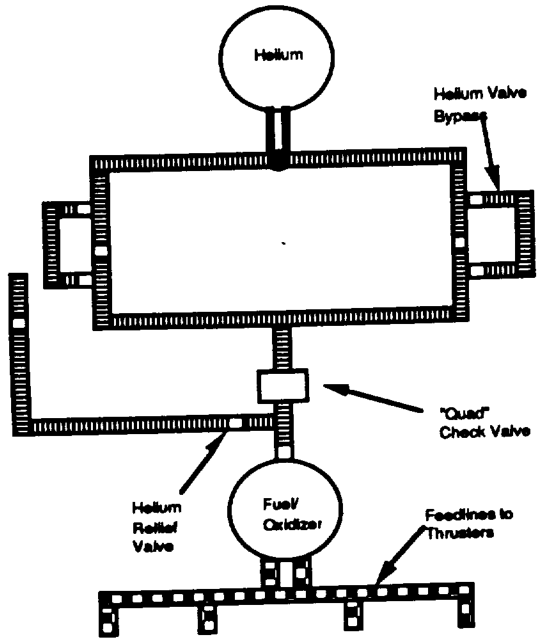
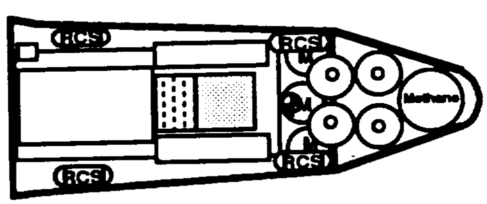



**Figure 9.5.3:** RCS pod feedline schematic

The thrusters may either be primary (3870 N thrust) or vernier (110N thrust). The proposed locations for the RCS pods for the MEV hab are shown below.

POD LOCATIONS - MEV BICONIC



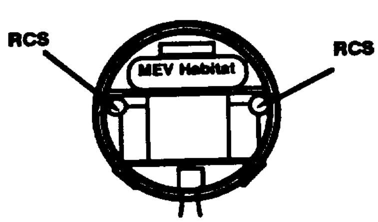




**Figure 9.5.4:** MEV RCS pod locations

In order to provide a better idea of how much thrust the pods can provide, below are two tables and a graph showing the relationship between the thrust per pod and the time to complete a 90 degree pitch, yaw, or roll maneuver. The following are the assumptions made in this calculation.

1. The biconic may be approximated as a solid cylinder (for moment of inertia calculations).
2. A mean vehicle mass of 60 metric tons (this assumes that the hab has broken orbit hence burning most of its fuel).
3. The pods act as couples in yaw and pitch maneuvers (ie when pod 3 fires, pod 1 fires, see Figure 9.5.4).
4. All pods fire in roll maneuver.
5. No external forces are acting on the HAB.

The following table is for yaw and pitch maneuvers through a 90° arc.

| Thrust per pod (N) | Mcg (Nm) | Angular Acc. ($s^{-2}$) | Time (sec.) |
|---|---|---|---|
| 110 | 1320 | .000615 | 71.47 |
| 3870 | 46440 | .0216 | 12.06 |
| 7740 | 92880 | .04328 | 8.519 |
| 11610 | 139320 | .0649 | 6.9547 |
| 15480 | 185760 | .0866 | 6.0 |

**Table 9.5.1:** MEV pitch and yaw response times

The following table is for roll maneuvers through a 90° arc.

| Thrust per pod (N) | Mcg (Nm) | Angular Acc. ($s^{-2}$) | Time (sec.) |
|---|---|---|---|
| 110 | 1760 | .003667 | 29.269 |
| 3870 | 61920 | .129 | 4.934 |
| 7740 | 123840 | .258 | 3.4895 |
| 11610 | 185760 | .387 | 2.849 |
| 15480 | 247680 | .516 | 2.46745 |

**Table 9.5.2:** MEV roll response times

Note that pitch and yaw are identical maneuvers, they only differ in reference frame. After some consideration, the Propulsion Group decided to utilize 15 thrusters per pod (five radially, and two banks of five tangentially). Adding more thrusters soes not significatly decrease the response time of the vehicle as seen in the graph on the following page. The estimated subsystem mass for the biconic RCS is 3mT.

### 9.5.4 MEV Crew Descent Stage RCS

The difficulty in designing the RCS for the MEV is deciding where to place the thrusters. Obviously, thrusters will be needed on the pod itself for ascent and docking maneuvers. However, during decent the pod rests nose down inside the aeroshell rendering the thrusters on the pod virtually useless. It is the opinion of the propulsion group that the best way to pitch, yaw, and roll the aeroshell/pod assembly to maintain stable descent is to place thrusters on the aeroshell itself.

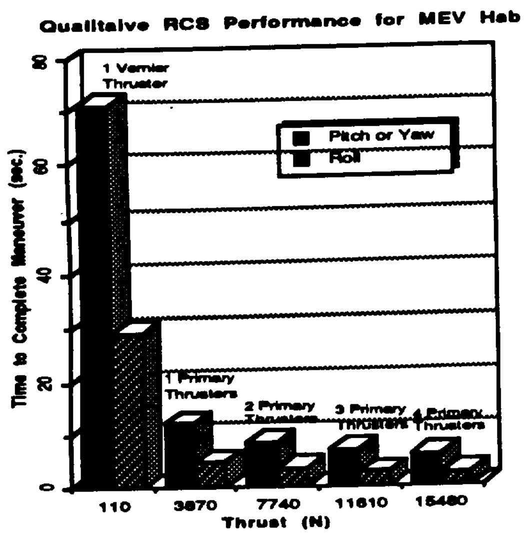



**Figure 9.5.5**: Graphical analysis of MEV response times

By placing four thruster pods at the locations shown in Figure 9.5.6, the thrusters should be able to provide adequate restoring moments to the aeroshell/pod assembly should it encounter any perturbations during its decent. If a single thruster fires, the MEV could perform a 45° pitch in 5.9 seconds. If three thrusters fire, this time would improve to 3.46 seconds. For redundancy and sufficient maneuver response time it is the recommendation of the propulsion group that pods located at the narrow ends of the aeroshell (the "top" and "bottom" in the following diagram) consist of 9 thrusters. Three will be oriented perpendicular to the plane of the aeroshell, and six parallel to the plane of the aeroshell (two banks of three opposite of each other. These two pods will provide pitch and yaw moments. The remaining two thruster pods will contain two thrusters each oriented perpendicular to the plane of the aeroshell to provide roll moments.

The propulsion group is estimating an RCS system mass of 600kg for this stage of the MTS. Please note that this is for decent only. The RCS for ascent and the MEV biconic will be discussed in the following section.

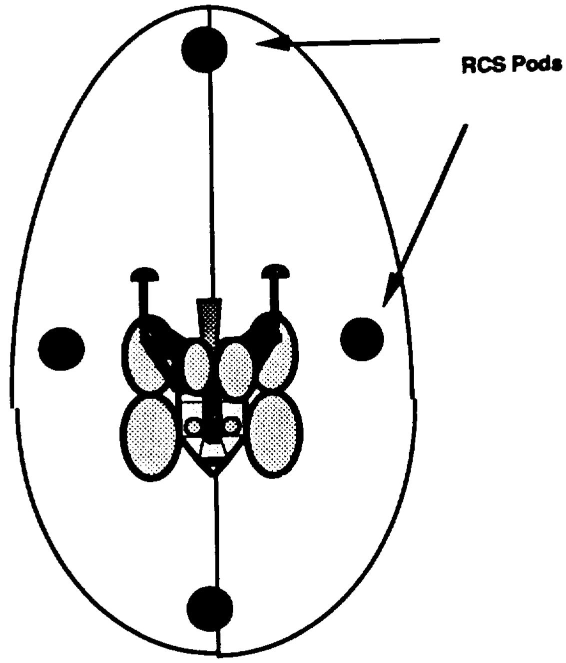



**Figure 9.5.6**: Aeroshell RCS placement

### 9.5.5 Crew Pod RCS

Since the pod is so small, the RCS pod concept suggested for the habitation biconic is not applicable. Instead of the thrusters being located beneath the hull, the thrusters will be located outside the hull on an arm as suggested by the Structures Group. The propulsion group suggests that two of these arrays be located as shown in Figure 9.5.6 and 9.5.7 below.

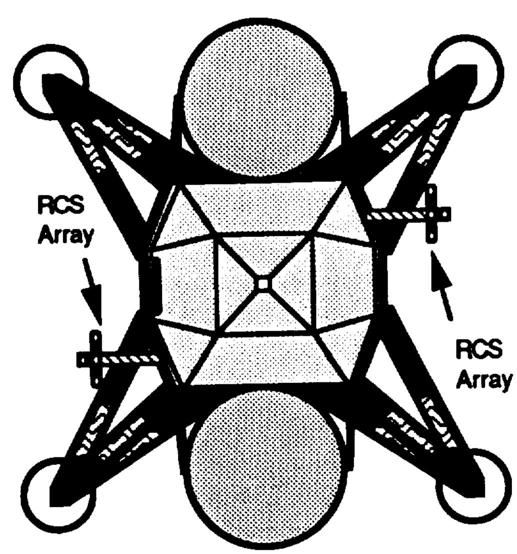



**Figure 9.5.7**: Top view: MEV crew pod

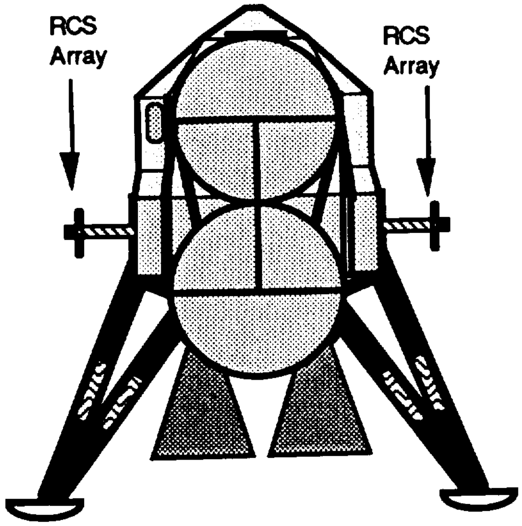



**Figure 9.5.8:** Side view: MEV crew pod

Each array will consist of 10 thrusters, two in each of the principle directions (see above). In a pitch/yaw maneuver with one thruster per array firing and both arrays firing as a couple, the lander could complete a 90 degree pitch or yaw in 4,58 seconds. Such a maneuver would utilize 3.14 kilograms of fuel. For saftey's sake, the propulsion group estimates that 300 kg of fuel should more than adequately cover all maneuvers during and after ascent. This amount of fuel would require approximately .3 cubic meters of tankage.

## 9.5.6 MTV RCS

This section of the report addresses the subject of Reaction Control for the MTV and its use in the generation of artificial gravity and course corrections.

### 9.5.6.1 Artificial Gravity During TMI

Two RCS pods will be placed on the truss to produce the moments needed for artificial gravity (see Figures 9.5.9 and 9.5.10).

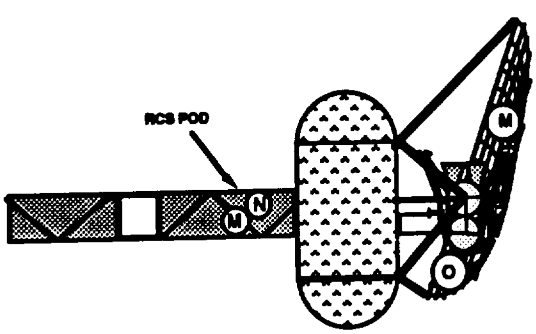



**Figure 9.5.9:** MTV forward RCS pod

To achieve a gravitation acceleration of 9.81 m/s^2 a rotation of about .36 radians per second (that is 3.45 revolution per minute) would be needed. If two pods fire as a couple to start or stop rotation with all thrusters in a pod firing then the time required to achieve a rotation of .36 radians per second can be calculated. The spin up time for TMI is 22.64 minutes. Using the parameters of the thrusters provided by reference one the system mass stands at 11 metric tons. As a consequence, the pod concept will be unusable for the MTV due to this high propellant mass. A decision was made to place the RCS propellant tanks at a central location to be determined by the layout and Structures groups. The propellant would then be piped to the thrusters from this location.

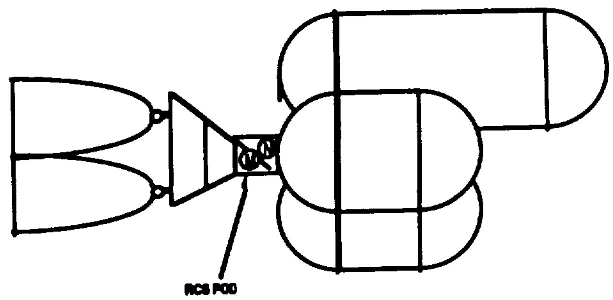



**Figure 9.5.10:** MTV aft RCS pod

The two thruster pods will contain a total of 8 thrusters. Two thrusters each for pitch and yaw in both directions. Only one thruster will fire in each maneuver, the second thruster is for redundancy.

### 9.5.6.2 Course Corrections

It is not be feasible to attempt a course correction during rotation. Any unbalanced thrust created by the RCS system would cause a parametric type motion about the center of gravity of the MTV. This would serve only to confound the original course problem. Hence, after the Venus swingby, an additional spindown will be needed to make a two degree course correction to account for the plane change between Mars and Earth. After the spindown, the plane change will be executed using the main engines. After the plane change, the MTV will spin up again and resume TMI. This maneuver will require another 4 metric tons of RCS propellant.

In the event that the MTV looses its course during an injection, there is little that can be done, fuel for another spin up and spin down is a costly mass penalty as seen above. The propulsion group offers the following solution to this problem. In the event a course correction is needed, the MTV will continue on its course, spin down early, make its correction using the main engines, and proceed with the rest of the injection without artificial gravity. How soon the MTV should spin down depends on the severity of the course change. A large correction would require spin down well before the destination is reached forcing the crew to go without artificial gravity for a certain amount of time.

## 9.6 Propellant Mass and Volume Estimates

This section will discuss the important topic of propellant masses and volumes. The discussion will be divided roughly into three parts. At first, comparison among four different scenarios for the first, second and third mission. Second, is to show the mass requirement for all three missions using each scenario independently. Third is to choose the most feasible option and scenario for the entire mission.

### 9.6.1 Defining the Options

The Orbital Mechanics group made calculations for the delta Vs required for each mission. Each mission has a different launch time and a different required delta V. All reasonable launch times within each mission will be referred to as options. Launch Window #1 (2016) contains 13 different options, launch window #2 (2018) contains 5 different options and launch window #3 (2022) has 7 different options.

### 9.6.2 Defining the Scenarios

The Propulsion Group developed a spread sheet to calculate the propellant mass and volume requirement for each mission. Four different scenarios were considered for this mission:

Scenario 1: EOI using the NTR and save Truss and MTV Habitat.

Scenario 2: EOI braking MEV crew Pod.main engines (refueling at Martian orbit after return is required)

Scenario 3: EOI using MEV crew pod main engines (no refueling required in martian orbit)

Scenario 4: EOI using MEV crew pod aerocapture using an aerobrake.

### 9.6.3 Comparison of Scenarios.

The following three figures show a comparison among the different scenarios defined above for each mission. Each figure set follows the same trend for each mission. However, the graph characteristics and the number of options within each mission are different. The reason for showing the comparison below is to have a solid argument once the most feasible scenario is selected.

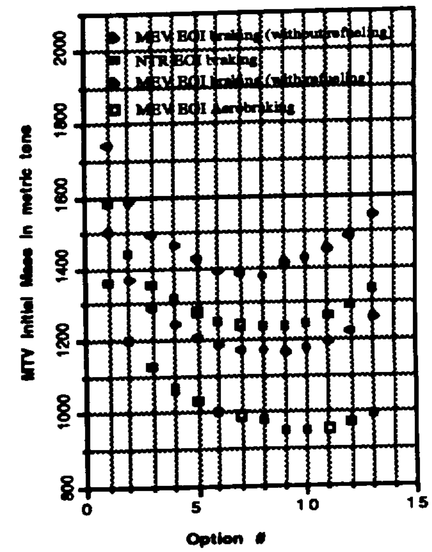



**Figure 9.6.1:** First mission comparison

Looking at the figure above, one can clearly conclude that the scenario utilizing the smallest amount of fuel for the entire mission is scenario 4. Scenario 4 is the one where the MEV will aerobrake in low earth orbit. 300 metric tons will be saved having chosen the last scenario in comparison with using the NTR to propulsivly brake. The disadvantage of such choice is loosing the truss, MTV Hab., and the NTRs. However, the propellant mass savings might overcome this disadvantage since the cost of bringing propellant and fuel tanks to low earth orbit is very expensive. The best options for the first missions are options 6 through 11 since the total mass required doesn't vary too much.

