SYSTEM DESIGN OF A
MARS ASCENT VEHICLE

by

SCOTT ALAN GEELS

S. B. Aeronautics and Astronautics
Massachusetts Institute of Technology
(1989)

Submitted to the Department of
Aeronautics and Astronautics
in Partial Fulfillment of
the Requirements of the Degree of
Master of Science

at the

Massachusetts Institute of Technology

January 1990

Signature of Author______________________________
Department of Aeronautics and Astronautics
January 1990

Certified by______________________________
Ben Clark
Martin Marietta Astronautics Group
Senior Scientist, Payloads & Sensors

Certified by______________________________
Professor Walter Hollister
Professor, Aeronautics and Astronautics
Thesis Supervisor

Accepted by______________________________
Professor Harold Y. Wachman, Chairman
Departmental Graduate Committee
Aeronautics and Astronautics

SYSTEM DESIGN OF A

MARS ASCENT VEHICLE

by

SCOTT ALAN GEELS

Submitted to the Department of Aeronautics and Astronautics
on January 19, 1989 in partial fulfillment of the
requirements for the Degree of Master of Science in
Aeronautics and Astronautics

ABSTRACT

The most mass-influential element in a manned Mars mission is the spacecraft which returns the Martian surface astronauts to Mars orbit. Therefore, the purpose of this study is to determine the approximate mass, energy, and volume required for a three-astronaut Mars Ascent Vehicle (MAV). This study, which is based on the Martin Marietta Astronautics Group's Manned Mars System Study (MMSS), also identifies enabling and enhancing technologies for this early 21st century manned Mars mission.

Through this study, it is found that the ascent portion of the Mars mission presents no significant enabling technology problems. Various enhancing technologies, which are described throughout this analysis, allow reduction of the overall vehicle mass.

Thesis Supervisor: Walter Hollister

Title: Professor of Aeronautics and Astronautics

Table of Contents

Title Page....................1
Abstract....................2
Table of Contents....................3
1.0 Introduction....................8
2.0 Missions....................10
2.1 Main Scenario....................10
2.2 Abort Scenario 1....................13
2.3 Abort Scenario 2....................13
3.0 Overall Vehicle Requirements....................15
4.0 Structures....................18
4.1 Introduction....................18
4.2 Requirements....................18
4.3 Structural Loading....................18
4.3.1 Earth Ascent Loads....................19
4.3.2 Mars Transit and Aerocapture....................19
4.3.3 Pressure Loads....................20
4.3.4 Ascent Loading....................20
4.3.5 Docking/Other Loads....................20
4.4 Options and Choices....................21
4.4.1 Material Selection....................21
4.4.2 Design Method....................22
4.5 Point Design....................22
4.5.1 Main Capsule....................22
4.5.1.1 Shell....................22
4.5.1.2 Stringers & Ribs....................25
4.5.2 Propulsion Structure....................26
4.5.2.1 Tank Structure....................27
4.5.2.2 Engine Structure....................29
4.5.2.3 Other Propulsion Structure....................29
4.5.3 Other Structural Elements....................30
4.5.3.1 MDV Connection....................30
4.5.3.2 Tank Structure....................30
4.6 Summary....................30
4.7 Recommendations....................32
5.0 Life Support System....................33
5.1 Introduction....................33
5.2 Requirements....................33
5.3 Point Design....................34
5.3.1 Consumables....................34
5.3.2 Air Circulation System....................36
5.3.3 Air Purification System....................38
5.3.4 Environmental Control....................39
5.3.3 Waste Management....................40
5.3.6 Acceleration Couches....................40
5.3.7 Spacesuits....................40
5.3.8 Reliability....................41
5.4 Summary....................41
5.5 Recommendations....................43
6.0 Propulsion System....................44
6.1 Introduction....................44
6.2 Requirements....................44
6.3 Propellant Choices....................45

6.3.1 Liquids vs. Solids....45
6.3.2 Cryogenics vs. Storables....46
6.3 Ascent Propulsion System Point Design....48
6.3.1 Propellant....48
6.3.2 Pressure- vs. Pump-Fed Systems....48
6.3.3 Staging....51
6.3.3.1 Drop Tanks....51
6.3.3.2 Full Staging....52
6.3.3.3 Summary....53
6.3.4 Engines....53
6.3.4.1 Description/Specs....54
6.3.4.2 Extendable Nozzle....55
6.3.4.3 Power....56
6.3.4.4 Helium....56
6.3.4.5 Reliability....58
6.3.4.6 Engine-Out Abort....59
6.3.4.7 Summary....60
6.3.5 Plumbing....61
6.3.6 Reliability....61
6.3.7 Configuration....61
6.3.8 Performance Summary....61
6.3.9 Mass Summary....63
6.4 OMS Propulsion System Point Design....64
6.4.1 Propellant Choice....66
6.4.2 Pressure- vs. Pump-Fed....66
6.4.3 Engines....66
6.4.4 Plumbing....66
6.4.5 Reliability....66
6.4.6 Performance Summary....67
6.4.7 Configuration....67
6.4.8 Mass Summary....67
6.5 Recommendations....68
7.0 Power System....69
7.1 Introduction....69
7.2 Requirements....69
7.3 Power Requirements....69
7.4 Options and Choices....71
7.4.1 Primary Batteries....72
7.4.2 Fuel Cells....73
7.5 Point Design....75
7.5.1 Power Source....75
7.5.2 Distribution System....77
7.5.2.1 Power Conditioning....77
7.5.2.2 Controller....77
7.5.3 Other Power Needs....78
7.5.4 Reliability....79
7.6 Summary....80
7.7 Recommendations....80
8.0 Avionics System....82
8.1 Introduction....82
8.2 Communications....82
8.2.1 Requirements....82
8.2.2 Point Design....84
8.2.2.1 Signal Power....84

8.2.2.2 Bandwidth 85
8.2.2.3 Antennae 87
8.2.2.4 Communication Interface 87
8.3 Guidance, Navigation and Control 88
8.3.1 Requirements 88
8.3.2 Point Design 88
8.3.2.1 Navigation Aids 88
8.3.2.1.1 IMU 88
8.3.2.1.2 Star Trackers 89
8.3.2.1.3 Ranging Equipment 89
8.3.2.2 Control System 90
8.4 Computer 90
8.4.1 Requirements 91
8.4.2 Point Design 91
8.4.2.1 Specifications 91
8.4.2.2 I/O Devices 92
8.5 Instrumentation & Other Avionics 94
8.5.1 Requirements 94
8.5.2 Point Design 94
8.5.2.1 Instrument Panels 94
8.5.2.2 Lighting 95
8.5.2.3 Cameras 95
8.6 Summary 95
8.7 Recommendations 98
9.0 Orbital Mechanics 100
9.1 Introduction 100
9.2 Requirements 100
9.3 Main Scenario Trajectory 100
9.3.1 Vehicle Characteristics 101
9.3.1.1 Drag Coefficients 101
9.3.1.2 Engine Characteristics 102
9.3.1.3 Vehicle Mass 103
9.3.2 Martian Atmosphere 103
9.3.3 Trajectory Description 103
9.3.4 Results 105
9.3.4.1 Thrust Profile 105
9.3.4.2 Trajectory Results 105
9.3.5 Effect of Different Drag Values 110
9.4 Abort Trajectories 110
9.4.1 Engine-Out On Pad 111
9.4.1.1 Trajectory 111
9.4.1.2 Results 11
9.4.2 Abort-on-Descent 116
9.4.3 Abort-to-Orbit 116
9.4.3.1 Orbit Synchronization 117
9.5 Summary 117
10.0 Thermal Control System 119
10.1 Introduction. 119
10.2 Requirements 119
10.3 Thermal Loads 120
10.3.1 Space 120
10.3.2 Martian Surface 120
10.3.3 Internal Heat Loads 122
10.3.4 Ascent Heat Loads 123

10.4 Options & Choices 124
10.4.1 Insulation System 124
10.4.2 Heat Collection and Rejection System 130
10.4.3 Coatings 132
10.4.4 Thermal Protection System (TPS) 132
10.5 Point Design 134
10.5.1 System Description 134
10.5.1.1 Heat Pipes 135
10.5.1.2 Ascent Heat Rejection System 135
10.5.1.3 Thermal Coatings 135
10.5.1.4 Insulation 138
10.5.1.5 Radiator 138
10.5.1.6 Main Propellant Vacuum System 140
10.5.1.7 Window 142
10.5.2 System Analysis 142
10.5.2.1 Space 142
10.5.2.2 Martian Surface 144
10.5.2.3 Martian Ascent 147
10.5.3 Summary 147
10.5.4 Reliability 148
10.6 Conclusions and Recommendations 149
11.0 Docking System 150
11.1 Introduction 150
11.1 Requirements 150
11.3 Options 150
11.4 Point Design 152
11.4.1 System Description 152
11.4.1.1 Probe 152
11.4.1.2 Drogue 153
11.4.2 Docking Scenario 153
11.4.2.1 Main Scenario 153
11.4.2.2 Abort Scenario 153
12.0 Other Systems 154
12.1 Payload 154
12.1.1 Requirements 154
12.2.2 Point Design 154
12.2.2.1 Sample Container Design 154
12.2.2.2 Location 154
12.2 Hatches 156
12.2.1 Requirements 156
12.2.2 Point Design 157
12.2.2.1 Locations 157
12.2.2.2 Summary 157
12.3 Windows 157
12.3.1 Location 157
12.3.2 Structure 157
12.4 Pyrotechnics 158
12.4.1 Requirements 158
12.4.2 Point Design 158
12.5 Tunnel to MDV 159
12.5.1 Requirements 159
12.5.2 Point Design 159
12.6 Summary 159
13.0 System Integration 162

13.1 Configuration 162
13.1.1 Subsystem Locations 162
13.1.2 Moments of Inertia 162
13.1.3 Center of Gravity 163
13.2 Comparison to Past Systems 165
13.3 Mass, Power, Volume Summaries 167
13.4 Mission Stages 168
13.4.1 Main Scenario 168
13.4.2 Abort Scenarios 168
13.5 Sensitivity Analysis 169
13.5.1 Crew Size 169
13.5.2 Payload Ratio 170
14.0 Conclusions 171
15.0 References 173
Appendix A: POST Input File 177
Appendix B: SINDA Input File 190
Appendix C: Table of Acronyms 195

1.0 Introduction

There has been an endorsement by the current presidential administration of placing a man on Mars early in the 21st century. The political and national pride obtained from landing the first man on another planet is enormous. Other benefits include the potential for scientific discoveries on Mars, and the pushing of the state of the art in technology, which will have "spin-off" technological benefits.

The purpose of this study is to make a point design of the spacecraft which transports the Martian astronauts from the surface of Mars to Martian orbit. Because this Mars Ascent Vehicle (MAV, see Figure 1-1) must descend to and ascend from the the Martian surface as well as make the entire journey to Mars, it is the most mass sensitive element in a manned Mars mission scenario. For each MAV kilogram that ascends back into Areosynchronous Mars Orbit (AMO), up to 20 kilograms of mass in Lower Earth Orbit (LEO) is required to reach that point. Because of this extreme mass sensitivity, the MAV is an important design element.

The MAV in this paper is designed based on a manned Mars mission scenario from the Manned Mars System Study (MMSS) at Martin Marietta Astronautics Group. All of the non-MAV spacecraft and scenarios are based on this study. Even though this makes the MAV point design specific to one mission design, the overall vehicle design should carry over into any manned Mars mission in the near future with only minor modifications.

This study first does some parametric studies in each subsystem in order to justify the choices made in the point design, minimizing mass whenever feasible. The point design gives figures for total masses, powers, and volumes required as well as pinpointing problem spots in MAV design and areas requiring further research. The design employs current available and state of the art technology, giving a baseline design into which technological advances can be integrated at a later point.

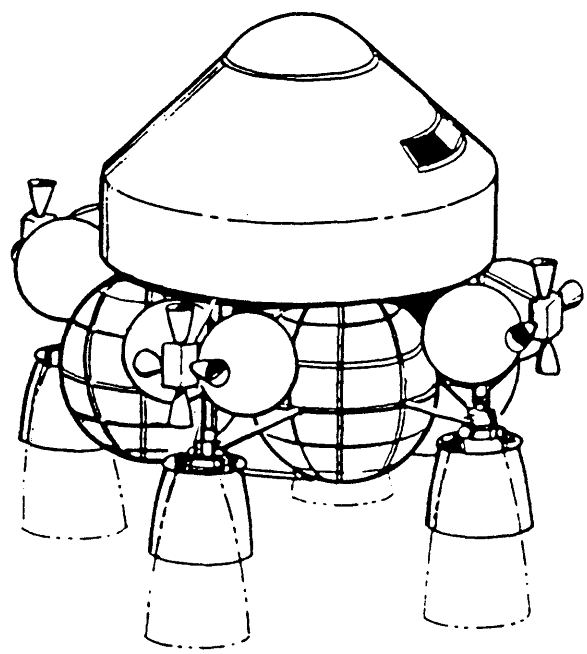

Figure 1-1: Mars Ascent Vehicle, External View

2.0 Missions

2.1 Main Scenario

In the baseline manned Mars mission, five astronauts make the journey from Earth to Mars in the Mars Transfer Vehicle (MTV, see Figure 2-1) in the year 2005. If the mission is conjunction class, this trip takes a maximum of a year, although different orbital mechanics paths, such as with sprint and opposition class missions, take less travel time.

When reaching Mars, the MTV aerocaptures into a 250 km by 33850 km areosynchronous orbit with a 37° inclination. At this point, three astronauts descend in the Mars Descent Vehicle (MDV) to an equatorial landing site on the Martian surface. The MDV employs an aerobrake to slow its path through the Martian atmosphere. This aerobrake is discarded before landing, allowing the MDV to propulsively lower to the surface. In case an abort becomes necessary on descent, the astronauts are seated in the Mars Ascent Vehicle (MAV), which is attached to the top of the habitation module of the MDV (see Figure 2-2).

The astronauts spend the equivalent of one Earth-year on the Martian surface, exploring, collecting samples, and running experiments. During this stay on the surface, the MAV is always operational and ready for immediate abort-to-orbit in case of an emergency. The samples are stored in the MAV because of this abort possibility.

After the year-long stay on the Martian surface, the astronauts ascend in the MAV back into the aerosynchronous orbit, and rendezvous directly with the Mars Orbiting Vehicle (MOV), the remaining section of the MTV. The MAV then docks with the MOV with the help of the on-orbit astronauts, permitting the transfer of the surface astronauts and samples to the MOV. In order to help with the return flight payload capability, the MAV is abandoned in the Martian orbit as the MOV returns to Earth.

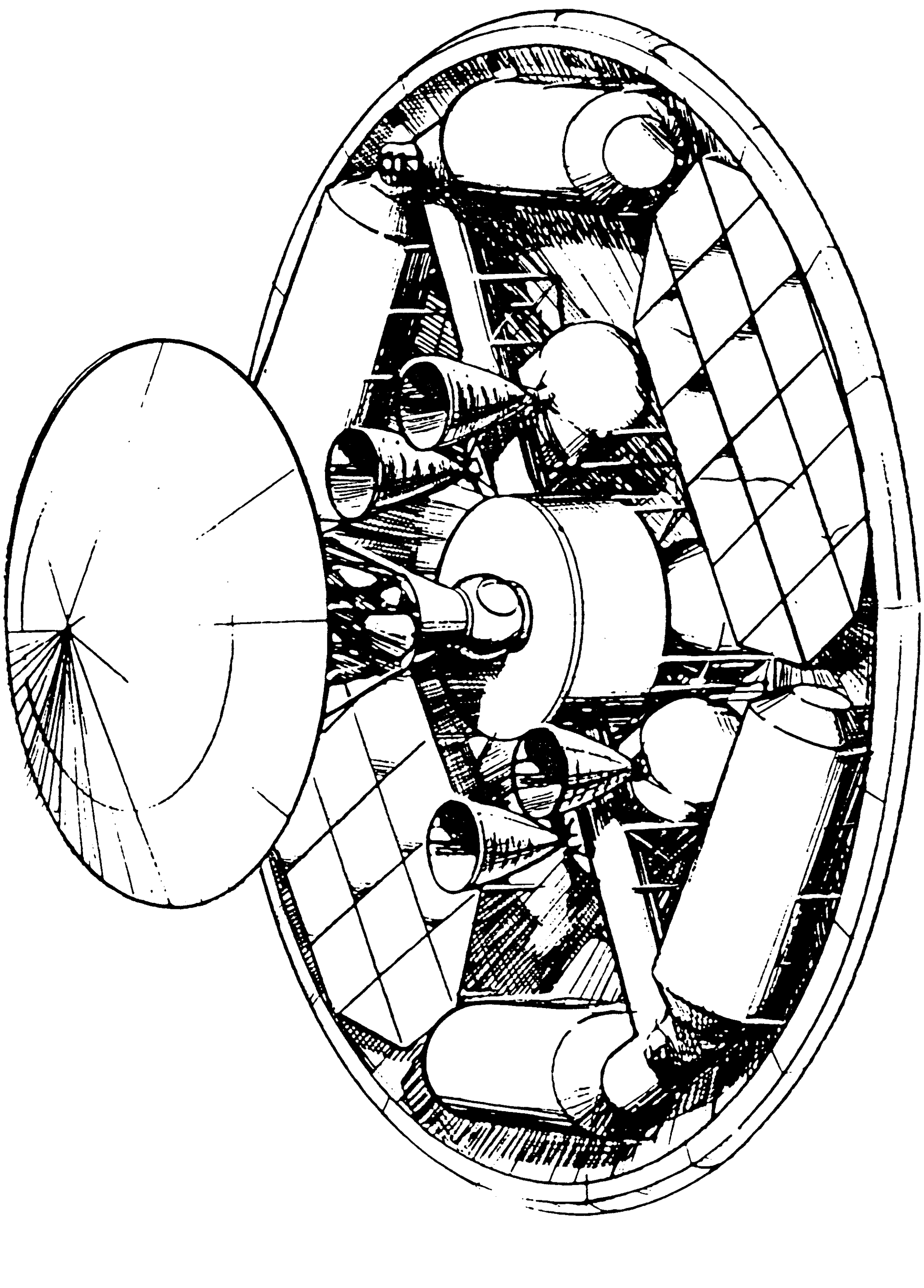

Figure 2-1: Mars Transfer Vehicle (MTV)

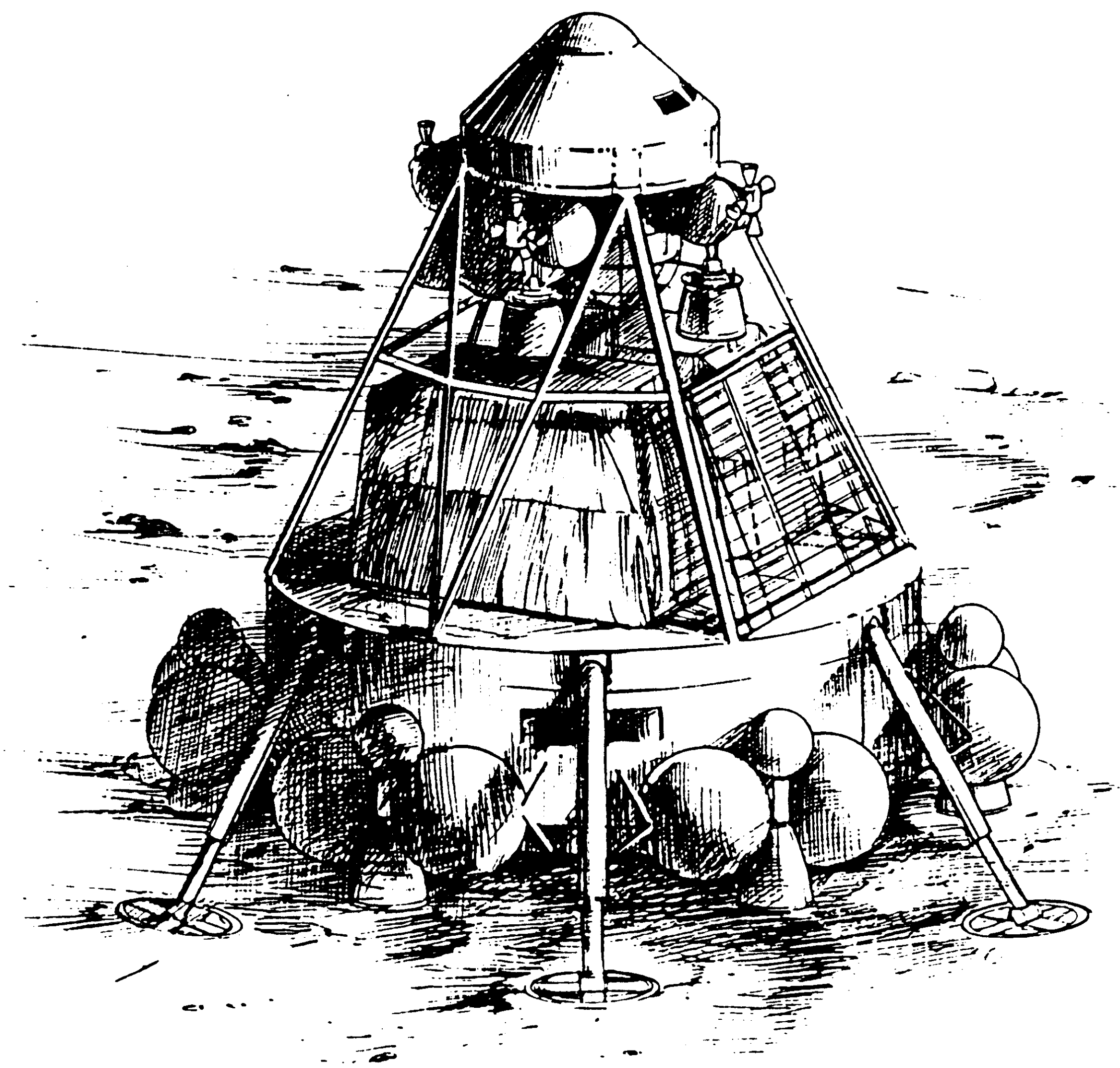

Figure 2-2: Mars Descent Vehicle (MDV)

2.2 Abort Scenario 1

The first abort scenario provides an abort during descent to the Martian surface. This could become necessary under several conditions, including MDV system failure and failure to locate a safe landing site. Unless there is a backup MDV, this means an overall mission failure as well as a Mars landing abort.

If an emergency occurs on descent, the MAV detaches from the MDV habitation module. The MAV engines then ignite, and the MAV ascends back into the orbit from which it had descended. Unfortunately, the MAV and the MOV are no longer orbit synchronized at this point, making rendezvous impossible.

To solve this problem, the MAV first ascends into a 250 km circular orbit. It then thrusts into an elliptical orbit with the same perigee location as the MOV. The apogee of the orbit is determined by how much the MAV lags behind the MOV in the orbit, and is fixed such that the two vehicles will be in the same location at the next perigee passage. At this point, the MAV thrusts to match velocities with the MOV, and the rendezvous and docking sequence occurs. This scenario should always take less than one sol (Martian day = 24.66 hours).

The only time this scenario does not allow abort on descent is when the MDV is so close to the Martian surface that the MAV cannot detach before a collision with the Martian surface occurs.

2.3 Abort Scenario 2

A second abort scenario is an abort-to-orbit sequence from the Martian surface. This scenario allows the surface astronauts to ascend back to the rendezvous orbit from the surface landing site at any time during their stay on the surface.

Before the ascent phase begins, the MAV must be able to ascend into the MOV orbit plane. This means that the MAV may have to wait up to 12.33 hours on the Martian surface before the landing site aligns with the ascending or descending node of the MOV orbit. When this occurs, the ascent sequence begins.

The ascent sequence is a combination of the first two scenarios. The normal detachment from the MDV habitation module occurs, and the MAV ascends into a 250 km circular orbit. Once again, the MAV and MOV must synchronize their orbits.

The only difference between this synchronization of orbits and the one described in the previous section is the possible location of the MOV. In the first abort scenario, the MOV is guaranteed to be in front of the MAV, assuring that rendezvous and docking can occur in a maximum of one sol. As a worst case in this abort scenario, however, the MOV could lag the MAV by less than 112 minutes, the period of a 250 km circular orbit. This would mean that it would take 26.5 hours (1 sol + 112 minutes) in order to synchronize the two orbits plus another 12.33 hour maximum stay on the surface before the orbit nodes align. This worst case rendezvous time of 38.8 hours sets the upper bound for the necessary life support system capability of the MAV.

3.0 Overall Vehicle Requirements

Before presenting the MAV subsystem designs, it is first necessary to establish the requirements on the entire vehicle. These requirements establish the requirements on the subsystems, which in turn determine the design of these subsystems.

By looking back at the described manned Mars scenarios (see Section 2.0), it is clear that the MAV must perform two major functions:

- transport 3 suited astronauts from the Martian surface to the MOV orbit
- provide a survivable environment for 38.8 hours (see 5.0 for details)

In order to successfully complete its mission, the MAV must be able to reach the MOV areosynchronous orbit. To accomplish this task, the MAV must:

- ascend to a 37° inclination, 250 km x 33850 km altitude orbit
- ascend from a 0° latitude, 0° longitude, Martian surface site
- have a propulsion system and a guidance, navigation and control system (GN&C) capable of reaching this orbit

Once the spacecraft reaches orbit, it rendezvous and docks with the MOV. To perform these functions, the MAV must have:

- Orbital Maneuvering System (OMS) capable of rendezvous
- approach $\Delta V < 0.1$ m/s between the MOV and the MAV
- docking mechanism compatible with the MOV
- pressure compatible with the MOV (or with airlock on the MOV)

To provide a man-rated ascent to the MOV orbit, the following safety requirements are specified:

- no wait MAV safe haven on the Martian surface
- abort-to-orbit on descent to the Martian surface

- .96 overall vehicle reliability (.995 subsystem reliability)
- no single point failures in design
- constant communication link between MAV and MOV
- possible shielding against radiation
- component shelf-life of 2 years

In addition to carrying the astronauts into orbit, the MAV also has to transport any samples that are being returned to Earth for further study. These include soil samples, atmospheric samples, and film. Therefore, the MAV must provide:

- any special environments required for cargo (see Section 12.0)
- capability to carry up to 100 kg of cargo on ascent only

Since the MAV will be connected to the MDV habitation module, the MAV must be fully compatible with it (see Figure 3-1). This means that the MAV must provide:

- entrance to the MDV habitation module (see Figure 3-1)
- pressure compatible with the MDV to avoid prebreathing (5 psia)
- geometry that fits inside the aerobrake inpingement cone of the MDV upon entering the Martian atmosphere (60°, see Figure 3-1)
- volume for the manned rover underneath the spacecraft

Generally, the mass should be minimized due to the sensitivity of the MAV to the mass in LEO. This minimization, however, is constrained by the need for reliability and safety in a man-rated spacecraft.

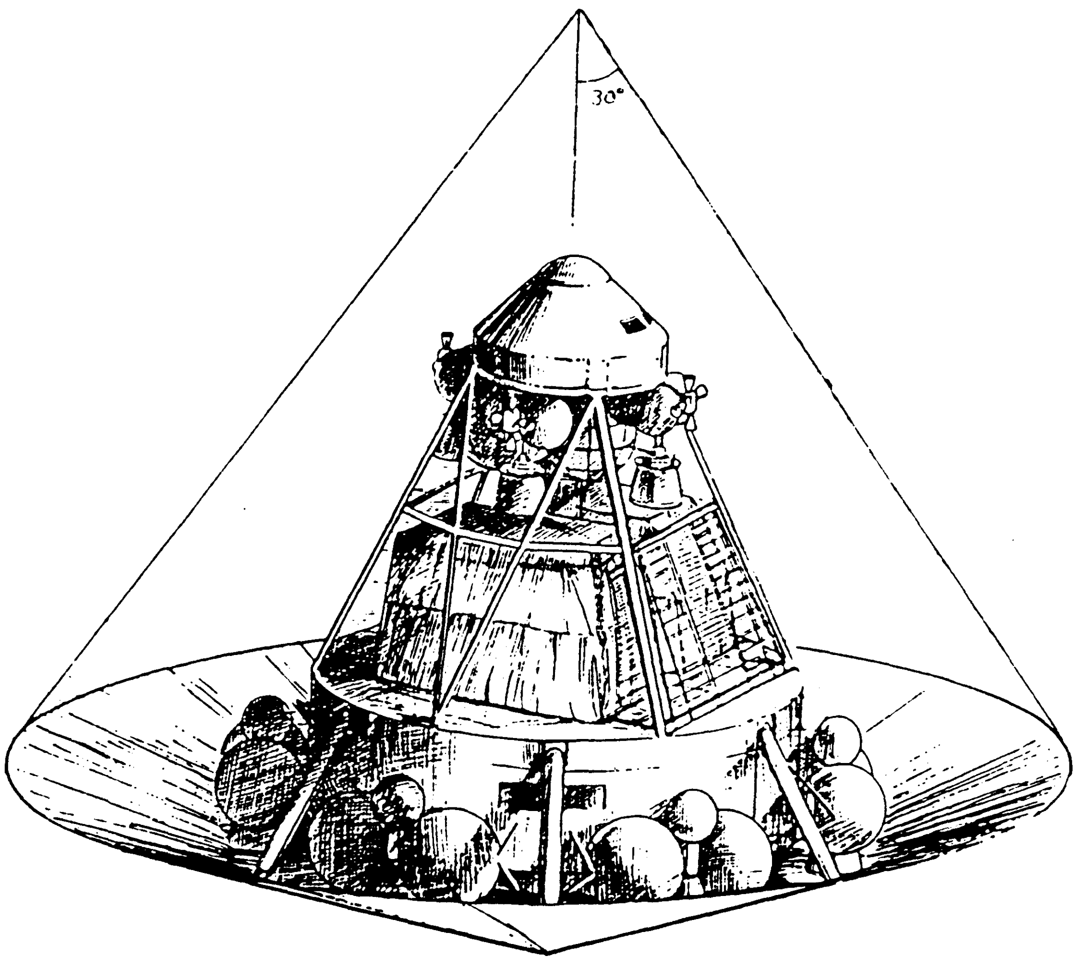

Figure 3-1: Aerobrake Impingement Cone

4.0 Structures

4.1 Introduction

Structural elements are required to withstand internal pressure loads, as well as landing, docking, and ascent loads. Additional structure is required for the propulsion system, and for the numerous pressurized tanks throughout the spacecraft.

4.2 Requirements

Each of the MAV structural components must withstand the loads placed upon it:

- acceleration loads - Mars ascent, Earth ascent, Mars aerocapture
- aerodynamic loads on Mars ascent
- internal pressure loads
- thermal loads on Martian ascent
- vibrational and acoustic loads on Earth-to-orbit ascent
- docking loads
- Mars landing loads

Additionally, the MAV structure has other requirements:

- withstand internal pressure and temperature loads (for both capsule and tanks)
- factor of safety = 2.0 (from ultimate failure stress)
- avoid corrosion effects
- shelf-life = 2 years

4.3 Structural Loading

The MAV must withstand structural loads on Earth ascent, Mars transit, Mars landing, and Mars ascent and docking. Additional loads stem from internal cabin pressures, as well as from pressures on propellant, helium, and oxygen tanks.

4.3.1 Earth Ascent Loads

Before the MAV makes its voyage to Mars, it must first be placed into Earth orbit. Assuming that it is launched in the STS, the MAV expects to see a maximum constant acceleration load of about 3 g's, less than the Mars aerocapture loads (see Figure 4-1). Vibrational loads will be significantly higher, however.

To avoid sizing the MAV structure based on Earth loadings, the MAV could be braced on ascent to withstand the higher STS vibrational loads. The MAV cabin would also need to be open to the atmosphere, to avoid having to size the spacecraft skin for the 67 kPa (9.7 psia) difference between Earth sea level pressure and MAV cabin internal pressure.

4.3.2 Mars Transit and Aerocapture

Trans-Mars injection acceleration to escape Earth orbit are expected to be about 0.2 to 2.0 g's, which won't affect MAV structural design. Mars aerocapture loads, on the other hand,

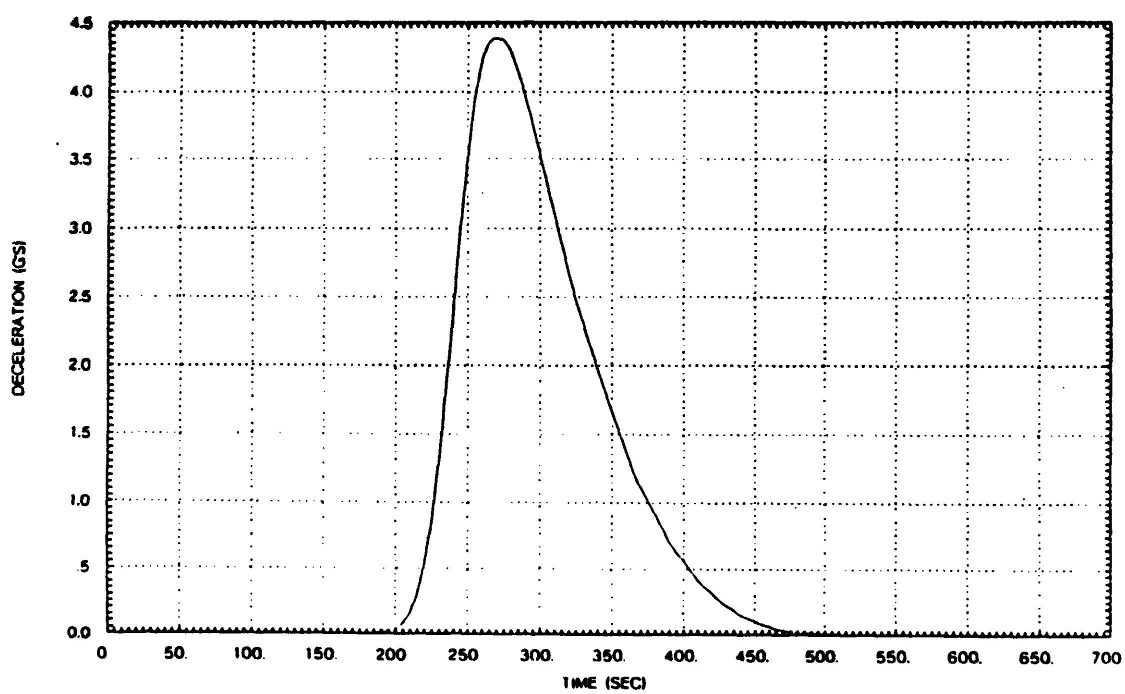

Figure 4-1: Typical Mars Aerocapture Loads (MMSS, 1989)

approach 4.5 g's (see Figure 4-1).

4.3.3 Pressure Loads

Internal pressure loads for the MAV cabin, and propellant, helium, and oxygen tanks are summarized in Table 4-1. Information is drawn from Sections 5.0 and 6.0.

Table 4-1: Internal Pressure Loads

Structural Element	Internal Pressure, MPa (psia)
cabin pressure	0.035 (5)
NTO tanks	0.345 (50)
MMH tanks	0.207 (30)
N_2H_4 tanks	2.069 (300)
He tanks	31.03 (4500)
O_2 tanks	3.448 (500)

4.3.4 Ascent Loading

The Mars ascent loads are described in Figures 9-4 and 9-11. Peak loads are approximately 2 (Earth) gee's. Propulsion structure must be sized for the maximum single engine thrust of 53.3 kN (12000 lbf), and the maximum total thrust of 133.4 kN (30000 lbf).

Additional loads come from dynamic pressure and thermal loads on ascent. Dynamic pressure peaks at approximately 810 Pa (17 psf; Figure 9-6), while surface temperatures are maintained at about 205 °K, which is well within the material plastic limits of the MAV skin.

4.3.5 Docking/Other Loads

Docking loads are significant due to their applied location. All of the previously mentioned acceleration loads are applied at the mass locations, producing a downward force with respect to the MAV center line.. The docking load, on the other hand, is applied at the front of the MAV, causing an upward (+ Z axis) compressive force on the MAV.

The Apollo Lunar Excursion Module (LEM) was sized for a 4 g docking impact. It is expected, that with state-of-the-art guidance, navigation, and control (GN&C) systems, this impact loading can be reduced to about 1 g.