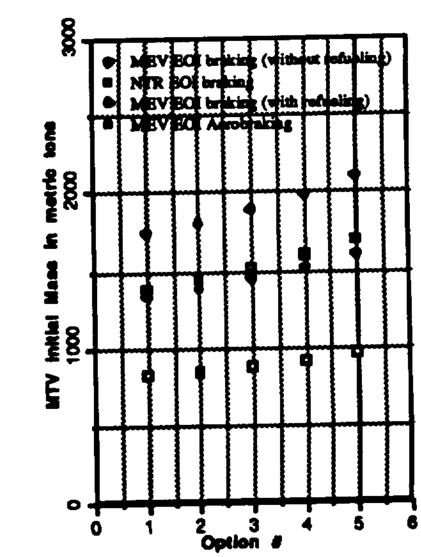



**Figure 9.6.2:** Second mission comparison

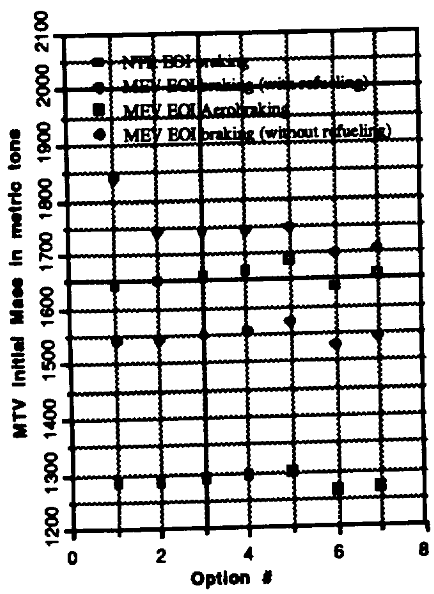



**Figure 9.6.3:** Third mission comparison

The same argument made for the first mission could be applied on the second and the third missions, but the total mass required is the higher for both of these missions. As a result, choosing the MEV aerocapture scenario is the best for all three missions.

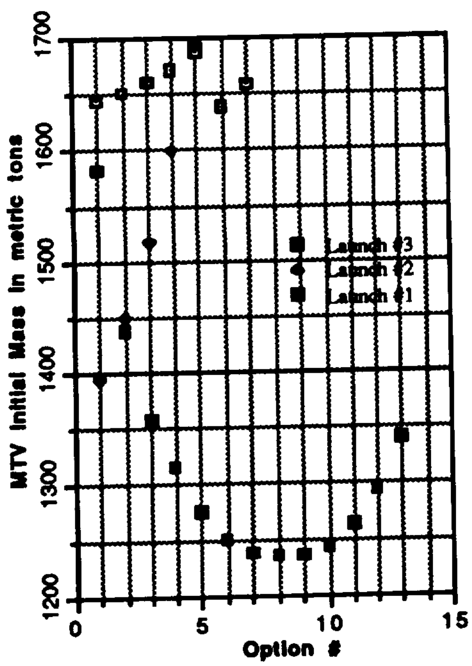



**Figure 9.6.4:** MEV EOI (Scenario 1)

Using the NTR EOI scenario for all three missions, mission three is about 500 metric tons higher than mission 1.

The overall mass for each mission went beyond a reasonable mass due to the requirement of propulsivly braking at Earth with everything included.

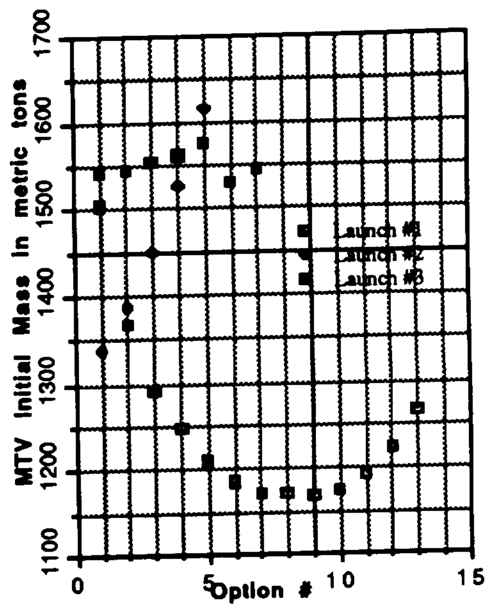



**Figure 9.6.5:** MEV EOI (Scenario 2)

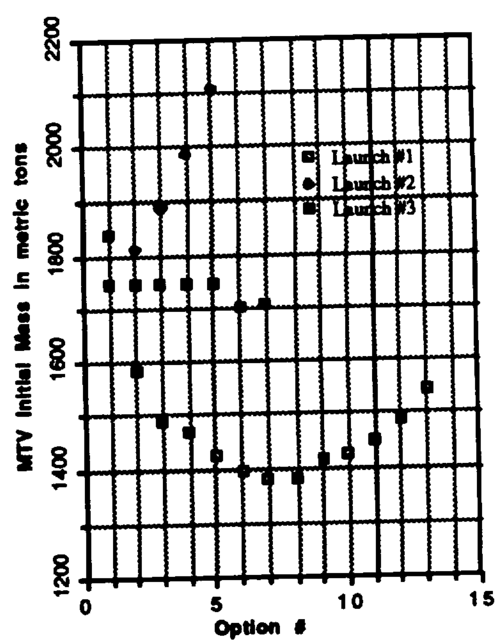



**Figure 9.6.6:** MEV EOI (Scenario 3)

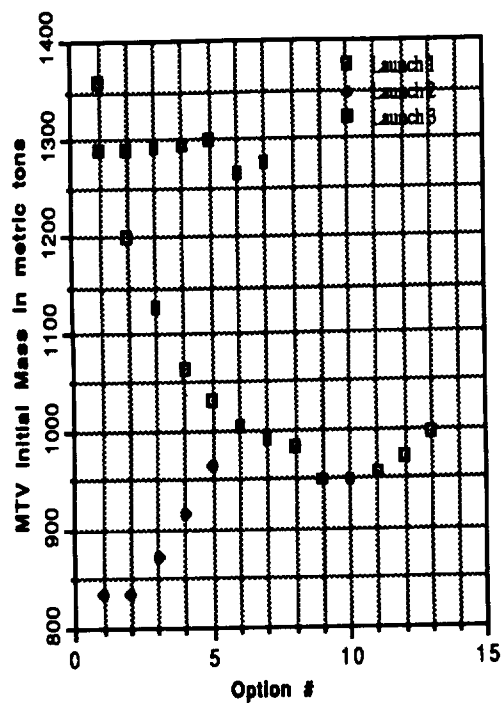



**Figure 9.6.7:** MEV aerocapture (scenario 4)

Notice from all figures above that the best propellant saving Scenario is to aerobrake at earth. By choosing this scenario for all three mission the fuel saving might reach a 1000 metric tons in comparison with scenario 1 or 3.

## 9.7 Burn Times

Below is a table showing the burn times necessary to achieve the required delta Vs for the first MTS mission.

| **Maneuver** | **Burn Time** |
|---|---|
| MTV: | |
| TMI (Option 9) | 8703.51 sec |
| TMI (Option 9) | 4569.98 sec |
| TEI (Option 9) | 3037.91 sec |
| MEV Biconic: | |
| Deorbiting | 734.67 sec |
| Orbital Escape | 1333.33 sec |
| Braking | 688.27 sec |
| MEV Descent Stage: | |
| Deorbiting | 427.12 sec |
| Orbital Escape | 765.79 sec |
| Braking | 151.08 sec |
| MEV Ascent Stage: | |
| Achieve Circular Orbit | 629.75 sec |
| Match MTV Orbit | 62.85 sec |

## 9.8 Conclusion

This concludes the analysis of the propulsion systems for the MTS. To recap the major points of this section, a Low Pressure NTR was chosen as the main engine for the MTV, a $LO_2$-$LCH_4$ rocket engine was chosen for the main engines of the MEV, a modular chemical RCS system modeled after the space shuttle was used for all stages of the MTS, and finally, it was decided that after EOI the crew would aerobrake at Earth to end their mission.

# 10.0 STRUCTURAL ANALYSIS

## 10.1 Introduction

The Structural Analysis discipline was responsible for the following tasks over course of the past nine months:

- Selection of materials
- Micrometeorite protection
- Thrust structure
- Truss design
- Subsystems mounting
- Propellant tank sizing
- MTV crew quarters
- Aerobrake structure
- Landing gear
- Mars ascent / descent module

Other tasks included thermal protection, particularly for the propellant tanks, and docking mechanisms, for attachment to Space Station Freedom and passage between various components of the vehicle. Thermal protection was dealt with primarily by Thermal Analysis, so it will not be discussed here except as it pertains to propellant tank sizing. Docking mechanisms were not dealt with due to a lack of time, but there is no reason to believe that there will be any difficulty in performing any of the scheduled docking maneuvers.

## 10.2 Selection of Materials

### 10.2.1 Selection Process

The first task which was undertaken was research into materials which would be used on the vehicle. Many different references were used to obtain a variety of materials, then the temperatures, stresses, and other factors were considered in order to determine which materials would be appropriate for each component of the vehicle.

### 10.2.2 Recommended Materials

Following is a list of some of the components and the materials selected for each of them. Explanations of the reasoning behind the material selection, as well as alternative materials, are also discussed.

**Truss**

Material: Boron-Aluminum MMC
Alternatives: Titanium or TiAl alloys

Although aluminum is recommended for almost any application on this mission because of its low density and good strength, it cannot be used on components which will be exposed to extremely high temperatures or stresses. For high-temperature applications, titanium or titanium-aluminum alloys should be used, while Boron-Aluminum Metal Matrix Composite (MMC) is ideal for high-stress situations. Since the original aluminum truss design was unfeasible, the material was changed to B-Al MMC, as discussed in Section 10.3.

**MTV Habitat**

Material: Aluminum
Alternatives: Boron-Aluminum MMC

Boron-Aluminum Metal Matrix Composite would be almost ideal for this application (simple cylindrical structure) because it is extremely strong, stiff, and lightweight, but it is also extremely expensive to work with - the boron fibers alone cost $700/kg to produce.

**Propellant Tanks**

Material: Kevlar-49
Alternatives: None recommended

As stated below in Sections 10.3 and 10.6, Kevlar-49 is used for this application because of its high strength, resistance to micrometeorite damage, and low density. Original plans called for the tanks to be made of E-Glass, but concern about the reaction of this material with liquid hydrogen forced the change to Kevlar.

**Micrometeorite Protection**

Material: Nextail (if necessary)
Alternatives: Betacloth

Micrometeorite protection will most likely not be necessary except during the construction phase of the mission, which is discussed in Section 10.3. One proven material for these purposes is Betacloth, a Teflon-impregnated glass cloth which was used most notably during the Viking mission to Mars.

**Aerobrake Structure**

Material:Graphite/Epoxy Composite
Alternatives: Boron-Aluminum MMC, Aluminum

As in many other applications, Boron-Aluminum Metal Matrix Composite would be ideal for this low-temperature, high-strength area. However, a lack of quantitative information about B-Al MMC forced the change to Graphite/Epoxy for both the aeroshell and biconic structural designs.

## 10.3 Micrometeorite Protection

For the journey to Mars one fact is clear: the MTV *will* be hit by micrometeorites. The good news is that they are not likely to penetrate any of the pressure vessels using the current materials and thicknesses. We make this statement after examining NASA's experiments on the subject. It was found that, using the equation[10.10]:

$$t = K_1 \, r^{0.1667} \, m^{0.352} \, V^{0.875}$$

a meteorite with a mass m of $10^{-6}$ kg (the average size) traveling at an average velocity V of 20 km/s will not penetrate a sheet of aluminum that is thicker then 0.052 cm. The wall of the MTV Habitat is 0.5 cm of aluminum, almost a factor of ten greater then the threshold of penetration. The Kevlar fuel tanks are 2.5 cm thick and should not sustain damage because of improved impact qualities over aluminum (lower $K_1$ value).

Protection is needed from orbital debris in low earth orbit during the construction phase. The vehicle will be built at Space Station Freedom, and NASA is planning to maneuver the station to avoid large objects. For smaller objects the station will be built to survive these with occasional touch up maintenance. To protect the MTS from this smaller threat, two ideas were presented. The first was to wrap the MTS with a micrometeorite cloth Nextail (preferred over Betacloth because it is more resistant to high temperatures). However, this adds weight that is not needed for the journey. A preferred method of protection is to construct a hangar out this cloth. This could be accomplished by hanging the fabric over the existing SSF truss which will be used to assist in the construction of the MTS.

This will require approximately 14.7 mT of material to make the four 160 m by 40 m sides of the hangar. Because of the low atmospheric density at that altitude, the drag force from this added area is minimal. Since masses fall at the same rate in a vacuum environment the added mass should not effect the orbital path of SSF. The space station is scheduled to be reboosted every 90 days in any case[10.14]. Because of the hangar it should take slightly more fuel to reboost the station back into its proper orbit.

## 10.4 Thrust Structure

Figure 10.3 shows the present design for the thrust structure. It consists of a 0.05 m thick 10 m-equilateral triangular slab of Boron-Aluminum Metal Matrix composite[10.15]. The triangle was sized so that the nozzles of the engines can be vectored a maximum of five degrees. This criteria was set so that if one engine fails the other two nozzles can be vectored so that the thrust will still be applied through the center of mass of the MTS.

This triangular slab is connected to the truss by three bars. The three which connect the mounting block to the truss are the truss are 6.22 m long. There are also three bars connecting the midpoints of these members. This arrangement has safety factors of 13.8 in tension and 10.03 in compression, with a buckling safety factor of 2.99. Saving mass was the important consideration, so the bars' radius and thickness were minimized to 0.062 m and 0.003 m respectively. The mass of each bar at this size is only 82 kg, which brings the total mass of the thrust structure to 6.5 mT.

## 10.5 Truss Design

The initial design for the truss had a length of 300 meters. A dynamic analysis of this truss under rotation to produce artificial gravity showed that it would fail at a location 101 meters from the MTV Habitation Module. To solve this problem two changes were implemented. The first was shortening the truss to a length of 150 meters, and the second was switching material to Boron-Aluminum MMC. With these two changes the oscillations that occurred during rotation damped out.

Though the overall length was changed, the same configuration was used for the truss members: a 3 m equilateral triangular cross section and truss members at 45 degree angles supporting the longitudinal members. This has proven to be an exceptionally strong design. A finite element model on I-DEAS™ of the truss in tension shows that it stretches only 25 cm over its entire length. When the FEM solved for the firing of the main engines it showed a compressive deformation of no more than 7 cm.

As seen from the above numbers the tension load is the dominant load and was the main driver in the truss design. The reason for this high tensile load is the 471 mT of fuel being swung around during rotation. With the estimated fuel mass the truss members were evaluated and had to be increased in size in order to take the load, which then increased the mass of the truss by a factor of five. The mass estimate for the truss is 15.7 mT. All members are hollow tubes with a radius of 0.13 m and a thickness of 0.002 m. This gives safety factors of 1.3 in tension and 65.5 in compression.

In each 30 m section of the truss the members are rigidly connected. The sections will then be connected to each other using precision titanium ball joints[10.11]. One of these will be used to join each longitudinal member to the corresponding member of the next section. This arrangement will then require three joints to join two sections together. With this setup and the low thermal expansion of Boron-Aluminum MMC it is expected to expand thermally (due to a 100° C temperature change) a distance of 0.09 m over its entire length.

The whole point of the truss is to provide a means of artificial gravity by rotating two masses at opposite ends. After TMI a gravity of 1 G was chosen for the comfort of the crew. This gravity is produced by a rotation rate of 2.87 rpm. After TEI a gravity of 0.5 G was chosen by Crew Systems, which is produced by a rotation rate of 3.25 rpm. The lower gravity is a reflection of the lower mass that counters the mass of the MTV habitation module. Both of these rotation rates are within the comfort zone as outlined by Crew Systems. The radius arms for these are 108.45m and 42.2 m respectively.

The first four natural frequencies of the truss are 5.0, 32.1, 88.45, and 170.42 Hz. The only way to raise these would be to shorten the truss. Crew Systems does not approve of this solution because it would reduce the amount of artificial gravity available for most of the trip, possibly eliminating it entirely. After consulting with

Avionics and Propulsion a dynamical system has been devised to counteract these oscillations. Accelerometers will be mounted on the truss, and when the oscillations get close to the first natural frequency the RCS system will fire to damp the oscillations. These oscillations will only occur during the start-up and stopping of rotation for production of artificial gravity. Once the system has reached an equilibrium in rotation, there will be no oscillations.

## 10.6 Subsystems Mounting

Subsystems mounting deals with the attachment of various components of the vehicle, including the biconic, aeroshells, and lander. Different disciplines dealt with the attachment of different components, so this section will deal solely with the components dealt with by the Structures discipline.

### 10.6.1 Docking Tunnel

The docking tunnel will bridge the gap between the habitation module and the lander. This tunnel has to be large enough in diameter to allow a fully suited astronaut to crawl through. Crew Systems set the diameter at 1.2 m. The tunnel has to be 3 m in length to bridge the gap. Once this part of the design was set an analysis was performed on the structure integrity.

The tunnel is to be made out of Boron-Al MMC. This provides the characteristics of light weight and high strength. With the previous dimensions it was determine a thickness of .002 m will be adequate to provide the need strength. The safety factor for buckling on this design is 2063. Safety factors for compression and tension follow along at 19,953 and 292 respectively. This yields a mass of 401.1 kg.

Inside the tunnel will be a pulley system to assist the astronauts in crawling through to the lander.

### 10.6.2 Aeroshell and Biconic Supports

These supports are to be used to connect the aeroshell and the biconic to the truss. The material chosen for these supports is also Boron-Al MMC again because of its light weight and its strength. To ease the cost of manufacturing and tooling it was also decided to make all these members with a common radius and thickness. The radius is .056 m and the thickness required is .003 m. The worst case scenario was analyzed. This case happens with the biconic with its added weight. The members were found to have safety factors in compression, tension, and buckling of 5.00, 4.39, and 2.06 respectively. By examining these safety factors it is easy to see the main design driver was the buckling and the members were sized according to this failure mode.

The preliminary arrangement of the members is a pyramidal shape for each component. For the Earth aeroshell there is a member 3 m in length that runs straight down from the top of the aeroshell to the middle of the truss. The next two members are 20 m in length and run from the top of the aeroshell to the edge of the truss. This arrangement occurs at both ends of the aeroshell.

The biconic has a similar arrangement as the aeroshell except for the lengths of the members. The straight members are only 1 m in length and the other two members are 20 m in length to account for the added distance they transverse to reach the top side of the truss. The arrangement can be seen in Figure 10.4.

## 10.7 Propellant Tank Sizing

The sizing of the fuel tanks has become a spreadsheet exercise which has allowed the Structures group to easily calculate optimal tank dimensions. A full description of the spreadsheet can be found in Appendix A10.1.

The large main engine fuel tanks are cylinders with hemispherical endcaps. All other fuel tanks (MADM and biconic tanks, RCS, etc.) are spherical. The six large main engine tanks all have the same diameter for both layout and manufacturing reasons. In addition to the six cylinderical main tanks there is an additional spherical tank which carries TEI fuel. This was necessary because of the large fuel requirements of the TEI burn and a limited launch capacity. The tank sizes were constrained by shroud size and launch capacity of the anticipated heavy launch vehicles (30.4 m long by 14 m diameter, 220 mT maximum payload). The tanks are sized to contain enough fuel for options 5 through 13 of the first mission, to include extra fuel to account for boiloff between each tank's launch into LEO and the time that its fuel is burned, and to account for the changes in mass of the MTS during the course of the mission (tanks are jettisoned when they are emptied, loss of components at Mars, etc.). Exact dimensions and masses are provided in the Systems Layout section of this report. For the second mission there are only five main fuel tanks and for the third mission there are again six main fuel tanks.

The tanks are made out of Kevlar-49 because of its high strength, ability to dissipate kinetic energy (minimize micrometeorite damage), and low density. Super-Insulation is used on the inside of the Kevlar shell to minimize the boiloff of the liquid hydrogen and prevent any possible reaction between the fuel and the tank.

## 10.8 MTV Habitation Module

After many discussions with Crew Systems and Thermal Analysis a shell and storm shelter for the crew habitation module that should satisfy all concerned disciplines has been developed. The storm shelter actually fulfills a number of requirements at once: it provides radiation protection during solar flares, acts as the habitation module's airlock, and is the structure that bears the load of the Mars Ascent/Descent Module (MADM) during transit. The storm shelter is a box 2 m long by 5 m wide by 2.3 m high that as located in the center on the lower level (see the Crew Systems section for the exact location).

Thermal Analysis set the requirement of 7.5 cm of aluminum to provide the necessary shielding against solar flares. They also set the criteria that the overall shell of the crew habitation module should be made so as to not interfere with the galactic cosmic radiation, i.e. be thin enough to minimize secondary particle emissions. This contributed to the shell being made of 0.5 cm of aluminum.

The storm shelter will contain two docking ports. One, on the side directly opposite the truss, will be occupied by the MADM on the trip to Mars and the MEV ascent module on the return voyage. Another docking port will be located on a "side" of the crew habitation module. This docking port is to provide access to the crew habitation module when the MTS is docked at Space Station Freedom.

The storm shelter is also the major load-bearing structure for the MADM and its aerobrake. The structure will bear a compressive load during the main engine firing, a moment during the initiation and spindown of artificial gravity, and a tensile load due to the rotation necessary for artificial gravity. The MADM and its aerobrake are stabilized by additional small trusses connected between the ends of the aerobrake and the ends of the cylinderical part of the habitation module. These small trusses reduce the moment applied to the primary load bearing member, the docking tunnel.

## 10.9 Aerobrake Structure

After the Mars Orbital Injection burn places the MTV in an elliptical orbit around Mars, two vehicles will land separately on the surface: the Mars Habitation Module (with rover and cargo) and the Mars Ascent / Descent Module. The majority of each vehicle's deceleration will be done with the aid of an aerobrake. The term aerobrake refers to any of several types of bodies which are capable of aerodynamically decelerating a payload into a planetary atmosphere. Each of the two landing vehicles will use a different type of aerobrake. The MHM will be contained inside a biconic aerobrake, and the MADM will be attached to an aeroshell. This section gives an overview of the design considerations which will be necessary for each type of aerobrake.

### 10.9.1 Aeroshell Design

Several references[10.2,12,13] have been extremely helpful in developing a conceptual design, though all of them idealized the aeroshell as being flat. Once the design parameters were finalized, the actual construction of a model proved extremely helpful in correcting the errors due to the flat aerobrake assumption.

The basic design of an aeroshell is fairly simple, and is shown in Figure 10.5. Typically, there is a "transitional truss" on which the vehicular components are mounted. This transitional truss is then connected at a certain number of nodes to the aerobrake truss. It is advantageous to attach these two together at as many points as possible, since this minimizes the internal member stresses of the aerobrake truss. Then, on the outer surface of the aeroshell support truss, hexagonal stiffener panels are attached, which fit together like honeycomb cells. These panels need to be stiff yet lightweight (though they typically account for more than 70% of the structural mass in any case), so are made of thin face sheets kept apart by a thick layer (>10x the face sheet thickness) of a honeycomb material. Finally, the outer surface of the hexagonal panels is covered with some sort of Thermal Protection System.

There are several differences between the designs proposed in the references and the design which we are proposing. The references assumed the aerobrake would be assembled in space[10.2], so they tended to stress minimizing assembly complexity, i.e., using as few pieces as possible. The design constraint for one of them was built solely around the panel size, which was simply the largest size which would fit inside the Space Shuttle bay. This large panel necessitated long truss members, which in turn determined that the elements themselves would have to be very thick to withstand buckling.

It is in fact much better to use small panels should it be possible to do so, for several reasons. First, since the truss member length and panel diameter are inherently linked ($a = \frac{L_s}{\cos 30°}$), reducing the panel size would reduce the strut length, and hence the strut diameter necessary to avoid buckling. Second, and more importantly, reducing the distance that each panel must span reduces the panel thickness. Since the panels account for the majority of the mass, this creates the greatest mass savings. Since the aerobrake will be assembled on earth and shipped up within a single shroud, it seems that the added complexity associated with smaller heatshield panels would be more than made up for by the mass savings.

One further difference is that there is no plan for a transitional truss to be employed with the

MADM; it will be attached directly to the aerobrake truss. This does not necessarily cause any difficulty in the design of the truss, but it means that the method of attachment must be much stronger than the rest of the truss, and care must be taken to minimize the loads applied to the lander.

Although the aeroshell was not modeled on a computer (due to both complexity and lack of time), the final dimensions of the structure have been decided upon. The same strut diameter will be used for both the Mars and the Earth aerobrakes, because even though the Earth aeroshell will be smaller, it will most likely be subject to a higher pressure loading due to the greater density of Earth's atmosphere. Since the actual size of the Earth aeroshell was chosen by what "seemed right" - simply 2/3 the size of the Mars aeroshell - this does not seem like a large concern, especially when the masses of these aerobrakes are a small fraction of the total vehicle mass.

The Mars aeroshell will have a nominal strut length of 1.5 m. Each strut will be made of 12-ply -10°/+10° Graphite-Epoxy laminate (r = 1744 kg/m$^3$) yielding a wall thickness of 1.525 mm, with a strut diameter of 10 cm. Each strut will have a mass of 1.25 kg, and the truss will extend a normal distance of 1.3 m from the aeroshell, meaning that the aeroshell will effectively be 1.7 m in depth rather than the nominal value of 3 m. This strut length gives a heatshield panel diameter of 1.732 m. Each face sheet will be made of 10 cm of aluminum honeycomb (r = 64 kg/m$^3$) sandwiched between two 0.5 cm sheets of aluminum (r = 2700 kg/m$^3$), yielding a 0.11 m thick panel with effective density of 304 kg/m$^3$. Each panel will thus have a mass of 65.16 kg.

Aeroshell truss design typically deals with concentric rings of tetrahedrons, and using these lengths, this design has between 4 and 7 rings, the variation being caused by the asymmetry of the aeroshell. Assuming the structure is approximately represented by a 5-ring truss, an initial estimate of the mass of the Mars aeroshell was 5740 kg without TPS.

The Earth aeroshell was designed in the same manner as the Mars aeroshell, and since it is simply 2/3 the size, the strut lengths will also be 2/3 those of the Mars aeroshell, or 1 m in length. This gives masses of 0.836 kg/strut and 28.96 kg/panel, yielding a initial mass estimate of 2740 kg without TPS.

Both of these masses were likely to be low, since neither accounted for the curvature of the aeroshell. In addition, these models only accounted for a hexagon-shaped aeroshell, ignoring the areas near the edges. Assuming an extra 35% of mass due to these considerations, more accurate mass estimates would be 7750 kg and 3700 kg for the Mars and Earth aeroshells, respectively. In fact, these match very well with estimates of 7746 kg and 3690 kg, which were obtained after building a physical model of the aerobrake structure.

**10.9.2 Biconic Aerobrake**

The structure is composed of three main parts: skin, stringers, and rings. This is shown in Figure 10.6. The skin of the biconic is made of aluminum honeycomb (1 cm thickness, r=64 kg/m$^3$) sandwiched between two aluminum face sheets (0.05 cm thickness each, r=2700 kg/m$^3$). This skin is not designed to sustain an internal pressure; instead, it was assumed that the Mars Habitation Module, which occupies a portion of the interior of the biconic, was self-contained. In other words, the MHM needs to withstand an internal pressure, but the only loads it transfers to the biconic are due to its mass.

The rings and stringers will be made of Graphite/Epoxy composite (r=1744 kg/m$^3$), with cross-sections as shown in Figure 10.1. The stringers run axially along each of the tapered sections of the biconic, with a spacing of 0.25 m at the smaller end of each section. Due to the tapering of the biconic, these stringers will be slightly farther apart at the wider end of each section. This spacing calls for 91 stringers in the main body and 13 more in the nose area, excluding the actual nose cone.

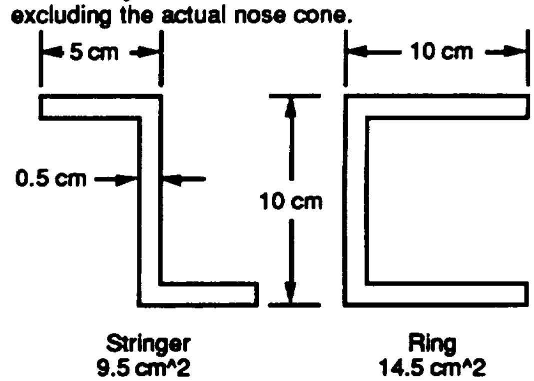



**Figure 10.1:** Stringer & ring cross-sections

The final part of the design is the "back side" or rear end of the vehicle. This was designed in the same manner, with a 9.5 m (diameter) section of honeycomb/facesheet supported by stringers spaced 0.5 m apart along a diameter. This, in addition to allowing for slightly more material for the nose cone, added approximately 500 kg to the total mass.

The masses for the various components are:

| | |
|---|---|
| Total Ring Mass | 2320 kg |
| Body Stringer Mass | 2460 kg |
| Rear Stringer Mass | 230 kg |
| Rear Skin Mass | 250 kg |
| Body Skin Mass | 1670 kg |
| **Total Mass** | **~7000 kg** |

This does not allow for any irregularities in the structure (such as holes for retracting engine nozzles), nor does it include thrust structure for the engines or specific support structure for the MHM, rover, fuel tanks, etc.

## 10.10 Landing Gear

As this mission has obvious relation to the Apollo missions, research was started on the landing gear systems for the Apollo lunar lander.

### 10.10.1 Energy Absorption

The first requirement to a safe landing is a gentle touchdown. The kinetic energy must be absorbed into the system without repercussions. Hydraulic shock absorbers are very unreliable in a space environment because the extreme temperatures cause valve and viscosity problems. Hydraulics are often very heavy, especially with the very high pressures imparted during landing[10.4] As these problems are definitely unacceptable, a honeycomb-core shock absorber was investigated. Crushable honeycomb shock absorbers are a very reliable and lightweight means of absorbing the impact energy. They can be readily made to crush at any desired pressure. They are a one use only absorber, exactly as is required.

Once the weight of the vehicle, maximum G loading desired, impact area, and touchdown velocity are known, an exact weight and displacement can be determined[10.7]. First, the minimum core thickness can be determined from the formula:

$$t_c = \frac{1}{0.7}\frac{v^2}{2gG}$$

where g is the gravitational constant of Earth, and G is the Earth G loading. This formula assumes that 70% of the honeycomb can be crushed. It is simply the equation for position vs. acceleration. Crush strength is given in terms of pressure, and is thus calculated as:

$$f_{cr} = \frac{mgG}{A}$$

where A is the impact area. From this value a standard honeycomb material can be selected. Performing this calculation for the lander with m=25,990kg, G=3.0, v=2m/s, and A=0.38m$^2$, the stopping distance was calculated at 0.097m, with a $f_{cr}$=2.00 MPa. A honeycomb with this crush strength will have a density of approximately 64 kg/m$^3$ (4 lbs/cu ft). This yields a honeycomb mass of about 3.65kg (8.05lbs).

Note: the lander has been designed with a core thickness of 1.00 m which will absorb energy at a constant 3G's for touchdown velocities up to 3.58 m/s.

### 10.10.2 Static Stability

Static stability is the amount of stability the lander will have once it has come to rest on the uneven Martian surface. It is very easy to determine the stability using a graph which charts static stability angle versus the ratio of center of gravity height to landing gear radius. For acceptable static stability, a good design estimate is to have the landing gear radius greater than or equal to the center of gravity height. This was a key driving factor in the design of the lander, and is detailed in Section 10.10.1.