Additional loads are derived from descent impact, which is expected to be approximately 5 g's (MRSR, 1989).

4.4 Options and Choices

4.4.1 Material Selection

The MAV structural design utilizes three materials; high-strength, isotropic graphite/epoxy, titanium, and aluminum. Table 4-2 illustrates the material characteristics employed in this structural analysis and for thermal analysis (see Section 10.0).

Table 4-2: Material properties

Properties	Gr/Ep (isotropic, high strength)	Titanium (Ti6 A1-4 V)	Aluminum (7075-T6)
Ult. tensile stress, MN/m^2	724	1034	523
Ult comp. stress, MN/m^2	690	1034	523
Youngs modulus, GN/m^2	83	110	71
Density, kg/m^3	1490	4430	2800
Conductivity (W/m-°K)	1.5	7.4	134
Specific heat (J/kg-°K)	800	837	502

4.4.2 Design Method

The propulsion structure was designed such that engine thrust would not create additional loads on the main capsule structure. It is a self-contained structure which allows the engines to first lift the propellant tanks before exerting any forces on the MAV capsule. Therefore, the main cabin need only take the pressure, acceleration, and docking loads. The propulsion structure absorbs all engine thrust loads.

The MAV cabin utilizes a two hull, stringer-rib construction. This gives a lightweight capsule structure. The skin is sized in order to take the pressure loads, and the stringers absorb the acceleration and docking loads.

4.5 Point Design

4.5.1 Main Capsule

The rib, stringer, and shell design utilized for the MAV main cabin is illustrated in Figure 4-2. To facilitate attachment of avionics, wiring, and ducting to the inside of the spacecraft, the inner shell and stringers are made up of aluminum. To reduce the outer shell mass, it is constructed of lightweight graphite/epoxy coated with a thin metallic coating, such as aluminum (to prevent leakage).

4.5.1.1 Shell

The MAV utilizes a double shell, to protect against hull rupture. To size the minimum required thickness of the cylindrical portion of the pressure shells, the hoop stress is determined:

$$t = \frac{P\,r}{\sigma_t}$$

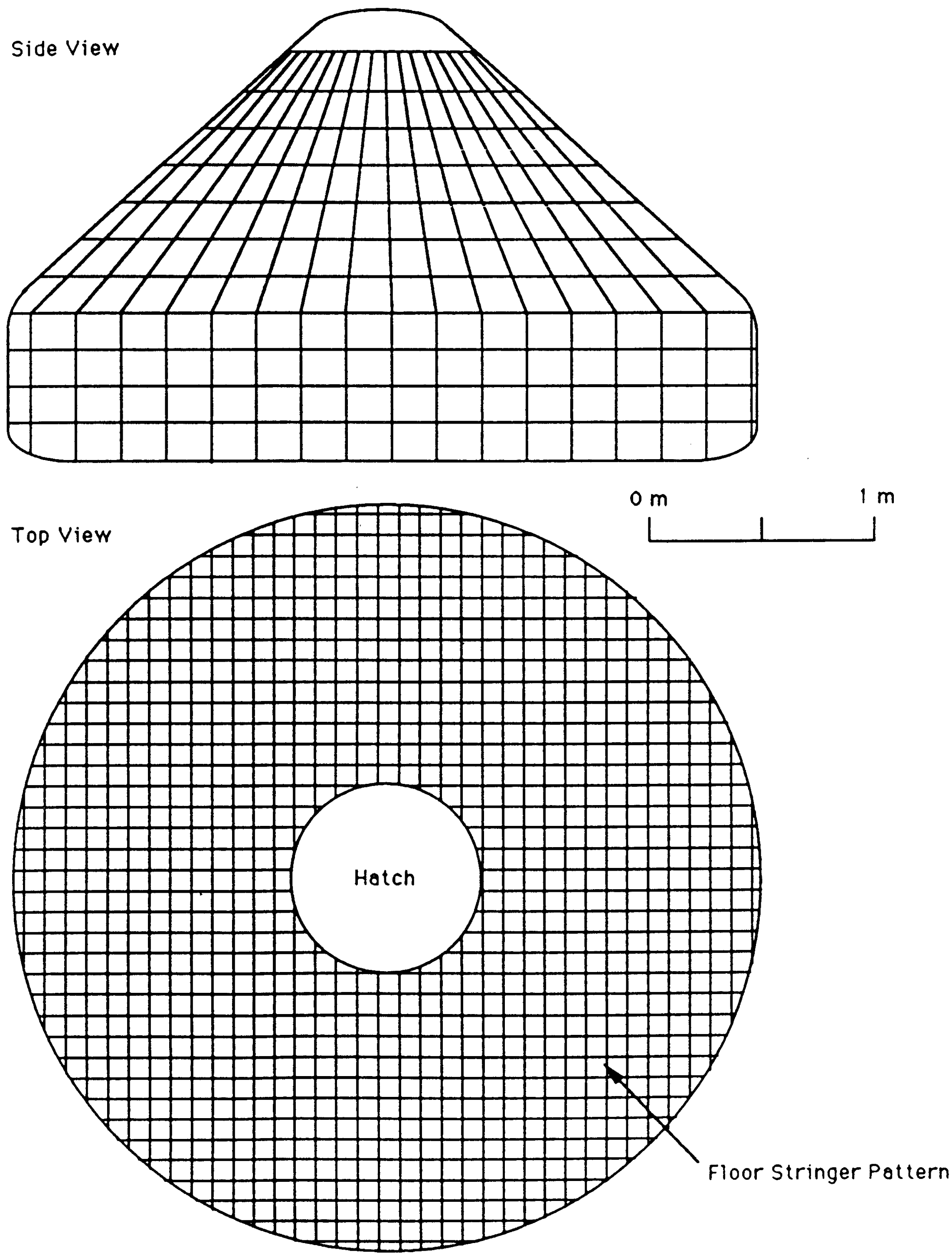

Figure 4-2: Main Capsule Structure

t = thickness of shell

P = internal pressure = 0.0345 MPa

r = radius of shell = 1.68 m

σ_t = material tensile strength = 523 (aluminum) or 724 (Gr/Ep) MN/m^2

Including the safety factor, the aluminum shell only needs to be 0.222 mm (0.0087 in) and the Gr/Ep shell requires a 0.16 mm (0.0063 in) thickness. Because the lateral pressure forces on the conic section are less than the axial pressure forces, the conic section also requires this minimum thickness. Dynamic pressure forces are insignificant compared to the internal pressure forces.

This required thickness is insufficient to eliminate the danger of accidental puncture. Therefore, the inner shell is sized for a 1.27 mm (0.05 in) thickness, while the Gr/Ep outer shell, which is not exposed to the astronauts, is sized for a 0.762 mm (0.03 in) thickness. The Gr/Ep shell is coated with a thin aluminum layer to minimize outgassing.

The aluminum floor pressure shell of the MAV is 1.4 mm (0.055 in). It is sized to withstand the pressure forces, plus the force of an astronaut standing at the center of one of the square panels. The shell thickness determines the distance between the stringers:

$$t = \left(\beta \frac{Pb^2}{\sigma_t}\right)^{0.5} \quad \text{(Rourk, 1954)}$$

β = constant = 0.50 (for square panels)

b = panel width

P = pressure load + astronaut standing on panel

For an 82 kg astronaut on Mars, and a desired floor thickness of 1.4 mm (0.055 in), the stringers must be (3 in) apart. This translates to a total face plate pressure of (12.4 psi)

The outer Gr/Ep shell is sized only for the internal atmospheric pressure loads, and is 0.762 mm (0.03 in) thick.

4.5.1.2 Stringers & Ribs

32 cabin wall stringers and 12 ribs absorb all of the non-pressure loads. The most significant loading occurs during docking, when the stringers' principle failure mode is buckling:

$$P_{cr} = \frac{\pi^2 EI}{L^2}$$

P_{cr} = failure load = MAV dry mass * 1 gee acceleration = 26.6 kN

E = Young's modulus = 71 GN/m^2

I = stringer minimum moment of inertia

L = stringer length

Assuming the stringers are end-fixed by the hoops (see Figure 4-2), the stringer length is 0.1778 m (7 in). If the docking load is off-center, so that only half of the stringers are employed, the required moment of inertia (with FOS) for each stringer is 1.5×10^{-10} m^4. With square stringers (side = 6.5 mm), this translates to 8.34 kg of stringer mass. The mass could be further reduced by utilizing hat or I-beam stringers.

Each of the 12 hoops must be able to absorb the docking load transferred from the stringers. By utilizing square cross-sectional hoops, the ribs must be 3.81 mm (0.15 in) along the axial direction, and 1.9 mm (0.84 in) along the lateral direction. This results in a total of 2.03 kg of hoop mass.

Because the propulsion structure supports the floor of the MAV, not a great deal of floor stringer mass is required.

4.5.2 Propulsion Structure

Figure 4-2 demonstrates the propulsion structural system.

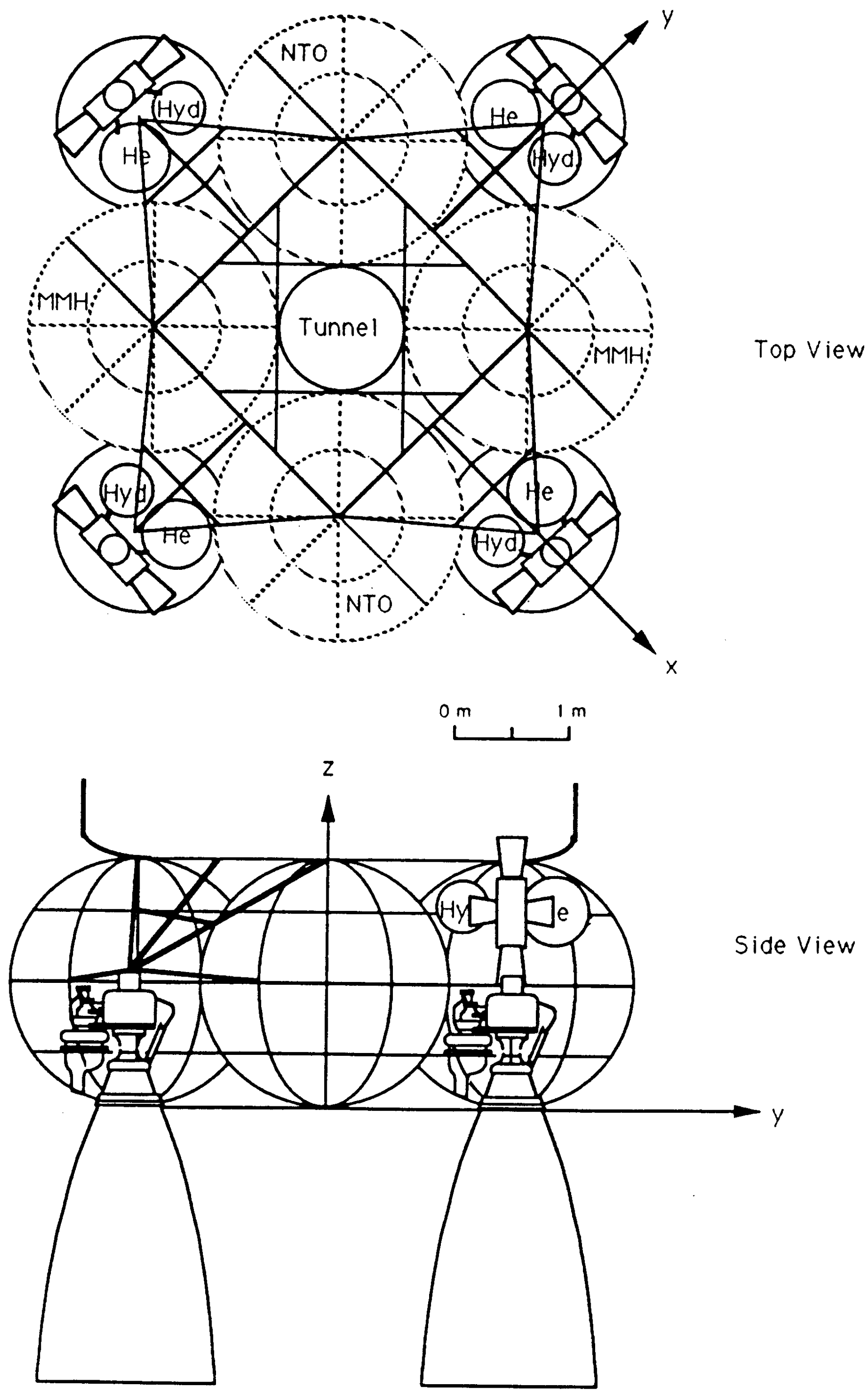

Figure 4-2: Propulsion System Structure

The propulsion structure is designed such that only acceleration forces are exerted on the main capsule structure.

4.5.2.1 Tank Structure

The tank structure consists of a rib, stringer, and shell external design, with internal supports to absorb the majority of the propellant mass loads (see Figure 4-3). To avoid structure-propellant compatibility problems, titanium is utilized for the propellant tank structure.

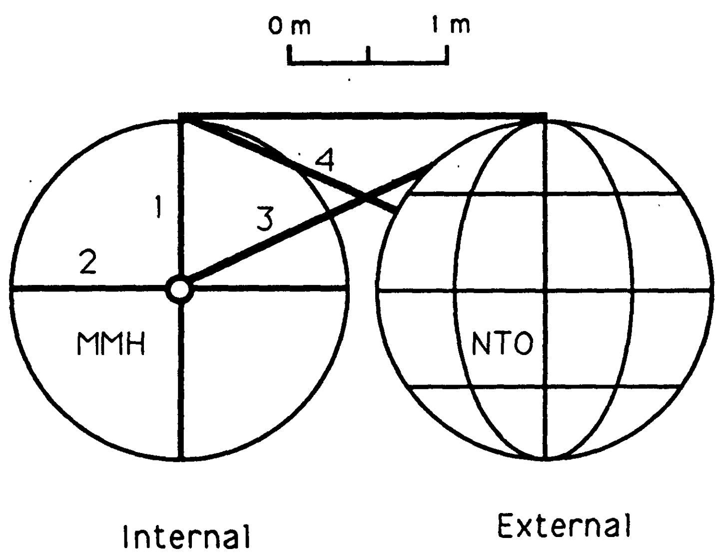

Figure 4-3: Tank Structure

The outer tank shell supports only the pressure loads. As with the main cabin shells, however, this required thickness is not enough to eliminate the possibility of tank accidental puncture. Using the hoop equation for spherical shells, it is found that the MMH tanks require only a 0.214 mm (0.008 in) thickness, while the NTO tanks need to be 0.356 mm thick (0.014 in):

$$t = \frac{P\,r}{2\sigma_t}$$

To avoid accidental puncture, the tank shells are sized for a 0.762 m (0.03 in) thickness.

The internal vertical strut (#1, see Figure 4-3) is designed to hold the entire propellant mass tension force during maximum mission loading:

$$A = \frac{F}{\sigma_t}$$

A = cross-sectional area of strut

F = propellant mass force = m_{prop} * 5 gee's

Therefore, each NTO vertical strut has a total mass of 3.28 kg, while the MMH strut has a mass of 1.7 kg. Because of these supports, the external structure of stringers and hoops need only support the local propellant masses, and not the total propellant mass. This support structure can thus be extremely lightweight.

The horizontal internal tank supports (#2) and the internal supports that connect the tanks together (#3 and #4) are expected to maintain the MAV propellant "self-contained" structure. If an engine-out occurs, these supports hold up the tanks that are no longer being lifted by the main engines, preventing forces from being exerted on the MAV capsule.

Each of these supports must not buckle or compressively fail while supporting the 2 gee force on the adjacent propellant tanks. To accomplish this, each support that is absorbing the NTO propellant mass (#3) must have an area moment of inertia of 2.53×10^{-7} m^4, while each MMH support (#4) must have a 1.26×10^{-7} m^4 area moment. By utilizing hollow pipe supports, 25 mm in external radius, 16 mm internal radius, each of the four NTO supports has a mass of 12 kg. Similarly, the 22 mm external, 16.4 mm internal radius MMH supports have a mass of 7 kg each.

The MMH horizontal internal supports (#2) have a required area moment of 2.6×10^{-8} m^4, translating to 2.76 kg for each of the four 15 mm external, 11.5 mm internal radius hollow cylinder supports. The NTO horizontal internal supports require a 5.2×10^{-8} m^4 moment, translating to a mass of 3.75 kg for each 18 mm external, 14 mm internal radius strut.

4.5.2.2 Engine Structure

The upper main engine structure is sized for a combination of buckling and compressive failure. Worst-case loading occurs if two engines are operating at maximum 53.3 kN thrust levels. In addition, if the engine is gimballing, the majority of the loading occurs on only one of the support struts (see Figure 4-2).

Each engine support must have an area moment of inertia of 9.8×10^{-8} m^4. By again employing pipe supports, each of the three main supports has a mass of 5.68 kg, for a 20 mm external, 14 mm internal radius strut.

Additional structure, required to sustain the attachment of the engine to the propulsion system, is connected to the tank horizontal internal supports. These structural element primarily withstand the engine tension forces of 53.3 kN, and have a total mass of about 5 kg.

4.5.2.3 Other Propulsion Structure

The remainder of the propulsion structure consists of a truss structure located on the top and bottom of the propulsion system (see Figure 4-2). The top truss must withstand the compression forces of the main engines, and is thus sized to avoid buckling and compressive failure.

As before, the worst-case loads are for maximum thrusting engines during the engine out scenario. The truss members, to which the engines are attached, are sized for a worst-case compressive loading of 42.6 kN (gimbaled engine, full thrust). This results in an approximate required area moment of 3.1×10^{-7} m^4. Utilizing a 30 mm external radius, 25.3 mm internal radius pipe strut, the top truss has a mass of 65.1 kg. The bottom truss does not take direct engine loadings, and thus requires less mass.

4.5.3 Other Structural Elements

4.5.3.1 MDV Connection

The MAV must be connected to the MDV habitation module. This structure (see Figure 2-1) fails through compression and buckling loads. If the MDV lands off-center, each of the eight support struts must have a 2.84×10^{-6} m^4 area moment. This translates to 302 kg of total mass for 60 mm external, 55.3 m internal radius supports.

Fortunately, this mass is not taken with the MAV when it ascends into Martian orbit.

4.5.3.2 Tank Structure

Tank masses throughout this design were determined using hoop stresses and utilizing a 2.0 non-ideal tank factor (Redd, 1989).

4.6 Summary

Table 4-3 summarizes the information in this section. Propellant tank masses are included in the propulsion system mass in Section 6.0 and are not included in the total structural mass.

Table 4-3: Structure Summary

Element	Number	Total Mass (kg)	Energy (W-hr)	Volume (m^3)
cabin structure:				
inner shell (conic)	1	64.8	0	0.023
inner shell (floor)	1	34.5	0	0.012

Table 4-3: Structure Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
outer shell (conic)	1	20.7	0	0.014
outer shell (floor)	1	10.0	0	0.007
stringers (conic)	32	8.3	0	0.003
stringers (floor)	64	6.8	0	0.002
hoops	12	2.0	0	0.001
welds, attachments		30.0	0	0.010
propulsion structure:				
NTO shell	(2)	(96.6)	(0)	(0.022)
MMH shell	(2)	(96.6)	(0)	(0.022)
internal vert. struts	4	10.0	0	0.002
other int. supports	16	102.0	0	0.023
engine structure	4	73.2	0	0.017
top truss	1	65.1	0	0.015
bottom truss	1	10.0		0.007
OMS support	4	10.0		0.007
He support	4	5.0		0.004
MAV connections		10.0		0.007
MDV connections		15.0		0.011
welds, attachments		30.0		0.021
prop. structure	34	330.3	0	0.112
capsule structure	112	177.2	0	0.072
TOTAL	146	507.5	0	0.185

4.7 Recommendations

This was a first cut analysis, assuming that the main failure mode was the only possible failure. Further analysis should be performed to determine the effects of torsion and shear stresses.

Propulsion structural mass could be further reduced by utilizing advanced materials for all propellant tank structure. If the propellant contamination and out-gassing problems are resolved (possibly by coating the Gr/Ep with a non-reactive metal), the tanks themselves can be made up of isotropic Gr/Ep.

5.0 Life Support System

5.1 Introduction

The life support system (LSS) of the MAV must provide a liveable environment by supporting three astronauts for a period of 38.8 hours. This includes maintaining an acceptable temperature, pressure, and humidity, as well as providing the astronauts with the needed food, water, and air for the duration of the ascent to Mars orbit.

5.2 Requirements

To provide a liveable environment for the astronauts, the following requirements must be met:

- humidity between 25% - 75% (NASA-STD-3000, 1987)
- temperature between 21°C and 27°C (NASA-STD-3000, 1987)
- pressure between 24.7 kPa (100% Oxygen) and 101 kPa (21% Oxygen)
    - compatible with MDV habitation module (34.4 kPa)
    - compatible with MOV (34.4 kPa)
- air revitalization - removal of CO_2, pollutants
- waste management system for 38.8 hours

The following consumables have to be provided for the 38.8 hour mission:

- food for 38.8 hours
- water for 38.8 hours
- air for 38.8 hours
- FOS = 1.15 (44.6 hours)

In addition to these requirements on consumables and environment, there are a number of other requirements based on preserving the safety of the astronauts during the mission:

- spacesuits for all astronauts in case of hull breach or air supply leakage

- 1.5 m^3 per astronaut for working space / sanity
- acceleration limits (NASA-STD-3000, 1987, Figure 5.3.3.1-1)
    - 5 g's sustained (> 10 minutes)
    - 10 g's peak (< 1 minute)
- acceleration couches if necessary
- rudimentary first aid equipment
- fire protection equipment
- shelf-life of 2 years
- no single point failures; dual-fault tolerant in most systems
- reliability of .995

5.3 Point Design

Figure 5-1 illustrates the LSS layout. All elements are located so that they are within reach by the astronauts, permitting repair accessibility throughout the 2-year mission. Due to better reliability, lower power requirements, and reduced research and development costs, an open LSS is baselined for this short duration, 38.8 hour mission.

5.3.1 Consumables

To determine the quantity of required air for a 38.8 hour mission, a 34.4 kPa (5 psia) pure oxygen environment is baselined. This pressure level, which has been previously employed on Apollo, Gemini, Mercury, and Skylab missions, is compatible with both the MDV habitation module and the MOV (MMSS, 1989), eliminating any EVA pre-breathe requirements. Opting for 34% of Earth atmospheric pressure also reduces the MAV structural mass and air leak rates. Furthermore, a pure oxygen environment simplifies the air distribution system (see Section 5.3.2)

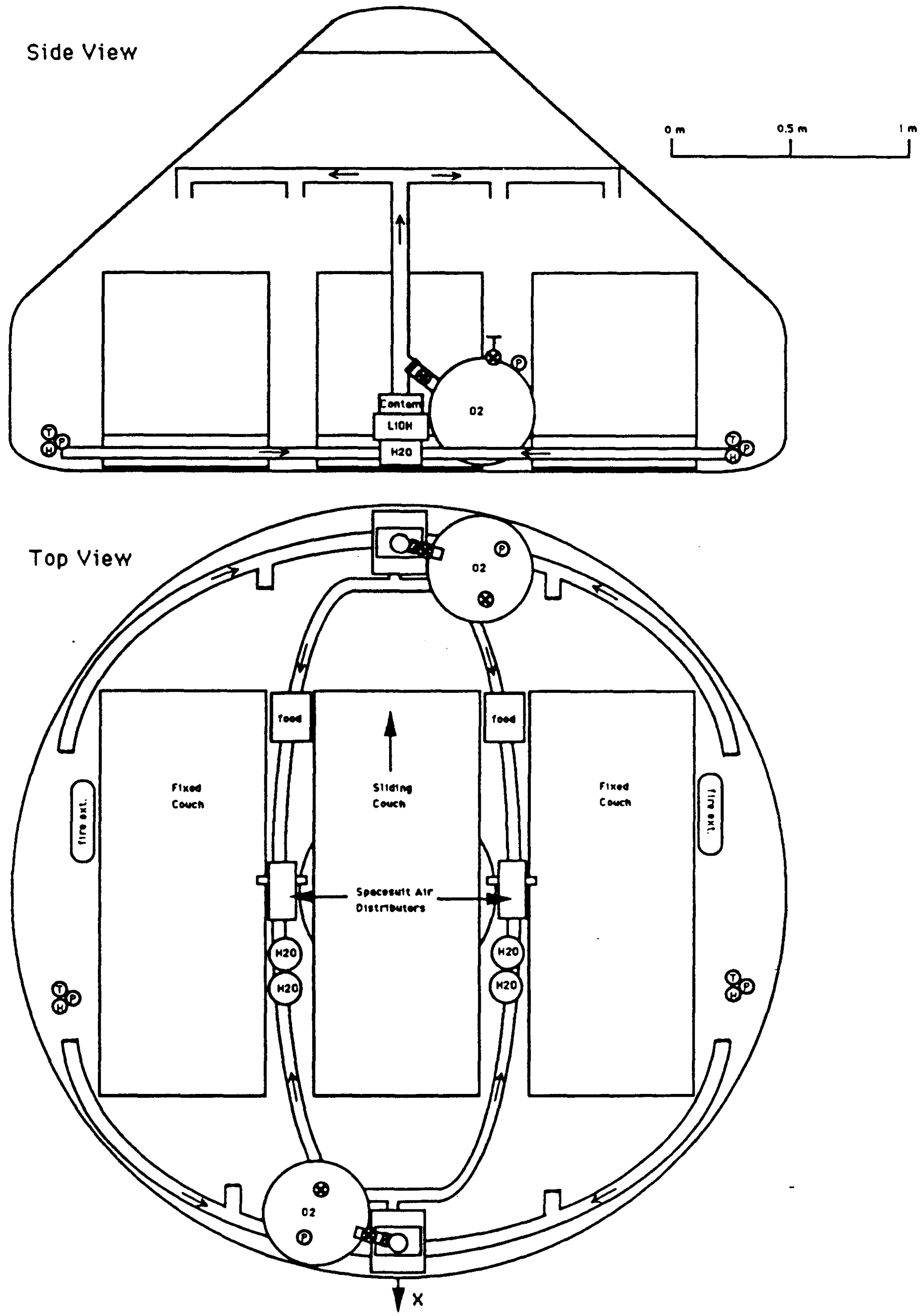

Figure 5-1: LSS Layout

Since the necessary partial pressure of O_2 for alveolar oxygen levels is only 24.7 kPa (3.6 psia), the 34.4 kPa oxygen environment provides a 1.39 factor of safety. In general, for missions over a couple of weeks, an inert gas must be provided (NASA-STD-3000, 1987), but this not necessary for the short-term MAV mission. Oxygen is utilized at a rate of 0.84 kg/p(person)-day (MMSS, 1987).

For the maximum duration mission, extensive quantities of food are not required, since an astronaut could easily survive without food for 38.8 hours. High-caloric packaged food is provided based on a 0.617 kg/p-day consumption rate and an additional food storage factor of 0.45 kg/p-day rate (MMSS, 1987).

Water is nominally provided to sustain a consumption rate of 3.63 kg/p-day (MMSS, 1987) by the astronauts, but 0.4 kg/p-day (Clark, 1988) is sufficient for survival. On-board canned water provides this minimum quantity, with the byproduct fuel cell water producing a useable water surplus.

5.3.2 Air Circulation System

A system of fans and ducts circulate the air throughout the cabin (see Figure 5-1 and Figure 5-2). The air is captured near the floor by fans, purified, supplemented with oxygen , and returned to the cabin through vents located on the walls of the spacecraft. The oxygen, stored in 3.45 MPa (500 psia) tanks, is injected at a rate that maintains the total cabin pressure at 34.4 kPa.

Two identical circulation systems prevent single-point failures. Each of the two systems is capable of sustaining the correct environment in the MAV cabin for half of the maximum mission length. In case of a circulation system failure, the oxygen can be manually released from vent valves located on the oxygen tanks.

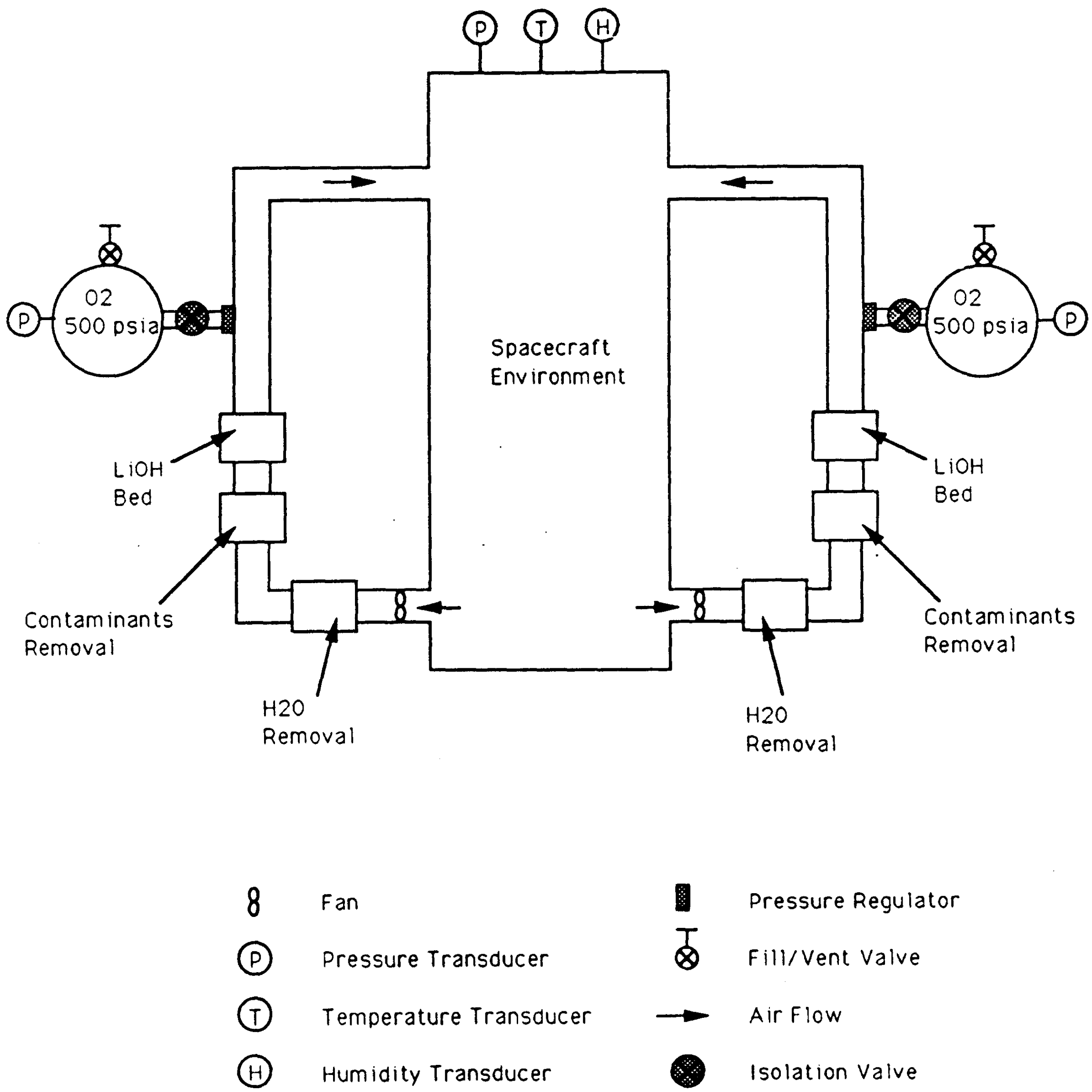

Figure 5-2: LSS Air Circulation Schematic

As an additional backup system in case of hull rupture, the stored oxygen can be pumped directly to the astronauts' spacesuits through a secondary ducting system (see Figure 5-1). If this system fails, or if any EVA is required, a one-hour, 0.035 kg supply of bottled 2.1 MPa (300 psia) oxygen is maintained in each of the spacesuits. This supply allows completion of the main scenario mission.

5.3.3 Air Purification System

The air purification system removes exhaled CO_2 as well as trace contaminants from the cabin environment. It utilizes a lithium hydroxide bed, charcoal filtering system, and a hopcalite bed.

Lithium hydroxide (LiOH), located in a bed in the main air circulation loop, reacts with the CO_2, separating it from the oxygen:

$$2LiOH + CO_2 \rightarrow Li_2CO_3 + H_2O$$

To maintain simplicity in the LSS, and because of the power required to reverse this exothermic reaction, the LiOH is not recycled, and must therefore be supplied at a rate of 1.35 kg/p-day (Purser, 1964).

Figure 5-3 displays a schematic of an Apollo-based trace contaminant removal system (Purser, 1964), consisting of a charcoal filter and hopcalite bed. The charcoal bed removes most of the contaminants from the air, including odor producing substances, particulate matter, and toxic substances, consuming charcoal at the rate of 0.059 kg/p-day (Purser, 1964). Periodically, the air flow is diverted through the hopcalite bed, which primarily removes carbon monoxide and hydrogen through catalytic combustion.

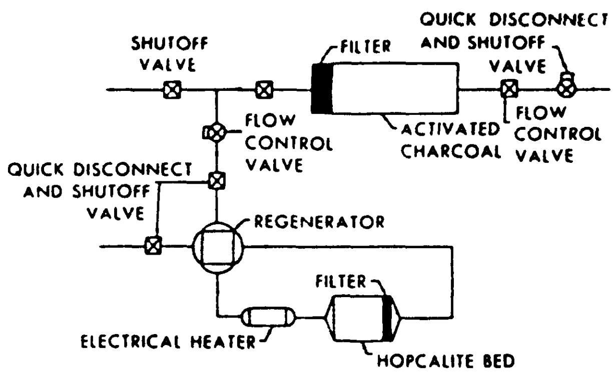

Figure 5-3: Apollo Contaminants Removal Subsystem (Purser, 1964)

5.3.4 Environmental Control

The MAV thermal system variable conductance heat pipes (see Section 10.0) maintain the correct temperature range of 21° to 27 ° C. To advise the astronauts of thermal system failure, the cabin temperature is constantly monitored by thermistors in the spacecraft.

To control the humidity level, H_2O needs to be removed from the air. A water separator located in the air circulation loop condenses the water out of the air stream, passively maintaining the required humidity level. For monitoring and check-out purposes, humidity detectors are located in the main cabin.

Maintaining the correct oxygen level keeps the cabin pressure at a constant 34.4 kPa. Pressure regulators release oxygen into the air stream in order to maintain this total pressure. As before, pressure transducers located in the capsule notify the astronauts of any changes in cabin pressure.

Because of the pure oxygen environment, there is a substantial fire hazard problem. To prevent the initiation of fires, electronic equipment is designed to avoid the possibility of sparks, arcs, or corona discharge. Also, fire-resistant materials are used throughout the spacecraft. If a fire does ignite, hand-held fire extinguishers can douse the fire. As a final backup, Halon, which is also provided on the STS, is automatically dumped into the cabin atmosphere to extinguish the fire.

5.3.5 Waste Management

Storage of human waste is not needed for the main scenario, which lasts approximately 30 minutes. In the abort scenarios, however, the storage of urine and fecal matter becomes necessary. To accommodate this necessity, the astronauts utilize waste elimination bags located inside their spacesuits.

5.3.6 Acceleration Couches

The maximum acceleration force the MAV astronauts experience is the impact force during final descent to the Martian surface. This force is expected to be approximately 5 gee's (see Section 9.0).

On account of these relatively small acceleration forces (about half of what Apollo astronauts experienced during re-entry at Earth), padded acceleration couches are not essential. Consequently, the couches are simply webbing strapped to a lightweight aluminum frame. Two of these couches are attached firmly to the MAV floor. The third seat must either slide away from the bottom hatch, or be removeable in order to allow the astronauts to enter from the MDV habitation module (see Figure 5-1). For this design, a slideable couch has been baselined to avoid problems with couch reattachment after the astronauts enter the MAV.

5.3.7 Spacesuits

In the main mission scenario, all crew activities are performed in a shirt-sleeve environment, including descent, ascent, crew transfer between the MDV and MAV, and crew transfer between the MAV and MOV. Thus, spacesuits are required only in certain abort situations.

To protect against the possibility of hull rupture or LSS malfunction, the astronauts are suited during both ascent and descent. Also, if either docking or shirt-sleeve transfer fails, astronaut transport between the MAV and the MOV is accomplished through EVA.

Because the spacesuits are required only in abort scenarios, lightweight, 34 kPa (5 psia) Gemini-like suits are used. These suits are not as sophisticated as those used by STS astronauts today, but such sophistication is not necessary for a single, short EVA.

5.3.8 Reliability

The allocated minimum LSS reliability of 0.995 should be readily achievable with the air circulation system redundancy. In the main scenario, the three non-linked subsystems (two main air circulation systems and a spacesuit backup) only need a 0.83 reliability in order to meet the overall LSS reliability criterion. It is expected that significantly better subsystem reliability than 0.83 can be obtained.

5.4 Summary

Table 5-1 summarizes the mass, power, and volume requirements of the MAV life support system. These data are approximated from historical sources, including Gemini, Apollo, STS, and other NASA programs (Purser, 1964; NAS9-1100, 1965; NASA R 17076, 1966; NASA-STD-3000, 1987; JSC-32025, 1987), and from information in this section. Some of the volume approximations are obtained by assuming an electronics specific gravity of 1 (density of 1000 kg/m^3). Consumables and power are based on a 38.8 hour mission with a relatively low FOS (factor of safety) of 1.15. The basis for this small margin is that the mission is a maximum of 38.8 hours, and will nominally be significantly less than that amount of time (~45 min).

Table 5-1: LSS Summary

LSS Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
food		3.4	0	0.005
food storage		2.5	0	0.003
water		2.2	0	0.002
water storage		4.1	0	0.004
oxygen		4.7	0	0.005
oxygen tanks	2	4.0	0	0.002
suit O_2 tanks (filled)	3	0.2	0	0.003
pressure regulator	4	1.4	446.0	0.001
pressure transducer	4	0.2	312.0	0.0002
fill/vent valve	2	0.5	0	0.001
relief valve	4	0.6	0	0.001
LiOH		7.5	0	0.008
LiOH bed	2	4.0	0	0.004
charcoal bed	2	4.0	0	0.004
charcoal		0.3	0	0.0002
flow control valves	4	1.0	223.0	0.001
shutoff valves	10	2.5	10.0	0.003
regenerators	2	2.0	0	0.002
hopcalite bed	2	4.0	0	0.004
electrical heater	2	2.0	12.0	0.002
fans	8	8.0	892.0	0.008
spacesuit ducts	4	6.0	0	0.040
isolation valves	2	3.2	0	0.003

Table 5-1: LSS Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
intake ducts	4	6.0	0	0.040
outflow ducts	2	5.0	0	0.035
humidity control	2	5.0	892.0	0.005
humidity detector	2	2.0	223.0	0.002
temp. transducer	2	0.1	134.0	0.0001
fire detectors	4	2.0	446.0	0.005
fire extinguishers	2	6.0	0	0.006
Halon		4.0	0	0.004
crew, 50% man	3	246.0	0	4.500
spacesuits	3	45.0	0	0.045 (stowed)
first aid kit	1	2.0	0	0.002
couch structure	3	10.0	0	0.010
webbing	3	4.0	0	0.004
straps	3	2.0	0	0.002
urine bag	6	5.0	0	0.010
fecal bag	6	5.0	0	0.010
TOTAL (w/out crew)	100	171.6	3590.0	0.284
TOTAL (w/crew)	103	417.6	3590.0	4.784

5.5 Recommendations

Mass savings are not easily obtained in this system. The only possible mass reductions come from elimination of redundancies, but the life-critical LSS system should contain numerous backups.

6.0 Propulsion System

6.1 Introduction

The propulsion system is the most mass-influential subsystem of the MAV, consisting of 90% of the overall mass. A propulsion system is needed for ascent to the MOV orbit, as a control system, and for rendezvous and docking maneuvers.

6.2 Requirements

Main propulsion system:

- minimize mass
- minimize complexity (subject to non-excessive mass penalty)
- provide thrust capability for ascent from the Martian surface (.38 g's)
- provide propellant for ascent and rendezvous with MOV
- allow engine throttling to remain within acceleration limits (see Section 5.0):
    - 5 g's sustained
    - 10 g's peak
- provide rendezvous capability with MOV after MDV descent abort

OMS/RCS:

- provide rendezvous and docking capability with MOV
- pitch, roll, and yaw control
- allow midcourse, post-thrusting corrections

Overall:

- no credible single point failures
- shelf life = 2 years
- reliability = .995

6.3 Propellant Choices

There are numerous propellant options for use in the main ascent propulsion system. Table 6-1 summarizes various possibilities, tabulating fuel to oxidizer ratios and propellant performance capabilities. The nuclear-based propulsion system is presented only for comparison, and was not considered for the MAV due to considerable required research and development.

Table 6-1: Propellant Options

Fuel	Oxidizer	Mass ratio (O:F)	Nominal I_{sp} (sec)
LH_2	LOX	6:1	460
Nuclear (LH_2)	--	--	850
MMH	LOX	1.4 to 1.6:1	380
MMH	H_2O_2	4:1	340
MMH, pumped	NTO	2:1	340
MMH, pressure-fed	NTO	1.6:1	320
Aerozine-50	NTO	2:1	310
N_2H_4 monoprop.	--	--	220
RP-1	LOX	2.6:1	330
Cold gas (N_2)	--	--	80
CH_4	LOX	3 to 4:1	380
C_3H_8	LOX	3.2:1	380
MPD ion engine (low-thrust)			6000

6.3.1 Liquids vs. Solids

Liquid propellants were selected early in the design process. Solid propellants have specific impulses between 200 and 300 seconds, significantly lower than can be obtained from most liquid propellants, translating into a substantial increase in the overall vehicle mass. Additionally, solid propellants do not permit throttling or engine shut-down and restart, which are necessary in this single-stage spacecraft.

6.3.2 Cryogenics vs. Storables

Having chosen liquid propellants for the MAV propulsion system, a second choice between cryogenic propellants and room temperature storable propellants is necessary. This is a decision between the better performance of cryogenic propellants (see Table 6-1) and the storability advantage of storable liquid propellants (see Table 6-2). Due to previous spaceflight utilization, the two propellant combinations considered were the MMH/NTO (monomethyl hydrazine/nitrogen tetroxide) storable propellants and LH_2/LOX (liquid hydrogen/liquid oxygen) cryogenic propellants.

Table 6-2: Propellant and Oxidizer Physical Properties

Compound	Density (kg/m^3)	Freezing Point (° K)	Boiling Point (° K)
LOX	1141	54.3	90.4
LH_2	70.8	13.7	20.4
NTO	1431	262	294
MMH	870.1	221	361
CH_4	422.9	90.9	111
C_3H_8	579.9	83.7	231
H_2O_2	1430	271	421
UDMH	790	216	336
RP-1	807	225	490
N_2H_4	1008	275	387

From strictly a performance standpoint, there is an obvious advantage to employing a LOX/LH_2 propulsion system over a storable, pump-fed MMH/NTO system. For a 3000 kg dry-weight MAV, 1.77 times more propellant mass is required for the storable system than for the cryogenic system.

However, as can be inferred from Table 6-2, the main difficulty with cryogenic propellant utilization is the problem of LH_2 storability on the Martian surface ($T_{av} \approx 214°$ K at the equator (see Section 10.0)). If propellant boil-off were to reach 43% in a LH_2/LOX system, the performance advantage of using cryogenics is eliminated. To reduce this boil-off, a vacuum system with multilayer insulation (MLI) is expected to be used to minimize heat losses. Preservation and maintenance of this vacuum present a number of problems, however.

One problem contributing to propellant boil-off is heat loss through structural contact points. Because the MAV structure takes both ascent and descent loadings, there are several necessary contact points on the propellant tanks. Heat circumvents the insulation layer through these points, dominating the overall tank heat loss.

Under typical circumstances, the boil-off due to this heat loss is expected to be contained under a few percent (Allen, 1989) for the year-long stay on the surface, indicating that the cryogenic propellant system is probably feasible. However, this does not address the problem of possible vacuum system failure.

If a catastrophic leak (i.e., one that cannot be evacuated by the pump system) occurs in the vacuum system, or the vacuum pump system fails, the cryogenics propellants will boil off. Even if this irrepairable leak is identified immediately, the MAV must still wait up to a maximum of 12.33 hours on the Martian surface before an emergency abort-to-orbit can occur. During this period, the LOX and LH_2 would freely boil off. If the cryogenics boil-off beyond the factor of safety built into the propulsion system, the MAV will be unable to reach the MOV orbit.

An additional penalty of utilizing cryogens derives from the low LH_2 density. Despite the assumed greater mass (not including thermal system mass differences) of the MMH/NTO system, it takes up 1.84 times less volume than the LOX/LH_2 system. With the limited MAV storage space available, this volume savings is highly advantageous.

Because of these reasons, storable MMH/NTO is baselined for the MAV design. If in the future, research indicates that long-term cryogenic storage on the Martian surface is viable, and that there is a controllable catastrophic boil-off risk, than cryogenics should be utilized to for probable vehicle mass savings.

6.3 Ascent Propulsion System Point Design

Schematics of the MMH and NTO feed systems (see Figures 6-1 and 6-2) illustrate the main MAV propulsion system. Actual system and subsystem dimensions and configurations are illustrated in Figure 6-8.

6.3.1 Propellant

A MMH/NTO bipropellant system is baselined for the MAV design. This propellant combination, which has been employed on the Apollo missions and the STS OMS (Orbital Maneuvering System), has the best Martian surface storability of any typically utilized system, although insulation and some heating is required (see Sections 7.0 and 9.0).

6.3.2 Pressure- vs. Pump-Fed Systems

There are two propellant feed system possibilities, pump- and pressure-fed. Pump-fed systems, although generally having higher performance than pressure-fed systems, have much lower reliability. A typical pressure-fed system has 0.9999 reliability, while pump-fed systems can approach only about 0.997 reliability.

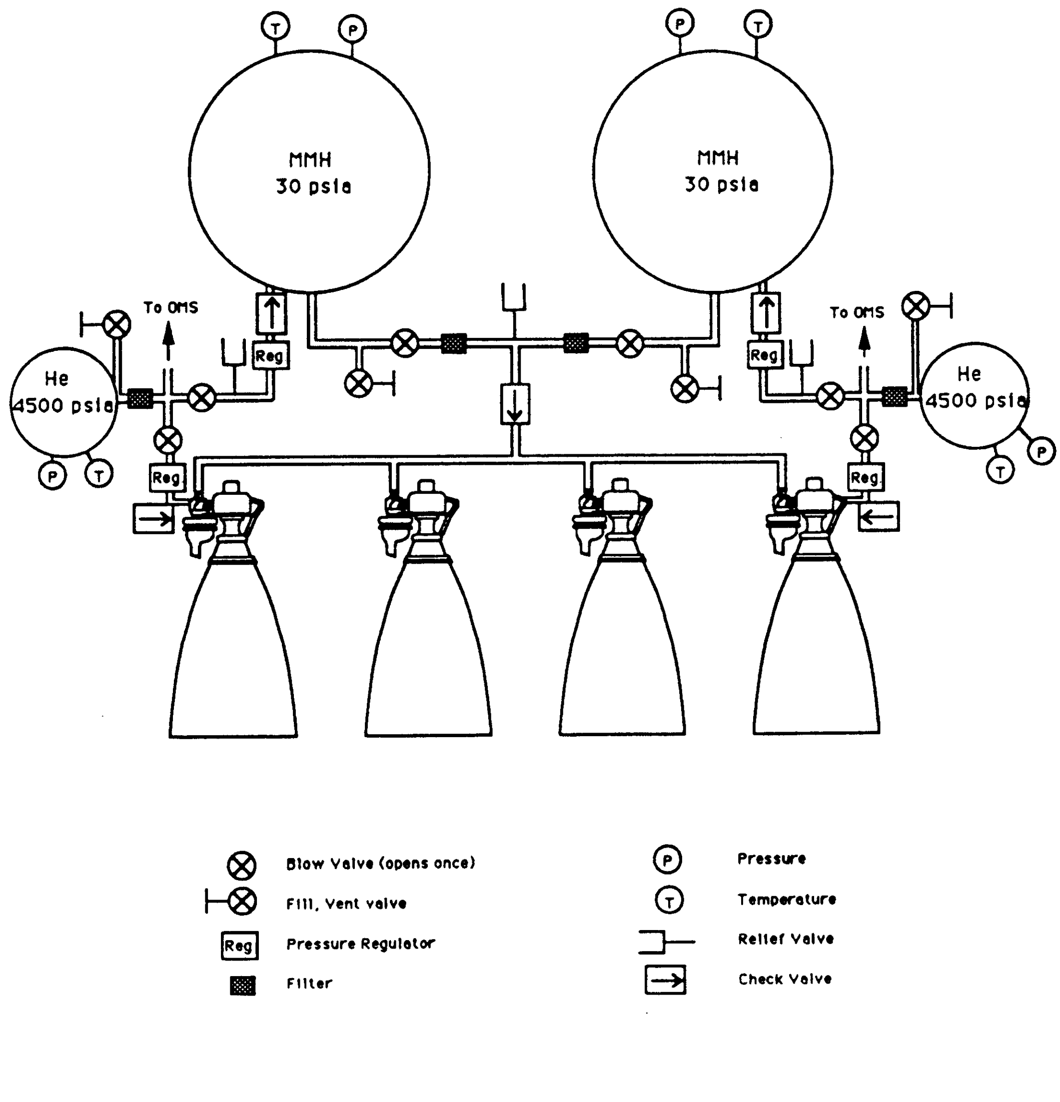

Figure 6-1: MMH Distribution System Schematic

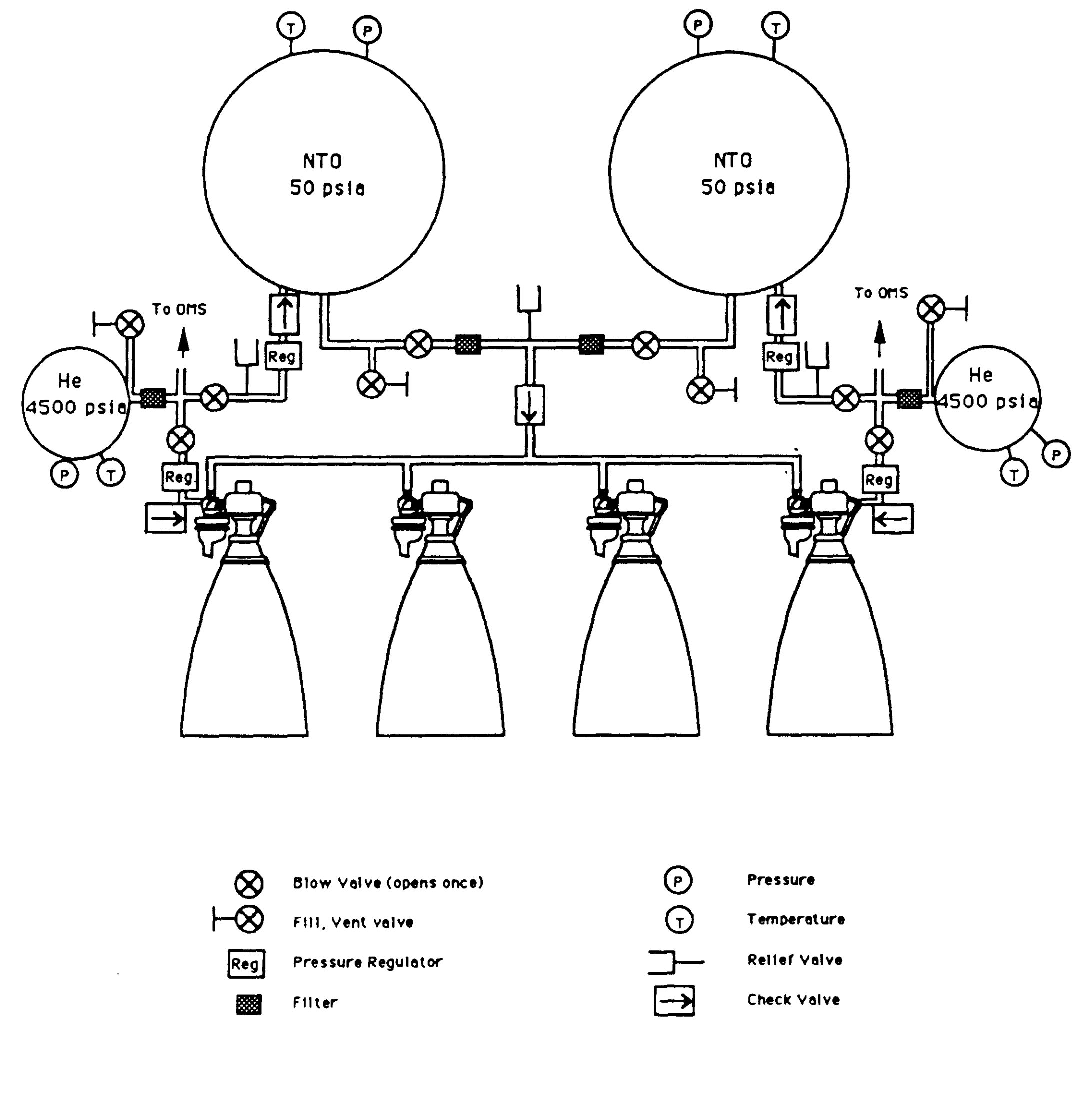

Figure 6-2: NTO Distribution System Schematic

An analysis was performed to compare the two principle feed system possibilities for an MMH/NTO propulsion system. The pressure-fed system assumed a 2.07 MPa (300 psia) propellant tank pressure with an engine I_{sp} of 320 seconds, while the pump-fed system consisted of 0.345 MPa (50 psia) tanks and a 340 second I_{sp}. Both systems utilized 31.1 MPa (4500 psia) helium as a pressurant gas.

Due to a 500 kg difference in helium pressurant storage tank mass, a 600 kg difference in propellant tank mass, and the variation in engine performance, the pressure-fed MAV had a mass of 26,300 kg as compared to the 15,000 kg pump-fed MAV.

Clearly, it is mass-advantageous for the MAV to employ pump-fed propellants. The problem of lower engine reliability is eliminated by allowing engine-out capability (see Section 6.3.4).

6.3.3 Staging

Spacecraft staging is typically employed when the ΔV requirement exceeds the engine exit velocity. In this design, the engine exit velocity is 3.36 km/s while the ascent ΔV requirement is 5.209 km/s, indicating that staging would be advantageous.

Generally, assuming the same performance for all stages, discarding mass during ascent reduces the overall vehicle mass. However, this mass savings must be balanced with reliability problems associated with the added complexity. Also, the volume constraints present in this design may make staging infeasible.

There are two primary methods of staging, drop tanks and full staging.

6.3.3.1 Drop Tanks

For drop-tank staging, the same engines are utilized for both stages, while only the propellant tanks are staged. However, in this pump-fed MAV design, the propellant tanks are extremely lightweight (see Table 6-5). Even if the propellant tank support structure (see Section 4.0) is included, there is simply not enough discardable mass to warrant tank staging. The added plumbing, stage connection structure, and separate thermal protection systems required for drop tanks eliminate the mass advantage of staging.

An additional problem with tank staging is derived from the MAV geometry. If drop tanks were to be specified, either the engines would need to be placed further away from the center of mass, or the tanks would be placed outside of the engines. In either situation, the total frontal area of the MAV would increase, resulting in difficulty in MAV storage within the aerobrake impingement cone.

6.3.3.2 Full Staging

Three separate propulsion system point designs were analyzed to determine the advantage of tank and engine staging, including a single-stage pump-fed design, a two-stage all pump-fed design, and a pump-fed first stage, pressure-fed second stage design. All pump-fed engines were assumed to have an I_{sp} of 340 seconds, while pressure-fed performance was 320 seconds. Four pump-fed engines were prescribed for all vehicle first stages, while a single engine was baselined for each second stage.

As mentioned before (Section 6.3.2), the single-stage pump-fed MAV has an overall mass of 15000 kg. Surprisingly, the two-stage mixed-feed MAV had a mass of 15200 kg. The mass savings obtained through staging was eliminated due to the lower performance of the second-stage engine and the excessive tank masses of the pressure-fed stage. This indicates that a two-stage all pressure-fed system also provides no mass advantage over the single-stage pump-fed MAV design.

The two-stage, all pump-fed MAV had an overall vehicle mass of 14100 kg, 900 kg less than the single-stage vehicle. Due to the extra engine, as well as the additional structure and thermal system, this mass savings is lower than would be expected. Also, there remains the previously mentioned storage problem with this two-stage MAV. This design results in increased MAV volume, and eliminates the possibility of a bottom hatch and tunnel to the MDV habitation module.

6.3.3.3 Summary

A single-stage pump-fed propulsion system is baselined for the MAV. Although the overall vehicle mass is greater than with the two-stage propulsion system, there is not a significant enough difference to warrant increased MAV complexity.

6.3.4 Engines

The MAV engine system requires a maximum of 133.4 kN (30000 lbf) of thrust (for an initial T/W of 2.54), a minimum thrust level of 62.3 kN (14000 lbf), and an approximate reliability of 0.999 to meet the overall 0.995 propulsion system reliability. Unfortunately, these characteristics do not exist in any single, current MMH/NTO pump-fed engine (see Table 6-3 and Section 6.3.4.5).

Table 6-3: Pump-fed MMH/NTO Engines

Engine	Maximum thrust (kN)	Isp (sec)	Throttling Ratio
U/R OME (modified)	53.3	342	up to 3:1
XLR 132, ox. cooled	44.5	345	none
Transtar, fuel cooled	33.4	342	none
Advanced Agena	52.9	336	N/A

Because of their superior throttling capability, four Aerojet U/R OMEs (uprated Orbital Maneuvering Engines) are utilized. Each of these engines have a thrust level of 26.7 kN (6000 lbf), but can be uprated to 53.3 kN (12000 lbf) by increasing chamber pressure.

In the main scenario, each of these four engines operate at 33.4 kN (7500 lbf) thrust level for the first stage of the MAV trajectory. At a specified time (see Section 9.0), two of the engines shut down and the remaining two throttle to 31.1 kN (7000 lbf). In the engine-out abort scenario, two engines begin at a 53.3 kN (12000 lbf) thrust level, and throttle down to 31.1 kN (7000 lbf).

6.3.4.1 Description/Specs

The modified U/R OME (see Figure 6-3) has a chamber pressure of 700 psia, resulting in a 53.3 kN thrust capability. It operates on MMH/NTO storable propellant at a mass mixture ratio of 1.93.

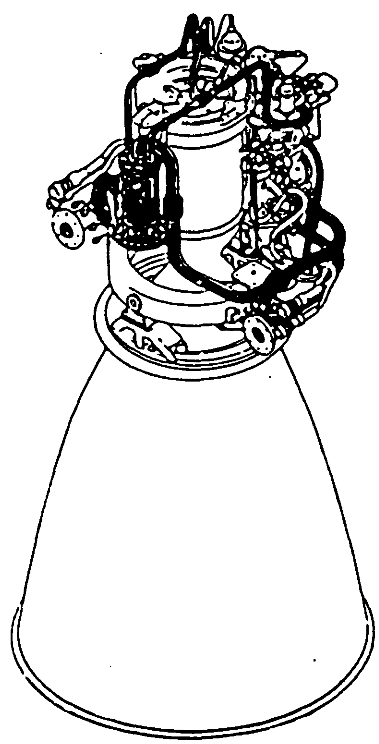

Figure 6-3: U/R OME (Aerojet, 1988)

This fuel-cooled 342 I_{sp} (at 12000 lbf thrust) engine, including gimbal actuators and engine controllers, has a projected mass of 124 kg. To maintain the required thrust level, nozzle dimensions of 1.75 m (69.0 in) diameter and 3.2 m (126 in) length are projected (using elementary nozzle theory) with an exit to throat area ratio of approximately 430:1.

Each engine has a gimbaling capability of up to ± 7° pitch and ± 8° yaw. MMH and NTO are pumped in from tanks at pressures of 50 psia and 30 psia respectively (see Figure 6-4).

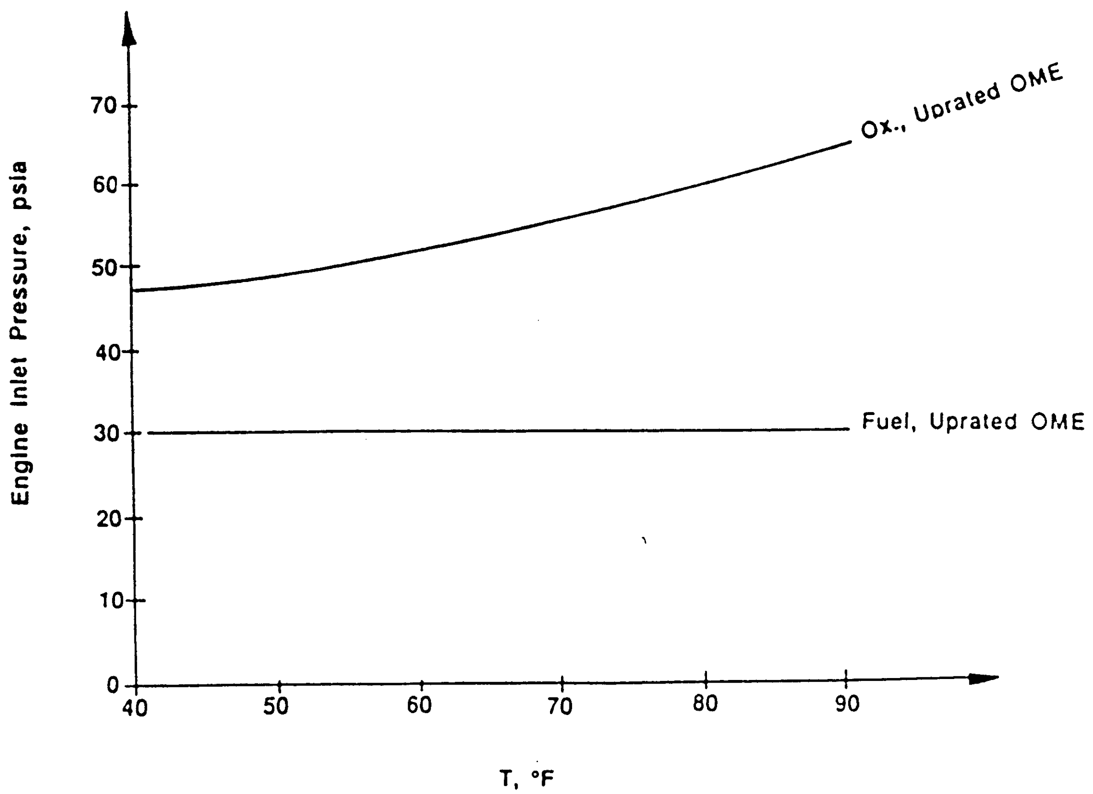

Figure 6-4: Propellant Storage Pressures (Aerojet, 1988)

6.3.4.2 Extendable Nozzle

Clearly, the OME length presents a packaging problem. One possible solution is to sacrifice some engine performance by decreasing the area ratio, and hence the nozzle diameter and length. A second solution, which maintains the high engine performance, is to utilize an extendible nozzle (see Figure 6-5). In stored position, the nozzle is roughly half of its

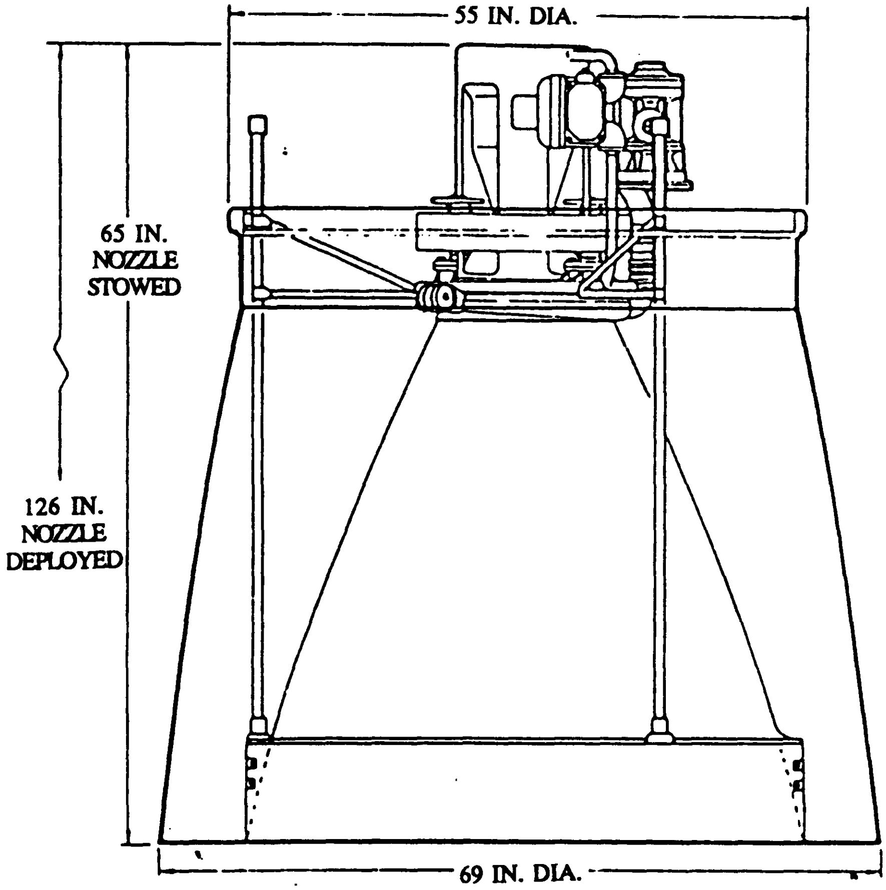

Figure 6-5: Extendible Nozzle

extended length. The OMEs extend after the MAV disconnects from the MDV habitation module. An extendible nozzle is currently used on the PeaceKeeper missile.

6.3.4.3 Power

The OME current requirements for the engine controls and gimbal actuators are summarized in Figures 6-6 and 6-7. These quantities are used to calculate the total required propulsion system power (see Section 7.0).

6.3.4.4 Helium

High pressure helium gas is required both for pressurizing the propellant tanks and for various engine tasks. Table 6-4 summarizes the helium requirements for a single OME.

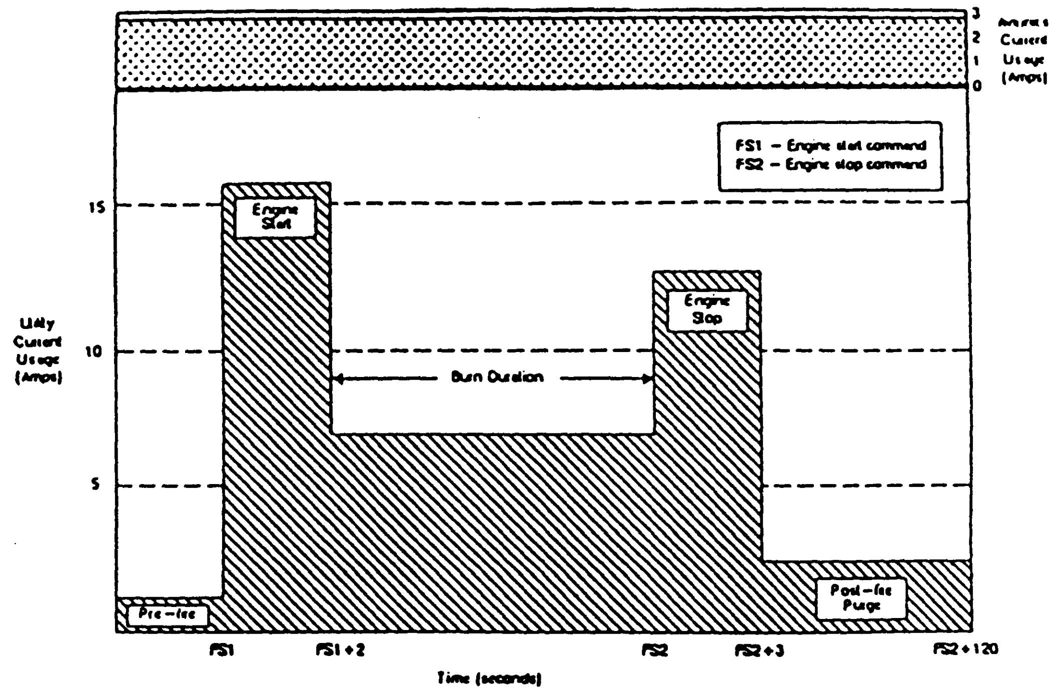

Figure 6-6: Avionics and Utility Current Requirements (Aerojet, 1988)

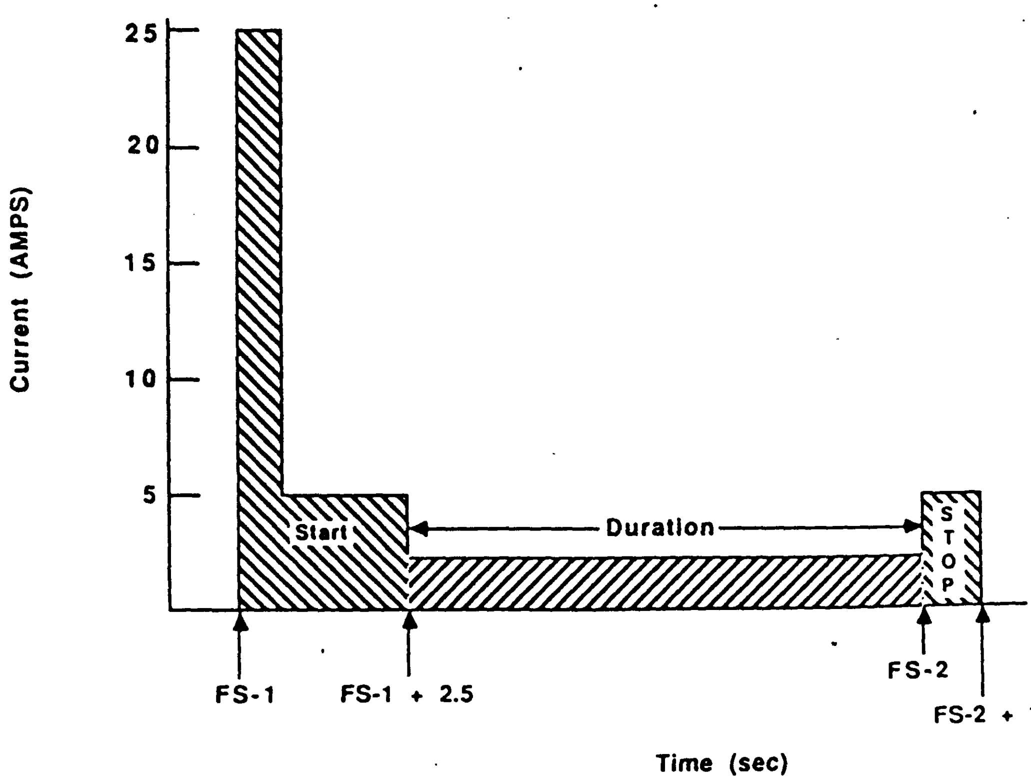

Figure 6-7: Gimbal Actuator Current Requirements (Aerojet, 1988)

Table 6-4: Engine Helium Requirements (Aerojet, 1988)

Engine Task	Helium Mass (kg), 338 sec., 2 cycles
Interpropellant Seal	.140
Turbine Seal Purge	.035
Fuel Line Purge	.007
Fuel Bearing Purge	.007
Gas Generator Purge	.012
Turbine Start	.120
Valve Actuation	.001
TOTAL (1 engine, total flight)	.279

Two of the main engines operate for a total of 341 seconds (see Section 9.0) in the main mission scenario. The other two engines require 0.207 kg of helium for only 154 seconds of engine operation time. Including a 1.5 factor of safety, a total of 1.458 kg of helium are required for the main ascent engines.

In addition to these helium requirements, 8.86 kg of helium (including a 1.5 FOS) are required to pressurize the MMH and NTO. All of the helium is stored at a pressure of 31 MPa (4500 psia).