### 10.10.3 Strut Design

In designing the main struts for the landing gear of the biconic and lander, both weight and reliability needed to be considered. To maintain a low weight for the biconic, the core thickness of the honeycomb absorbtion system was increased in order to decrease the touchdown G force, thereby decreasing the cross-sectional area and mass. However, simultaneous protection against Euler buckling and thin-wall buckling is difficult with this area and length, as a cylinder with the required diameter (34cm) and area is only 1.1mm thick. This led to the cross-section shown in Figure 10.2. Thin-wall buckling is eliminated, with a minimal mass cost, as the walls are not considered to be axial load-bearing members. This cross section also proved to be useful on the lander, where the radius/thickness ratio is around 60 for a circular profile.

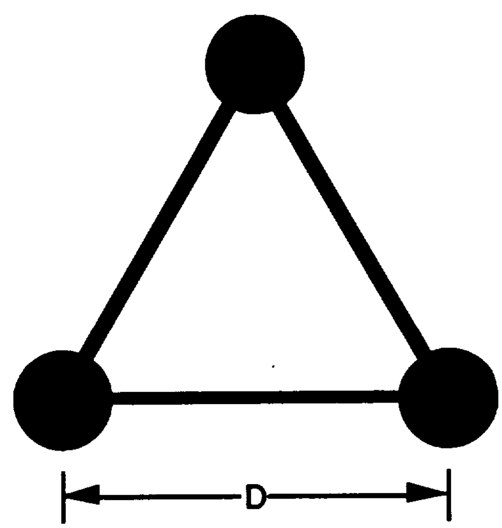



**Figure 10.2**: Cross section of landing gear strut

### 10.10.4 Specifications

The final specs for the landing gear for the lander and biconic are shown in Table 10.1:

| | Lander | Biconic |
|---|---|---|
| Material | 6061-T6Al | 6061-T6Al |
| Touchdown V | 2 m/s | 2 m/s |
| Vmax | 3.58 m/s | 3.24 m/s |
| "G" Force | 3.0 | 1.0 |
| D (See Figure 10.2) | 0.183 m | 0.262 m |
| Total Strut Mass | 42 kg | 57 kg |
| S.F. (Yielding) | 2 | 2 |
| S.F. (Buckling) | 5 | 5 |

**Table 10.1**: Landing gear specifications

## 10.11 Mars Ascent/Descent Module

The purpose of the Mars Ascent and Descent Module (MADM) is to bring the crew down to the Martian surface and back to Mars orbit with samples of the Mars environment. As the cabin and cargo bay of the MADM will be present on all of the injections, the fuel mass of the entire mission is very sensitive to their masses. The dimensions of the craft will affect the dimensions of the aerobrakes, which will also affect the fuel mass.

### 10.11.1 Configuration

The final configuration must satisfy the following requirements:

• The width and height should be roughly equal for the optimal aerobrake structure. The thickness of the craft (on its back) should also fit within the wake flow of the 3m high aerobrake. The largest aerobrake allowable is one that will fit inside of the ETO shroud.

• The cabin must hold the six astronauts and their suits fairly comfortably, since they may be in orbit for several days upon their eventual return to Earth.

• It must be statically stable on landing and able to withstand the 3 G landing force. There must also be some means of providing dynamic stability on landing.

• An escape hatch at least 0.81m$^2$ will need to be provided for the crew.

• A 5m$^3$ cargo bay for returning the samples must be included.

• The landing gear and descent systems must be detached for ascent from Mars.

Our initial configuration was staged, with the crew cabin above the ascent tanks and engine, which were above the descent tanks and engine. Cylindrical fuel tanks were required to make everything fit, with a complex landing gear structure to surround the tanks. This configuration met the requirements, but supporting the tanks would prove to be a problem, and propulsion felt that it would be difficult to pump the fuel from the tanks. Also, the the landing gear structure was not very sound, and the center of gravity of the vehicle was rather high, which required retractable landing gear in order to stay within the aeroshell requirement and maintain static stability. This is a difficult problem, as hydraulics are very unreliable in a space environment.

Succesive iterations led to the design is shown in Figures 10.7 - 10.9. The tanks were moved out from underneath the cabin to the sides, lowering the center of gravity and decreasing the landing gear complexity. The cylindrical tanks were turned on their ends and finally made spherical for the optimum volume to surface area ratio. Separate engines for ascent and descent were no longer required, and this lowered the center of gravity of the lander sufficiently so that the landing gear no longer needed to retract.

The landing gear consist of one main and two supporting struts with honeycomb shock absorbtion on each. This allows three degrees of freedom for each landing pad on impact, increasing their stability and integrity on landings with high tangential velocity.

The shorter and simpler design also allows for a lighter and simpler tank harness.The cargo bay, sitting directly above the engines, was also made larger and simpler, to allow for 2.5m$^3$ of ECLSS in addition to that stored above the crew.

As there was no reason for carrying the de-orbiting fuel tanks down to the surface, these *very* large tanks (4m Dia x 4) were left on the aeroshell. The fuel required for final braking and touchdown were then so small that they were absorbed into the ascent tanks with virtually no change in the ascent tank diameter.

### 10.11.2 Specifications

The descent module will travel from Earth to Mars and down to the Martian surface. It is shown in detail with dimensions in Figure 10.9. The current specifications for this vehicle are as follows:

| | |
|---|---|
| Height: | 7.0 m |
| Width: | 7.8 m |
| Thickness: | 7.0 m |
| Escape hatch: | 0.81 m$^2$ |
| Main bulkhead: | 0.8 m x 1.5 m |
| Cargo bay: | 5.0 m$^3$ |
| Structural G limit: | 6.0 |

During Mars takeoff the landing gear will detach from the ascent vehicle, which will travel from the Martian surface to Mars orbit. Once docked at the MTV, the fuel tanks will detach from the Earth return vehicle (Figure 10.10), which will travel from Mars orbit to low Earth orbit. Its dimensions are:

| | |
|---|---|
| Height: | 6.1 m |
| Width: | 3.0 m |
| Thickness: | 3.4 m |

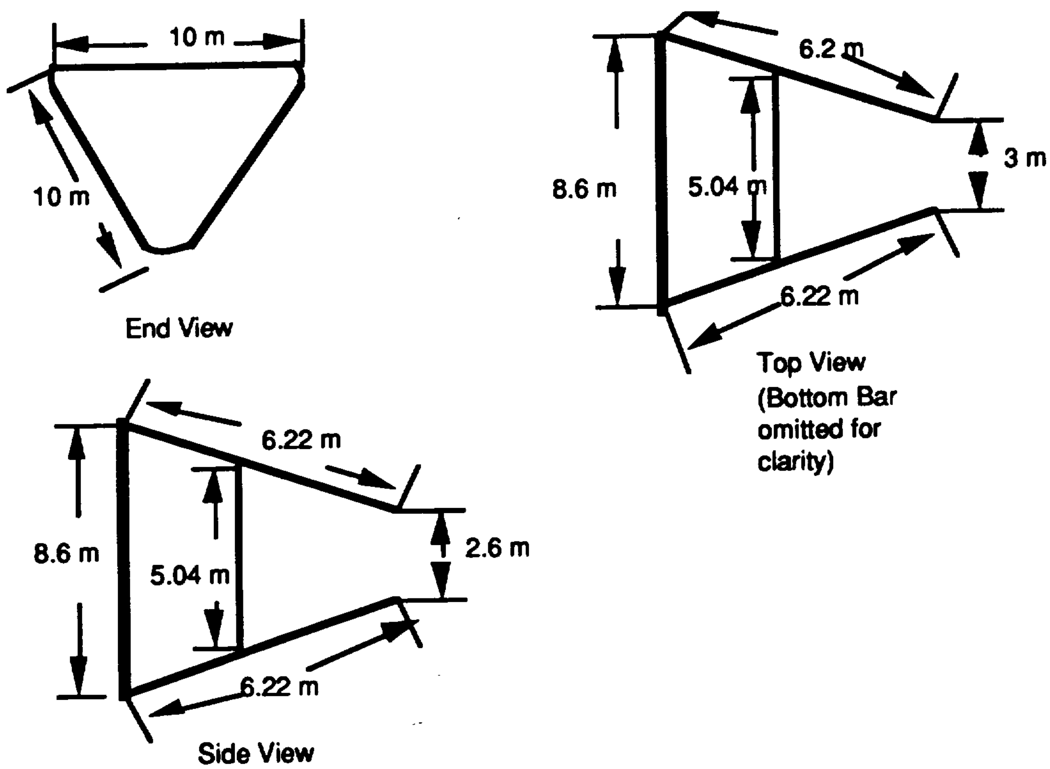
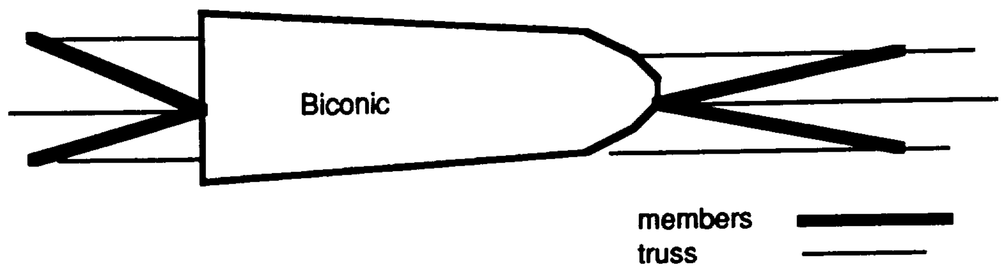
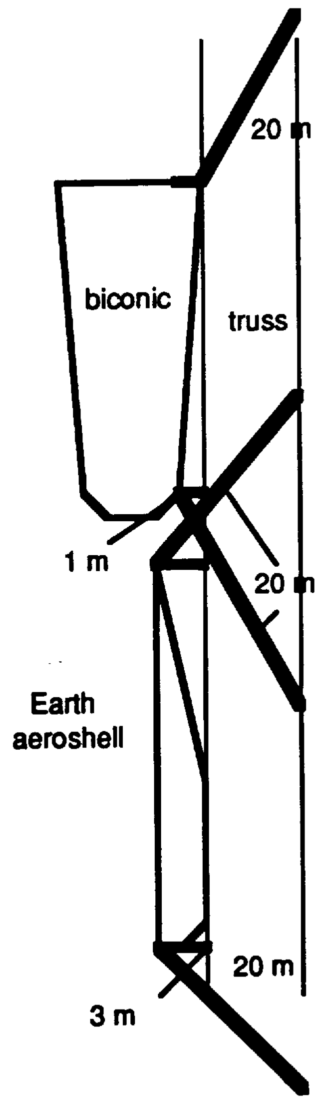