6.3.4.5 Reliability

The U/R OME with redundant valves has a predicted reliability of 0.9982 (Aerojet, 1988). If single engine-out is acceptable, than the only failure mode occurs if one of the four engines fails and a second engine fails once the MAV has switched to the engine-out abort mode. This results in an engine system reliability of 0.999974, easily meeting the engine reliability requirement.

If single engine-out on ascent is not acceptable, the engine system has a reliability of only 0.9928.

6.3.4.6 Engine-Out Abort

There is some concern about spacecraft rotations if an engine-out occurs on ascent. Because the engines do not thrust directly through the spacecraft center of mass, engine failure induces a spin rate about the spacecraft x- or y- axis (see Figure 6-8).

The MAV is held down to the MDV habitation module for the first 3 seconds (see Section 9.0) of engine ignition, eliminating the spin problem in the case of pad engine-out. During the period with 4 OMEs operating, engine-out is also not expected to be a problem. For the 0.76 seconds (expected to be 0.5 seconds by OME development completion; Boyce, 1989) that is required for the engine opposite the failed engine to shut down, the remaining two engines can gimbal and increase thrust to compensate, maintaining the MAV on its trajectory.

If engine-out occurs when only two engines are operating, the remaining two engines must start up, gimbal, and compensate for the induced tumble rate. The viability of this option can be analyzed:

$$\alpha = \frac{T}{I_{xx}} = \frac{r \times F}{I_{xx}} = 4.08 \frac{\text{rad}}{\text{sec}^2}$$

α = spacecraft angular acceleration

T = induced moment about rotation axis

r = moment arm to engine = 2.44 m (see Section 6.3.7)

F = average engine thrust over shut down = 15.568 kN

I_{xx} = moment of inertia about rotation axis at end of two-engine run = 9302 kg-m^2 (see Section 13.0)

Utilizing this angular acceleration, the total spacecraft rotation angle during this engine failure can be estimated, assuming that the remaining engines don't start attenuating the angular rotation rate until 0.66 seconds (simultaneous engine start and gimballing time; Boyce, 1989):

$$\omega = \omega_0 + \alpha t_{stop}$$

$$\theta = 0.5\ \alpha t_{stop}^2 + \omega\ (t_{start} - t_{stop})$$

ω = angular velocity after engine thrust stop

ω_0 = initial angular velocity = 0

θ = total angular displacement

t_{stop} = engine stop time

t_{start} = engine start time

Because the engine start time (0.66 sec. to 66% thrust; Boyce, 1989) is less than the engine stop time, the angular velocity contribution to the angular displacement can be neglected (neglecting failure detection time). Therefore, the total angular displacement after engine-out is 51°, which is expected to be correctable (MAV is outside of the Mars atmosphere by this time). Additionally, if the engine stop-time is reduced to the predicted 0.5 seconds, and the start-time is reduced to ~0.5 seconds (Boyce, 1989), the rotation angle will be reduced to 29°.

If any rotation is unacceptable, engine-out during the two engine run can be eliminated as an acceptable failure mode. This would reduce the engine system reliability to 0.996.

6.3.4.7 Summary

To summarize the MAV engine system, 4 modified U/R OMEs with extendible nozzles are utilized, each having a maximum thrust level of 53.3 kN (12000 lbf). In the main scenario, all four engines start the ascent trajectory at 33.4 kN (7500 lbf) thrust level, reducing at 151 seconds to two engines at 31.1 kN (7000 lbf) thrust.

In the worst-case, pad engine-out scenario, two engines start up at 53.3 kN thrust and reduce to 33.4 kN after 186 seconds. Engine-out capability is provided throughout the MAV ascent.

6.3.5 Plumbing

To maintain high system reliability, one-shot blow valves are used throughout the propulsion system (see Figure 6-1 and 6-2). Each of these valves opens before engine ignition, and remains open throughout the ascent. When the engines require the shut-off of helium or propellant flow, valves within the engines themselves close. Further reliability is obtained by placing isolation valves in the engines in case of catastrophic engine failure.

6.3.6 Reliability

If single engine-out is allowed, the plumbing system simplicity, combined with the 0.999974 engine system reliability, is expected to maintain the system reliability over 0.995.

6.3.7 Configuration

The main propulsion system, as well as the OMS, is illustrated in Figure 6-8. Spherical tanks are baselined for all propellant and helium supplies, to minimize both structural mass and surface area (for thermal control purposes (see Section 10.0)). The propellant system is configured compactly to remain within the aerobrake impingement cone.

Four main propellant tanks are specified for mass symmetry, with one helium tank pressurizing a single propellant tank.

6.3.8 Performance Summary

The main propulsion system is sized for a ΔV of 5259 m/s. This already contains a significant margin of error (3.8%) for the main mission scenario, so no added propellant factor of safety is included.

An additional ullage factor of 5% (Redd, 1989) of the total propellant mass is included for unusable/gaseous propellant, while another 3% is included for attitude control. For the

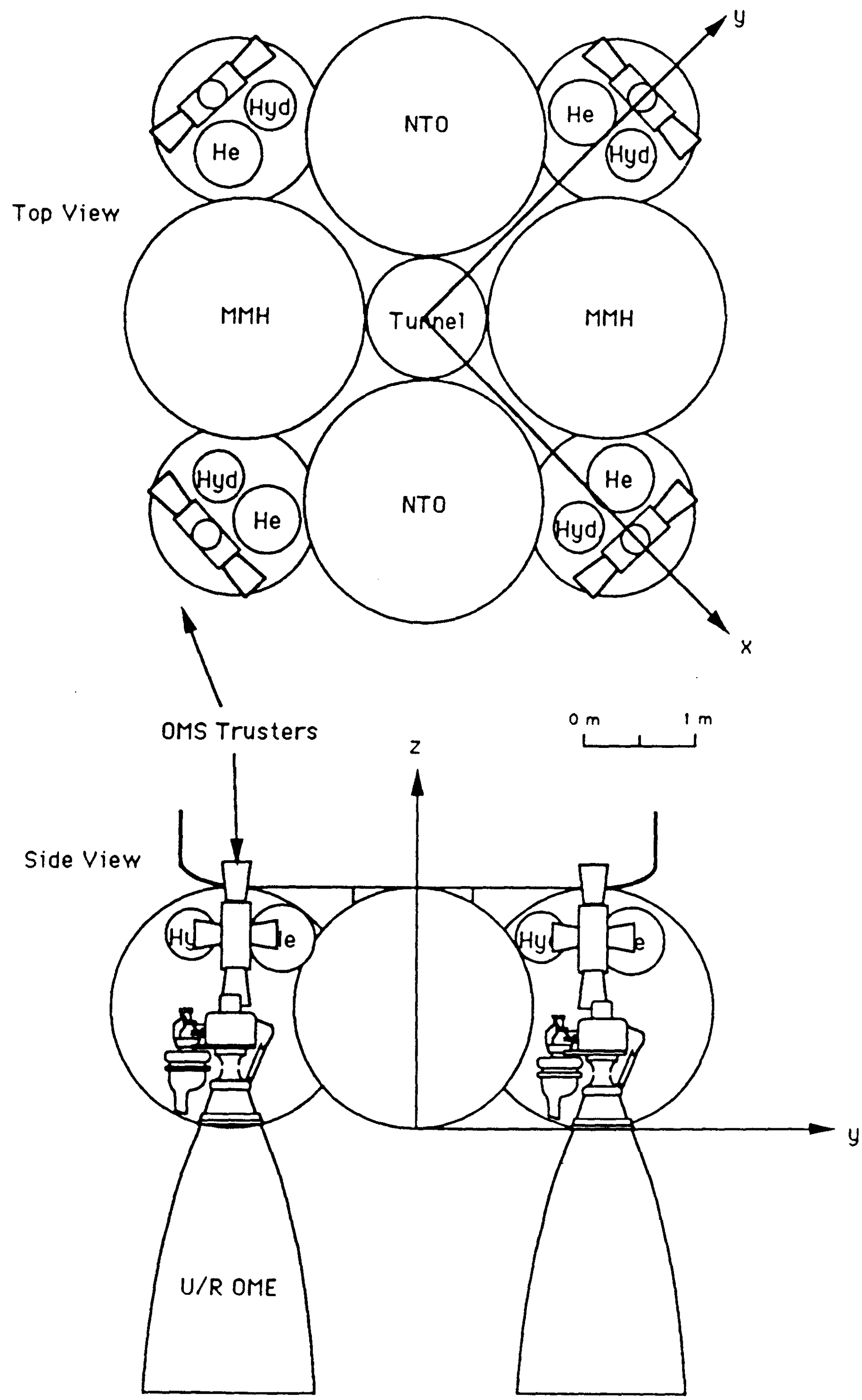

Figure 6-8: Propulsion System Configuration

spacecraft dry mass (including OMS propellant) of 2760 kg, this gives a total MMH/NTO load of 11333 kg for a total vehicle mass of 14093 kg.

6.3.9 Mass Summary

Table 6-5 summarize the main propulsion system, including mass, energy, and volume requirements. Energy needs are based on an engine pre-purge operating time of 30 minutes. Helium tank masses include containment of required OMS helium.

Table 6-5: Main Propulsion System Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
MMH tank (inner shell)	2	96.6	0	0.022
NTO tank (inner shell)	2	96.6	0	0.022
GHe tank	4	70.1	0	0.018
GHe (for engines)	4	1.5	0	0.007
GHe (for tank pressure)	4	8.9	0	0.043
insulation	see thermal section (10.0)			
heating units	see power section (7.0)			
main engines	4	497.0	529.0	12 (stored)
nozzle extender	4	20.0	0.2	0.020
fill, vent valve	8	2.2	0	0.008
blow valve	12	4.8	0.1	0.008
pressure transducer, low	4	0.2	0.05	0.0001
pressure transducer high	4	0.2	0.05	0.0001
regulator	8	5.6	0.2	0.006
filter, He	4	1.8	0	0.002
filter, propellant	4	1.8	0	0.002

Table 6-5: Main Propulsion System Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
relief valves, He	4	1.2	0	0.001
relief valves, propellant	4	1.2	0	0.001
check valves, helium	4	2.0	0	0.002
check valves, propellant	4	2.0	0	0.002
temp. transducer, He	4	0.2	0.05	0.0001
temp. transducer, Prop	4	0.2	0.05	0.0001
lines, cabling, electronic		60.0	0	0.060
ΔV MMH	2	3581.4	0	4.056
ΔV NTO	2	6912.4	0	4.763
ACS MMH	2	107.4	0	0.122
ACS NTO	2	207.4	0	0.143
ullage/unusable MMH	2	179.1	0	0.284
ullage/unusable NTO	2	345.6	0	0.333
TOTAL (Dry)	98	874.1	529.6	15.224
TOTAL (Wet)	104	12207.5	529.6	24.925

6.4 OMS Propulsion System Point Design

The MAV OMS (orbital maneuvering system) allows the spacecraft to dock with the MOV. Figure 6-9 shows a schematic of this hydrazine-based system and Figure 6-9 illustrates overall configuration. Four OMS thruster groups, each with four engines, permit pitch, roll, and yaw control for the on-orbit spacecraft.

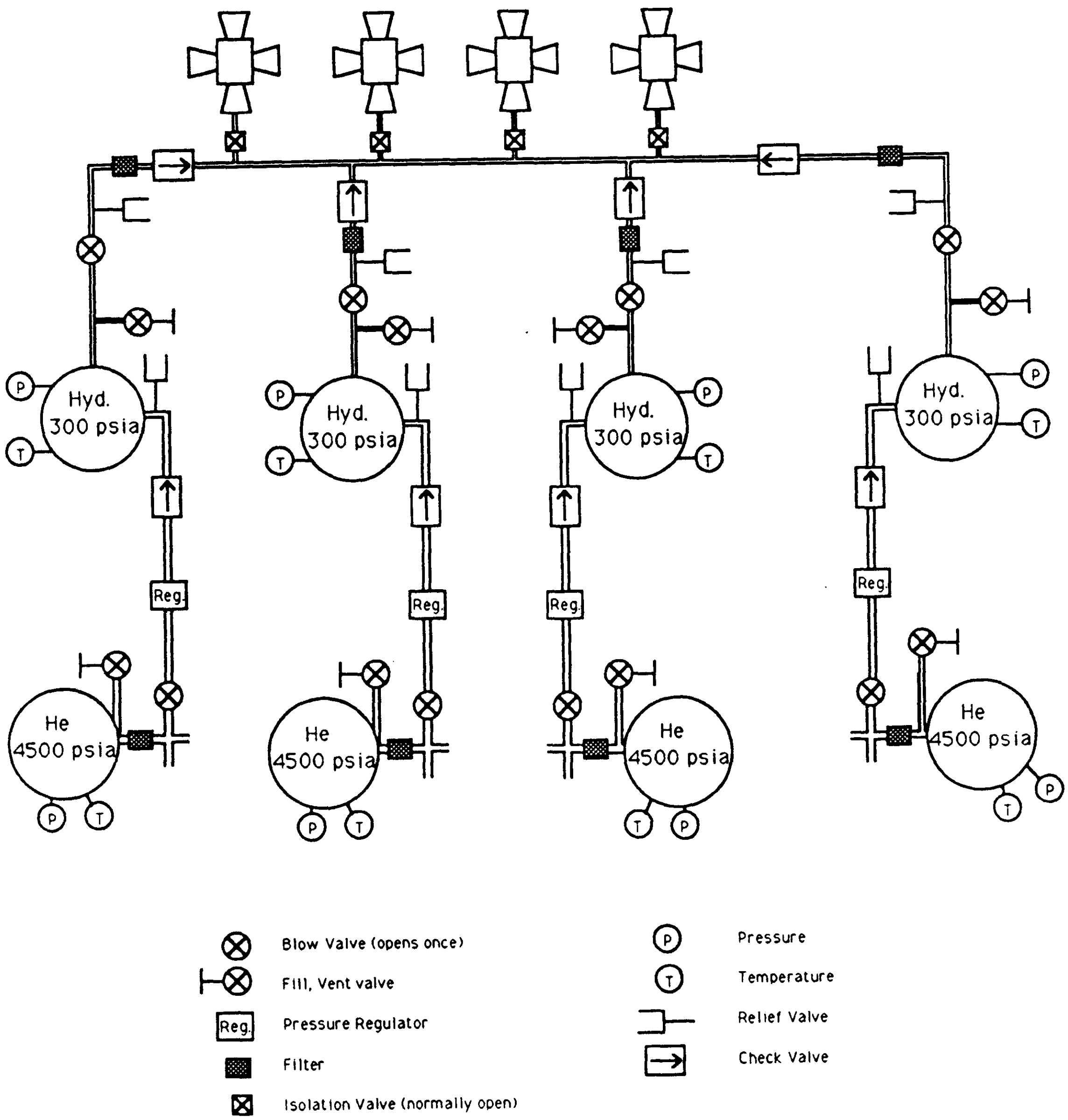

Figure 6-9: OMS Schematic

6.4.1 Propellant Choice

Because the main ascent engines are responsible for attitude control on ascent, the OMS only needs to support the 50 m/s docking ΔV requirement. As a result of this, a high performance propellant system is not required. Therefore, to maintain OMS simplicity, a N_2H_4 monopropellant is baselined as the OMS propellant. For a pressure-fed system, hydrazine has a 220 second I_{sp}, resulting in 64 kg of required hydrazine.

6.4.2 Pressure-vs. Pump-Fed

For such a small propellant load, pressure-fed system simplicity is more important than pump-fed performance. Also, system masses eliminate the pump-fed performance advantage for low propellant masses. Consequently, the hydrazine is pressure-fed to the OMS engines at a pressure of 2.07 MPa (300 psia).

6.4.3 Engines

A total of 12 engines are required to give the MAV total pitch, roll, and yaw control. For reliability, four clusters of four 111 N (25 lbf) thrust engines are provided on the MAV. If OMS impingement on the main engines is a problem, the ascent engines can be discarded before rendezvous.

6.4.4 Plumbing

The hydrazine tanks are accessible by all four sets of engines. This allows the entire hydrazine propellant load to be utilized, even if an engine cluster fails. If an engine cluster fails, isolation valves prevent propellant loss through the defective system (see Figure 6-9).

6.4.5 Reliability

Each of the clusters has an overall reliability of 0.9996 (engine reliability of 0.9999), translating to an overall engine system reliability of 0.99998 (with allowance of single engine cluster failure).

6.4.6 Performance Summary

For the required docking ΔV of 50 m/s, 64 kg of hydrazine is required. An additional ullage factor of 7% (Redd, 1989) is also included in the total propellant load, giving a total propellant load of 68.48 kg.

6.4.7 Configuration

To give the maximum control over the spacecraft, OMS engines are located away from the spacecraft center of mass (see Figure 6-9). Each hydrazine tank is placed near a helium tank to minimize high-pressure pressurant gas lines.

6.4.8 Mass Summary

Table 6-6 summarizes the OMS mass, energy, and volume requirements. Helium is stored with the main propulsion system pressurant gas at a pressure of 31 MPa (4500 psia).

Table 6-6: OMS Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
OMS engines	16	48.0	0.2	0.096
engine struct.	4	10.0	0	0.010
N_2H_4 tanks	4	4.0	0	0.001
GHe	4	0.4	0	0.002
insulation	see thermal section (10.0)			
heating unit	see power section (7.0)			
regulators	4	2.8	0.2	0.008
pressure transducer	4	0.2	0.05	0.0001

Table 6-6: OMS Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
fill, vent valve	4	1.1	0	0.004
blow valve	8	3.2	0.03	0.006
isolation valve	4	6.4	0.1	0.004
filter	4	1.8	0	0.002
relief valve, helium	4	1.2	0	0.001
relief valve,propellant	4	1.2	0	0.001
check valve	8	4.0	0	0.001
temperature transducer	4	0.2	0.05	0.0002
plumbing, electronics		30.0	0	0.030
ΔV N_2H_4	4	64.0	0	0.063
N_2H_4 ullage	4	4.5	0	0.004
TOTAL (Dry)	76	114.6	0.3	0.167
TOTAL (Wet)	84	183.1	0.3	0.234

6.5 Recommendations

Substantial MAV mass savings are obtainable if cryogenic fuels are utilized. However, for this to be possible, long-term storage capability of LH_2/LOX on the Martian surface needs to be developed. Cryogenic boil-off rates, as well as reliability issues, must be analyzed to determine the feasibility of a MAV cryogenic propulsion system.

If cryogenic use is not viable, a highly reliable, 133.4 kN (30000 lbf), throttleable MMH/NTO engine should be developed. Use of a single engine on ascent would eliminate the performance disadvantage and safety problems of the required engine-out capability in the four-engine MAV.

7.0 Power System

7.1 Introduction

The MAV power system must accommodate a wide range of power demands, from peak power requirements (5.6 kW) that occur on ascent, to a significantly lower steady-state power requirement (500 W).

7.2 Requirements

Power system must:

- supply power to all subsystems
- supply power for a maximum of 38.8 hours
- meet peak power demands of 5.6 kW
- meet sustained power demands of 500 W
- minimize mass
- have no single-point failures
- have a distribution system
- reliability = .995
- shelf-life = 2 years

7.3 Power Requirements

The MAV power system needs are summarized in Table 7-1, with more detailed energy requirements appearing in the other subsystem technical sections. Total energy requirements are based on the maximum length 38.8 hour mission, with only 913 W-hr of energy actually required in the principal scenario.

Table 7-1: Subsystem Power Requirements

Subsystem	Peak power (W)	Total Energy (W-hr)
life support system	104	3590
propulsion system	4996	530
power system	45	2008
avionics system	354	14142
thermal system	50	2231
TOTAL	5549	22501

This total energy is distributed over the mission length as illustrated in Figure 7-1. The peaks, lasting for periods of about 2 seconds, occur at engine startups and shutdowns. Peak locations in the power schedule vary if an abort scenario occurs, but the total energy for ascent remains the same independent of scenario.

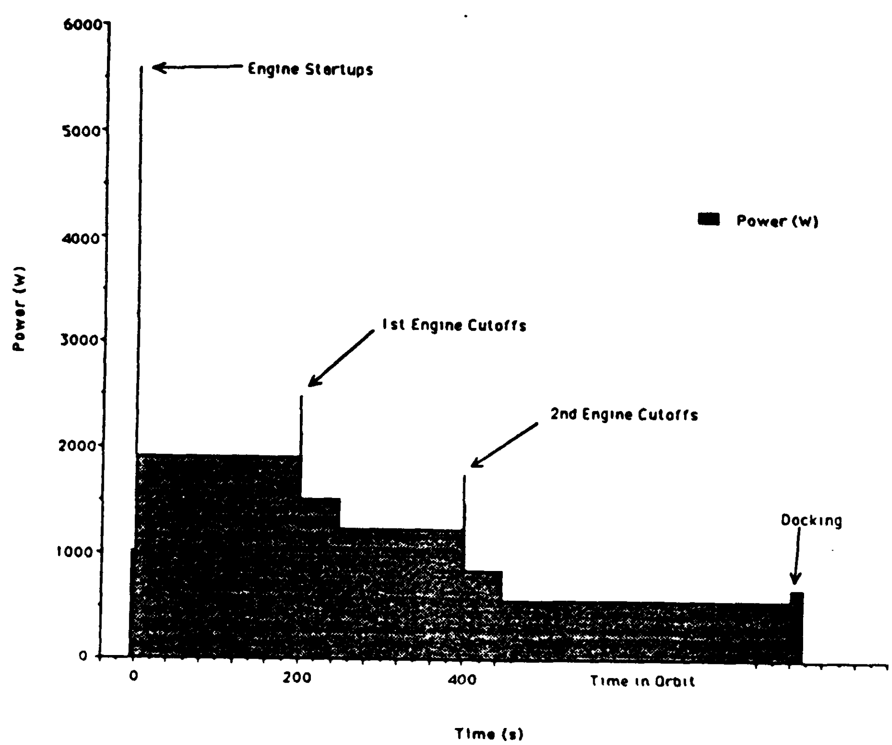

Figure 7-1: Required Power

In addition to the mission electrical power requirements, thermal energy must be provided to prevent the storable propellants from freezing throughout the 2-year mission. Because of the vacuum system insulating the propellants, only 83 W of constant power is required for this purpose (see Section 10.0). An additional 438 W of thermal power is necessary to sustain the room temperature environment of the MAV capsule.

7.4 Options and Choices

There are two viable power system alternatives for this application, fuel cells and primary batteries. The large system masses of historically long-term power-supply systems, including solar cells, nuclear reactors, and isotope-based systems make these options infeasible (see Figure 7-2). The following sections examine both fuel cells and batteries, providing a basis of selection between the two options.

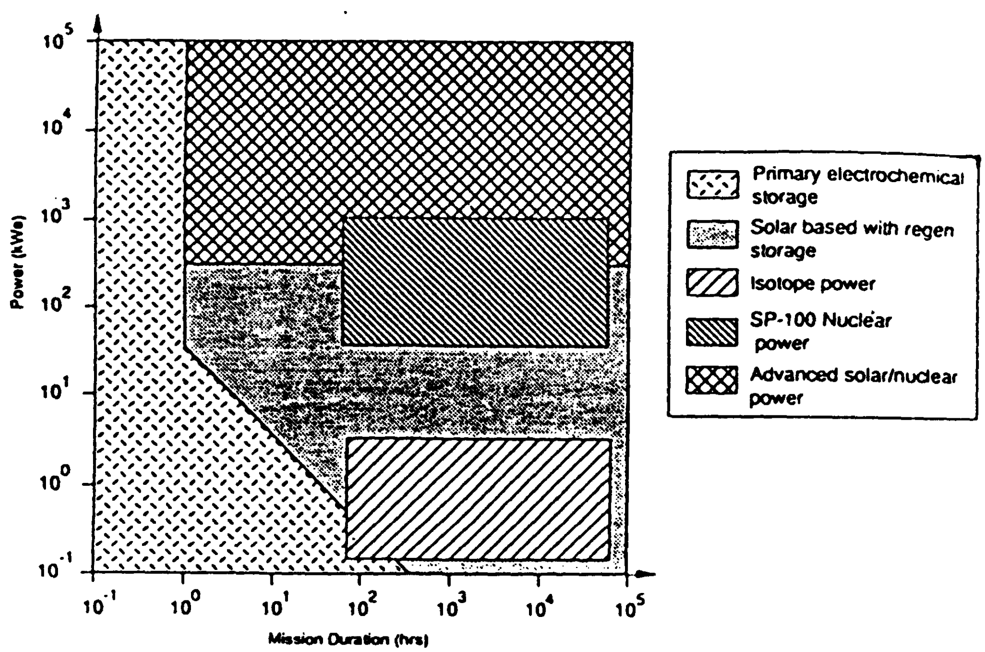

Figure 7-2: Power Source Comparison for Different Mission Lengths (NASA TM, 1988)

7.4.1 Primary Batteries

The utilization of battery-stored energy is the less complicated of the two power system alternatives. Required power is directly extracted from a system of batteries, drawing on a 2-year old store of chemical energy.

Table 7-2 enumerates the state-of-the-art specific energies of several candidate battery systems. In addition to these specific energies, shelf-life also influences battery selection.

Table 7-2: Battery Comparisons

Battery	Specific energy (W-hr/kg)	Comments
$LiSOCl_2$	250 (NASA TM 88174, 1985)	long shelf life,best specific energy,untested
NiH_2	31.5 (MRSR, 1989)	good cycle life
AgZn	150 (NASA TM 88174, 1985)	used on all past manned space missions
NiCd	28.6 (MRSR, 1989)	low specific energy,well-tested
$LiTiS_2$	83 (MRSR, 1989)	not space proven

From the standpoint of specific energy, $LiSOCl_2$ and AgZn batteries are unquestionably superior to the other possibilities. The remaining three batteries mentioned are typically employed because of their long cycle life, which is not a concern in this situation, due to the fact that the MAV power system operates only once.

Because of the necessity of a 2-year shelf-life, the utilization of AgZn batteries could be difficult. The present-day AgZn battery has a shelf-life of approximately 2 years (NASA TM 88174, 1985), as compared to the $LiSOCl_2$ 8 year storability (NASA TM 88174, 1985). The silver-zinc battery life may not allow a suitable factor of safety.

For this reason, as well as the differences in specific energies, it is beneficial to space-qualify the lithium-based battery. By opting for the $LiSOCl_2$ battery, the system mass, excluding the power distribution equipment, would be 90 kg for the specified 22500 W-hr.

7.4.2 Fuel Cells

Fuel cell systems have been previously utilized during Apollo, Gemini, and STS missions, generally delivering higher specific energies than primary battery systems for long-term missions (>24 hours).

Past spacecraft fuel cell systems have all operated on a hydrogen/oxygen mix. As discussed before (Section 6.0), this presents a problem with storability, specifically the risk of accidental excessive boil-off. Because of the life-critical nature of the power system, the utilization of cryogens for fuel cells should be avoided. If, in the future, research indicates that cryogens are viable for the main propulsion system, then they can also be used for power.

Because of the presence of monomethyl hydrazine (MMH) and nitrogen tetroxide (NTO) in the propulsion system, it is advantageous to also use this fuel/oxidizer combination for fuel cell power. The reactants can be stored with the rest of the propellant, eliminating the need for additional tankage. The main problem with this type of fuel cell is that it hasn't been fully developed for space applications. Hydrazine-based systems exist (Schmidt, 1984; Linden, 1984), but they use air or hydrogen peroxide as the oxidizer. It is not expected to be a problem, however, to modify these systems to use NTO as an oxidizer.

Since MMH/NTO fuel cells have not been flown previously, the performance of these fuel cells is estimated from compound heats of formation:

$$CH_3NHNH_2\ (l) + N_2O_4\ (l) \rightarrow 2H_2O\ (l) + N_2\ (g) + H_2\ (g) + CO_2\ (g) + \text{heat}$$

$$\Delta H_f\ (CH_3NHNH_2\ (l)) = \quad 54.81\ \text{kJ/mol}$$

$$\Delta H_f\ (N_2O_4\ (l)) = \quad -19.66\ \text{kJ/mol}$$

$$\Delta H_f\ (H_2O\ (l)) = \quad -238.49\ \text{kJ/mol}$$

$$\Delta H_f (CO_2 (g)) = \qquad - 393.30 \text{ kJ/mol}$$

$$\Delta H_f (N_2 (g)) = \Delta H_f (H_2 (g)) = 0 \text{ kJ/mol}$$

Utilizing these values, the maximum chemical energy obtainable is determined:

$$\Delta H_{f\ total} = 2\ \Delta H_f (H_2O\ (l)) + \Delta H_f (CO_2\ (g)) - \Delta H_f (CH_3NHNH_2\ (l)) - \Delta H_f (N_2O_4\ (l))$$

$$= - 905.42 \text{ kJ/mol}$$

$$\Rightarrow 0.183 \text{ kg/kW-hr MMH}, \ 0.365 \text{ kg/kW-hr NTO}$$

This results in a MMH/NTO fuel cell theoretical maximum specific energy of 1.833 kW-hr/kg of reactant. This chemical energy must then be converted to electrical energy, taking into account inefficiencies based on side reactions and electrode degradation. To determine a probable efficiency, the Apollo LH_2/LOX fuel cell system is evaluated.

By performing a similar analysis to the one performed on the MMH/NTO system, it is found that the theoretical specific energy of a LH_2/LOX system is 3.6 kW-hr/kg. The actually achieved specific energy of the system is 2.564 kW-hr/kg (Linden, 1984), which indicates a 71% chemical-to-electrical energy conversion efficiency.

Assuming a more conservative efficiency of 50% for the MMH/NTO system reduces the fuel cell specific energy to 0.913 kW-hr/kg. The fuel cell, therefore, requires 8.15 kg of MMH and 16.30 kg of NTO to produce 22 kW-hr of energy. Scaling the system mass up from a 300 W hydrazine/air system (Linden, 1984), the fuel cell total dry mass is about 20 kg, yielding a total system mass of 49.5 kg (dry mass and propellant). With additional mass savings attained from 5 kg of water produced, which is expected to be purifiable, the fuel cell system has a net mass of about half of that of the battery system.

This mass savings translates into an overall vehicle mass savings of about 300 kg, when propellant loads are included. Some batteries are still needed, however, to handle the peak loads, to initially fire-up the fuel cell, and for periodic MAV system checks during the 2-year mission.

7.5 Point Design

The power system, featuring a MMH/NTO fuel cell system, is illustrated in Figure 7-3.

7.5.1 Power Source

The majority of the power needs (22000 W-hr) are provided by the MMH/NTO fuel cell system. The reactants are forced into the 0.05 m^3 cell (Linden, 1984) from the main propulsion tanks at a pressure of 1.01 x 10^5 Pa (1atm; Linden,1984). The fuel cell itself is located outside the MAV cabin (see Figure 7-3) to avoid contamination of the cabin air by the byproduct fuel cell gases.

A schematic of the fuel cell system is shown in Figure 7-4. A solid electrolyte, zirconia, is used to avoid corrosion problems usually associated with hydroxide electrolytes. This is particularly critical in this system, since a hydroxide electrolyte would present problems during the two years the power system is dormant. The fuel cell anode and cathode are sintered platinum electrodes.

Since MMH has a positive heat of formation, batteries provide power to start up the reaction. After the fuel cell is running, the heat needed to break up MMH is provided by the fuel cell itself. The byproduct water is condensed, purified, and pumped into storage containers located in the MAV cabin, where it can be accessed by the astronauts.

To bootstrap the fuel cell and the entire power system, handle the peak power requirement, allow periodic system checks during the 2-year mission, and allow completion of the main scenario mission without fuel cells, 9.5 kg of $LiSOCl_2$ batteries are provided. This provides 913 W-hr of energy and allows 5.6 kW power peaks for up to a 2 second period. Redundant power controllers can access both power sources in case of distribution system failure.

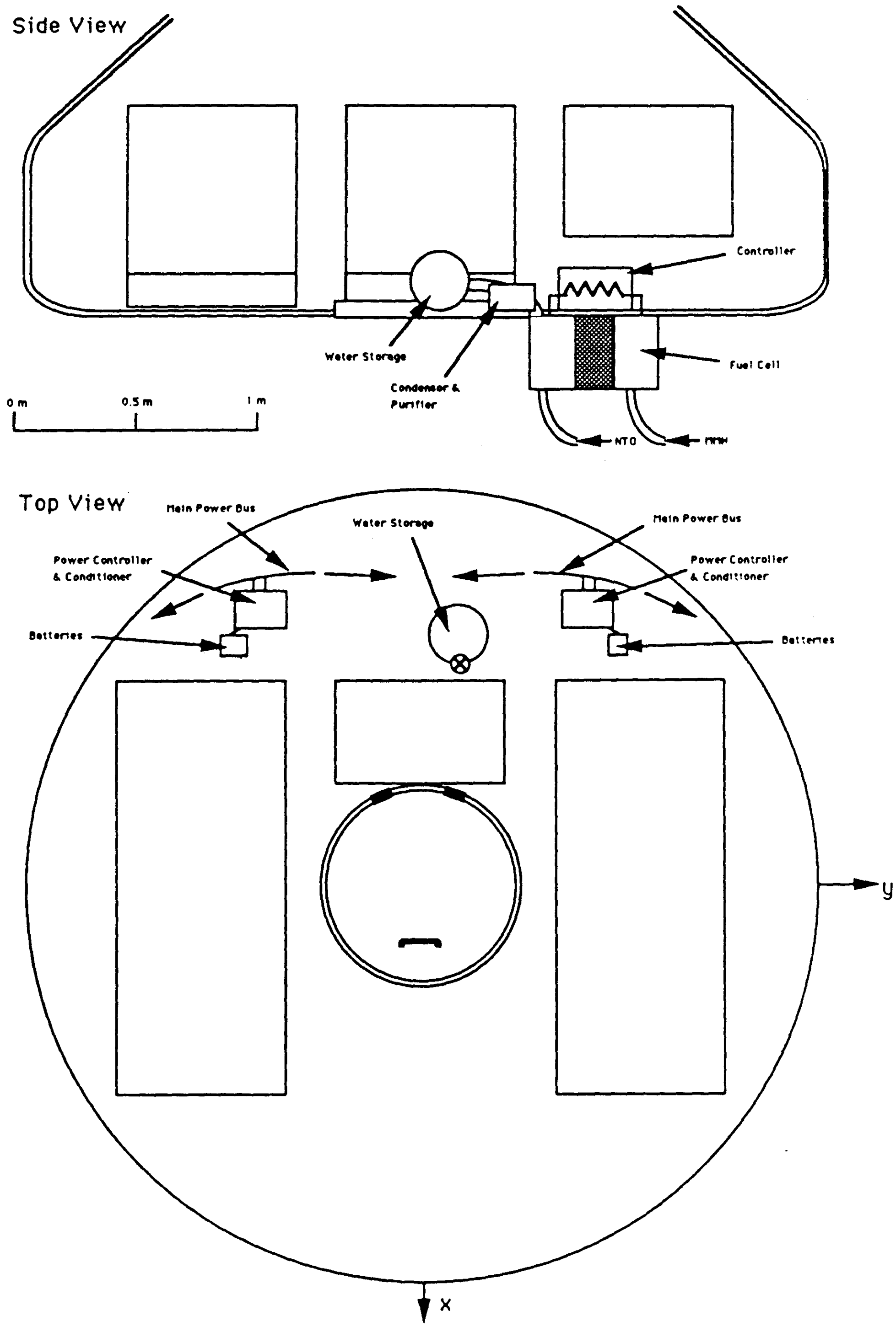

Figure 7-3: Power System

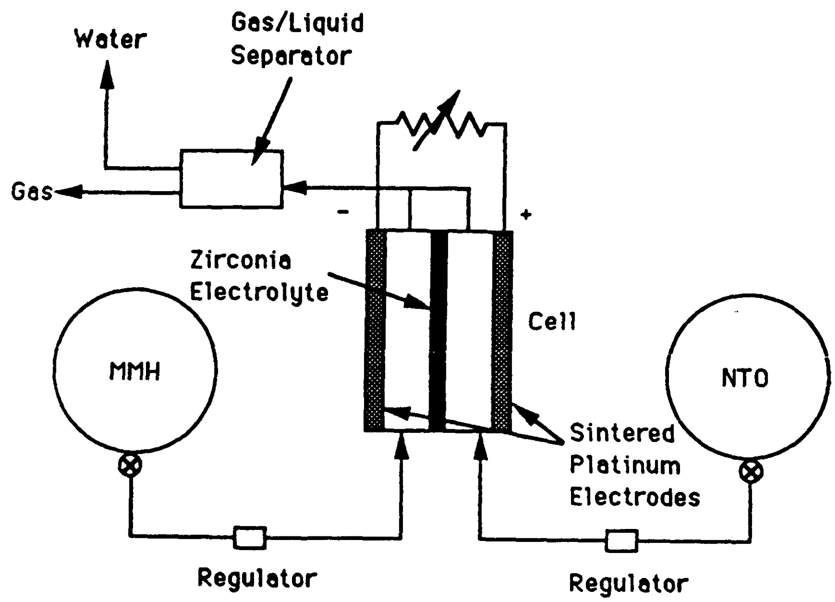

Figure 7-4: Fuel Cell System Schematic

7.5.2 Distribution System

In order to distribute the fuel cell and battery power around the MAV, the power must first be converted to a 28 V DC source. A power controller, which is in turn monitored by the main computer system, determines where the power is needed and monitors the power system. A wiring system (with solid state circuit breakers and switching) then distributes the power to the various subsystems of the MAV.

7.5.2.1 Power Conditioning

The power obtained from the fuel cells and batteries is converted to a 28 V direct current (DC) source (no alternating current (AC) is required). Each subsystem component that needs a different voltage internally modifies the 28 V power in order to meet its needs.

7.5.2.2 Controller

The power controller monitors (with the main computer) the subsystem power needs in order to route the correct amount of current to the various subsystems. The controller also monitors the fuel cell power, and converts the MAV to battery power if a fuel cell failure occurs.

7.5.3 Other Power Needs

To allow immediate, emergency abort-to-orbit (when possible from an orbital mechanics standpoint), thermal energy must be provided to keep the propellants in the liquid phase, and the MAV cabin at room temperature.

During the Earth-Mars journey, the MAV is expected to be part of the MTV spacecraft environment, providing additional living and storage space for the journey to Mars. Therefore, the MAV cabin is maintained at room temperature as a result of its connection to the conditioned air of the main spacecraft.

While on the Martian surface, the MAV cabin requires 438 W average (see Section 10.0) of thermal energy to maintain a room temperature environment. This heat is expected to be provided by the power system of the MDV, and thus does not affect the MAV power system design. If this presents too much of a load for the MDV power system, a separate system can be set up on the Martian surface specifically for the purpose of heating the MAV.

The power needed by the propellant could also be supplied by the MDV, while the MAV is on the surface, and by the MTV, while in space. However, that would require a separate heat venting system, because the MAV propellant tanks are not directly connected to the MAV cabin.

To prevent the propellants from freezing, 83 W of thermal power is required (see Section 10.0). This power is supplied by small radioisotope heating units (RHUs) that are located at the center of the propulsion tanks. Each of the RHUs weighs 43 grams and supplies 1.1 W of heat (Putnam, 1989). To provide the required 83 W of heat, 76 RHUs are placed inside the propellant tanks; 15 in each NTO tank, 11 in each MMH tank, and 6 in each N_2H_4 tank.

The RHUs are based on the Galileo heating units. Each RHU consists of a plutonium isotope (^{238}Pu, 89 year half-life), surrounded by iridium and a graphite housing. The graphite housing prevents the plutonium from scattering in the atmosphere if an accident should occur on ascent to Earth orbit.

Because these RHUs are being utilized in a manned spacecraft, there is some concern about the radiation hazard to the astronauts. Each unshielded RHU results in 9×10^{-3} mRem/hr of gamma radiation dose-rate and 4×10^{-3} mRem/hr of neutron radiation dose-rate at a distance of 1 meter (Zocher, 1989). With the 83 W source, this amounts to 8.65 Rem/yr exposure at a 1 meter distance.

Since the astronauts are not located near the MAV for any extended period of time during the flight to Mars, this is not a problem during the Earth-Mars transit. While on the surface, the astronauts spend most of their time in the MDV habitation module, approximately 2.5 meters from the RHUs. Therefore, at this distance, the radiation hazard drops to 1.38 Rem/yr, which is well within the 5 Rem/yr specified limit for man-made radiation (OSHA, 1989). The radiation exposure level is further attenuated by the propellant surrounding the RHUs, shielding the astronauts from some of the radioactive emissions.

7.5.4 Reliability

Required reliability is obtained through redundancy. Two separate power systems connected to the same bus, with separate power controllers and conditioners, are baselined for the power system. Both systems can run off of either the fuel cells or the batteries. To obtain the required system reliability, each of these separate systems must have 0.99 reliability. For the main scenario, it is suggested that the MAV operate off of the lithium batteries.

7.6 Summary

Table 7-3 summarizes the mass, volume, and energy of the MAV power system. As in Section 4.0, these estimates are obtained from information in this section and from historical data. Energy requirements are based on a 38.8 hour worst-case mission with a 1.15 time factor of safety. Additional batteries are located on the MDV to heat up the MAV for the maximum 12.3 hr wait on the surface.

Table 7-3: Power System Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
fuel cell MMH		8.2	0	0.009
fuel cell NTO		16.3	0	0.011
fuel cell stack (dry)	1	20.0	0	0.050
plumbing		3.0	0	0.003
water separator	1	2.0	223.1	0.002
battery	2	9.5	0	0.010
controller	2	15.0	892.4	0.015
power conditioner	2	5.0	892.4	0.005
RHUs	76	3.3	0	0.003
RHU tank structure	8	4.0	0	0.004
misc. structure	1	2.0	0	0.002
wiring & connectors		25.0	0	0.040
TOTAL	93	113.2	2007.9	0.154

7.7 Recommendations

Further investigation and eventually development of MMH/NTO fuel cells is required if they are to be utilized on this mission. Specific energies, efficiencies, and stack masses must be determined, and the fuel cells must be flight-tested and space-qualified.

If such research and development is too expensive, or the MMH/NTO fuel cell turns out to be infeasible, additional mass penalties for high-energy batteries will be incurred. Previously used argon-zinc battery systems are not viable on this mission, unless advanced development provides adequate progress in extending shelf-life.

8.0 Avionics System

8.1 Introduction

The avionics system consists of communications, data systems, instrumentation, and guidance, navigation, and control (GN & C). Each of these four subsystems are described in detail in this section. The avionics system layout is shown in Figure 8-1.

Overall, the entire avionics system must have a 0.995 reliability to meet the vehicle reliability requirement. This means that the four subsystems that make up the avionics system must each have a 0.999 reliability.

8.2 Communications

The MAV communication system must provide a link between the MAV and the MOV at all times for successful rendezvous to occur. This involves keeping a link available while the MAV is on the surface, on ascent, and while in orbit. A secondary link to Earth may also be desirable, if the penalty in power and mass for providing this link is not prohibitive.

8.2.1 Requirements

Communication system requirements:

- constant comm. link between MAV and MOV
- at least a voice link, possibly up to color images
- receiving and sending antenna
- omni antennae on the MAV for assured link on ascent
- high-gain antenna on MOV and comm. satellites for power advantage
- comm. gear accessible to all crew
- possible backup comm. link with Earth (through communication satellite)

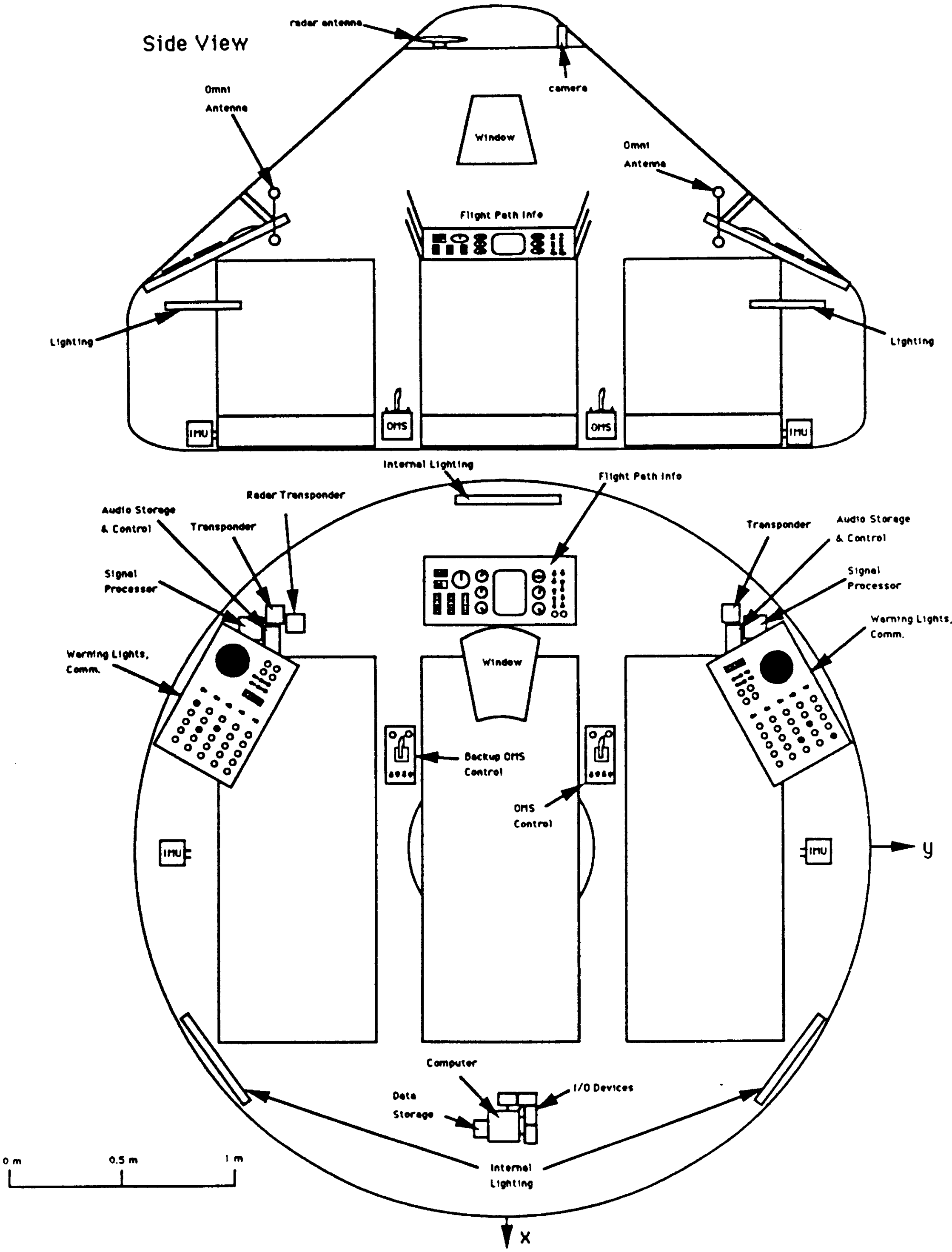

Figure 8-1: Avionics System Layout

- reliability = .999
- shelf-life = 2 years

8.2.2 Point Design

The MAV must maintain a constant communication link with the MOV. During the abort scenarios, the MOV may be out of the line-of-sight of the MAV, so the MAV signal must be relayed through areostationary (18000 km altitude) communication satellites located over the surface site.

The communication subsystem is shown in Figure 8-1, essentially consisting of two sets of redundant omni antennae, each facing in opposing directions, and spacecraft communication interface equipment.

8.2.2.1 Signal Power

To determine the required MAV transponder signal power, the noise power must first be estimated:

$$P_r = kTB$$

k = Boltzmann constant = 1.38×10^{-23} J/°K

T = noise source absolute temperature = 300 °K (NASA TM, 1988)

B = communication bandwidth = 20 kHz (see Section 8.2.2.2)

With an assumed signal to noise ratio of 20 dB (100), the MAV and MOV (or communication satellite) must have a received power of 8.28×10^{-15} W in order to process the voice and command communication signals. To determine the transmitted power required to convey this received power level, the spacecraft communication system must first be defined, including operational frequencies, antennae gains, and necessary communication path lengths.

For this study, the X-band frequency range has been selected. The MAV transmits at a frequency of 8.4 GHz while the MOV (or communication satellite) transmits at 7 GHz to avoid signal interference. The X-band range readily permits the 40 kb/s bit rate required by the MAV, and is well-tested for space applications.

To maintain a constant communication link, omni antennae are utilized by the MAV. These antennae transmit to a high gain antenna located on the MOV (or comm. satellite). Using this information, the required MAV transmitted power can be determined:

$$P_t = \frac{P_r\,(4\pi r)^2}{G_r\,G_t\,\lambda^2}$$

P_t = transmitted power

P_r = received power = 8.28×10^{-15} W

r = maximum path length = 18000 km

G_r = receiving antenna gain = 30000

Gt = transmitting antenna gain = 1

λ = signal wavelength = 0.0357 m

To transmit information to the MOV, therefore, the MAV requires a maximum of 11 W of constant power. Power requirements for high-rate video data transfer are detailed in Section 8.2.2.2.

8.2.2.2 Bandwidth

To determine the bandwidth required by the MAV communication signal, the transmitted data type is first specified. At the minimum, single channel 20 kb/s (see Table 8-1) voice communication is required. Bandwidth is also necessary for command and range/range rate information (see Section 8.3.2.1.3), for which an additional 20 kb/s is provided.

Table 8-1: Data Rate Requirements (NASA TM 4075, 1988)

Data type	Data rate (Mb/s)	Description
high-rate video	100	• 1 channel, color, 512 x 512 pixels, 8 bits/pixel, 30 frames/sec
low-rate video	0.20	• 1 channel, monochrome, 512 x 512 pixels, 8 bits/pixel, 0.1 frames/sec
voice	0.02	• 1 channel
science telemetry	to 10 to 300	• low duty cycle spectral scanning w/storage • no data storage, spectral scanning
engineering	0.2 0.002	• per manned spacecraft • per unmanned spacecraft
telerobotics	0.2 200	• command channel, per rover • stereo, high-rate video
command	to 0.002	• per spacecraft or science platform/site
data load	to 1.0	• Earth to manned vehicle

Using Shannon's limit, the theoretical required bandwidth for a 40 kb/s data rate is determined:

$$B = C\left(\log_2\left(1 + \frac{S}{N}\right)\right)^{-1}$$

C = data bit rate = 40 kb/s

S/N = signal-to-noise ratio = 100

Therefore, a minimum 6 kHz bandwidth is required for a 40 kb/s data rate transfer. To allow for modulation, deviations from the theoretical bandwidth potential, and additional bit-rate transfer, the MAV system is sized for a bandwidth of 20 kHz.

If uncompressed high-rate video is desired, X-band frequency does not provide sufficient data-rate capability. Even if a K_a (~32.4 GHz) frequency range is used for this purpose, the MAV would require a minimum of 110 kW of transmitting power. With the MAV omni antennae, high-rate video transfer is simply not feasible. Nor, for the same reason, is a direct link to Earth viable. An Earth-link, if needed, must be maintained through a communication satellite.

8.2.2.3 Antennae

The MAV communication system is based on 4 omni antennae. Each antenna covers the entire area in front of the antenna plane. This translates to an antennae unit gain:

$$G \approx \frac{30000}{\theta^2} \text{ (Agrawal, 1986)}$$

$$\theta = \text{coverage angle} = 180°$$

Two antenna face forward and two backwards with respect to the center line of the MAV, permitting 360 degree redundant coverage. Only one antenna is nominally utilized to transfer the signal to the MOV.

8.2.2.4 Communication Interface

MOV voice communications are received by the astronauts through redundant speakers located on the communication boards (see Figure 8-1). As a backup, the astronauts can also receive and send communication signals through individual headsets. If hull rupture occurs, or for communication during emergency EVA, the spacesuits are also equipped with headsets.

All communications are taped and stored in the MAV. Redundant transponders, communication boards, and signal processors are provided for reliability.

8.3 Guidance, Navigation and Control

The GN & C system in the MAV must monitor and control the MAV trajectory on ascent, rendezvous with the MOV, and during docking maneuvers.

8.3.1 Requirements

The MAV guidance, navigation, and control system requires:

- ability to monitor MAV trajectory data
- inertial navigation system
- guidance and sufficient computer power to allow rendezvous with MOV
- avionics to control the main propulsion system, as well as the OMS
- reliability = .999
- shelf-life = 2 years

8.3.2 Point Design

To maintain the MAV trajectory, the GN & C system must monitor the MAV location with respect to that trajectory using a number of navigation aids. It then corrects for any deviations from the prescribed trajectory, utilizing the control system.

8.3.2.1 Navigation Aids

To determine the location of the MAV with respect to its pre-programmed trajectory, inertial measurement units (IMUs), a star tracker, and radar ranging equipment are employed.

8.3.2.1.1 IMU : Two lightweight, redundant Honeywell GG1320 IMUs are selected to monitor the spacecraft's inertial position in space. Each IMU consists of three GG1320 ring lazer gyros (RLGs) to monitor MAV pitch, roll, and yaw, three Sunstrand Superflex accelerometers to measure acceleration, and a TMS 320 IMU processor (see Figure 8-2).

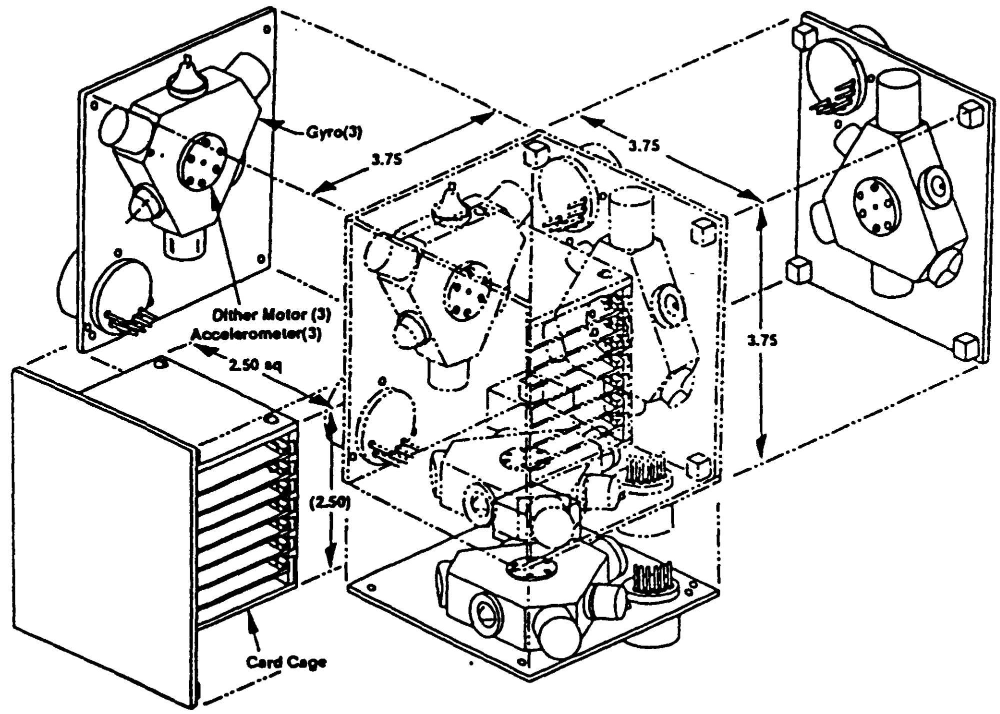

Figure 8-2: Honeywell GG1320 IMU (Honeywell, 1989)

The entire IMU is 0.0953 m (3.75 in) square, has a mass of only 0.76 kg, and takes less than 7 W of constant power.

8.3.2.1.2 Star Trackers: A star tracker is required only to initially align the MAV IMUs. Since this will occur on the Martian surface, the star tracker need not ascend with the rest of the MAV. The measurements can be taken from inside the MDV habitation module, and transmitted to the MAV for IMU alignment.

If this is infeasible, a star tracker is relatively lightweight (7.7 kg for the STS star tracker) and could be mounted and retained onboard.

8.3.2.1.3 Ranging Equipment: Range and range rate data is required for rendezvous and docking with the orbiting MOV. For most of the ascent from the Martian surface, the communication equipment can be utilized for this purpose. By measuring the turnaround time for communication signals between the MAV and the MOV, range and range rate data can be obtained. Due to the processing time on both ends of the communication link, this is a course estimating method for this information, but that is all that is required for the majority of the MAV trajectory (> 1km distance between spacecraft).

As the MAV approaches the MOV for docking, more accurate ranging information is required. A radar signal projected from an antenna in the nose cone of the MAV gives the precise ranging information required for the docking maneuver. Utilizing a modified Apollo rendezvous radar system, a 9.833 GHz signal is projected from a 0.254 m (10 in) antenna. Since this maneuver occurs at a distance of less than 1 km from the MOV, and the beamwidth is ~10°, no pointing mechanism is required.

8.3.2.2 Control System

Because two engines are always being utilized during Martian ascent, a separate reaction control system (RCS) for ascent is not required. The main ascent engines can both differentially throttle and gimble to control the MAV flight path. All of these maneuvers are controlled by the main computer, which continuously compares the pre-programmed trajectory to the measured real-time trajectory.

For docking maneuvers, the OMS (see Section 6.0) provides smaller thrust levels. This system is astronaut-controlled using redundant joysticks inside the MAV. Additional manual controls are provided for emergency main engine and OMS shutdown.

8.4 Computer

The main MAV computer must monitor all of the subsystem functions to warn the crew of possible spacecraft malfunction. Additionally, the computer controls the main ascent propulsion system, monitors the MAV trajectory, and controls certain MAV subsystem functions.

8.4.1 Requirements

The MAV computer system must:

- control some subsystem functions
- monitor all subsystems
- monitor ascent data
- control flight path
- reliability = .999
- shelf-life = 2 years

8.4.2 Point Design

The computer selected for this mission is the Harris R3000 CPU-based computer, which is also being studied for possible use in the the Mars Rover Sample Return (MRSR) mission.

8.4.2.1 Specifications

The main computer consists of three separate redundant processor nodes, redundant power supplies, and sufficient telemetry memory (3 Mbit) to monitor the main trajectory (see Figure 8-3). Input/output information from the subsystems are obtained through the I/O devices wired to the redundant system buses. Each processor node consists of a R3000 CPU chip capable of 20 MIPS (mega-instructions per second), 256 Kbytes of instruction and data memory RAM, and a R3010 FPU chip.

The system is protected against single point failures through CPU cross-checking and has single event upset (SEU) protected memory. This, along with the double redundancy, allows a system reliability of 0.9994 (Harris, 1989). The entire assembly has a mass of 2.3 kg and requires a maximum of 40 W of power.

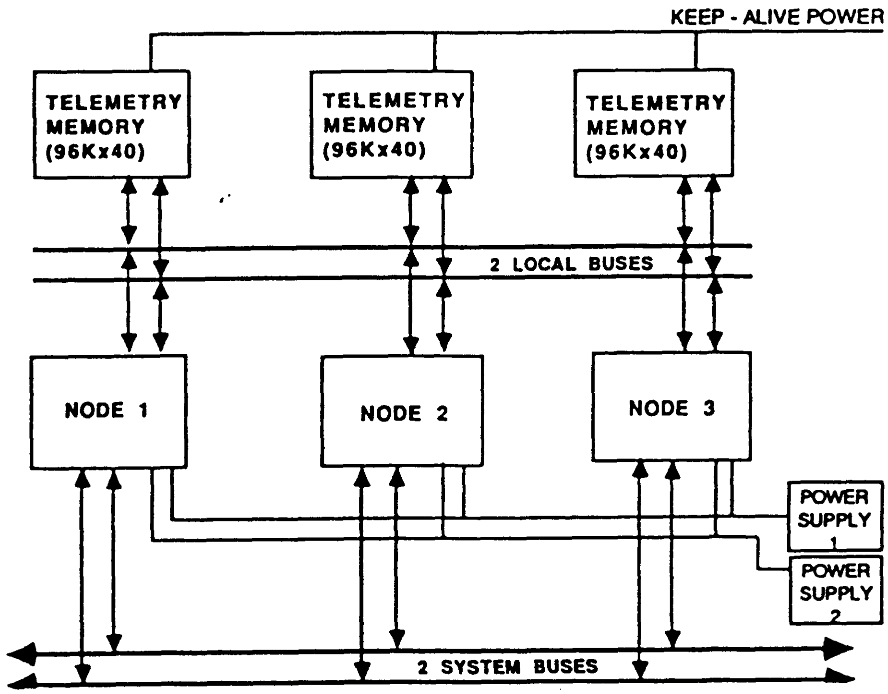

Figure 8-3: MAV Computer Schematic (Harris, 1989)

8.4.2.2 I/O Devices

Input/Output devices are utilized to monitor and control the various subsystem functions. Figure 8-4 and 8-5 illustrate the LSS, propulsion system, and GN & C computer schematics. All of the other subsystems, including power and thermal control, are passive systems, and thus are only monitored by the computer system.

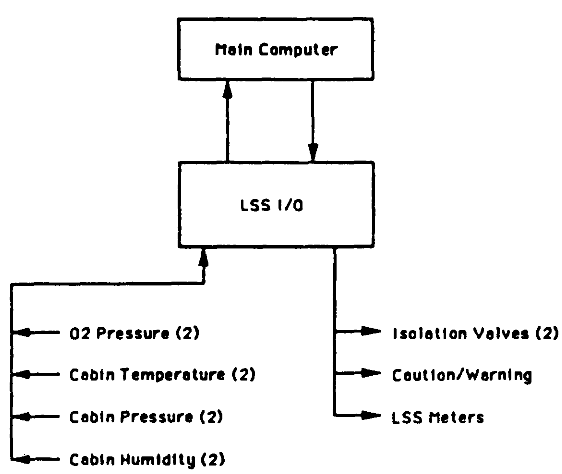

Figure 8-4: LSS I/O Schematic

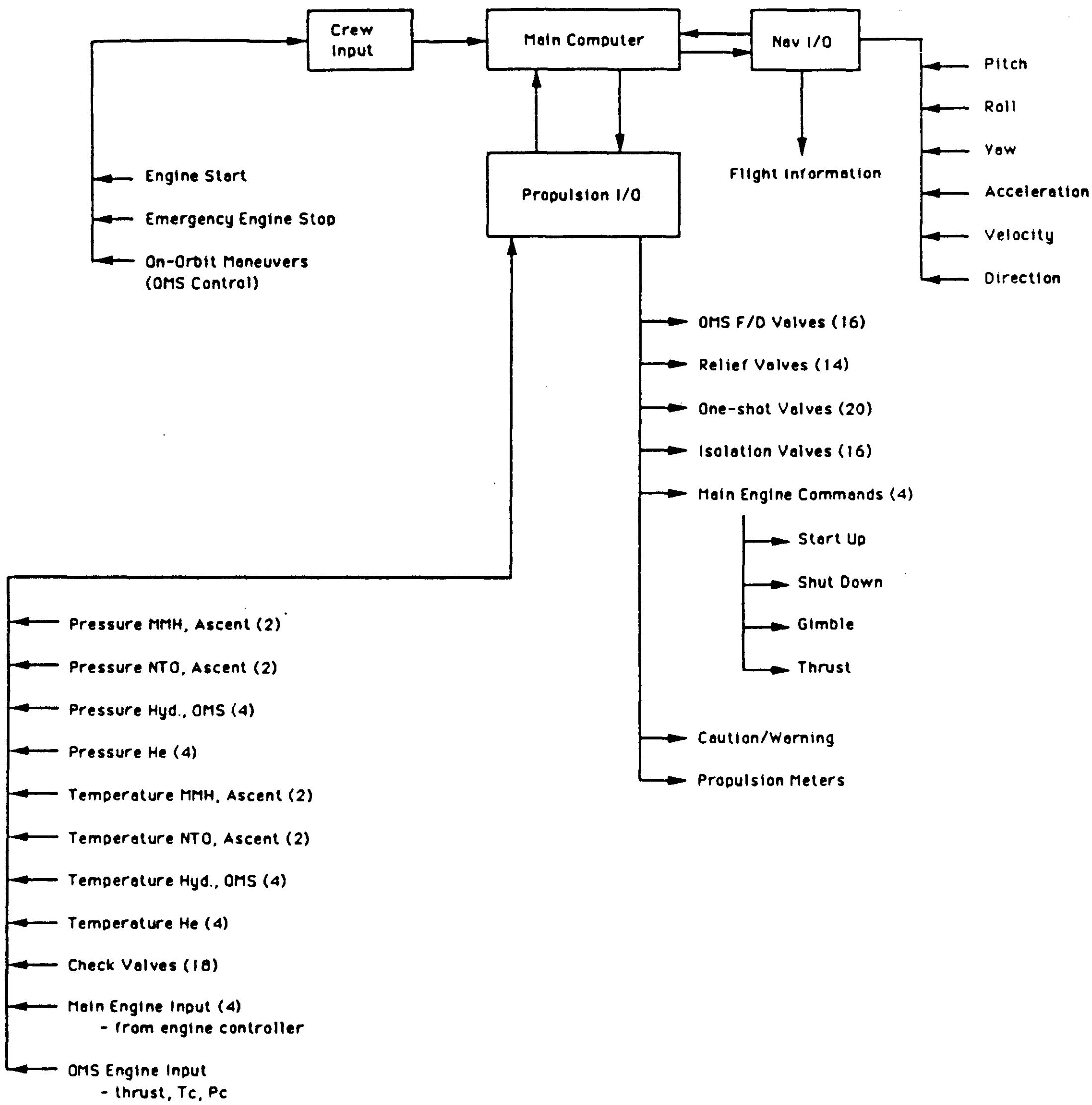

Figure 8-5: Propulsion and GN&C I/O Schematic

8.5 Instrumentation and other Avionics

The remainder of the avionics system consists of lighting, both external and internal, displays and instrumentation, and cameras.

8.5.1 Requirements

- instrumentation/controls accessible by all crew
- displays/warnings for all life critical systems
- emergency detectors for life critical systems
- display flight path information
- provide cameras for PR purposes
- sufficient lighting for internal and external viewing needs
- reliability = .999
- shelf-life = 2 years

8.5.2 Point Design

8.5.2.1 Instrument Panels

Information is displayed on one of three main panels accessible to the astronauts; two communication and warning panels, and a flight information panel (see Figure 8-1).

The communication/warning panels contain LSS warning lights for cabin pressure, temperature, and humidity, fire warnings, subsystem and overall power meters, helium pressurant meters, propellant levels, audio comm. speakers, and communication controls. Additionally, if automatic switching fails, the astronauts can manually switch to a redundant system using control switches on this board.

An integrated display is located in the center of the flight panel, giving information on navigation data, including velocity, range to target, acceleration, and pitch, roll and yaw data. Actual trajectory data is reported on-screen and compared to pre-programmed trajectory data. Surrounding the display are propellant meters, event and mission timers, and main engine emergency shut-down and startup controls.

8.5.2.2 Lighting

Lighting is required for internal viewing of instruments and controls and for docking with the MOV. Internal lighting parameters are based on the Contingency Earth Return Vehicle (CERV) study (JSC-32025, 1987).

For external lighting, a high intensity tracking light, similar to the one used on Apollo is utilized. This allows visual sighting of the MAV from a distance of up to 259 km, permitting MAV viewing from the MOV during the entire main mission scenario. When the MAV approaches within 60 m of the MOV, incandescent lights are used to allow visual docking.

8.5.2.3 Cameras

Cameras are located on both the front and rear of the MAV, providing images of the ascent from the Martian surface. These pictures are stored and sent back to Earth once rendezvous has occured.

8.6 Summary

Table 8-2 summarizes the mass, energy, and volume requirements of the avionics system. Data is taken from historical sources, and from information in this section. Energy requirements are based on the maximum 38.8 hour mission with a 1.15 factor of safety. Power amplifier efficiencies are assumed to be 35% (NASA TM 88174, 1985).

Table 8-2: Avionics System Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
audio:				
audio controller	2	7.0	892.0	0.007
audio storage unit	2	5.0	446.0	0.005
crew headsets	3	0.9	0	0.001
X-band:				
transponder	2	6.0	1403.0	0.006
power amplifier	3	6.0	892.0	0.006
signal processor	2	6.8	669.0	0.007
omni antennae	4	4.0	0	0.004
radar antenna	1	2.0	0	0.002
radar transponder	1	3.0	25.0	0.003
Data:				
central processor	1	2.3	1785.0	0.002
mass storage	1	2.0	446.0	0.002
time/freq. unit	1	2.0	446.0	0.002
I/O devices	5	10.0	892.0	0.010
wiring and cables		15.0	0	0.015
warning, comm panel:				
panel	2	10.0	0	0.010
panel support structure	2	6.0	0	0.006
LSS warning lights	12	3.0	223.0	0.003
fire warning	2	1.0	44.6	0.001
comm. controls	2	1.0	223.0	0.001

Table 8-2: Avionics System Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
power meter	2	1.0	44.6	0.001
OMS pressure meter	4	1.0	89.2	0.001
ascent pressure meter	4	1.0	89.2	0.001
helium pressure meter	4	1.0	89.2	0.001
prop. temp. gauges	8	2.0	223.0	0.002
docking - yes/no	2	1.0	5.0	0.001
caution/warn. circuitry	2	10.0	0	0.010
toggle switches	30	3.0	223.0	0.003
rotary switches	4	1.2	223.0	0.001
switch circuitry	2	6.0	0	0.006
wall plugs	2	0.8	0	0.001
speakers	2	1.4	0	0.001
flight path panel:				
panel	1	5.0	0	0.005
panel support structure	1	3.0	0	0.003
integrated display	1	15.0	1785.0	0.002
OMS propellant meter	1	1.0	134.0	0.001
ascent propellant meter	2	2.0	268.0	0.002
event indicator	1	2.0	223.0	0.002
event timer	1	0.5	134.0	0.001
mission timer	1	0.5	134.0	0.001
toggle switches	8	0.8	134.0	0.001
main engine controls	4	4.0	10.0	0.004

Table 8-2: Avionics System Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
OMS controls:				
thrust controls	4	4.0	5.0	0.004
directional control	2	2.0	10.0	0.002
OMS shutoff	8	4.0	3.0	0.004
OMS control structure	2	2.0	0	0.002
OMS circuitry	2	8.0	0	0.008
GN & C:				
ring lazer gyro	2	0.8	0	0.001
IMU processor/struct.	2	0.7	625.0	0.001
accelerometers	2	0.3	0	0.001
star tracker	1	7.7	10.0	0.008
cabling		10.0	0	0.010
lighting:				
int. light assembly	3	6.0	1339.0	0.006
dimmers	2	2.0	446.0	0.006
external lighting	2	6.0	500.0	0.006
misc:				
cameras	2	4.0	892.0	0.004
TOTAL	169	213.67	14142.0	0.204

8.7 Recommendations

The avionics system utilizes almost 64% of the MAV energy budget. The average power requirement (~300 W) is not excessive. The extensive time period over which it operates causes the high energy need. Therefore, one possible method of reducing this power requirement is to shut down the non-life-critical elements of the avionics system during the on-orbit period of the maximum length mission.

9.0 Orbital Mechanics

9.1 Introduction

In order to size the propellant, and hence the spacecraft itself, the ΔV required to reach the MOV orbit needs to be analytically determined. This delta-velocity, which is effected by drag and gravity losses, must be optimized within the reliability, thrusting, and staging constraints in order to minimize the MAV mass.

9.2 Requirements

The MAV must:

- ascend to 250 km x 33850 km altitude orbit
- ascend to 37° inclination orbit with the same line of nodes as MOV orbit
- ascend from 0° Latitude, 0° Longitude landing site
- rendezvous with MOV - including synchronizing the orbits
- remain within human tolerable acceleration limits (see Section 5.0)
- have an abort-to-orbit capability from the Martian surface
- have an abort-on-descent capability

Others:

- minimize mass within thrust and staging constraints
- allow single engine-out on ascent

9.3 Main Scenario Trajectory

To evaluate the MAV trajectory, Program to Optimize Simulated Trajectories (POST; NASA1-18147, 1987) is used on the HP9000 computer. POST has been used in the past for evaluating Titan launches as well as optimizing STS trajectories. This program's generality permits it to solve orbital mechanics problems around any rotating oblate body and in any known atmosphere.

POST operates on a user-written input file which contains planetary, atmospheric, vehicle, and trajectory goal data. It functions by trying to optimize a number of user-defined variables, while meeting the trajectory criterion (see Table 9-2), also defined by the user. To determine a MAV trajectory, this input file was written and is listed in Appendix A.