**Figure 10.3:** MTV Thrust Structure




Top view of Biconic
Support arrangement




Side view of Biconic and Aeroshell

**Figure 10.4:** Earth Aeroshell and Biconic Support Structure

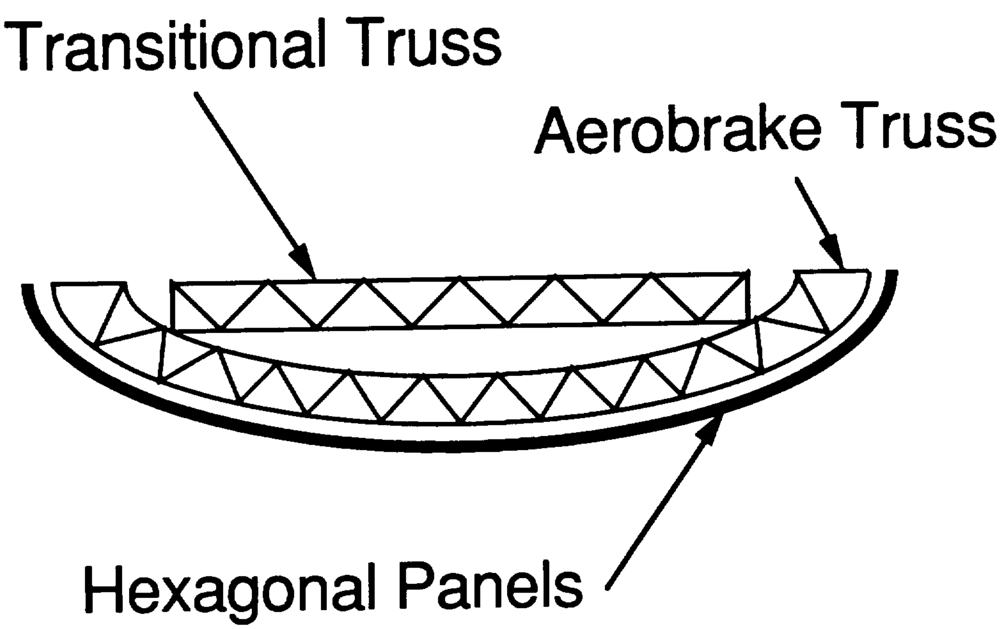



**Figure 10.5:** Typical Aeroshell Structure (Schematic)

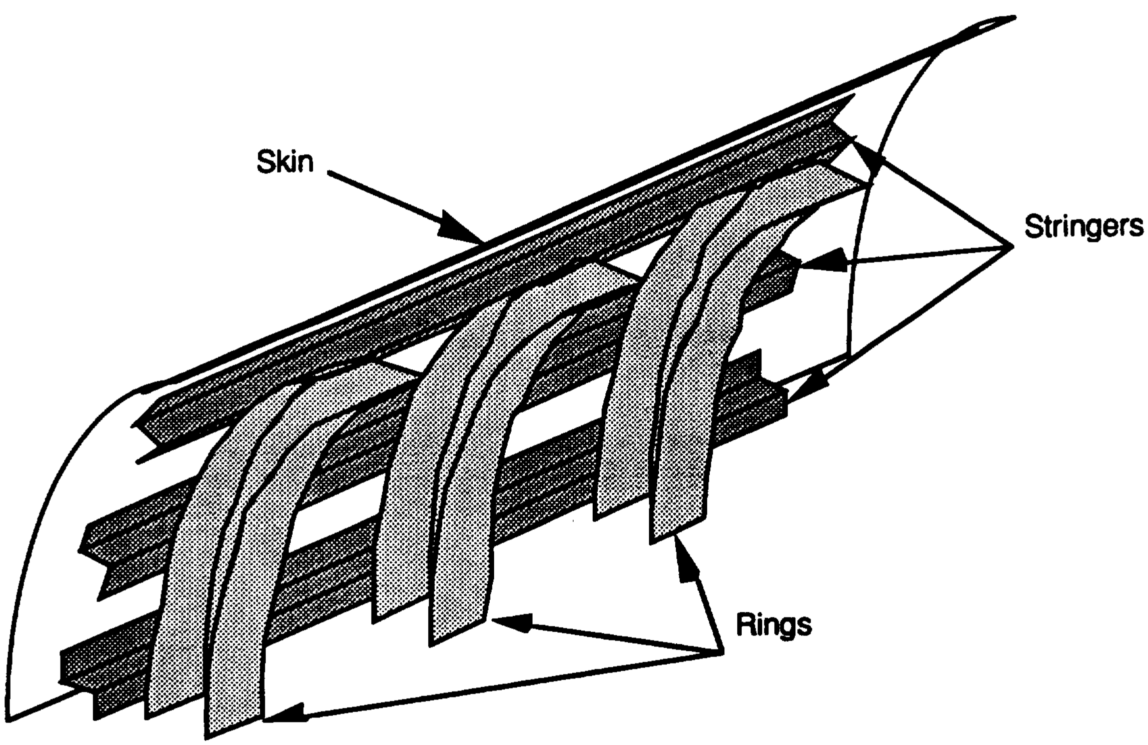



**Figure 10.6:** Biconic Structure (Schematic)

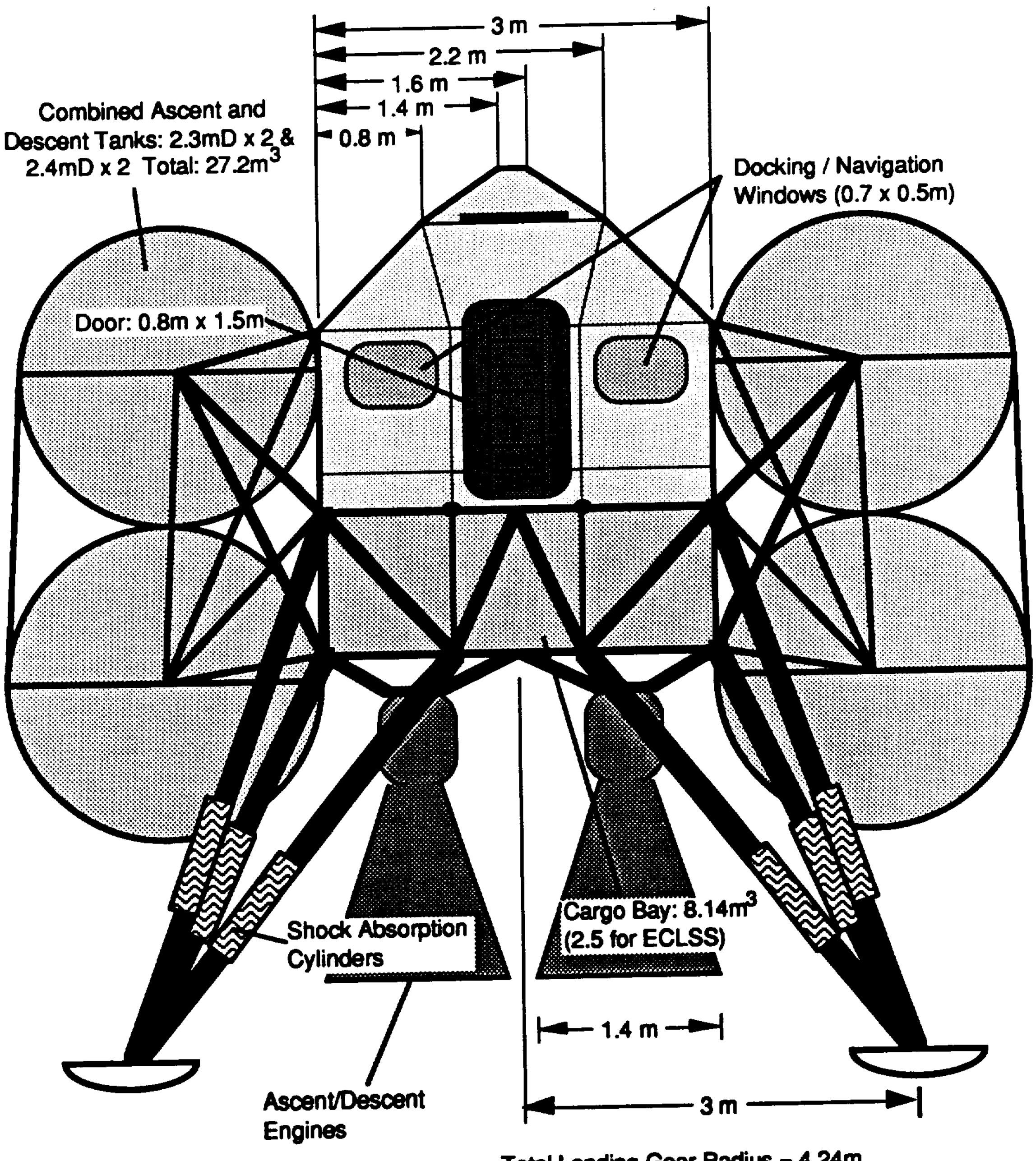



**Figure 10.7:** MADM Front View

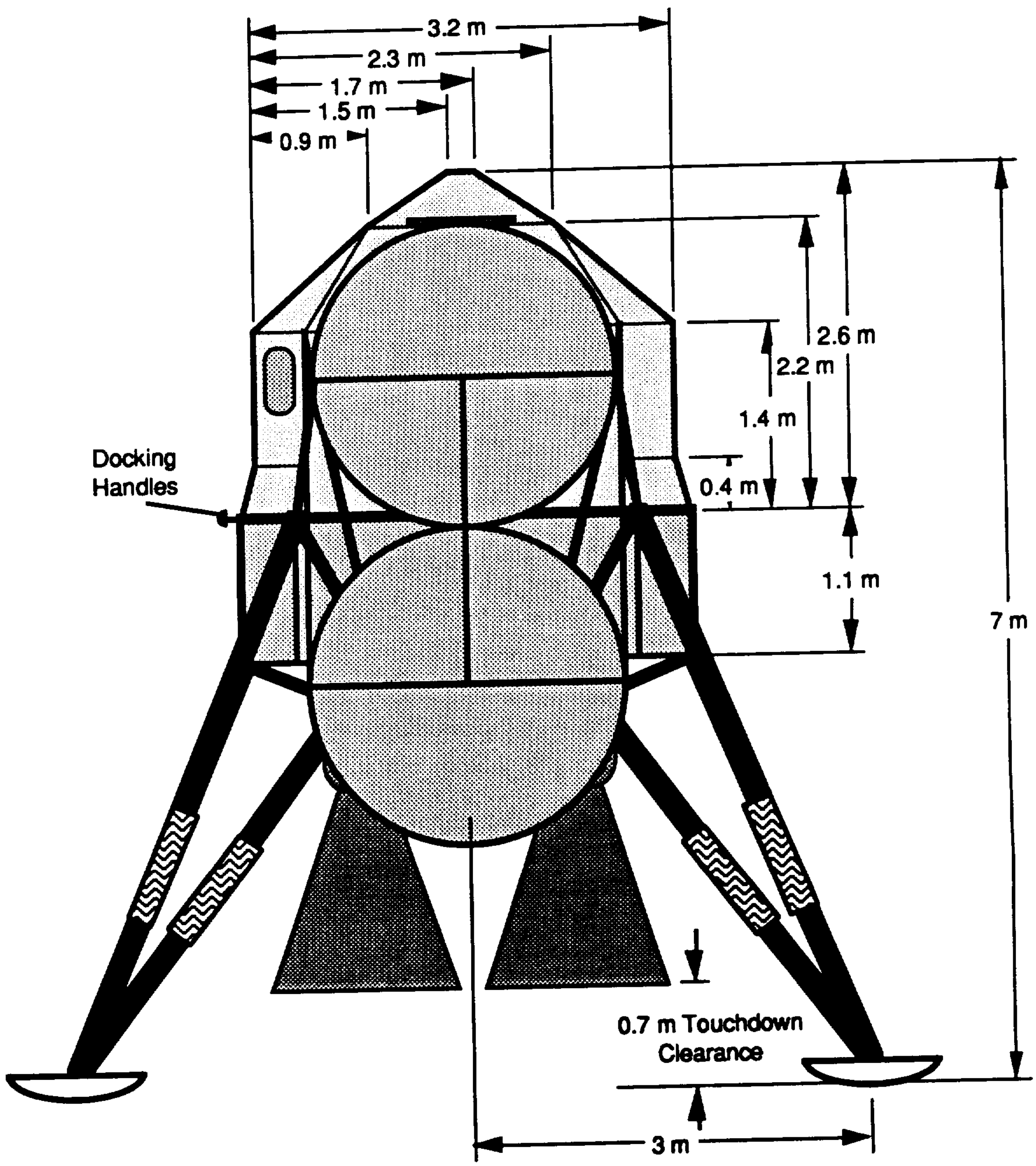



**Figure 10.8: MADM Side View**

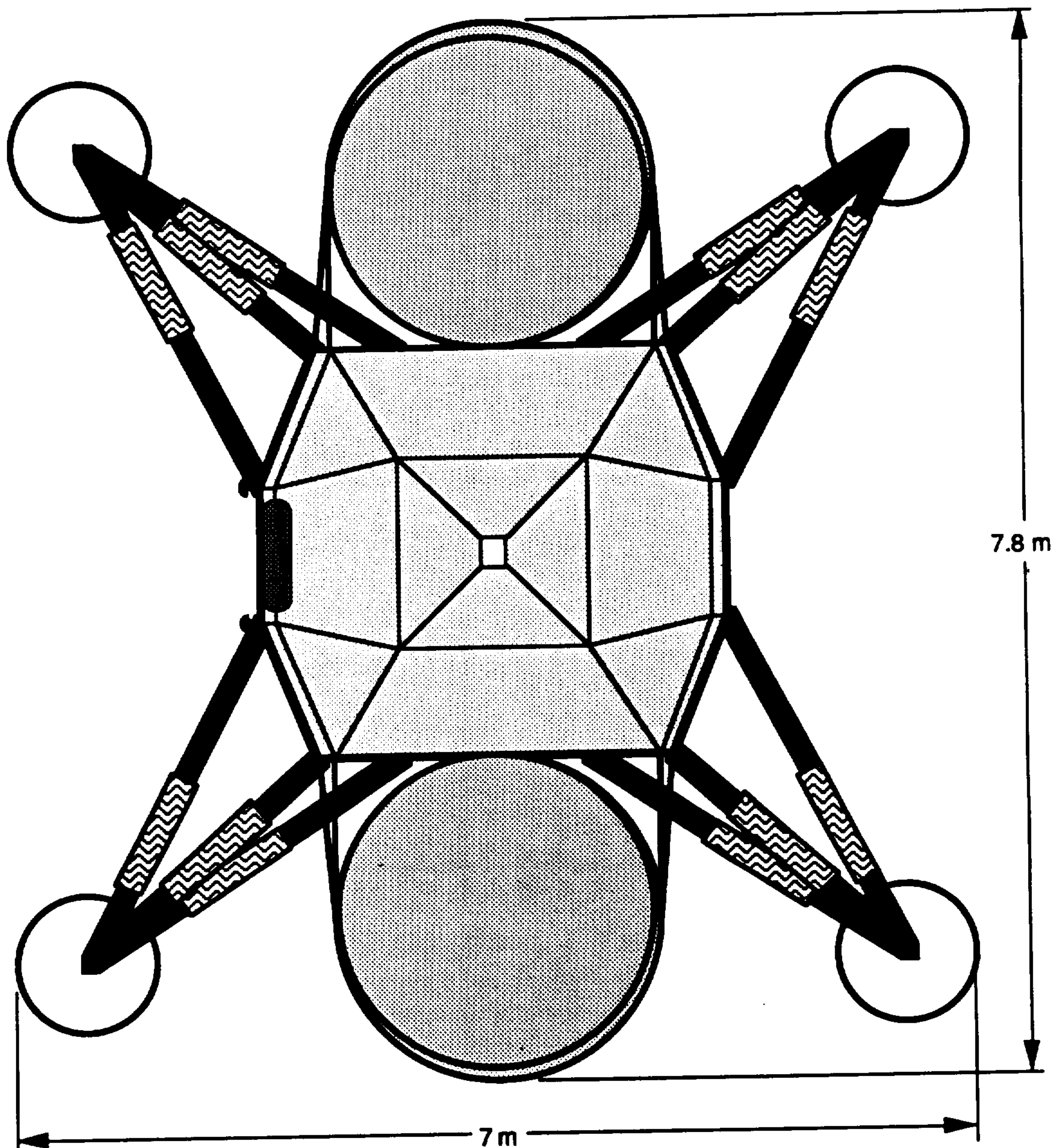



**Figure 10.9: MADM Top View**

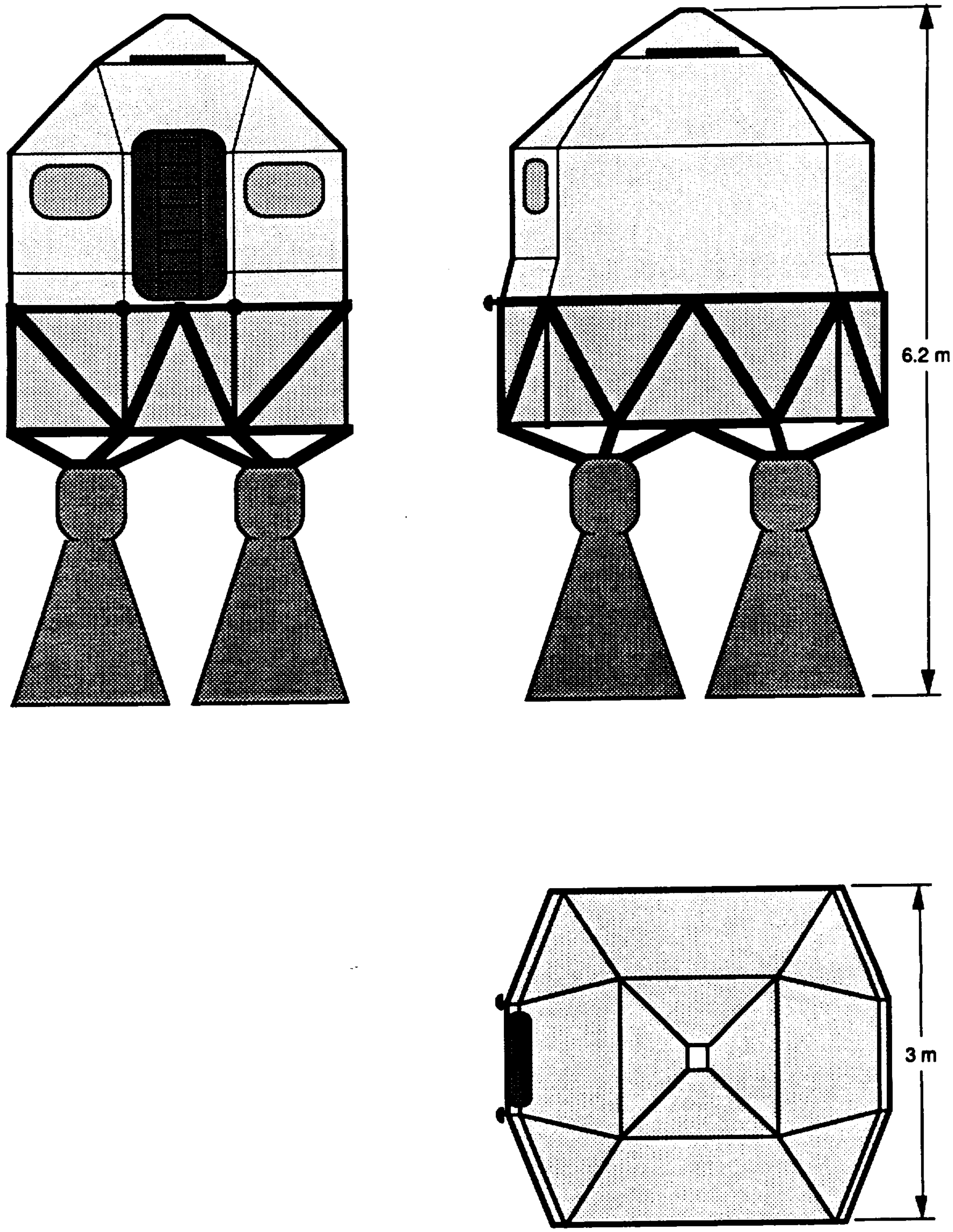



**Figure 10.10:** Earth Return Vehicle

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# Appendix 2.1: Mass summary of vehicle

| System | | MEV Biconic Mass(mT) | Discipline | % of MEV bic. |
|---|---|---|---|---|
| Ground Operations Equipment | | 7.15 | Mission Ops | 4.99 |
| Biconic A/B (structure only) | | 7.00 | Structures | 4.88 |
| MEV hab structure only | | 12.00 | Structures | 8.37 |
| Landing Structure/Gear | | 0.50 | Structures | 0.35 |
| Thrust Structure | | 5.00 | Structures | 3.49 |
| TPS | | 4.00 | Thermal | 2.79 |
| Power System | | | | |
| Nuclear Reactor Based System | | 3.00 | Avionics | 2.09 |
| Backup: H2-O2 Fuel Cell | | 1.00 | Avionics | 0.70 |
| Power Transmission / Power Conditioning | | 0.50 | Avionics | 0.35 |
| Computers | | 0.10 | Avionics | 0.07 |
| 1m Parabolic Dish | | 0.01 | Avionics | 0.00 |
| Control | | 0.05 | Avionics | 0.03 |
| Omni directional antenna | | 0.01 | Avionics | 0.00 |
| Consumables | | 4.28 | Crew Systems | 2.99 |
| Crew Quarters | | 0.46 | Crew Systems | 0.32 |
| Excerise | | 0.17 | Crew Systems | 0.12 |
| Health Care (medical) | | 5.01 | Crew Systems | 3.50 |
| Wardroom | | 0.08 | Crew Systems | 0.05 |
| Galley | | 0.98 | Crew Systems | 0.68 |
| Laundry | | 0.10 | Crew Systems | 0.07 |
| Maintenance | | 0.44 | Crew Systems | 0.30 |
| Life Support (ECLSS) | | 3.80 | Crew Systems | 2.65 |
| Misc. | | 2.95 | Crew Systems | 2.06 |
| RCS Subsystems (4) | | 1.20 | Propulsion | 0.84 |
| Main Engines(2) | fuel/tank(mT) | 0.50 | Propulsion | 0.35 |
| methane tank (3) (2.3 m Dia.) | 2.74 | 0.01 | Structures | 0.01 |

| | | | | | |
|---|---|---|---|---|---|
| | | MEV Biconic, cont | | | |
| methane tank (1) (3.4 m Dia.) | | 8.22 | 0.03 | | 0.02 |
| oxygen tanks (6) (2.5 m Dia.) | | 9.58 | 0.01 | Structures | 0.01 |
| Subtotal (dry) | | | 60.39 | | |
| Propellant | | | 73.92 | Propulsion | 51.56 |
| SubTotal (wet) | | | 134.31 | | |
| Contingency (15% of dry mass) | | | 9.06 | | **% of MTS** |
| **Total Mass for the MEV Biconic** | | | **143.37** | | 12.82 |
| | | | | | |
| | | | | | |
| | | | **MTV HAB** | | |
| **System** | | | **Mass(mT)** | **Discipline** | **% of MTV Hab** |
| Science | | | 1.19 | Mission Ops | 1.86 |
| Shell (outer wall only:Straight Al(.5cm thick))) | | | 5.86 | Structures | 9.17 |
| Docking mechanism | | | 0.50 | Structures | 0.78 |
| Main Power System (utilizing NTR reactors) | | | 1.50 | Avionics | 2.35 |
| Fuel Cell Reserve System | | | 0.50 | Avionics | 0.78 |
| Computers | | | 0.40 | Avionics | 0.63 |
| Control | | | 0.05 | Avionics | |
| Computer control system (phased array) | | | 0.10 | Avionics | 0.16 |
| Omni Antenna | | | 0.01 | Avionics | 0.01 |
| Storm Shelter (2mx5mx5m) | | | 10.40 | Crew Systems | 16.27 |
| Consumables | | | 20.22 | Crew Systems | 31.64 |
| Freezers | | | 1.24 | Crew Systems | 1.94 |
| Health and Exercise | | | 2.40 | Crew Systems | 3.76 |
| Greenhouse | | | 1.44 | Crew Systems | 2.25 |
| Galley/Storage | | | 1.20 | Crew Systems | 1.88 |
| Personal Hygiene | | | 0.43 | Crew Systems | 0.68 |
| Partitions | | | 0.90 | Crew Systems | 1.41 |
| Furniture (Chairs and Tables) | | | 0.16 | Crew Systems | 0.25 |

| | | | | |
|---|---|---|---|---|
| **MTV HAB, cont.** | | | | |
| Floors | 2.13 | | Crew Systems | 3.33 |
| Finishes and Misc. | 0.36 | | Crew Systems | 0.56 |
| Lighting | 0.07 | | Crew Systems | 0.11 |
| Hatches and Bulkheads | 0.69 | | Crew Systems | 1.09 |
| ECLSS | 3.82 | | Crew Systems | 5.98 |
| SubTotal | 55.57 | | | **% of MTS** |
| Contingency (15%) | 8.34 | | | |
| **Total mass for the MTV Hab** | **63.90** | | | 5.72 |
| | | | | |
| **Ascent/Descent Vehicle** | | | | |
| **System** | **Mass(mT)** | | **Discipline** | **% of Descent** |
| Landing Gear | 0.20 | | Structures | 0.74 |
| Thrust Structure | 0.10 | | Structures | 0.37 |
| Crew Cabin | 0.49 | | Structures | 1.82 |
| Cargo Bay | 0.20 | | Structures | 0.74 |
| Fuel Cell | 0.60 | | Avionics | 2.23 |
| Computers | 0.10 | | Avionics | 0.37 |
| Avionics/RCS | 0.10 | | Avionics | 0.37 |
| 1m Parabolic Dish | 0.01 | | Avionics | 0.02 |
| Omni Dish | 0.01 | | Avionics | 0.02 |
| Crew Systems | | | | |
| Descent Stage | 1.84 | | Crew Systems | 6.83 |
| Ascent Stage (1.84 + rocks) | 3.84 | | Crew Systems | 14.26 |
| RCS Subsystems | 0.30 | | Propulsion | 1.11 |
| Main Engines (4) | 1.00 | | Propulsion | 3.71 |

| | | | | | | |
|---|---|---|---|---|---|---|
| | | Ascent/descent vehicle, cont. | | | | |
| | | | | | | |
| Ascent: | | fuel/tank (mT) | | | | |
| methane tanks (2) (2.2 m Dia.) | | 2.35 | 0.02 | | Structures | 0.07 |
| oxygen tanks (2) (2.4 m Dia.) | | 8.23 | 0.02 | | Structures | 0.07 |
| Descent fuel is inside the Mars Aeroshell | | | | | | |
| | | | | | | |
| | Descent | Ascent | | | | |
| Subtotal (dry) | 5.02 | 6.82 | | | | |
| propellant | 21.16 | 21.16 | | | | |
| Subtotal (wet) | 26.18 | 27.98 | | | | |
| Contigency (15% | 0.75 | 1.02 | | | | |
| | | | | | | % of MTS |
| Total mass: | | | | | | |
| Descent vehicle before leaving the MTS | | | 26.93 | | | 2.41 |
| Descent vehicle after touchdown on Mars | | | 26.93 | | | |
| Ascent vehicle before leaving Mars | | | 29.00 | | | |
| Ascent vehicle after return to MTS | | | 7.84 | | | |
| | | | | | | |
| | | | | | | |
| System | | | Mass(mT) | | Discipline | % of MTS |
| Aeroshells (2) | | | | | | |
| Mars Aeroshell | | | | | | |
| Dimensions: 20mx12mx3m | | | 7.9 | | Structures | 0.71 |
| Descent fuel: | | fuel/tank (mT) | | | Structures | 0.00 |
| Methane (2) (2.5m Dia.) | | 3.39 | 0.02 | | Structures | 0.00 |
| Oxygen (2) (2.7m Dia.) | | 11.87 | 0.03 | | Structures | 0.00 |
| Earth Aeroshell | | | | | Structures | 0.00 |
| Dimensions: 13.3mx8mx2m | | | 3.90 | | Structures | 0.35 |
| | | | | | | |

| | | | | | | |
|---|---|---|---|---|---|---|
| | | Aeroshell, cont. | | | | |
| | | | | | | |
| Descent fuel: | | fuel/tank (mT) | | | Structures | 0.00 |
| Methane (2) (1.5 m Dia.) | | 0.69 | 0.01 | | Structures | 0.00 |
| Oxygen (2) (1.6m Dia.) | | 2.42 | 0.01 | | Structures | 0.00 |
| TPS | | | 2.40 | | Thermal | 0.21 |
| | | | | | | |
| Cylindrical passageway (3m x 1.2 m Dia.) | | | 0.40 | | Structures | |
| | | | | | | |
| **Engine** | | | | | | |
| 3 NTR Engines | | | 5.93 | | Propulsion | 0.53 |
| Thrust structure | | | 6.44 | | Structures | 0.58 |
| | | | | | | |
| **Primary RCS System** | | | 19.01 | | Propulsion | 1.70 |
| monomethyl hydrozine (2) (2.2 m Diam.) | | | 0.01 | | Structures | 0.00 |
| fuel/tank: | 4.75 | | | | | |
| nitrogen tetraoxide (2) (1.9 m Diam.) | | | 0.01 | | Structures | 0.00 |
| fuel/tank: | 4.75 | | | | | |
| | | | | | | |
| **Truss** | | | | | | |
| 150 m long (3m equilateral triangle) | | | 15.70 | | Structures | 1.40 |
| Navigation/Control/Communications | | | 0.10 | | Avionics | 0.01 |
| Parabolic Dish (3m-emergancy comm.) | | | 0.25 | | Avionics | 0.02 |
| Computer control system (phased array) | | | 0.10 | | Avionics | 0.01 |
| | | | | | | |
| **Fuel Tanks** | | | | | | |
| 7 tanks | | | 24.96 | | Structures | 2.23 |
| (3@20.1x11.2,2@18.2x11.2,1@30mx11.2,1@7.8m diam.) | | | | | | |
| MTS subtotal (dry) | | | 359.06 | | | 32.11 |
| | | | | | | |

| | | | | | | |
|---|---|---|---|---|---|---|
| **MTS** | | | | | | |
| Propellant | | | | | | |
| TMI | | | 341.66 | | Propulsion | 30.56 |
| MOI | | | 189.92 | | Propulsion | 16.99 |
| TEI | | | 173.64 | | Propulsion | 15.53 |
| Total Propellant mass | | | 705.22 | | | |
| MTS dry | | | 359.06 | | | |
| Subtotal MTS wet | | | 1064.28 | | | |
| Contingency (15%) | | | 53.86 | | | |
| **Total mass for the MTS configuration** | | | **1118.14** | | | |
| | | | | | | |
| | **MEV Biconic** | | | | **Asc./Des.** | |
| **System** | **Mass (mT)** | **% MEV Bic.** | | **System** | **Mass (mT)** | **% of As/Des** |
| Mission Ops | 7.15 | 5.00 | | | | |
| Structures | 24.50 | 17.13 | | Structures | 0.79 | 0.59 |
| Thermal | 4.00 | 2.80 | | Avionics | 0.81 | 0.60 |
| Avionics | 4.66 | 3.26 | | Crew Systems | 5.68 | 4.23 |
| Crew Systems | 18.26 | 12.77 | | Propulsion | 22.46 | 16.73 |
| Propulsion | 75.67 | 52.90 | | Subtotal | 26.18 | |
| Subtotal | 134.26 | | | Conting. (15%) | 0.56 | |
| Conting. (15%) | 8.79 | | | **Total** | **26.74** | |
| **Total** | **143.05** | | **MTV Hab** | | | |
| | | **System** | **Mass (mT)** | **% MTV Hab** | | |
| | | Mission Ops | 1.19 | 1.86 | | |
| | | Structures | 6.36 | 9.95 | | |
| | | Avionics | 2.56 | 4.00 | | |
| | | Crew Systems | 45.47 | 71.15 | | |
| | | Subtotal | 55.57 | | | |
| | | Conting. (15%) | 8.34 | | | |
| | | **Total** | **63.90** | | | |

# Appendix 2.2: Specification sheets

# SPECIFICATION SHEET

**UNIVERSITY OF MINNESOTA / NASA / USRA**

# MTS Overall Mission

**MARS TRANSPORTATION SYSTEM**

## Operational Purpose

**MTV:** Provide transportation from Earth orbit, to Mars orbit, and back to Earth orbit.

**MEV:** Provide transportation from Mars orbit to Martian surface, provide for 60 day stay period; then provide transportation back to Mars orbit.

## Mass

| | | |
|---|---|---|
| MTS: | dry | 359.1 mT |
| | wet (w/o conting.) | 1064.3 mT |
| | wet (with conting.) | 1118.1 mT |

## Center of Mass

| Time | x | y | z |
|---|---|---|---|
| Pre-TMI | 0.0m | 0.0m | 7.8 m |
| Post-TMI | 0.0m | 0.0m | 52.2 m |
| Post-MOI | 2.5 m | 0.0m | 67.2 m |
| Post-TEI | 1.8 m | 0.0m | 117.5 m |

## Dimensions

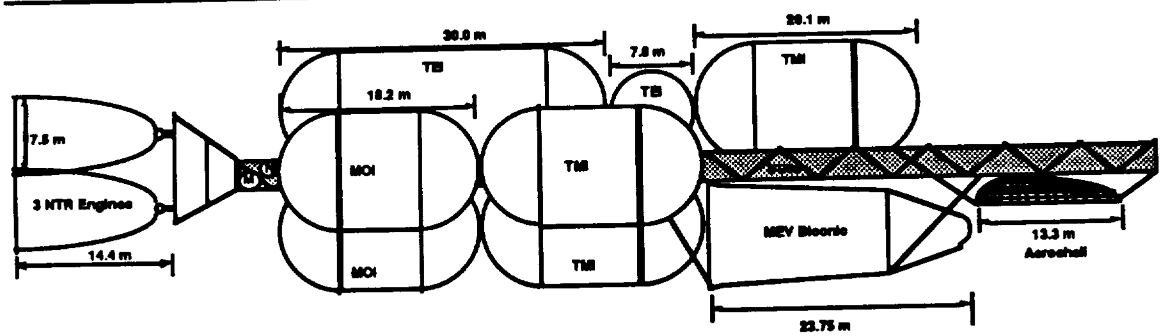
Truss length: 150.0 m

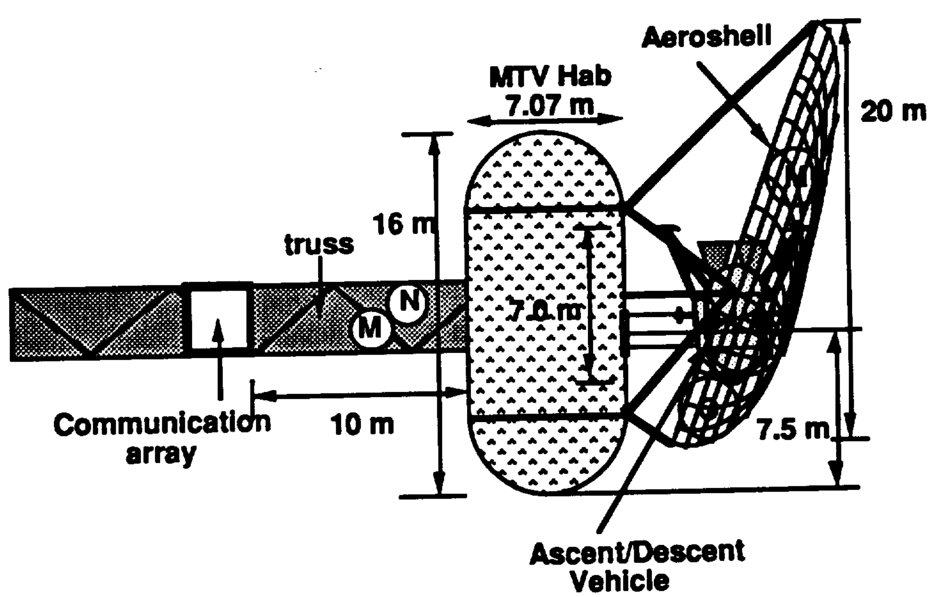
## Crew Systems

| | | |
|---|---|---|
| Crew size | | 6 |
| Rotation speed: | | |
| | TMI | 2.9 rpm |
| | post TEI | 3.3 rpm |
| Effective gravitation: | | |
| | Outbound | 1.0 g |
| | Inbound | 0.5 g |

## Trip Time

| | |
|---|---|
| Outbound: | 200 days |
| Inbound: | 250-270 days |
| Surface stay, all missions: | 60 days |
| Total trip time: | 510-530 days |



# SPECIFICATION SHEET

**Aeroshell & Ascent/Descent Vehicle**

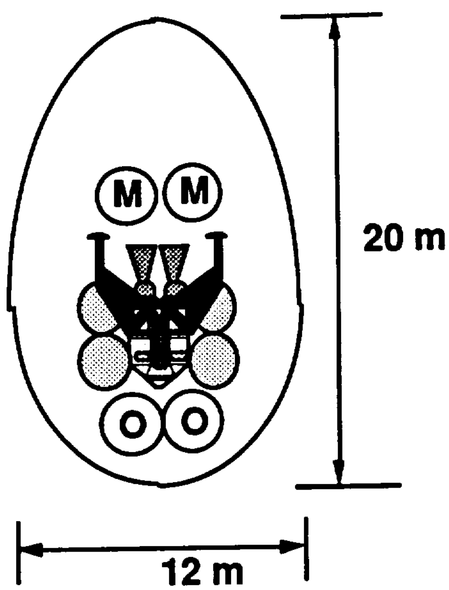
UNIVERSITY OF MINNESOTA / NASA / USRA — MARS TRANSPORTATION SYSTEM

## Mass

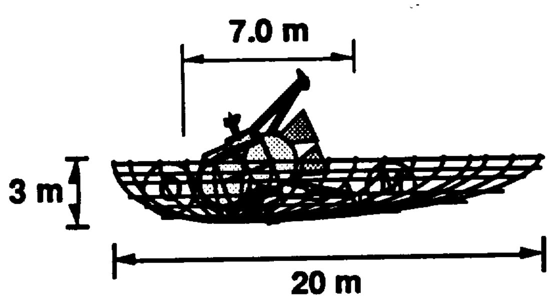
| | |
|---|---|
| Ascent/Descent Vehicle: | |
| Descent vehicle | |
| before leaving the MTS: | 26.9 mT |
| after touchdown on Mars: | 26.9 mT |
| Ascent Vehicle | |
| before leaving Mars: | 30.5 mT |
| after return to MTS | 9.4 mT |
| Aerobrake (Mars) | 39.7mT |
| Aerobrake (Earth) | 11.3mT |