In general, POST can optimize trajectories subject to variable thrust and stage weights. However, in this case, this is not really possible, since there are constraints imposed by permitting single engine-out and by using engines of limited throttleability. Therefore, a trial and error method is employed to "optimize" the trajectory within the given constraints. The following sections describe data that is used in the trajectory-determining input file.

9.3.1 Vehicle Characteristics

The physical characteristics of the spacecraft affect the trajectory. Data on drag coefficients, engine characteristics, and vehicle mass are required for the POST trajectory analysis.

9.3.1.1 Drag Coefficients

Data on drag coefficients for this spacecraft are taken from a previous study on the Manned Mars System Study (MMSS) contract (Peterson, 1989). These data were estimated with an ascent vehicle shaped roughly the same as the MAV. Each vehicle has a conic shape with a rounded front section, differing only in cone half-angles. The previous ascent vehicle had an approximate 35° cone half-angle, while the current MAV has a 45° cone angle. This fact means that the previously utilized values are somewhat optimistic. To address this problem, the effect of increasing drag coefficients is evaluated in Section 9.3.5.

The drag coefficient values utilized in the POST input file for various mach numbers are tabulated in Table 9-1.

Table 9-1: Drag Coefficients for Various Mach Numbers

Mach Number	Drag Coefficient	Mach Number	Drag Coefficient
0	.80	2	.90
.10	.80	3	.73
.25	.70	4	.64
.50	.65	5	.60
.75	.50	10	.48
.95	.62	15	.46
1.04	.90	20	.455
1.50	1.00	30	.45

In addition to the drag coefficients, the MAV frontal area is required to determine the effect of drag on the spacecraft. Based on a 1.67 m (5.5 ft) radius, the vehicle has a frontal area of 8.83 m^2.

9.3.1.2 Engine Characteristics

The throttling capability and maximum thrust required were determined after several POST runs. The maximum thrust level needed is 53376 N (12000 lbf) per engine. In order to allow engine-out, each engine must also be able to throttle from this maximum thrust down to 31.1 kN (7000 lbf). This translates to a 1.7:1 throttling ratio.

By utilizing the engines described in Section 6.0, with an I_{sp} of 342 sec, the mass flow at maximum thrust is 35 kg/sec per engine. At minimum thrust, this drops to 20.4 kg/sec per engine. The engine exit area, which is needed by POST to determine pressure losses (which are minimal in the Martian atmosphere), is about 2.48 m^2 per engine.

9.3.1.3 Vehicle Mass

The vehicle dry mass was also determined after several POST runs, since the propulsion system mass determines the overall vehicle mass. The vehicle dry mass is 2760 kg. The overall mass for the optimum ascent trajectory is 14093 kg, with a propellant mass of 11333 kg.

9.3.2 Martian Atmosphere and Planetary Data

To determine the effect of the Martian atmosphere on the MAV trajectory, Mars' density, pressure, and temperature profiles are needed. These data are found in Appendix A in the POST input file (JPL, 1978). The speed of sound in the Martian atmosphere is calculated from these quantities and also tabulated in Appendix A.

Also required by the POST input file are the Martian planetary and launch site data. These include the J_k, planetary spin rate, planetary mass, polar and equatorial radii, longitude, and latitude. These values are tabulated in Appendix A.

9.3.3 Trajectory Description

POST runs on a series of events which are specified by the user. Each event is started by the conclusion of the previous event and concluded when a dependent variable value is equal to a user-specified value. Independent variables are altered by the computer during optimization in order to minimize the spacecraft mass. These independent variables and optimized values are tabulated in Table 9-2.

The trajectory chosen is a two thrust-level, one-stage, gravity turn trajectory (see Figure 9-1). The MAV is first held down to the MDV habitation module until all engines are functioning.

The MAV then disconnects from the MDV, rises vertically for 0.9 seconds with all four engines running, and pitches over at a fixed rate for the next 4.34 seconds.

From that point, the MAV flies with a zero relative angle of attack. At a POST optimized time, the MAV throttles down and goes into a gravity turn, sending the spacecraft into an intermediate orbit whose apogee is the MOV perigee altitude. The engines shut down, and the spacecraft coasts to the MOV perigee altitude. When this altitude is reached, the engines start up again and the final orbit velocity is obtained. Docking occurs immediately at the MOV perigee.

Table 9-2: POST Independent Variables

Variable	Value	Phase in which variable is used
launch azimuth	48.921 deg	hold-down
pitchover rate	-4.625 deg/s	pitchover
apoapsis radius at burnout	3639.6 km	before coast phase
true anomoly at synch.	180 deg	after coast phase
final apoapsis radius	37249.1 km	final orbit

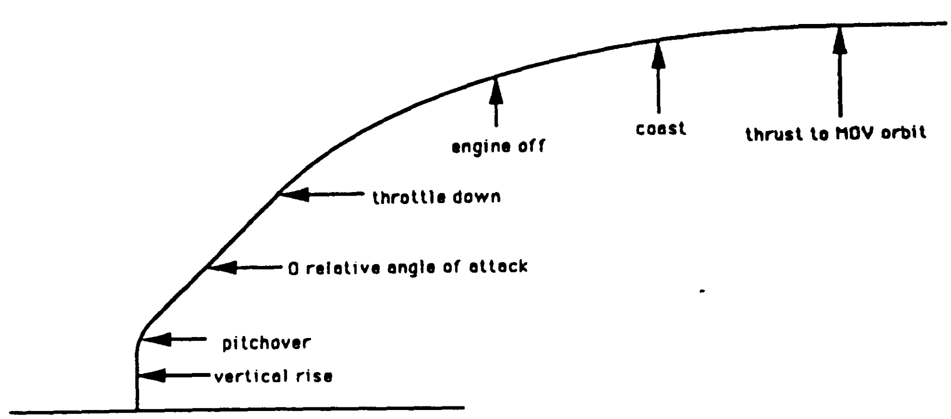

Figure 9-1: MAV Trajectory

9.3.4 Results

9.3.4.1 Thrust Profile

To obtain the minimum vehicle mass with this trajectory, the MAV starts out at 133.4 kN (30000 lbf) of thrust with all four engines operating at 33.4 kN (7500 lbf) each. At the throttle-down point, the thrust is reduced to 62.3 kN (14000 lbf), which translates to two engines running at 31.1 kN (7000 lbf). These values were obtained by trial and error "optimization", since the engines needed to allow the thrust values for the engine-out scenario (see Section 9.4.1). The engine throttling occurs after 6000 kg of propellant are consumed.

9.3.4.2 Trajectory Results

The final trajectory is detailed in Figures 9-2 to 9-8. Velocity, altitude, acceleration, mass, dynamic pressure and heat rates are all obtained from POST runs. The event times, also produced by POST, are shown in Table 9-3.

Table 9-3: Event Times

Event	Start Time (sec)	End Time (sec)
MAV hold down	-3.00	0.00
vertical rise	0.00	0.90
pitchover	0.90	5.24
thrust level 1	5.24	151.24
thrust level 2	151.24	337.57
gravity turn - coast	337.57	1520.15
MOV orbit burn	1520.15	1602.18
docking	immediately after MOV orbit attained	

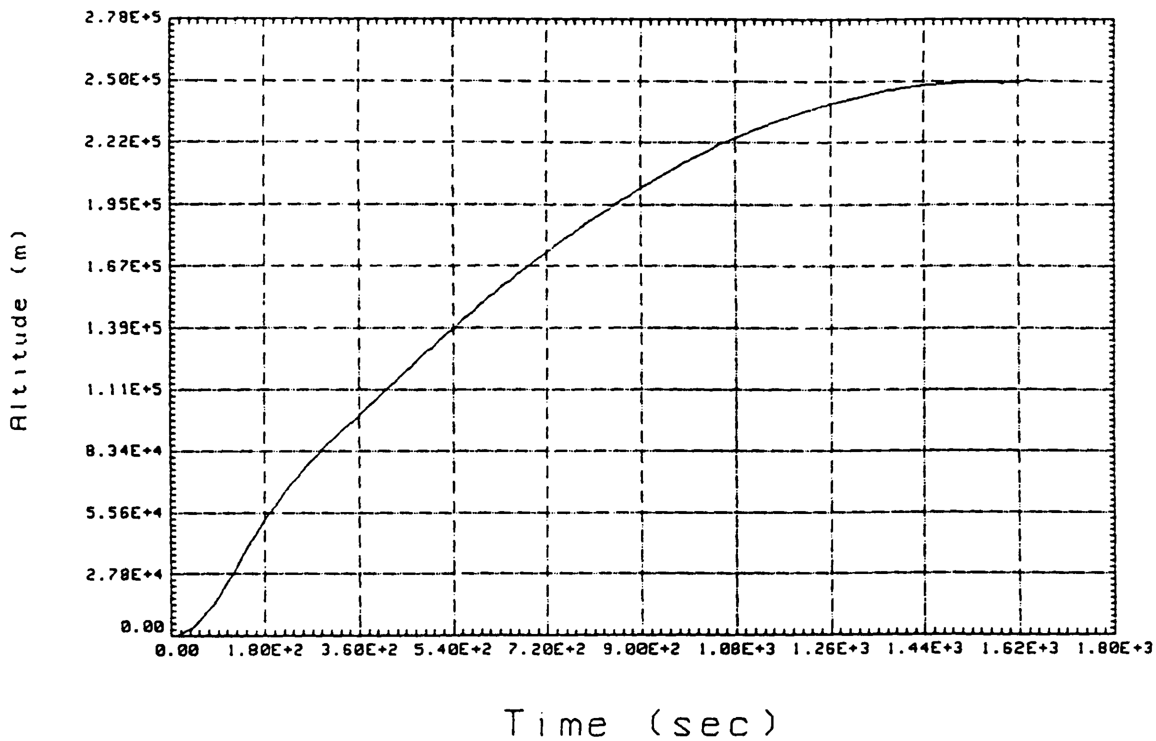

Figure 9-2: Main Scenario Altitude Profile

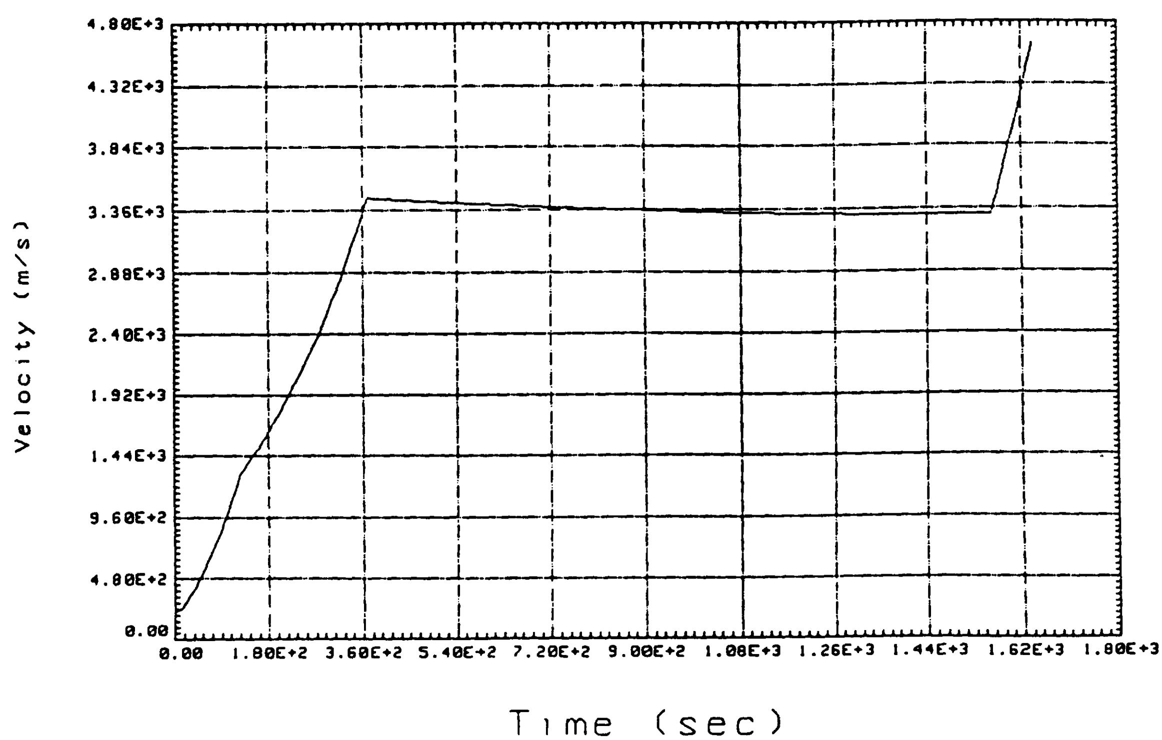

Figure 9-3: Main Scenario Velocity Profile

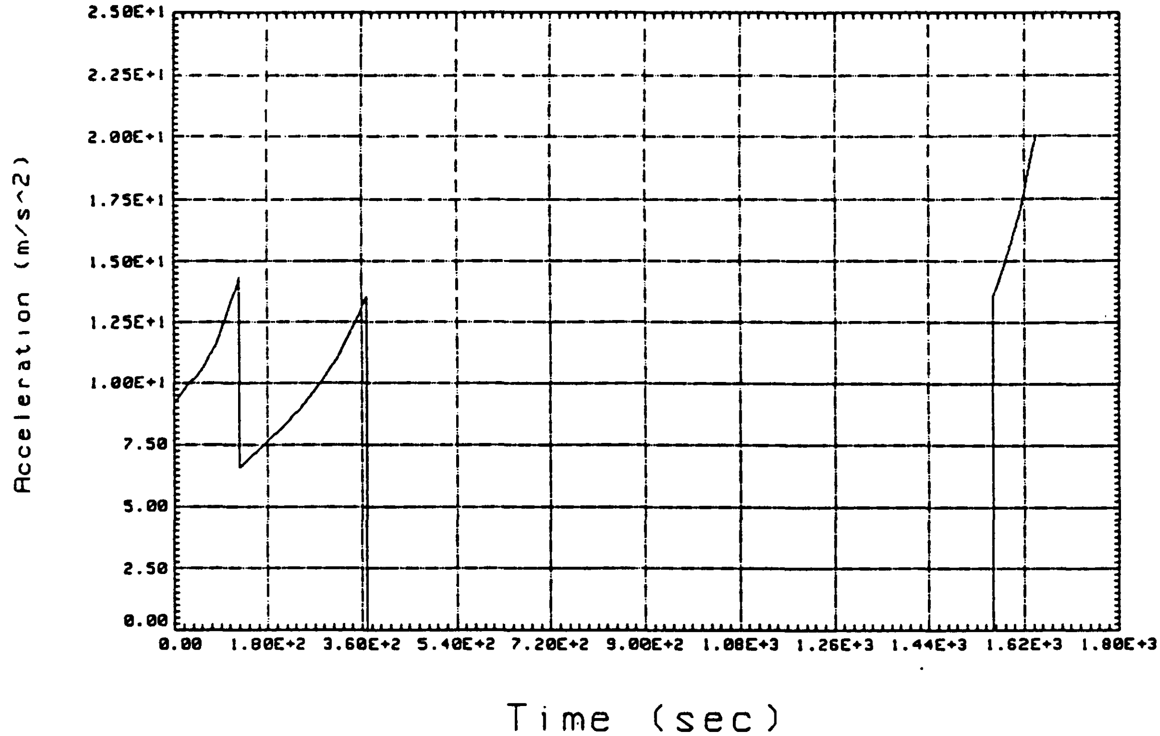

Figure 9-4: Main Scenario Acceleration Profile

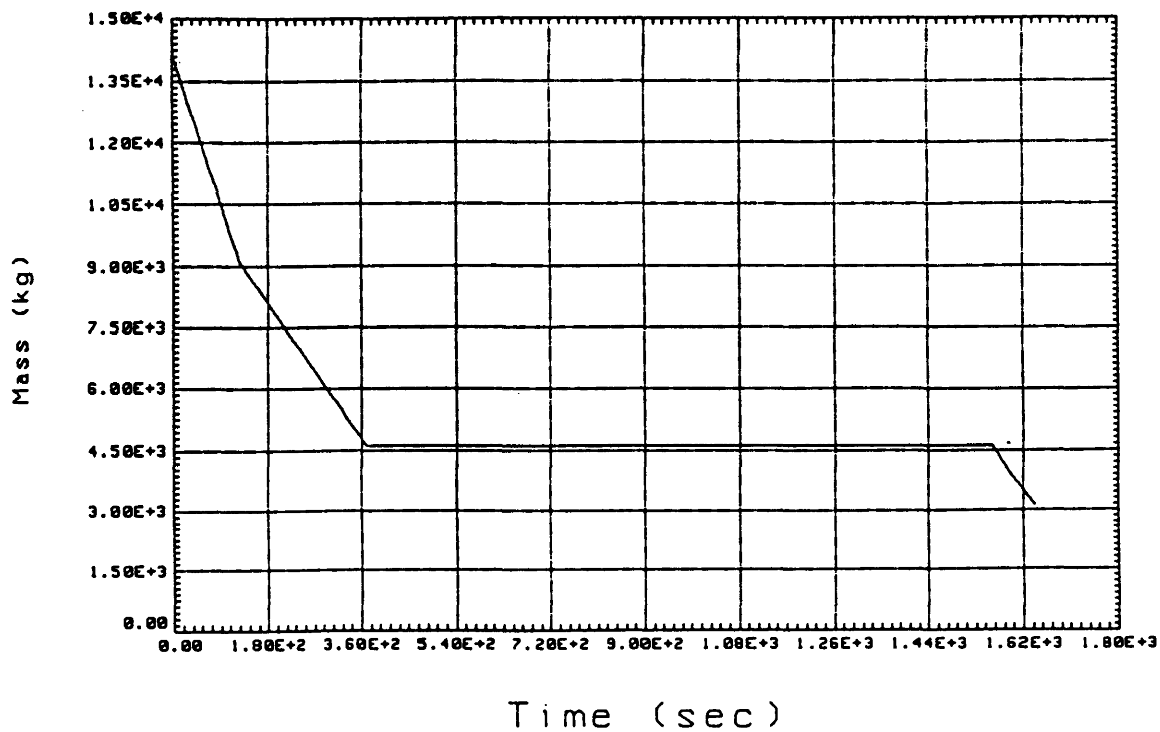

Figure 9-5: Main Scenario Mass Profile

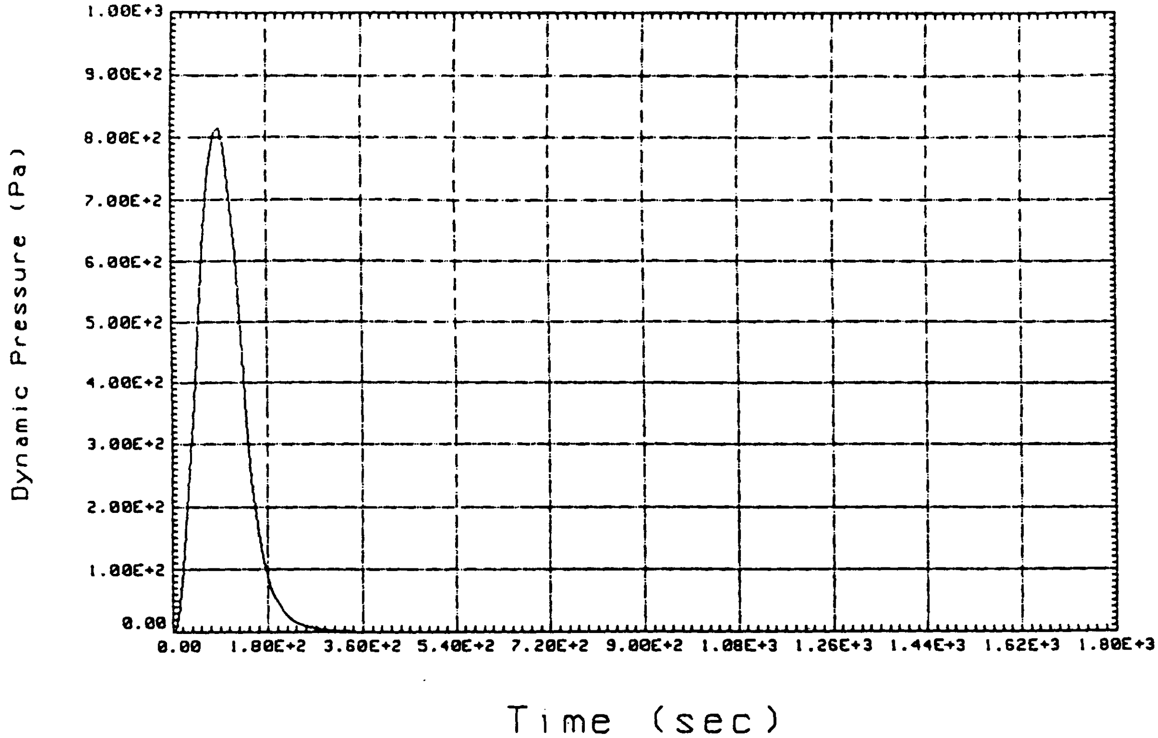

Figure 9-6: Main Scenario Dynamic Pressure Profile

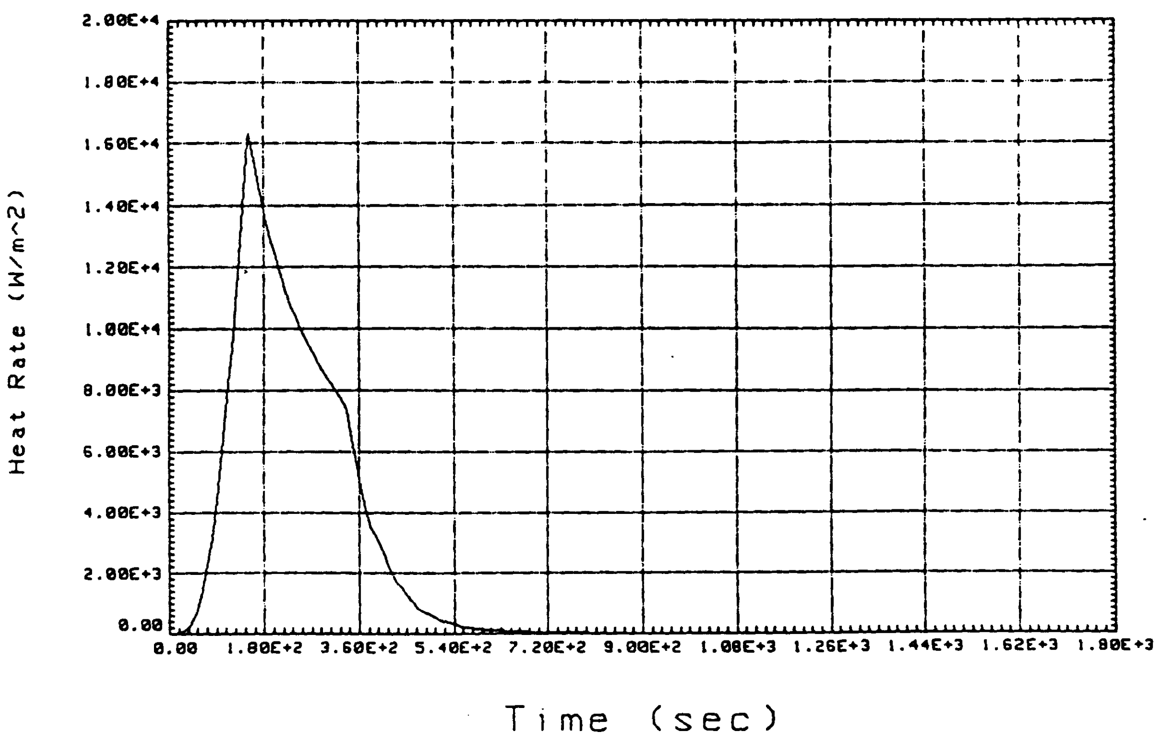

Figure 9-7: Main Scenario Heat Rate Profile

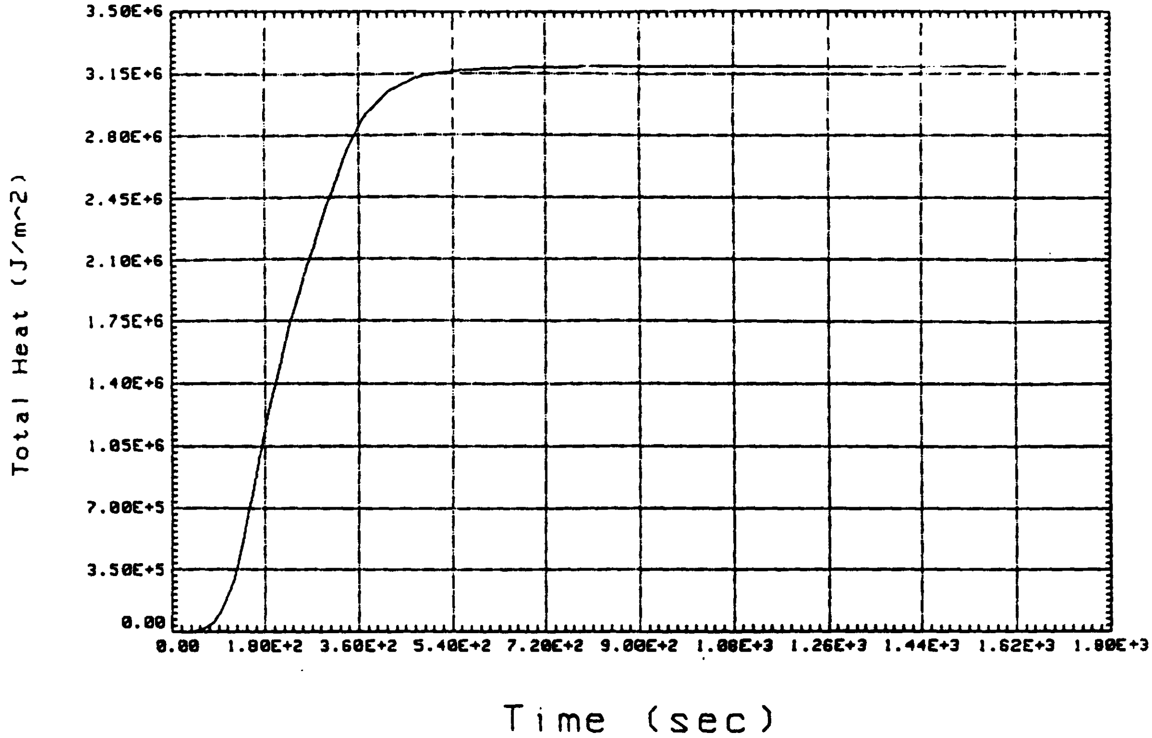

Figure 9-8: Main Scenario Total Heat Profile

The ΔV requirements, which are use to size the MAV propellant, are tabulated below.

Table 9-2: Cumulative ΔV Values

Event	ΔV_{Total} (m/s)	$\Delta V_{Gravity}$ (m/s)	ΔV_{Drag} (m/s)	ΔV_{Thrust} (m/s)
vertical rise	8.53	3.30	0	0
pitchover	49.99	19.29	.01	.20
thrust level 1	1865.81	295.58	56.02	.20
thrust level 2	3734.67	382.46	67.48	3.15
gravity turn	3734.67	382.46	67.48	3.15
MOV orbit	5067.36	384.02	67.48	3.26
docking	5117.36	384.02	67.48	3.26
Total	5117.36	384.02	67.48	3.26

This table summarizes ΔV requirements for the various phases of the MAV mission. The table also includes the ΔV penalties due to gravity, drag, and thrust vectoring.

9.3.5 Effect of Different Drag Values

Because of possible drag coefficient errors, several POST runs were performed with higher drag coefficients. These runs are summarized in Table 9-3. Four of the ΔV values are obtained by multiplying the previous drag coefficients by a constant factor. The other four values are obtained using a constant drag coefficient.

Table 9-3: ΔV for Various Drag Coefficients

Drag Coefficients	ΔV_{Total} (m/s)	$\Delta V_{Drag\ Loss}$ (m/s)
previous run	5117	67
1.5 x previous	5152	104
2.0 x previous	5194	149
2.5 x previous	5251	207
0.5 (constant)	5102	48
1.0	5152	104
1.5	5218	173
2.0	5366	281

This table shows that drag is not a very large percentage of the total ΔV no matter what the vehicle shape. This is very different from Earth, where drag greatly influences both the vehicle and trajectory shape.

9.4 Abort Trajectories

There are two abort trajectories; an engine-out scenario and an abort on descent to the Martian surface. The first abort trajectory evaluated is the engine-out scenario.

9.4.1 Engine-Out On Pad

Clearly, engine-out can occur at any time in the MAV flight. In order to bracket the performance losses in this scenario, the worst case of an engine failing on the pad is evaluated. All other engine-out cases give a ΔV value between the pad failure and the normal scenario.

9.4.1.1 Trajectory

Basically, this is the same trajectory as before, except that the thrust values and the pitchover rate change. If an engine fails on the pad, two engines operate at maximum capacity (53.3 kN or 12000 lbf each) during the first thrust level phase, and throttle down to 31.1 kN (7000 lbf) each for the second thrust phase. The new pitchover rate for this trajectory is -1.17 deg/sec. Also, a synchronization burn may be required since this path takes longer to traverse than the main scenario trajectory (see Section 9.4.3.1).

9.4.1.2 Results

The abort trajectory results are found in Figures 9-9 through 9-15 and in Table 9-4.

Table 9-4: Event Times

Event	Start Time (sec)	End Time (sec)
MAV hold down	-3.00	0.00
vertical rise	0.00	0.90
pitchover	0.90	5.24
thrust level 1	5.24	189.05
thrust level 2	189.05	385.27
gravity turn - coast	385.27	1891.13
MOV orbit burn	1891.13	1967.68
docking	immediately after MOV synchronization	