## Dimensions

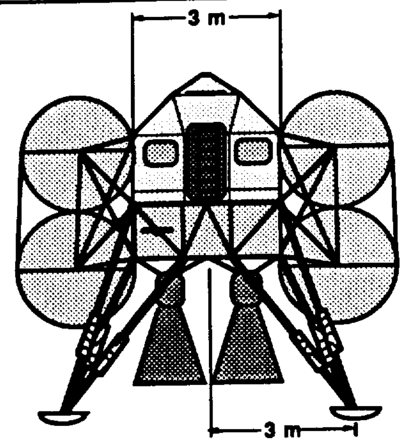
Aerobrake (Mars): 20.0 m x 12.0 m oval



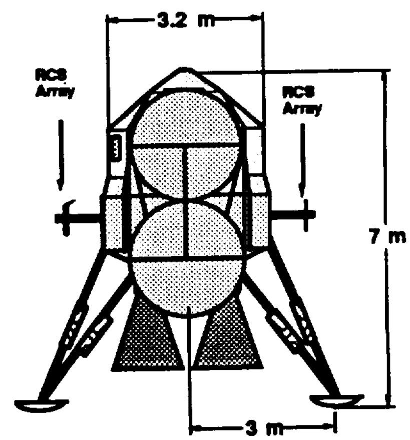

Top view of aeroshell with lander



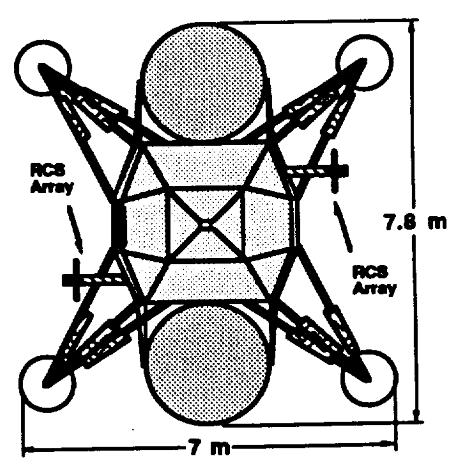

Side view of aeroshell with descent vehicle




Front view of ascent/descent vehicle
(6 m x 7 m)




Side view of ascent/descent vehicle
(6 m x 7 m)




Top view of ascent/descent vehicle
(7 m x 8.2 m)

# SPECIFICATION SHEET

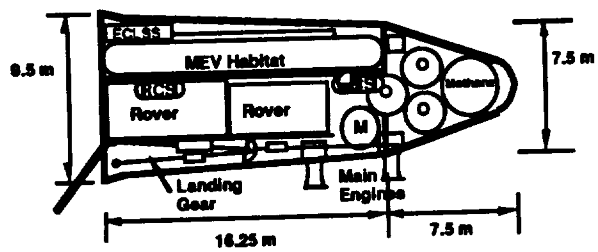
**UNIVERSITY OF MINNESOTA / NASA / USRA**

## MEV Biconic & MTV Hab

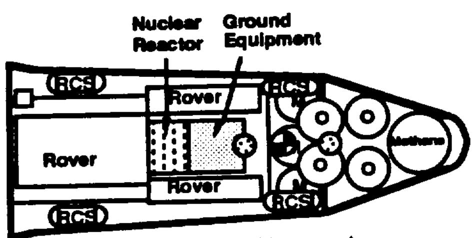
**MARS TRANSPORTATION SYSTEM**

### Mass

| | |
|---|---|
| MEV Biconic: | |
| dry: | 60.4 mT |
| wet: | 143.4 mT |
| MTV Hab: | 63.9 mT |
| Crew provisions: consumables | |
| MEV | 3.1 mT |
| MTV | 20.2 mT |



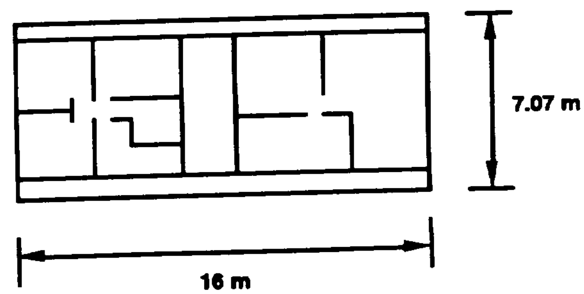

MEV biconic, Side View
23.75 m x 9.5 m



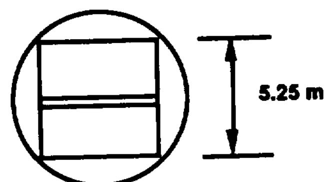

Top view of bottom layer

### Dimensions

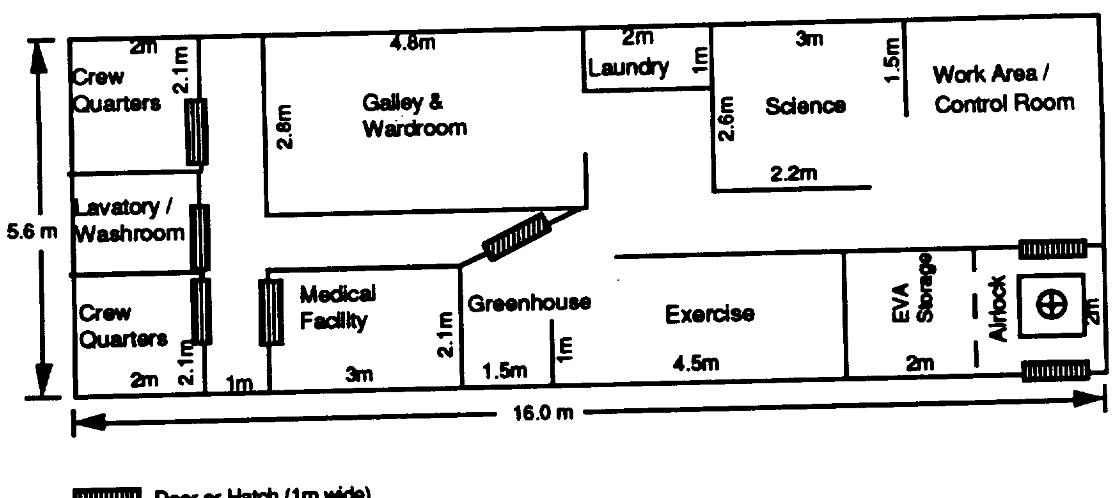
| | |
|---|---|
| Crew quarters: | |
| Volume: MEV biconic: | 487.6 $m^3$ |
| MTV HAB: | 173.4 $m^3$ |
| MTV Hab | 16 m x 7.07 m |




Top View of lower level




End View of HAB module




Top view of MEV Habitat

# SPECIFICATION SHEET Engine, Communications and Thermal

UNIVERSITY OF MINNESOTA/NASA/USRA — MARS TRANSPORTATION SYSTEM

## MTV Main Engines

Type: 3 @ LPNTR
Weight: 1.98mT
Thrust: 113.0 kN
Isp: 1100.0sec
Duty cycle: TBD

## Propellant

Total trip mass 707.5 mT

**TMI**
Fuel mass 324.6 mT
Tank mass: 3@3.8 mT
Tank size: 20.1 m x 11.2 m

**MOI**
Fuel mas: 189.9 mT
Tank mass: 2@3.3 mT
Tank size: 18.2 m x 11.2 m

**TEI**
Fuel mass: 73.6 mT
Tank mass: 1@6.6 mT & 1@0.4 mT
Tank size: 30.0 m x 11.2 m & 7.8 m diam.

Type:
TMI/TEIliquid $H_2$ base, volume: 8934.0 $m^3$
MEV Biconic: $LCH_4$& $LO_2$
Vol: 36.5$m^3$/47.5$m^3$
Ascent/Descent Mod: $LCH_4$ & $LO_2$
Vol: 5.6/7.2$m^3$ , 35/44.75$m^3$

## Reaction Control System

Type chemical
Propellant:
H: Monomethly Hydrazine
$NH_4$: Nitrogen Tetroxide

| vehicle | # pods | # thrusters/pod | fuel mass | fuel volume |
|---|---|---|---|---|
| ascent/ descent | 4 | 2x9 thrusters<br>2x2 thrusters | 450 kg | 0.045m3 |
| Biconic | 4 | 15 | 3 mT | 2.70 m3 |
| MTV | 2 | 6 | 10 mT | 10.0 m3 |

## Navigation Systems

Inertial Measurement Unit (IMU)
Star Mapper

## Communication Systems

Power required (ma) 2.0 kW
Band ka (32 GHz)
Channel capacity 100.0 Mbps

## Transfer Phase

Delay time (max) 26.0 min
Array dimensions 4.0 x 4.0 m
Back-up 2.5 m diameter
Parabolic dish

## On Station

Two (2) satellites aerosynchonus for most communication. Surface communication has 0.5 m dish for satellite uplink and omni directional antennas

## Power Systems

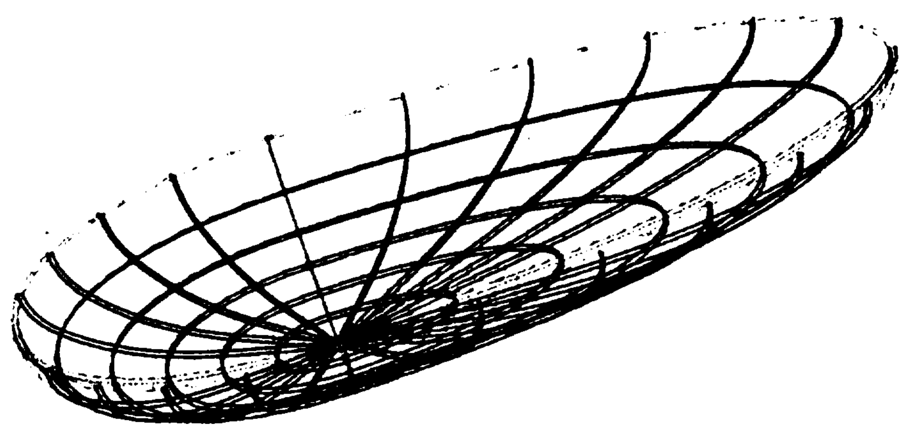
**MTV:**
Main type and rating:
Nuclear &Thermoelectric converter: 2.2 mT
Backup type and rating:
Regenerative fuel cell 2.4 mT

**MEV:**
Main type and ratiing:
15kW Regenerative fuel cell: 0.2 mT + 0.216mT/day

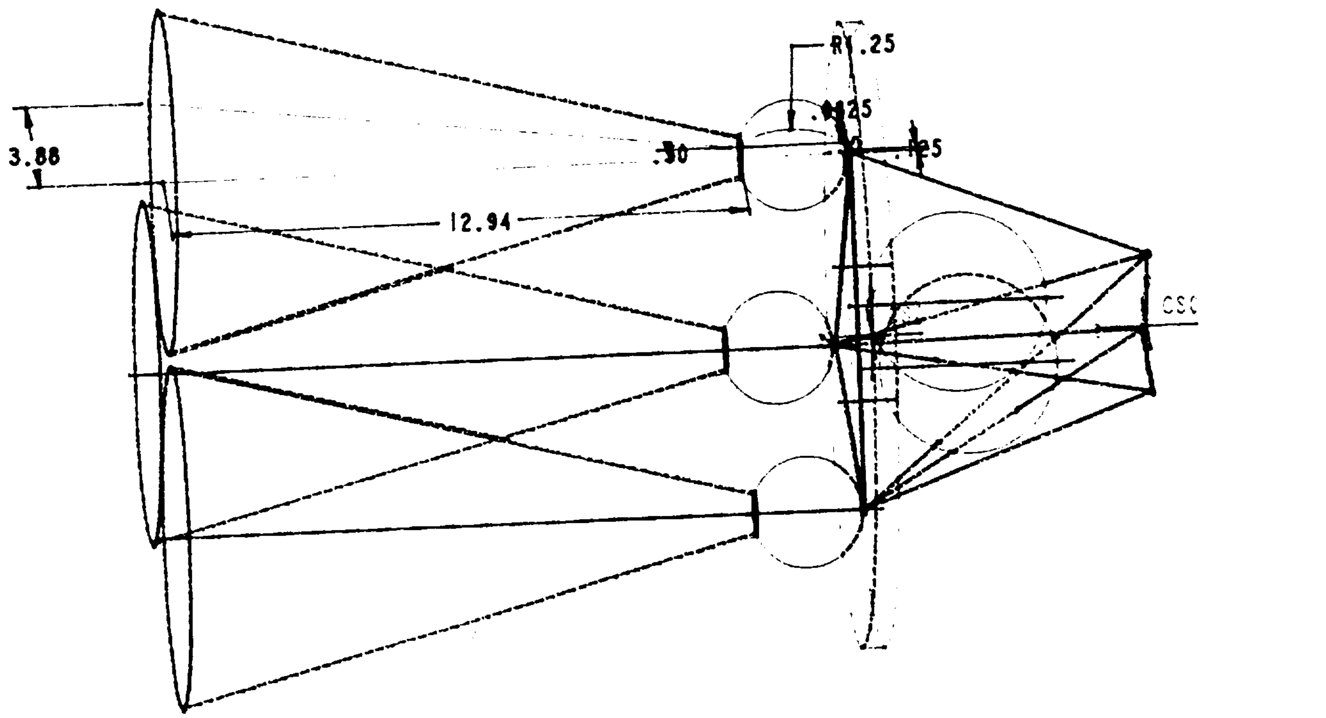
## Thermal Systems

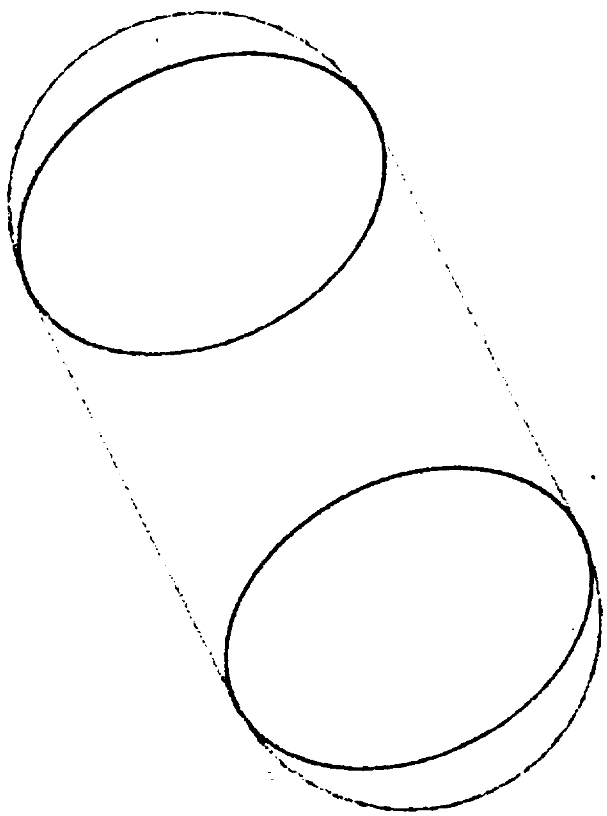
Biconic:
TPS 4.0 mT
Aerobrake:
TPS 5.2 mT