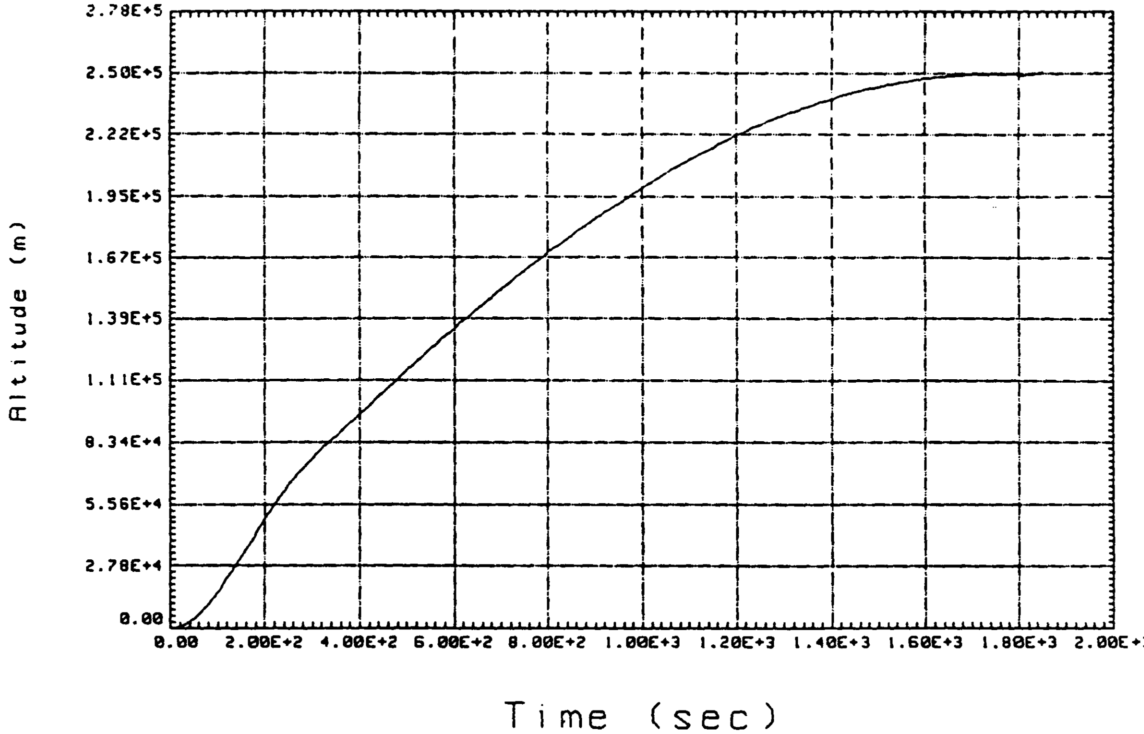

Figure 9-9: Engine-Out Scenario Altitude Profile

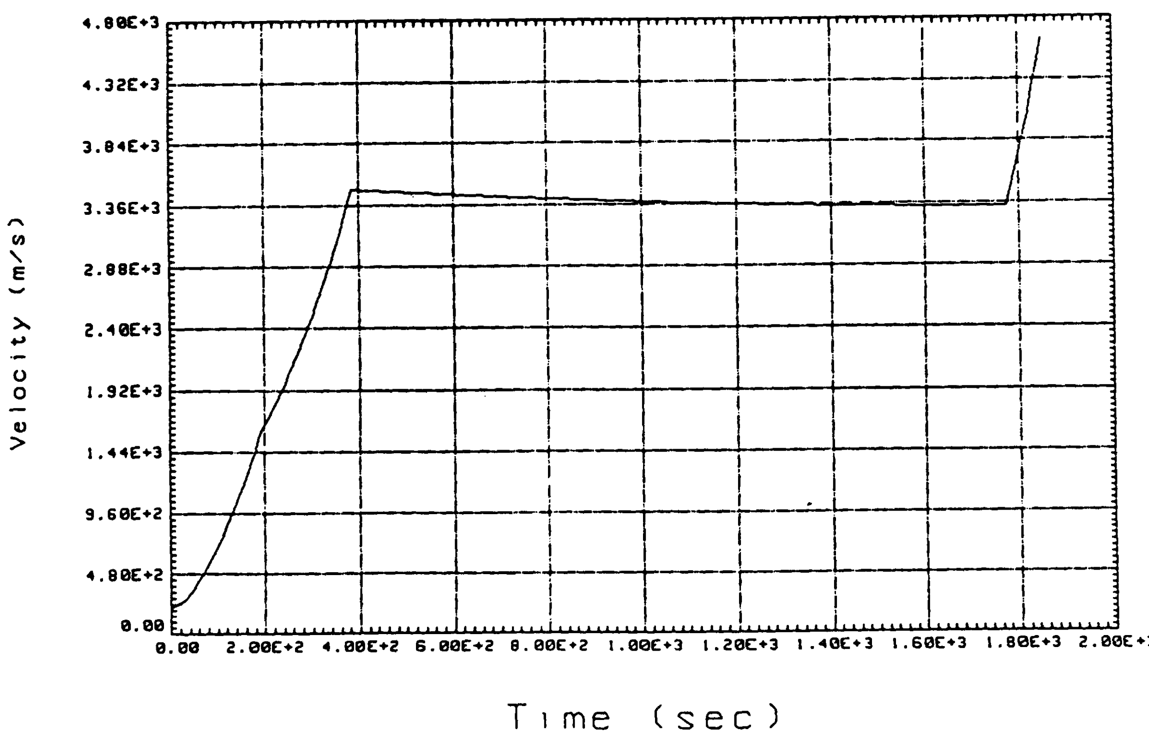

Figure 9-10: Engine-Out Scenario Velocity Profile

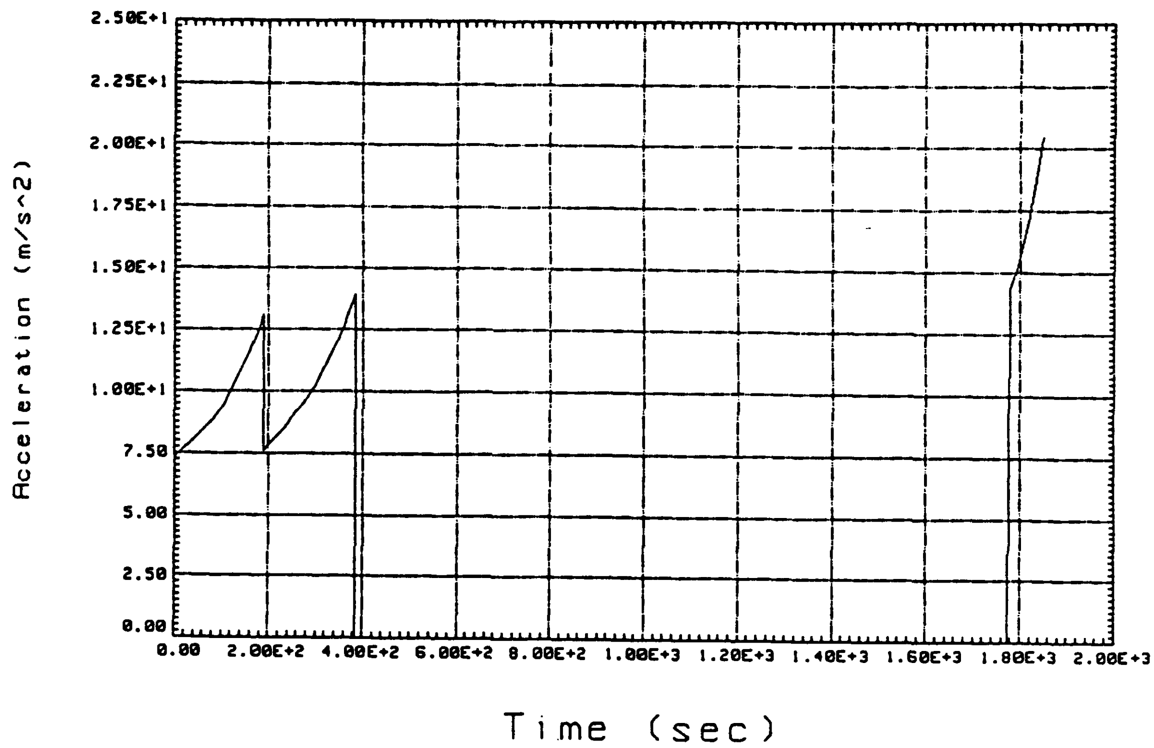

Figure 9-11: Engine-Out Scenario Acceleration Profile

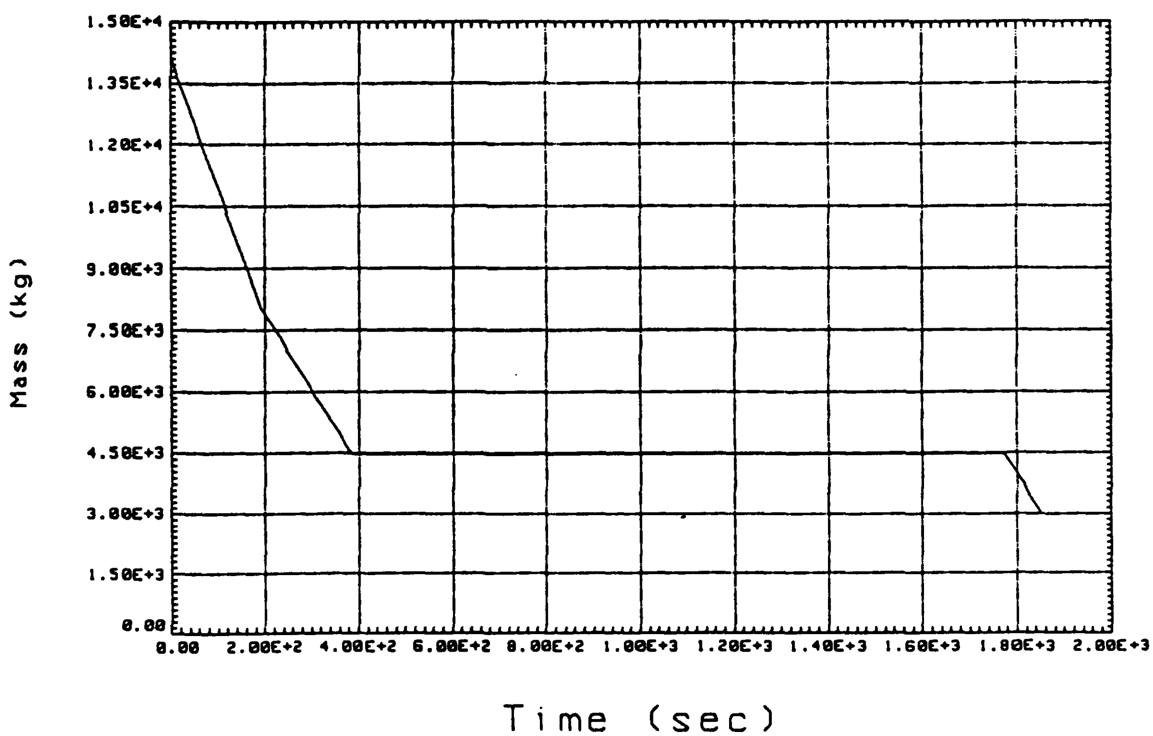

Figure 9-12: Engine-Out Scenario Mass Profile

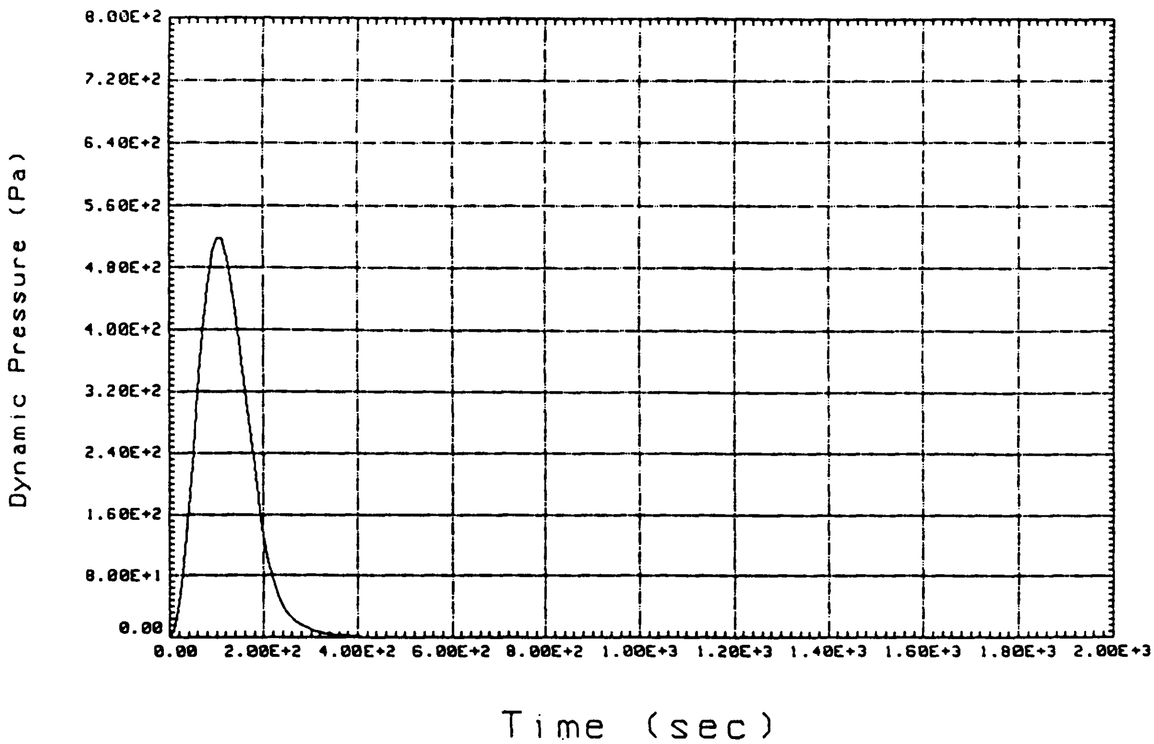

Figure 9-13: Engine-Out Scenario Dynamic Pressure Profile

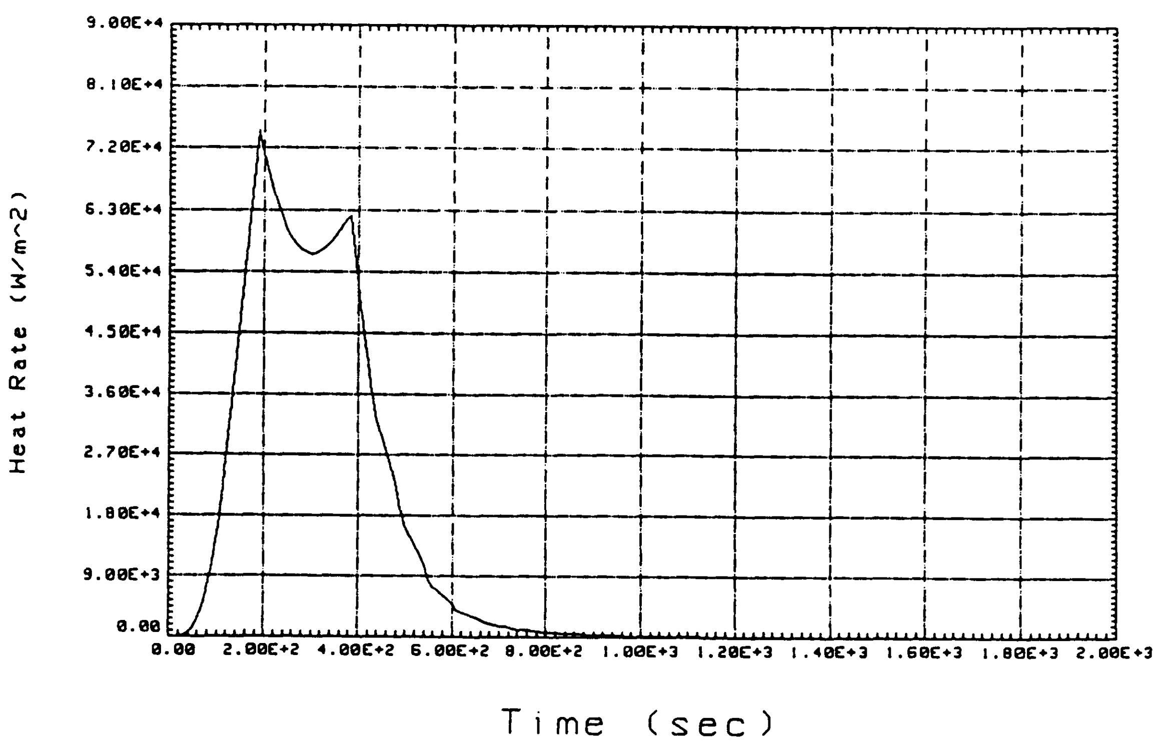

Figure 9-14: Engine-Out Scenario Heat Rate Profile

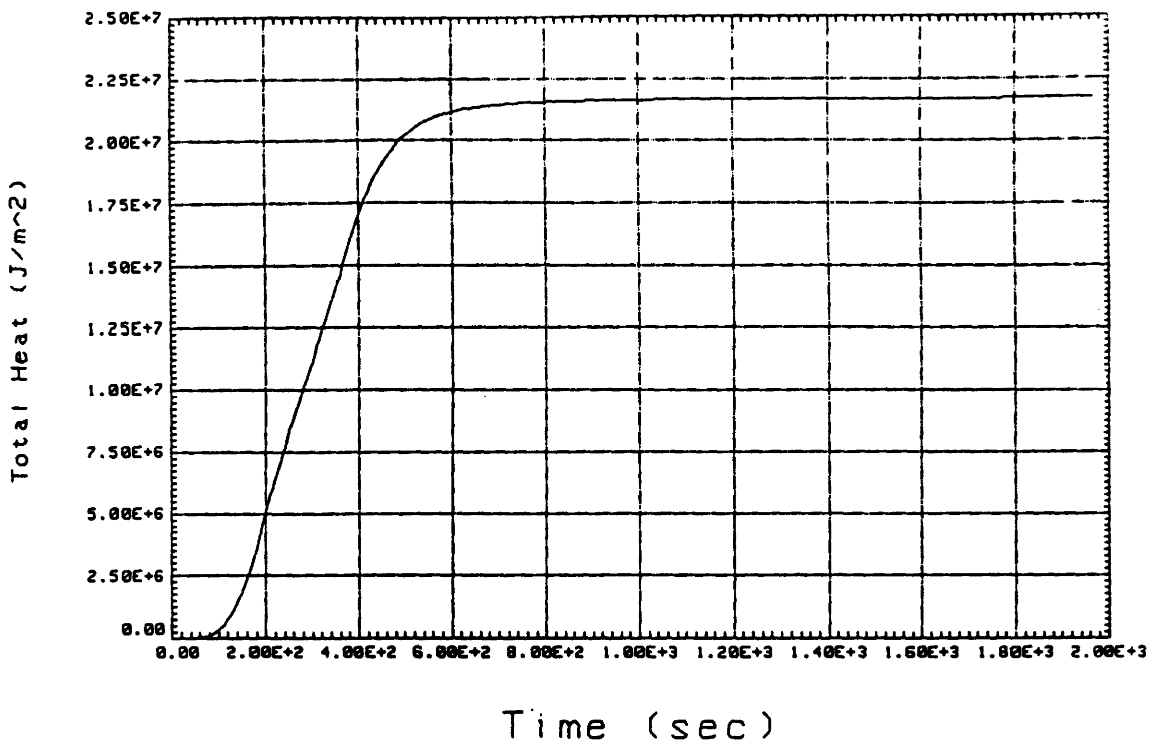

Figure 9-15: Engine-Out Scenario Total Heat Profile

As before, the ΔV chart is given below, with the losses due to various factors.

Table 9-5: Cumulative ΔV Values

Stage	ΔV_{Total} (m/s)	$\Delta V_{Gravity}$ (m/s)	ΔV_{Drag} (m/s)	ΔV_{Thrust} (m/s)
vertical rise	6.82	3.31	0	0
pitchover	39.93	19.42	0	0
thrust level 1	1865.77	412.92	37.90	.08
thrust level 2	3869.97	492.21	42.58	3.80
gravity turn	3869.97	492.21	42.58	3.80
MOV orbit	5156.21	493.74	42.58	4.22
docking	5206.21	493.74	42.58	4.22
Total	5206.21	493.74	42.58	4.22

By comparing Table 9-2 and Table 9-5, it is observed that the abort scenario incurs the loss of about 100 m/s in performance, mainly due to increased gravity loss. If each engine had 66.7 kN (15000 lbf) of thrust, and were throttleable to 31.1 kN (7000 lbf), that performance loss could be eliminated. However, this would involve designing an entirely new engine (see Section 6.0).

9.4.2 Abort-on-Descent

The second abort scenario is an abort-to-orbit on descent. In this scenario, the MAV detaches from the rest of the descending MDV and ascends into orbit. To allow this abort-to-orbit, the spacesuited astronauts descend to the surface while located in the MAV.

After detaching from the MDV, the MAV ascends back into the orbit from which it descended. The same path as the main scenario, gravity-turn trajectory is followed (the final burn to reach an elliptical orbit is delayed, however). This is possible until the MDV is too close to the surface to allow successful engine start-up and separation of the MAV from the rest of the MDV.

After orbit is attained, the MAV must synchronize orbits with the MOV, since, although their orbit planes will be the same, they will no longer be in the same location in that orbit (see Section 9.4.3.1). After this orbit synchronization occurs, the normal docking sequence follows.

9.4.3 Abort-to-Orbit

If an emergency occurs on the Martian surface, the astronauts have the option of ascending back into the MOV orbit. As described before (Section 2.0), the MAV must first wait until it aligns with the MOV orbit line of nodes. This occurs twice a day, translating to a maximum possible wait of up to 12.33 hours before launch is possible. In case the emergency is a MDV habitation module failure, the MAV will provide life support during this period.

After the MAV aligns with the MOV line of nodes, the MAV ascends into orbit, using the same trajectory described in the main scenario. After reaching orbit, the MAV must again synchronize with the MOV.

9.4.3.1 Orbit Synchronization

As described before (Section 2.0), the synchronization of the two orbits takes a maximum of 26.5 hours (1 sol + period of a LMO). This time period, along with the possible surface wait, determines the maximum mission length of 38.8 hours.

In order to synchronize the two orbits after an abort ascent, the MAV only ascends into a 250 km circular orbit. The MAV then makes two burns; one to make sure the two spacecraft align at the next MOV periapsis passage, and the second at the next MOV periapsis passage, to raise apoapsis for synchronization of the two spacecraft. This causes no performance loss, since the same energy is required as in the main scenario.

9.5 Summary

The worst case, engine-out abort scenario determines the minimum MAV ΔV requirement. In addition to that, a ΔV factor of safety is included.

A possible ΔV error could stem from errors in the atmospheric or planetary data, unforeseen propulsion system performance problems, or guidance system errors. The fact that the ΔV used to determine the propellant mass is already sized for an abort case assists in this matter. If any non-engine-out scenario occurs, the MAV has an automatic 84 m/s margin in ΔV. In addition to this margin, the MAV provides another 2% (103 m/s) margin in case the engine-out scenario occurs.

The 50 m/s needed for docking is not provided by the main propulsion system, since the main engines are too powerful to make this delicate maneuver. A separate orbital maneuvering system (OMS) is provided for this action. Therefore, the main propulsion system is sized for a ΔV of 5259 m/s.

10.0 Thermal Control System

10.1 Introduction

Thermal control of the MAV cabin is required to maintain a human-surviveable temperature throughout the astronauts' stay on the Martian surface. Additional thermal problems stem from control of ascent heating, and from maintaining both the OMS and ascent propellants in the liquid state.

10.2 Requirements

The thermal control system of the MAV must:

- maintain a liveable temperature (21° - 27° C) inside the capsule
- maintain the main ascent fuel and oxidizer in the liquid phase
- maintain the OMS fuel in the liquid phase
- minimize overall system mass and volume whenever feasible
- minimize external power requirements subject to reasonable mass
- have a shelf-life of 2 years
- have no single-point failures
- have a system reliability of .995

To provide an emergency abort-to-orbit capability, the spacecraft propellant must be maintained in the liquid phase throughout the surface stay, and the MAV cabin must be thermally controlled to a liveable temperature. Additionally, the MAV is thermally maintained to provide a safe haven for the astronauts in case of system failure of the MDV habitation module.

10.3 Thermal Loads

The MAV must be thermally controlled in a wide variety of environments. These include the vacuum of space, the cold Martian surface, and the harsh heat loads of ascent from the Martian surface.

10.3.1 Space

To determine the necessary thermal control for the MAV throughout its year-long journey to Mars and during ascent, the temperature of space is assumed to be 4° K. Additional space-based heat loads come from solar insolation, which varies from 1350 W/m^2 in Earth orbit to a 583 W/m^2 average at Mars.

For most of the trip to Mars, the MAV is eclipsed from the sun by the main body of the MTV (see Section 2.0). The MAV is also eclipsed at certain points in the Martian LMO and the Earth orbit.

10.3.2 Martian Surface

While the MAV is on the surface, some of the solar energy is attenuated by dust in the Martian atmosphere. Figure 10-1 illustrates average solar intensities throughout the Martian year for the MAV landing site. Figure 10-2 shows daily solar intensity variation.

To specify the remainder of the Martian thermal environment, Figures 10-3 and 10-4 illustrate both the average temperatures throughout the year and the daily temperature distribution at the surface site.

For determining the effects of convection, average Martian surface wind velocities are assumed to be 3 m/s (Ash, 1987).

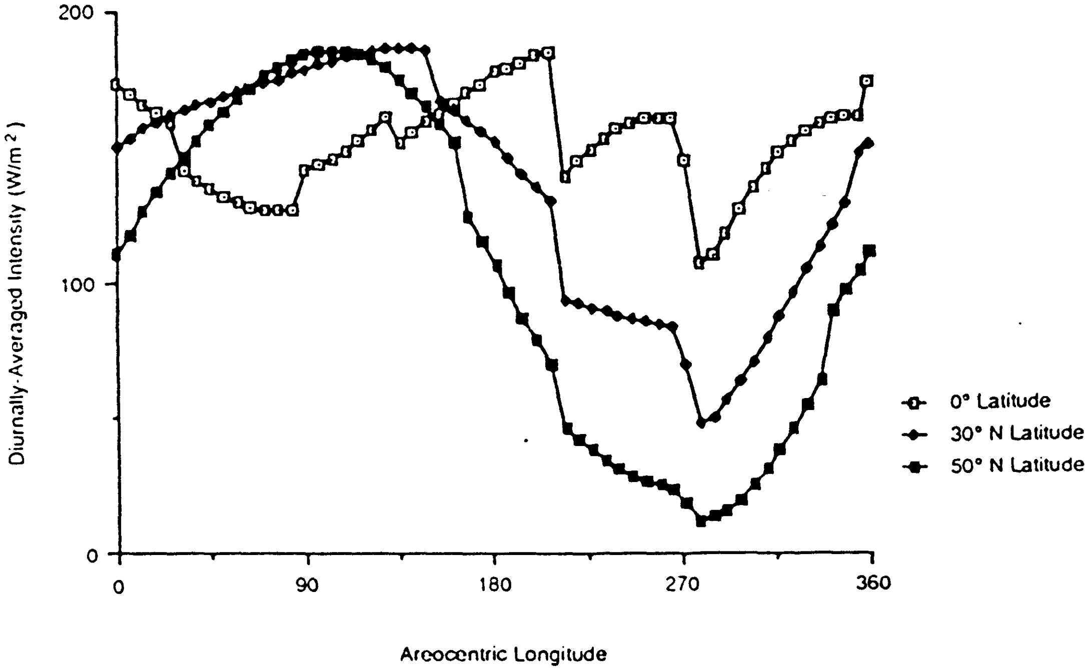

Figure 10-1: Average Solar Insolation on the Martian Surface

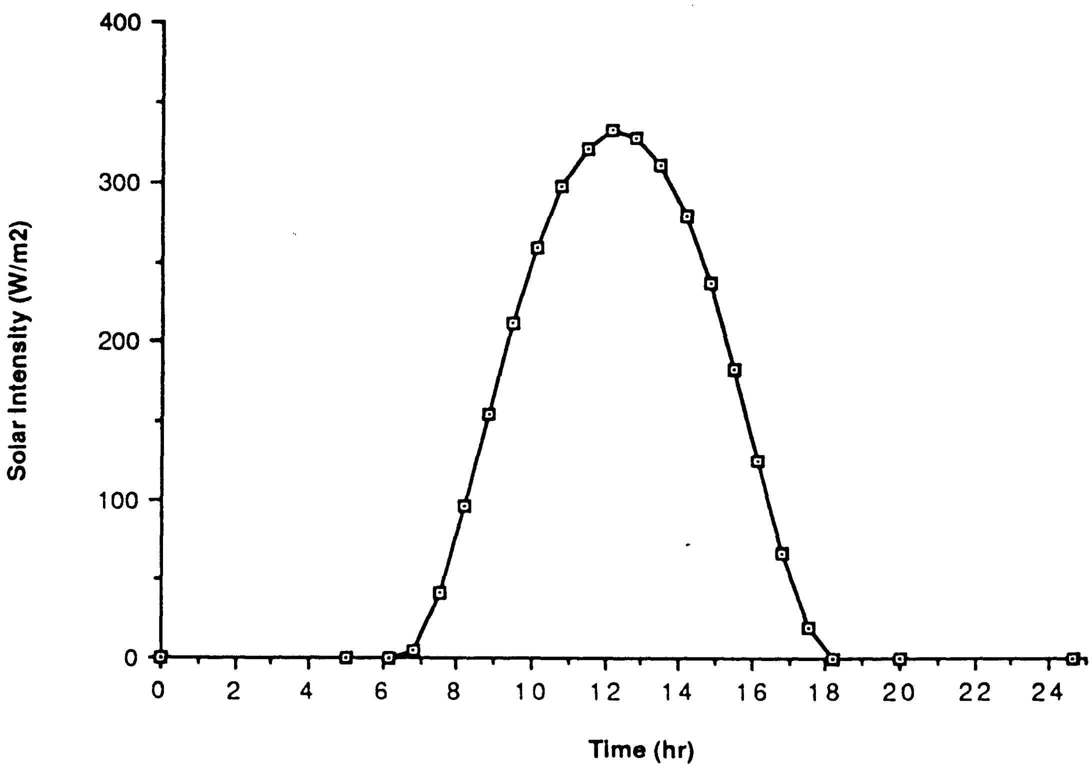

Figure 10-2: Daily Solar Flux Variation (Ls = 90°, Lat. = 0°)

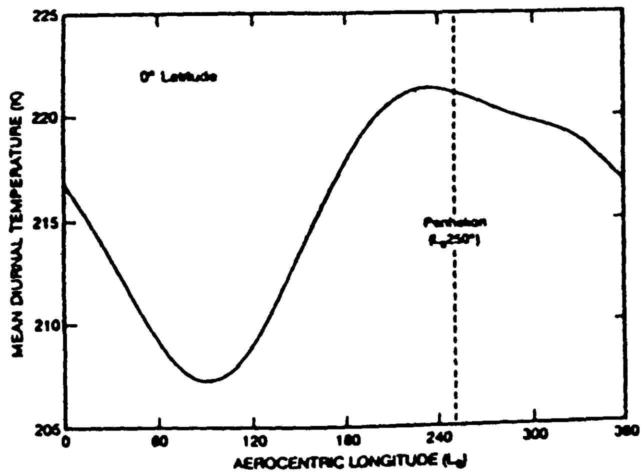

Figure 10-3: Average Diurnal Martian Surface Temperatures (Clifford, 1986)

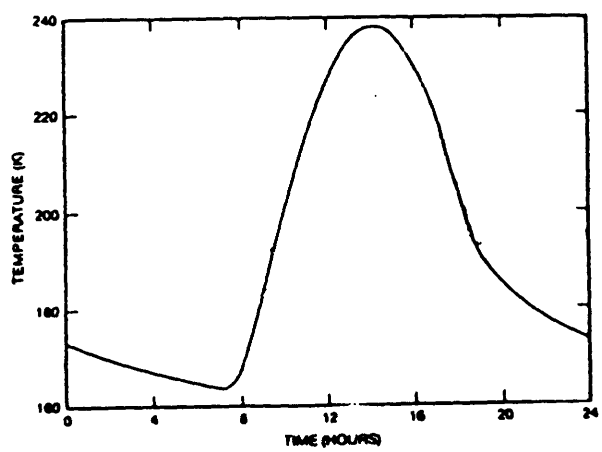

Figure 10-4: Daily Variation in Surface Temperatures (0° Latitude; Clifford, 1986)

10.3.3 Internal Heat Loads

Table 10-1 enumerates the total internal heat rates from the various subsystems. Because the propulsion system is external to the MAV cabin, it does not contribute to the thermal system heat load. Human metabolic heat output is assumed to be a constant 86 W per astronaut (Simonsen, 1988).

Table 10-1: Subsystem Heat Loads

Subsystem	Average Heat (W)	Peak Heat (W)
Power	30	30
Thermal	50	50
Propulsion	0	0
Avionics	277.21	297
Structure	0	0
LSS	88.6	108
Metabolic heat	258	258
TOTAL	703.81	743

10.3.4 Ascent Heat Loads

The main scenario stagnation point heating rate is illustrated in Figure 9-6, with the total stagnation heat load shown in Figure 9-7. Because this heat loading is only calculated for the stagnation point, a heat distribution is assumed in order to calculate the total heat over the MAV surface area (see Figure 10-5).

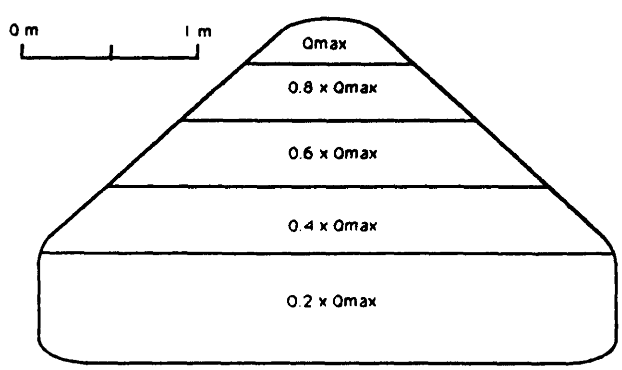

Figure 10-5: Assumed Ascent Heat Distribution

10.4 Options & Choices

There are several different options for each of the various thermal control elements. Possibilities for the insulation system, heat rejection system, thermal coatings, and thermal protection system are evaluated in the following sections.

10.4.1 Insulation System

Both the propellant system and the MAV cabin require insulation to maintain their temperatures within the specified ranges (see Section 5.0 and 6.0). There are two options for an insulation system, bulk insulation and a vacuum system.

The simplest insulation system, due to the fact that it requires no special pressure environment, utilizes bulk insulation. The two main possibilities for bulk insulation are foam insulation and non-evacuated powders (see Table 10-2).

Table 10-2: Insulations

Insulation	Density (kg/m^3)	Conductivity (W/m-°K)
Pure vacuum, $< 10^{-10}$ MPa	--	< .005
Polysterene foam	46	.026
Polyurethane foam	34	.023
Glass foam	140	.035
Non-evacuated powder (1 atm):		
Perlite	50	.026
Silica aerogel	80	.019
Fiberglass	110	.025
Evacuated powder ($1.3x10^{-8}$ MPa):		
Perlite	60-180	.080

Table 10-2: Insulations (cont.)

Insulation	Density (kg/m^3)	Conductivity (W/m-°K)
Silica aerogel	80	.0017-0021
Fiberglass	50	.0017
Opacified powder ($1.3x10^{-8}$ MPa):		
Al/Santocel	160	.00035
Cu/Santocel	180	.00035
MLI (1.3x10-9 MPa):		
Al foil and fiberglass	12-27 layers/cm	$3.5\text{-}7.0x10^{-5}$
Al foil and fiberglass	30-60 layers/cm	$1.7x10^{-5}$
Al foil and nylon net	31 layers/cm	$3.5x10^{-3}$
Al crinkled, Mylar film	35 layers/cm	$4.2x10^{-5}$

Most non-evacuated powders require some additional support structure, through which heat leaks occur. They also have substantial densities, translating to a high insulation system mass. Foam insulations, on the other hand, are lightweight and completely self-supporting. They are simply sprayed onto the spacecraft. For these reasons, foam insulation is chosen over the non-evacuated powders. The best practical foam insulation, from a thermal conductivity-to-density ratio standpoint, is polyurethane foam.

An analysis was performed to determine the foam insulation thickness required by the MAV cabin and main propellant tanks. For this first cut analysis (more detailed analysis in Section 10.5.1.3), the outer skin of the MAV was assumed to be at the average Martian surface temperature, ignoring convective and radiative heat transfer limitations. In addition, the foam thermal conductivity was assumed to be independent of temperature.

Therefore, MAV heating requirements were analyzed utilizing only the conductive heat losses through the foam:

$$Q = \frac{k\,A}{l} * (T_{int} - T_{ext})$$

k = thermal conductivity = .023 W/m-°K

A = MAV surface area = 27 m^2

l = insulation layer thickness

T_{int} = cabin temperature = 294° K

T_{ext} = average Martian temp. at 0° latitude = 214° K

Figure 10-6 illustrates the MAV cabin heating required for various polyurethane foam insulation thicknesses. Every centimeter of the foam insulation layer adds an additional 8.6 kg to the overall vehicle mass.

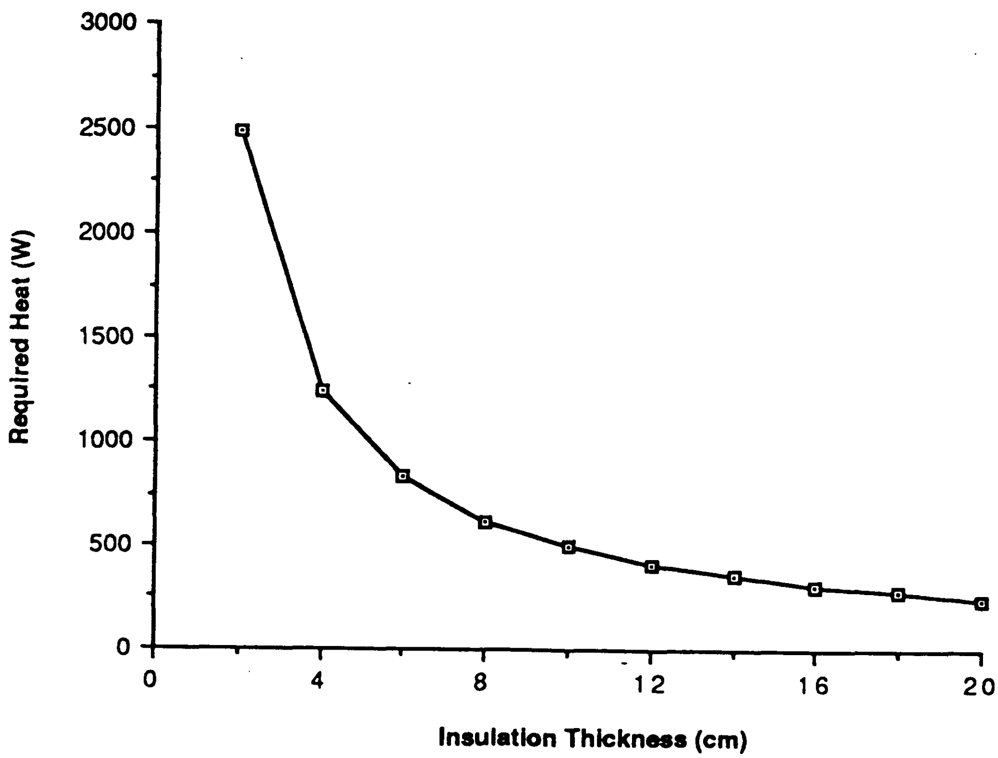

Figure 10-6: MAV Cabin Required Heat vs. Insulation Thickness

A similar simplified analysis was performed for the main propellant and oxidizer tanks. Each of the fuel tanks needs to be maintained at greater than 221 °K, and the oxidizer tanks must be kept at more than 262 °K (see Section 6.0). For the purpose of sizing the NTO insulation, the external wall temperature was assumed to be 214 °K, while the external temperature of the MMH tanks was assumed to be 200 °K (setting all temperatures above the MMH freezing point to 221 °K, see Figure 10-4). Figure 10-7 illustrates the total insulation thickness for two fuel-to-oxidizer thickness ratios, given the 57.2 m^2 of surface area.

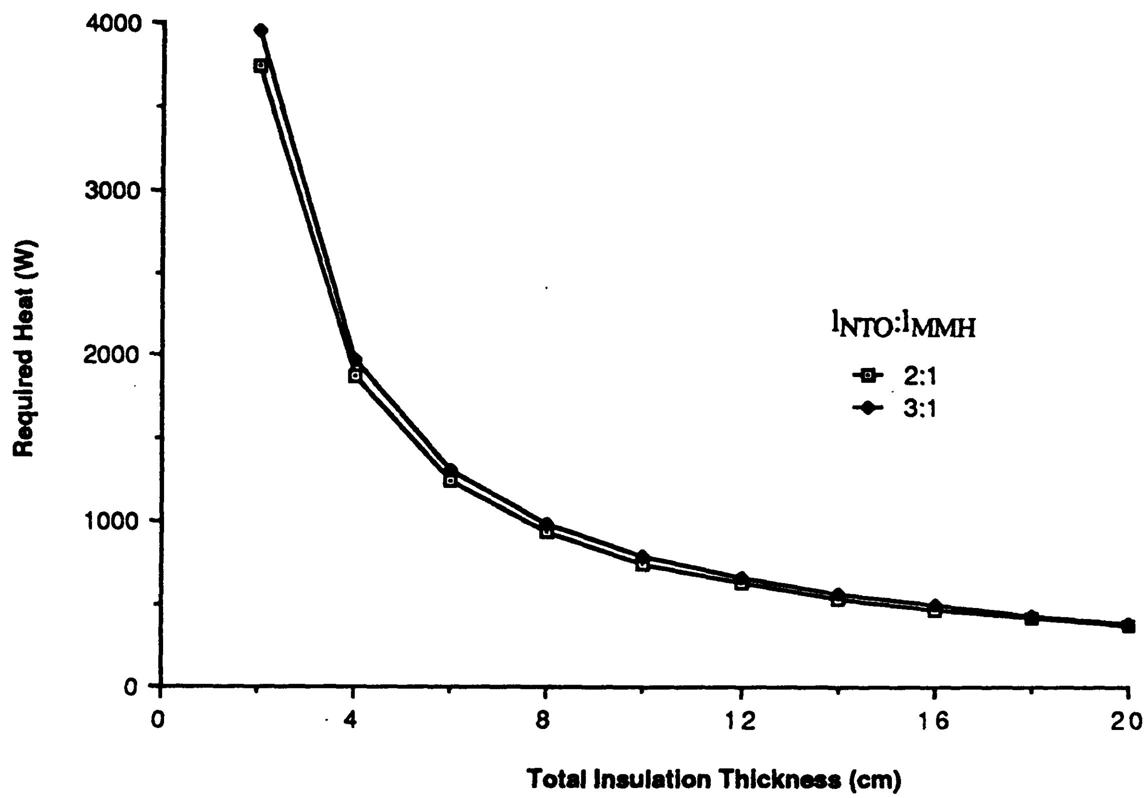

Figure 10-7: Propellant Heat Required vs. Total Insulation Thickness

For each centimeter of propellant insulation specified, an additional 9.2 kg of mass is appended to the overall MAV mass. Therefore, if bulk insulation is utilized throughout the spacecraft, about 212 kg (142 kg propellant, 70 kg cabin) of insulation is required to reduce the required heat supply to under 1 kW, which is still a fairly significant heat level. Therefore, alternative methods of insulation are evaluated to reduce both the insulation mass and heat requirements.