# Appendix 2.3: ProEngineer plots

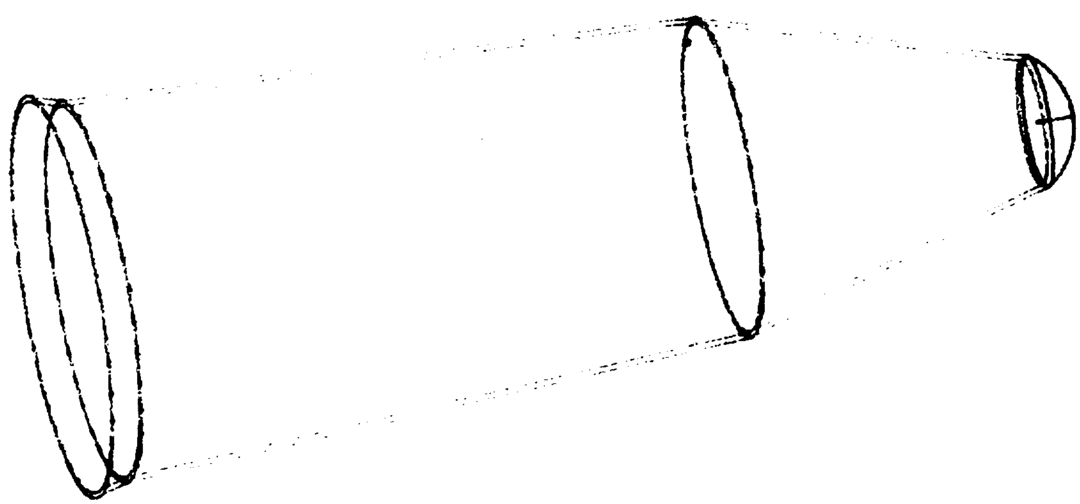


NAME : THEAEROBRAKE
TYPE : STD

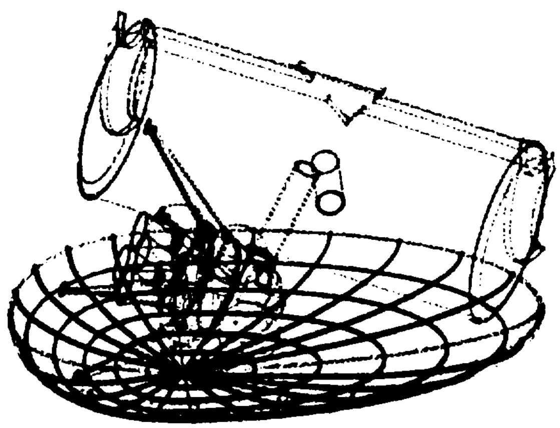

R1.25
.0125
.30
.125
3.88
12.94
CSC
NAME: ENGINEASSEMB



NAME: TEI-TANK



NAME: BICONIC



NAME: HABASSEMBLY

# Appendix 3.1: Mission Timeline

**Mission Timeline** valid for all missions. Dates refer to the first mission.

| Step/Phase | ΔV (km/s) | Step duration | Date |
| --- | --- | --- | --- |
| | | | |
| 0. ETO–Launches 1-9 | | 2 years | |
| | | | |
| I. TMI | | 200 days | |
| A. Leave Earth Orbit | 3.7-3.8 | 2.45 h | 2/15/16 |
| 1. Jettison tanks | | | |
| 2. Start artificial gravity | | | |
| B. Deep Space Maneuver/Plane change | .14-.17 | tbd | |
| 1. Stop gravity | | tbd | |
| 2. Start gravity | | tbd | |
| | | | |
| II. MOI | | tbd | 9/10/16 |
| A. Stop gravity | | tbd | |
| B. MTV enters elliptical parking orbit | 2.2-3 | 2.45 h | |
| 1. Jettison tanks ? (either now or before TEI) | | | |
| C. MEV descends | | 24 hours | |
| 1. Biconic descent | | ca. 8 hours | |
| a. Separation from MTV | | | |
| b. Change planes and enter circular parking orbit | 0.6 | | |
| c. Change from parking orbit to descent trajectory | 1.6 | 7 min. | |
| d. Initial entry into the atmosphere. | | \| | |
| e. Peak heating of the vehicle. | | \| | |
| f. Peak loading of the vehicle. | | \| | |
| g. RCS for trajectory correction. | | \| | |
| h. Deployment of the parachutes for stability. | | \| | |
| i. Propulsive system for final braking. | tbd | \| | |
| j. Landing of the vehicle. | | \| | |
| k. Verification of systems on the biconic | | V | |
| 2. Crew Lander descent | | ca. 8 hours | |
| a. Detach from the MTV. | | | |
| b. Change planes and enter circular parking orbit | 0.6 | | |
| c. Change from parking orbit to descent trajectory | 1.6 | 7 min. | |
| d. Initial entry into the atmosphere. | | \| | |
| e. Peak heating | | \| | |
| f. Peak loading | | \| | |
| g. Aerobrake jettisoned. | | \| | |
| h. RCS for trajectory correction | | \| | |
| i. Deployment of parachutes for stability | | \| | |
| j. Propulsive system for final braking. | tbd | \| | |
| k. Landing | | V | 9/12/16 |

| Step/Phase | ΔV (km/s) | Step duration | Date |
|---|---|---|---|
| | | | |
| III. MARS SURFACE OPERATIONS | | 60 days | |
| A. Initiate and verify power system, life support | | | |
| B. Set up communications equipment | | | |
| C. Unpack living quarters | | | |
| D. Unpack rovers and other equipment | | | |
| E. Initial site exploration/test equipment | | | |
| F. Rover expeditions | | | |
| 1. Send robotic rover out for local mapping, surveying | | | |
| 2. Long pressurized rover journeys: 10 5-day missions | | | |
| G. (Meanwhile) Mars surface experiments | | | |
| H. Pack up/clean up | | | |
| | | | |
| IV. TEI | | 270 days | |
| A. Crew ascends and rendezvous with MTV | | tbd | 11/12/16 |
| 1. Take-off | 2.25 | | |
| 2. Re-establish circular parking orbit and change planes | 2.25 | | |
| 3. Redock with MTV | 0.6 | | |
| B. Leave Mars orbit | 5.9-7.9 | 2.45 h | |
| 1. Start gravity | | tbd | |
| C. Venus swing-by | | tbd | |
| 1. Stop gravity | | tbd | |
| 2. Fine adjustments/Plane change | .16-.18 | tbd | |
| 3. Observe Venus | | tbd | |
| 4. Start gravity | | tbd | |
| | | | |
| V. EOI | | | 7/15/17 |
| A. Stop gravity | | tbd | |
| B. Crew inspects MTV for effects of wear | | tbd | |
| C. Crew enters Earth Return Vehicle2 | | | |
| D. Earth Return Vehicle leaves MTV | tbd | tbd | |
| E. Crew aerobrakes into lower orbit | 4.9-5.9 | tbd | |
| F. Crew module docks with space station | | | |

# Appendix 4.1: MHM Masses and Volumes

## MHM Masses and Volumes

| ITEMS | VOLUME ($m^3$) | MASS (kg) |
|---|---|---|
| crew quarters (2 rooms): | | |
| personal storage | 1.5 | 25.0 |
| finishings | 0.0 | 5.0 |
| partitions | 1.0 | 5.0 |
| beds: 2 x 3m bunks) | 10.1 | 420.0 |
| free space | 7.4 | 0.0 |
| TOTAL CREW QUARTERS | 20.0 | 455.0 |
| exercise | | |
| equipment | 1.1 | 174.0 |
| equipment storage | 2.5 | 0.0 |
| free space | 9.2 | 0.0 |
| TOTAL EXERCISE | 12.8 | 174.0 |
| health care & maintenance | | |
| health care | 6.5 | 1800.0 |
| shower & wash basin | 5.0 | 260.0 |
| waste management | 1.0 | 200.0 |
| medical supplies | 5.0 | 2480.0 |
| free space | 3.5 | 0.0 |
| spares | 2.3 | 273.8 |
| TOTAL HEALTH CARE | 23.3 | 5013.8 |
| wardroom: | | |
| entertain-ment | 1.0 | 50.0 |
| spares | 1.0 | 6.3 |
| free space | 13.8 | 0.0 |
| table | 1.0 | 20.0 |
| TOTAL WARDROOM | 16.8 | 76.3 |
| galley: | | |
| cooking preparation | 8.0 | 720.0 |
| freezer | 0.5 | 28.3 |
| refrigerator | 2.5 | 150.0 |
| pantry | 2.0 | 26.7 |
| spares | 0.4 | 24.8 |
| TOTAL GALLEY | 13.4 | 949.9 |
| laundry: | | |
| washer | 1.0 | 50.0 |
| dryer | 1.0 | 50.0 |
| TOTAL LAUNDRY | 2.0 | 100.0 |
| eva: | | |
| suits | 7.4 | 0.0 |
| spares | 1.5 | 180.0 |
| TOTAL EVA | 8.9 | 180.0 |
| maintenance | | |
| tools | 0.6 | 285.0 |
| workbench / area | 0.5 | 50.0 |
| supplies | 0.4 | 100.0 |
| TOTAL MAINTEN-ANCE | 1.5 | 435.0 |
| consumables | | |
| ambient food | 0.5 | 348.3 |
| frozen food | 0.3 | 210.7 |
| emergency food | 0.2 | 82.2 |
| startup water | 1.0 | 980.0 |
| emergency water | 0.2 | 200.0 |
| repress gas | 1.1 | 1056.8 |
| detergent | 0.1 | 5.9 |
| misc. | 0.1 | 37.9 |
| TOTAL CONSUM-ABLES | 3.5 | 2921.8 |
| life support: | | |
| eclss | 12.0 | 1910.0 |
| spares | 0.0 | 1890.0 |
| TOTAL LIFE SUPPORT | 12.0 | 3800.0 |
| misc.: | | |
| lighting | 0.0 | 170.0 |
| mobility aides | 0.0 | 244.0 |
| greenhouse | 33.8 | 2174.0 |
| free space | 11.9 | 0.0 |
| spares | 0.0 | 365.0 |
| TOTAL MISC. | 45.7 | 2953.0 |
| | | |
| TOTAL | 159.9 | 17058.8 |

# Appendix 10.1: Tank Sizing Spreadsheet

## A10.1 Tank Sizing Spreadsheet

This appendix is intended to explain the spreadsheet which was used to determine the propellant tank sizes. It has been combined with a spreadsheet created by Propulsion Systems which determined propellant masses, and with the Systems Layout mass spreadsheet. This one spreadsheet has been a great help in speeding up the design process.

A sample of an old portion of this spreadsheet is shown in Figure A10.2. For the entries in the data grid, the plain text values are material properties and should only be changed if more accurate information is obtained. *The values in italics are calculated from other entries in the grid, and should not be changed unless the formulas need to be changed.* **The bold values are entered by the user.** These are the parameters which will determine the tank characteristics.

The tank itself is modeled as a cylindrical body with spherical end caps, shown in Figure A10.1.

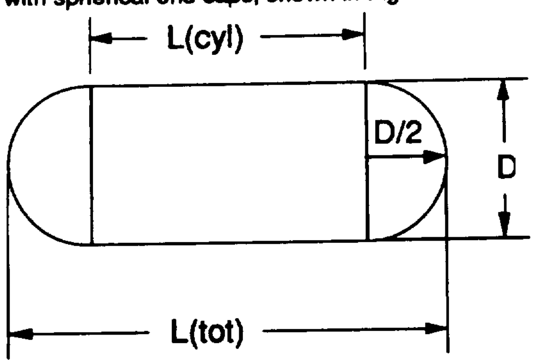



**Figure A10.1:** Tank sizing nomenclature

The propellant tanks are sized by volume, i.e. the volume of fuel to be contained by the tank is a constant, but the actual dimensions of the tank are changed to "optimize" the design by use of a parameter $l = L_{cyl}/D_{in} = L/D$. The primary design consideration is a low $m_{tank}/m_{fuel}$ ratio.

First, the actual amount of fuel necessary at launch must be calculated. This amount of fuel differs from the actual amount at burn due to boiloff. If a mass of fuel is launched, and is allowed to boil off at a certain percentage per month, then the amount left after m months is:

$$m_{burn} = m_{launch}\,(1 - \%bo)^{m}$$

This can be reversed to give the actual fuel mass at launch:

$$m_{launch} = m_{burn}\,(1 - \%bo)^{-m}$$

Given the mass of fuel to be contained by a given tank, the volume of the tank is a constant:

$$V = \frac{p\,D_{in}^2\,L}{4} + \frac{p\,D_{in}^3}{4} \quad \text{(body + ends)}$$

After some manipulation, this gives:

$$D_{in} = \left[\frac{12\,V}{(3l + 2)\,p}\right]^{1/3}$$

The thermal insulation is placed immediately in contact with the fuel, and has a thickness which is determined by the boiloff rate. Using a heat transfer formula provided by the Thermal group, the outer diameter of this insulating layer (also the inner diameter of the structural shell) is:

$$D_{out} = D_{in}\exp\left[\frac{2\,p\,k\,L_{cyl}\,(T_{in} - T_{out})}{Q}\right]$$

After these values are obtained, several others are easily calculated:

$$L_{cyl} = l\,D_{in}$$

$$L_{tot} = L_{cyl} + D_{out}$$

$$A_{end} = \frac{p\,D_{in}^2}{4} \quad \text{(perpendicular to axis only)}$$

The thicknesses of the structural shell at the sides and at the ends are then calculated from elastic theory:

$$S_{max} = \frac{P \cdot D_{out}}{4\,t_{end}} + \frac{P}{A_{end}} = \frac{P \cdot D_{out}}{2\,t_{side}}$$

$$t_{end} = \frac{P \cdot D_{out}}{4\left(S_{max} - \frac{P}{A_{end}}\right)}$$

$$t_{side} = \frac{P \cdot D_{out}}{2\,S_{max}}$$

And the insulation thickness is simply half the difference in diameters:

$$t_{ins} = \frac{D_{out} - D_{in}}{2}$$

The masses of the various parts of the tank are then calculated:

$$m_{tank} = m_{structure} + m_{insulation}$$
$$= r_t\,(t_{end} \cdot SA_{end} + t_{side} \cdot SA_{side}) + r_{ins}\,t_{ins}\,(SA_{end} + SA_{side})$$

$= r_t\, p\, D_{out}^2\, (t_{end} + l\, t_{side})$

$m_{tot} = m_{fuel} + m_{tank}$

$+ r_{ins}\, p\, D_{in}^2\, t_{ins}\, (1 + l)$

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| density(fuel) | | 68 | kg/m3 | (liquid hydrogen) | | | | k | 9.0E-9 | kW/mK |
| density(tank) | | 1460 | kg/m3 | (Kevlar-49) | | | | *Q* | *0.288* | *kW* |
| density(insul.) | | 133 | kg/m3 | ("super" insulation) | | | | **Tfuel** | **33** | **K** |
| *max stress* | | *3.3E+8* | *Pa* | **S.F. =** | **1.5** | | | **Tspace** | **227** | **K** |
| **Pressure** | | **1.3E+6** | **Pa** | **Fuel to burn** | | **146** | **mT** | **boiloff** | **1** | **%/mth.** |
| *V(const)* | | *2403* | *m3* | **No. of tanks** | | **1** | | *boiloff* | *6E-04* | *kg/s* |
| *m(fuel)* | | *163.37* | *mT* | *Total m(fuel)* | | *163.37* | *mT* | h(LH2) | 464 | kJ/kg |
| **# of months in space** | | | | **11.3** | | | | | | |
| | | | | | | | | | | |
| din | dout | L (cyl) | L (tot) | A(end) | t(end) | t(side) | t(ins) | m(tank) | m(t)/m(f | m(tot) |
| m | m | m | m | m2 | mm | mm | mm | mT | | mT |
| 14.21 | 14.21 | 5.7 | 19.9 | 158.6 | 13.9 | 27.7 | 1.5 | 23.28 | 14.3% | 186.66 |
| 13.99 | 14.00 | 6.3 | 20.3 | 153.8 | 13.6 | 27.3 | 1.7 | 23.49 | 14.4% | 186.86 |
| 13.79 | 13.79 | 6.9 | 20.7 | 149.4 | 13.4 | 26.9 | 1.8 | 23.68 | 14.5% | 187.05 |

**Figure A10.2:** Propellant tank sizing spreadsheet