One possible method for reducing the insulation mass is by utilizing a trapped CO_2 convection system. This can be accomplished either by using a separate external CO_2 containment mechanism, or by simply using the space between the two hulls as an insulating CO_2 gap. Unfortunately, the calculated heat loss of over 700 kW through the aluminum stringers makes the latter approach infeasible.

By utilizing an externally located Nomex honeycomb structure filled with Martian carbon dioxide (see Figure 10-8), convection and conduction losses through the MAV cabin insulation can be calculated:

$$Q = (R_{equiv})^{-1} (T_{int} - T_{ext})$$

$$R_{equiv} = \frac{R_{Nomex} R_{CO2}}{R_{Nomex} + R_{CO2}} \text{ (assuming } h_{CO2} = 0\text{; see below)}$$

$$R_{Nomex} = \frac{l}{k_{Nomex} A_{Nomex}}$$

$$R_{CO2} = \frac{l}{k_{CO2} A_{CO2}}$$

$k_{CO2} = .0184$ W/m-°K (at Martian atmosphere conditions)

$k_{Nomex} = .623$ W/m-°K

$A_{CO2} = 0.98 * 27 \text{ m}^2$

$A_{Nomex} = 0.02 * 27 \text{ m}^2$

$l = 1.3$ cm (honeycomb thickness)

h_{CO2} = free convection constant for cylinder = 0 if $N_{Grs} < 1000$

$$N_{Grs} = s^3 \frac{\beta g}{v^2} \Delta T \text{ (Wolf, 1983)} = 1.96$$

β = reciprocal of avg. absolute temperature $= \frac{1}{254°K}$

g = Martian gravity = 3.73 m/s^2

v = CO_2 viscosity = $.00072 \text{ m}^2\text{/s}$

s = honeycomb diameter = .95 cm

$\Delta T = 80°K$

By again assuming an external wall temperature of 214°K, the calculated heat loss through a single honeycomb layer is 5.1 kW, for a honeycomb mass of 13.34 kg. If four honeycomb layers are placed in series with each other, 1280 W of heat are still required for a 53.4 kg insulation mass. Clearly this is not an advantage over the bulk insulation system, which required only 70 kg of insulation to reduce the cabin losses to 500 W.

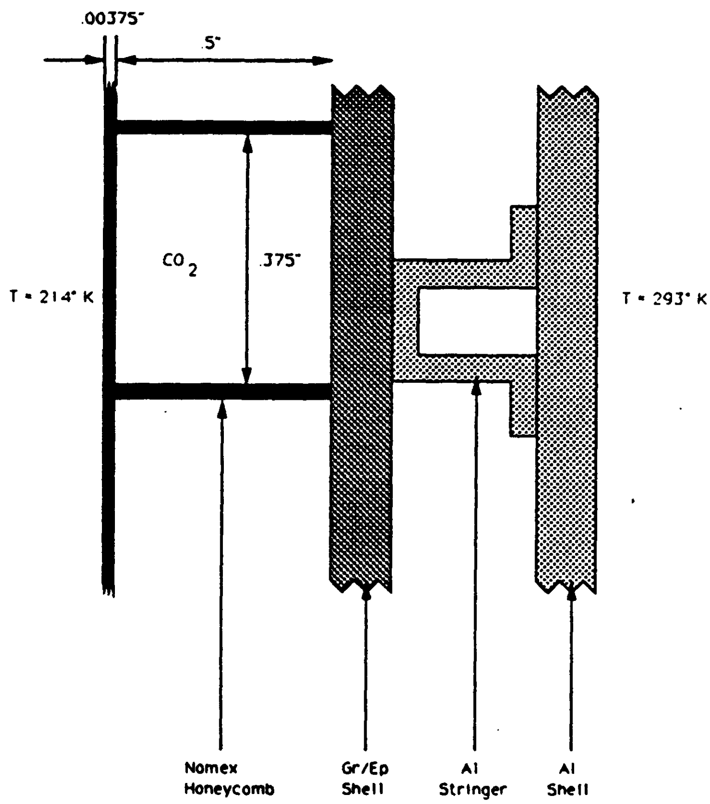

Figure 10-8: Honeycomb Insulated MAV Wall

A second alternative to bulk insulation is low pressure insulation systems. Multilayer insulations (MLI) have extremely low thermal conductivities if a vacuum environment can be maintained. If 1 centimeter of MLI is utilized for the MAV cabin, only 30 W of heat (assuming no heat loss through contact points) is required to heat the entire cabin. Similarly, if 1 cm of MLI is utilized for the propellant tanks, 58 W of heat is required.

To create a vacuum environment for the propellant tanks, a second, air-tight hull must be placed around the propellant tanks while minimizing the structural contact points between the two shells. Vacuum would then be obtained while the MAV was in space, and sustained while on the surface.

This type of system presents a spacecraft reliability problem, however. If the vacuum system were to fail, the propellants would freeze, preventing the spacecraft from ascending to orbit. To solve this leak problem, a vacuum pump needs to be installed on the MAV in order to maintain the vacuum in the gap between the two hulls. If an irrecoverable leak occurs, the MDV must provide the necessary power to maintain the propellants in the liquid state (until an abort-to-orbit or hull repair can occur).

MLI allows a substantial weight and power advantage over bulk insulation for the propellant tanks. However, for the smaller surface area of the MAV cabin, the vacuum system does provide a large mass advantage. Also, because the MAV skin must withstand ascent dynamic pressure, a MAV secondary hull requires numerous contact points through which heat leaks will occur.

Therefore, to avoid further reliability problems, bulk insulation is baselined for the MAV cabin. To lower the total required heat, the propellant tanks utilize a MLI-containing vacuum system, with a MDV heating and abort-to-orbit backup. For the small surface area OMS propellant tanks, bulk insulation is specified.

10.4.2 Heat Collection and Rejection System

During descent and ascent, the MAV's internally generated heat load of 704 W (average) must be rejected to maintain the spacecraft within the required temperature range. This requires both a heat collection and rejection system. There are two principle options for collecting internally generated heat, a coolant loop and heat pipes.

Coolant loops have been utilized on previous manned space flights, with Apollo, Gemini, and Mercury all employing water-based coolant systems. In these systems, a coolant fluid, which is circulated throughout the spacecraft, stores the heat until the fluid reaches a radiator. At that point, the fluid-stored heat is transferred to the radiator and radiated to space.

Although these systems are well-tested, they are generally quite heavy, containing coolant plumbing, pump systems, heat exchangers, condensers, and coolant fluids. They are also active systems, which could present a reliability problem for the two-year shelf-life MAV thermal system.

An alternate heat rejection method employs a passive, variable-conductance heat pipe system. A heat pipe is a closed pipe containing a small quantity of working fluid (see Figure 10-9). A capillary wick creates the pumping force required to move the fluid from the heat source, where the fluid is evaporated, to the heat sink, where it condenses. By introducing a fixed amount of non-condensible (at heat pipe operating temperatures) gas into the heat pipe to vary the condenser effective length, the source can be passively maintained at a constant temperature.

In past applications, ammonia, freon, and water have been used for working fluids. For the MAV application, water is baselined to avoid toxicity problems.

One disadvantage of this system is that the heat source must be physically under the heat sink so that the capillary action occurs even in a gravity environment. This means that the MAV radiator must be located on the upper cone of the external shell. Also, past heat pipe systems have typically dissipated around 50 W of heat or less. A large number of these pipes are required to reject the entire 700 W heat load.

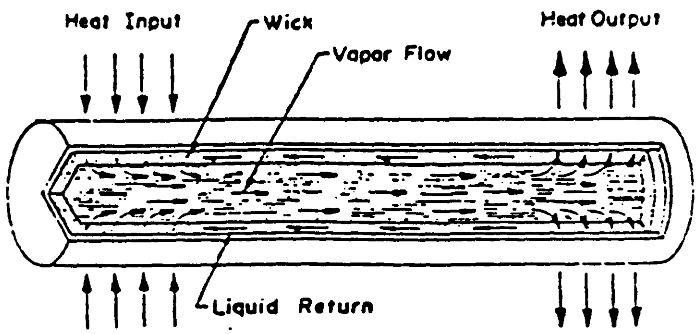

Figure 10-9: Heat Pipe Schematic

To reject the collected heat, the condenser ends of the heat pipes dump into a radiator. The radiators are mounted through the insulation to the upper conic shell of the MAV.

In summary, heat pipes are utilized to collect heat from the MAV internal sources. These heat pipes collect the heat at the source, and transfer it to a radiator system located on the external MAV surface.

10.4.3 Coatings

Table 10-2 shows solar absorptivities and infrared emissivities for possible MAV surfaces. Determination of the actual coatings employed is discussed in the the various thermal element point designs.

10.4.4 Thermal Protection System (TPS)

There are several options for protecting against or rejecting the ascent heat load, including: utilizing the already present heat pipe and radiator system, having a separate water coolant loop, and coating the outside of the MAV with an ablative material. Allowing the thermal

Table 10-2: Absorptivities and Emissivities of Various Materials

Material	Absorptivity (Solar)	Emissivity (IR)
White paint	.20	.85
Black paint	.94	.94
Silvered teflon	.08	.72
Aluminum paint	.66	.20
Aluminum	.44	.08
Gold	.20	.03
Titanium	.60	.20
Beryllium	.58	.08
Steel	.85	.36

mass of the spacecraft to absorb the 4.1 kW peak load is not an option, since the ascent heating has already heat-saturated the MAV structure. The internal peak heat load of 4.1 kW is the result of over 100 kW/m^2 of ascent heating.

Because the ascent heat rates create an internal heat load 5 times the normal MAV heat loads (see Section 10.5.2.3), the radiator would require significantly more surface area to reject this heat. The heat pipes would also need to increase in number (or total diameter) by a factor of five. Therefore, a separate thermal system is specified to dump ascent heating loads.

One possibility is a water evaporation system. Water pipes run through the MAV insulation, storing the ascent heat by water evaporation. The water vapor is then dumped into space. Evaporation of 1 kg of water absorbs 2.43 MJ of heat.

The main problem with this heat rejection system is that it adds a significant mass quantity to the MAV. A separate system, with a water pump, a water vapor rejection point on the spacecraft, and an additional plumbing system is required. In addition to this, there may be a problem in preventing the heat from circumventing the water pipes (not boiling the water), and heating the internal cabin. Furthermore, this method would not rid the spacecraft of the 800 W internal load.

An ablative coating on the external MAV surface is a simple way to absorb the excess heat from ascent. A typical ablative material absorbs 4.19 MJ/kg, which is nearly twice the performance of the water system. Also, the ablation method requires no additional system elements. Therefore, a thin ablative coating is baselined to absorb the excess MAV ascent heating.

One problem with either method of heat rejection is that the radiator, which is not covered with the ablator, will exceed the heat pipe operating temperature, preventing heat rejection. To solve this problem, a simple water evaporation loop is utilized in the radiator. This loop cools the radiator surface below 294 °K, allowing heat radiation of the internal loads during ascent.

10.5 Point Design

The MAV thermal system is designed to allow the MAV abort-to-orbit capability at all times on the Martian surface. Cabin temperature is sustained at a human compatible level on the surface, as well as during the ascent and orbit stay, and the propellants are maintained in the liquid state.

10.5.1 System Description

Figure 10-10 illustrates the MAV thermal system. Bulk insulation on the MAV cabin and a MLI vacuum system on the propulsion tanks are utilized to maintain the Martian surface thermal power requirement under 500 W. Heat pipes are used during the mission to collect the generated internal heat, and the radiator is designed to reject this heat load. An ablative material coats the outside of the spacecraft to absorb the ascent heat loads.

10.5.1.1 Heat Pipes

A total of 36 variable conductance heat pipes, each with a 50 W heat transfer capability, are utilized to reject the internally generated and solar absorbed heat load of 800 W (see Section 10.5.1.3). Twelve heat pipes (including 6 for redundancy) are employed to collect the avionics heat load of 300 W (see Figure 10-10), while 12 heat pipes are attached to each of the air circulation vent heat exchangers to remove the remaining 500 W of heat.

To remove the total internal heat load, only one of the air circulation systems must be operating. In addition to the 10 required heat pipes attached to this system, 2 more heat pipes are provided for redundancy. Redundancy is also provided for each of the avionics heat pipe clusters.

Water is utilized as a working fluid, evaporating at the heat source end of the heat pipe, and condensing at the heat sink. Each cluster of heat pipes spread out through the radiator, allowing heat transfer between the condenser end of the pipe and the radiative surface.

10.5.1.2 Ascent Heat Rejection System

To reject the ascent heat loads, 5.6 kg (with a 1.5 factor of safety) of ablative material is sprayed onto the MAV external skin. This amount of material absorbs the non-radiator ascent heat load of 15.6 MJ (see Section 10.5.2.3). An additional 5.4 kg (FOS = 1.5) of water is required to maintain the radiator at 290 °K (see Section 10.5.1.5), so that the internal heat loads can be rejected during ascent. Section 10.5.1.5 examines the water coolant system.

10.5.1.3 Thermal Coatings

The thermal coatings on the MAV cabin, radiator, and tanks must be specified in order to determine the spacecraft heat loads. In general, it is beneficial for the MAV to have a highly

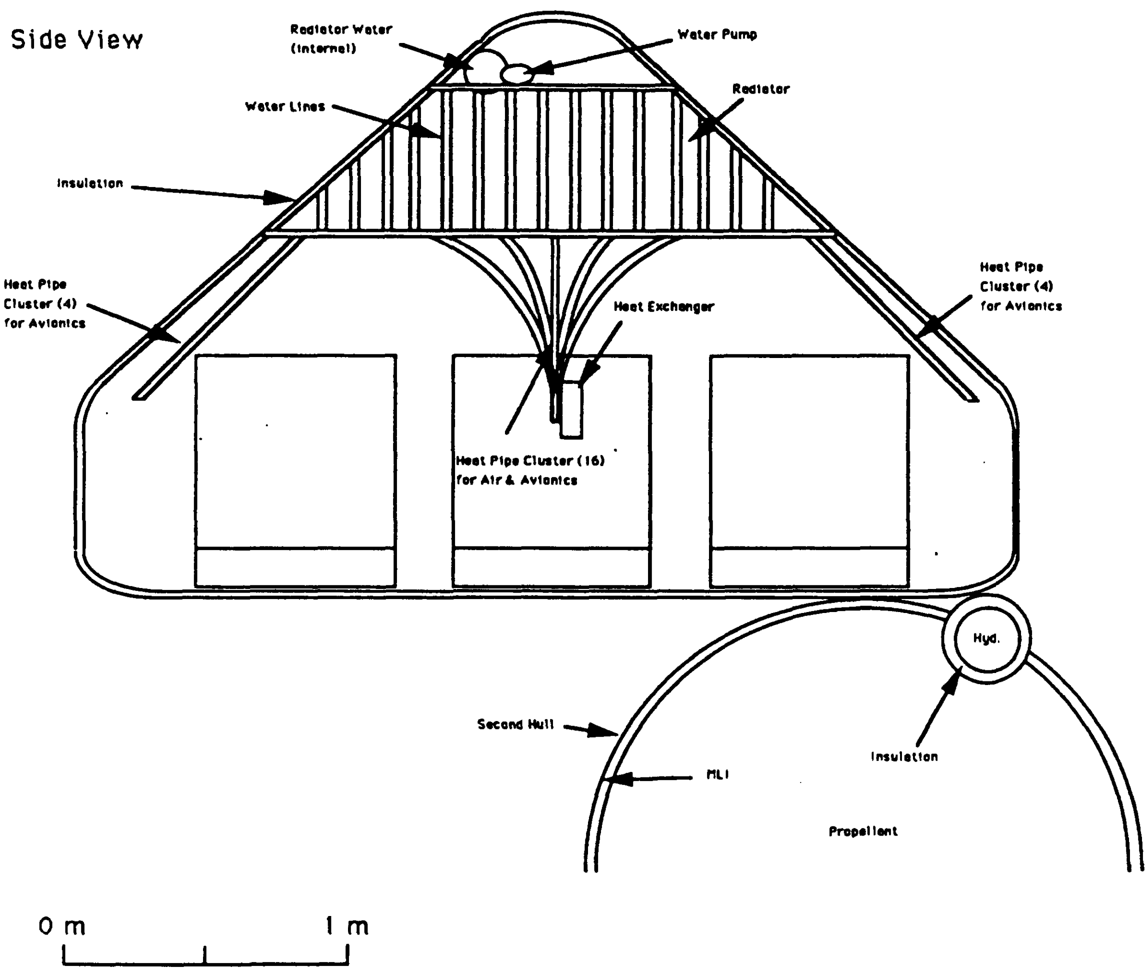

Figure 10-10: Thermal System Layout

solar absorptive coating while the MAV is on the Martian surface. The solar energy assists in reducing the overall MAV heating requirements.

To determine the required emissivity of the MAV outer surface, the time during Martian orbit solar eclipse is examined. This is the point where the internal spacecraft heat must sustain the MAV at 294 °K:

$$Q_{int} = \frac{k\,A}{l} * (T_{int} - T_{ext})$$

Q_{int} = internal heat = 704 W

k = MAV skin thermal conductivity const. = 0.0216 (see Section 10.5.1.4)

A = MAV surface area = 27.2 m^2

l = insulation thickness = 0.0762 m (see Section 10.5.1.4)

T_{int} = internal temperature = 294 °K

T_{ext} = external temperature

If a thermal balance is to be maintained in this case, the external surface temperature must be sustained at a temperature of 203 °K. Utilizing this temperature, the infrared emissivity can be calculated:

$$\varepsilon\, \sigma\, T_{ext}^4\, A = Q_{int}$$

ε = infrared emissivity

σ = Stefan-Boltzmann constant = 5.667×10^{-8} W/m^2-°K^4

Therefore, the MAV cabin infrared emissivity must be less than 0.269, or additional heat sources must be supplied on the MAV. To meet this emissivity specification, and to maintain the highest possible solar absorptivity, aluminum paint is baselined for the MAV skin. This paint has an emissivity of 0.2 and an absorptivity of 0.66. The aluminum paint coats the entire surface of the MAV, and is mixed in with the ablative material to sustain the needed surface emissivity after Martian ascent. Aluminum paint is also utilized on the hydrazine propellant tanks.

The main propellant tanks are sustained within an acceptable temperature range at Earth orbit, Mars orbit, and Mars surface by utilizing the optical properties of the aluminum coated tanks (see Table 10-2). No additional thermal coatings are required.

The MAV radiator is painted white to maintain the required high IR emissivity, while having a low solar absorptivity. This allows the MAV heat to be rejected, while increasing the radiator efficiency through reduced solar heat input.

10.5.1.4 Insulation

Polyurethane foam insulation is specified for both the MAV cabin and the hydrazine tanks. A 0.0762 m (3 in) layer of this lightweight insulation is sprayed onto the MAV cabin surface, reducing the total MAV cabin required heating to an annual average of 438 W (see Section 10.5.5.2). With a density of 34 kg/m^3, this amounts to a MAV cabin insulation mass of 70.5 kg.

A 0.1524 m (6 in) layer of insulation is also sprayed onto the hydrazine tanks, reducing the heat requirements to 25 W, or 6.25 W per tank (see Section 10.5.5.2).

Figures 10-11 and 10-12 illustrate the thermal properties of the polyurethane insulation BX-250.

10.5.1.5 Radiator

The MAV radiator is required to reject a maximum of 800 W of heat. To size this radiator, a radiator temperature of 290 °K (must be under 21 °C internal temperature) is specified in the following energy balance:

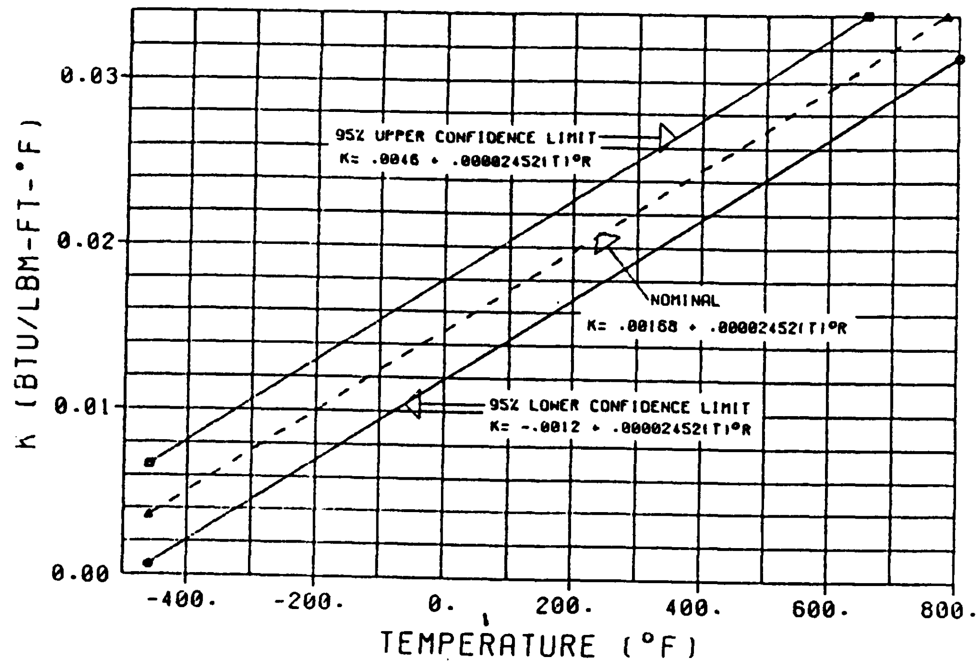

Figure 10-11: Polyurethane BX-250 Thermal Conductivity vs. Temperature

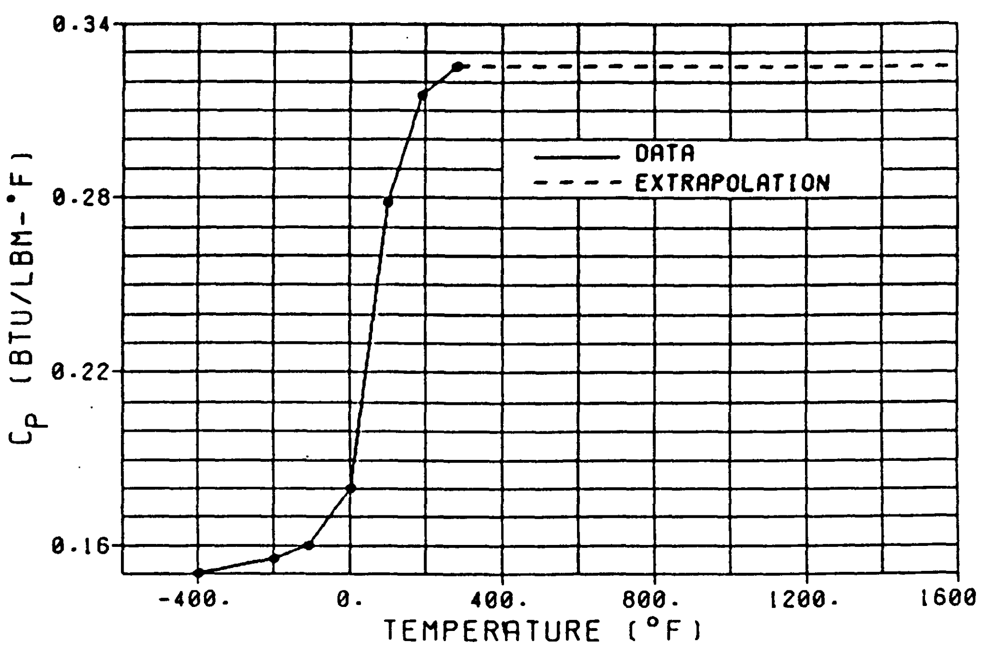

Figure 10-12: Polyurethane BX-250 Specific Heat vs. Temperature

$$\alpha_s S_m F A_r + Q_{max} = \varepsilon_r \sigma T_r^4 A_r$$

α_s = solar absorptivity = 0.20

S_m = solar intensity in Mars orbit = 583 W/m^2

F = worst-case radiator area projection factor = 0.707 (sin 45°)

A_r = radiator area

Q_{max} = maximum heat rejected = 800 W

ε_r = radiator IR emissivity = 0.85

T_r = maximum radiator operating temperature = 285 °K

Therefore a minimum of 3.1 m^2 of radiator area is required. For a factor of safety, and to take into account unused radiative area, a radiator area of 3.9 m^2 is specified.

The radiator itself is constructed of aluminum 6061-T6, which permits rapid heat transfer (thermal conductivity = 166 W/m-°K). The condenser portions of the heat pipes are threaded through the radiator, rejecting their heat to the aluminum surface. The radiator is located on the upper portion of the MAV cone (see Figure 10-10).

To maintain the MAV at the 290 °K operating temperature during the Martian ascent, a water evaporation loop is utilized (see Figure 10-13). The water is pumped through pipes in the radiator, reducing the radiator surface temperature by evaporating. The water vapor is then separated from the liquid water and discarded. A total of 8.7 MJ of heat is rejected from the radiator by 5.4 kg (FOS = 1.5) of evaporated water.

10.5.1.6 Main Propellant Vacuum System

To sustain the MMH/NTO propellants in the liquid phase, a multilayer insulation (MLI)-filled vacuum system is utilized. A 1 centimeter gap is filled with 15 layers of fiberglass/aluminum coated mylar foil MLI (0.23 kg/m^2), reducing the conductive loss in space to 58 W of total

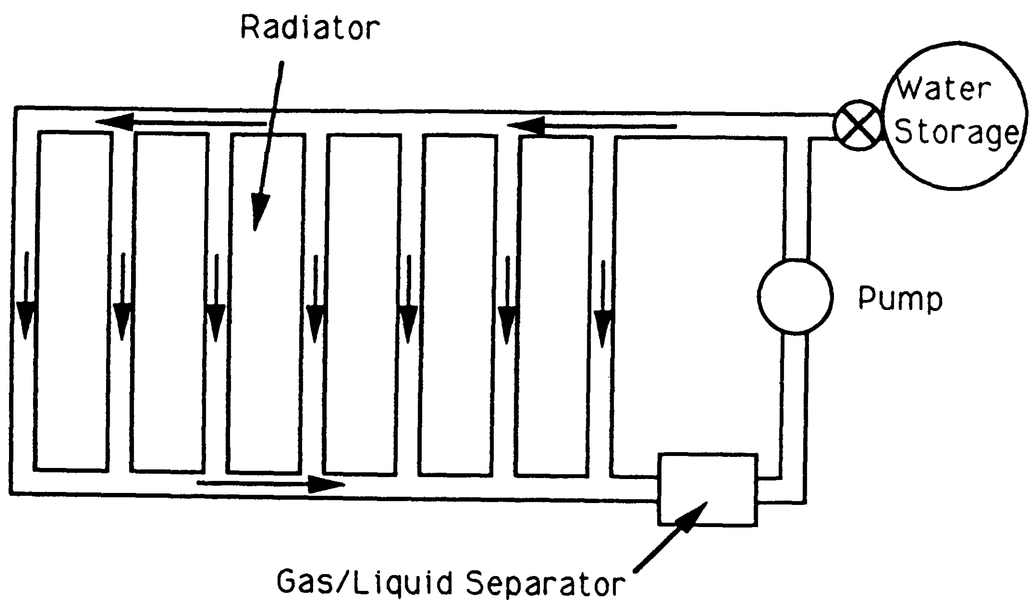

Figure 10-13: Radiator Water Evaporation Loop

heat, 17 W for each of the NTO tanks, and 12 W for each MMH tank (see Section 10.5.2.2). Slightly less heat is required on the Martian surface.

To create the vacuum environment, a 0.000762 m (0.03 in) graphite/epoxy layer, coated with a thin layer of aluminum, is employed as the second shell surrounding the propellant tanks. The second hull is sized to reduce the possibility of accidental puncture, since it actually requires only about $2x10^{-5}$ centimeters of Gr/Ep to take the loads from the 9 kPa Martian environment. The aluminum coating is used to reduce the vacuum leak rate.

Because the second structure can also readily withstand Earth pressures, the vacuum chamber can be evacuated either in space or before launch to Earth orbit.

In case of vacuum system leaks, a Balzer turbo-molecular pump is provided. This pump is efficient to a flow rate of up to 600 l/s, which will evacuate the entire propellant vacuum system in 0.93 seconds. Since this pump is only required on the Martian surface, it can be operated from and located on the MDV habitation module.

10.5.1.7 Window

To reduce heat losses through the window, a still air gap is maintained between two panes of glass separated by a 2 centimeter gap. This reduces the space heat losses to about 31 W, and the Martian surface heat losses to about 8.6 W. The external MAV insulation overlaps the outside of the window pane, reducing heat losses through the window support structure.

10.5.2 System Analysis

The MAV thermal system was analyzed on the HP9000 computer using SINDA (NAS9-17448, 1987), a thermal resistance-based analysis tool. The program operates on a user-defined input file, which is listed in Appendix B. It determines required internal heat values by analyzing convective, conductive, and radiative heat transfer.

10.5.2.1 Space

In order to analyze the MAV cabin and propellant tanks, a model of the MAV using thermal resistances and thermal nodes must be defined. Figure 10-14 illustrates this resistor network, with the resistance values appearing in Appendix B. The resistance values are defined as follows:

$$R_{conductance} = \frac{k\,A}{l} \qquad Q = R * (T_{node\ a} - T_{node\ b})$$

$$R_{convection} = h\,A \qquad Q = R * (T_{node\ a} - T_{node\ b})$$

$$R_{radiative} = \varepsilon\,A\,\sigma \qquad Q = R * (T_{node\ a}^4 - T_{node\ b}^4)$$

Each of the thermal nodes is defined by the product of mass and heat capacitance, which is also illustrated in Appendix B.

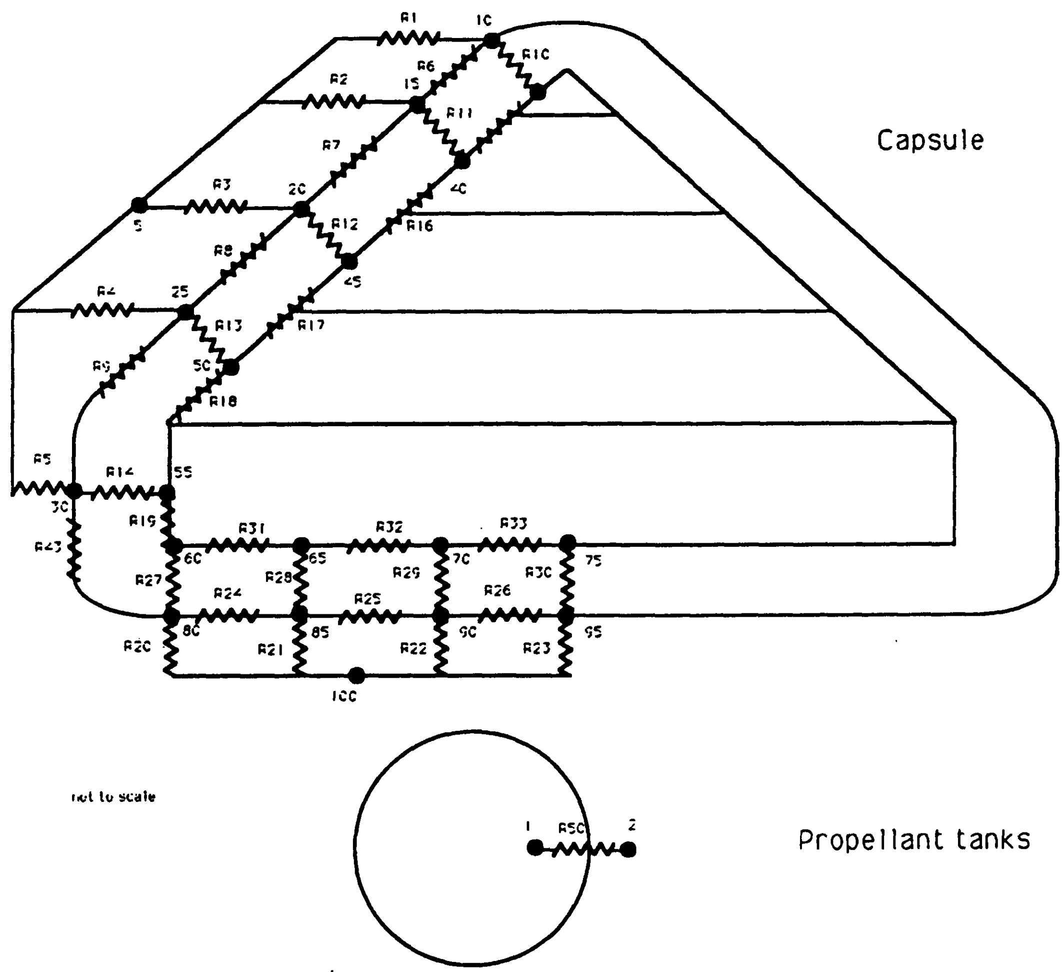

Figure 10-14: Thermal Resistor Network

By utilizing the insulation data in Figures 10-11 and 10-12, and the thermal coating emissivities and absorptivities, the required heats for the MAV cabin and propellant tanks in space were determined.

During the journey to Mars, the MAV cabin requires 608 W of thermal energy to maintain it at room temperature. This heat is expected to be provided by the MTV by virtue of the two vehicles shared air flow (see Section 7.0). While the spacecraft is operating in Martian orbit in full frontal solar view, 800 W of heat must be rejected from the MAV. If the MAV is eclipsed by Mars, only 96 W of heat must be rejected.

The propellant tanks require 93 W of total heat while in space. Assuming a MLI thermal conductivity of 7×10^{-5} W/m-°K, 17 W of heat is needed to maintain each of the NTO tanks at a temperature suitable for the liquid oxidizer. An additional 12 W of energy is required to sustain each MMH tank and another 6.25 W is required for each hydrazine tanks.

10.5.2.2 Martian Surface

To determine the surface heating requirements, the thermal forced and free convection constants for the MAV cabin and the propellant tanks can be estimated (Edwards, 1979 and Wolf, 1983):

$$h_{total} = h_{free} + h_{forced} \text{ (worst-case)}$$

$$h_{free} \text{ (sphere)} = \frac{k}{D} * (2 + 0.6\,(Gr_D\,Pr)^{0.25}) = 0.25 \text{ W/m}^2\text{-°K}$$

$$D = \text{sphere diameter} = 7 \text{ m}$$

$$Gr_D = D^3 \frac{\beta g}{\nu^2} \Delta T \text{ (as in Section 10.4.1)}$$

$$Pr = 0.71$$

$$h_{forced} \text{ (sphere)} = \frac{k}{D} * (2 + 0.3 Re_D^{0.6} Pr^{0.33}) = 0.596 \text{ W/m}^2\text{-°K}$$

$$Re_D = \frac{D\,u}{\nu}$$

$$u = \text{Martian air velocity} = 3 \text{ m/s}$$

$$h_{free} \text{ (vertical cylinder)} = \frac{k}{D} * 0.67 Ra_L^{0.25} (1 + (\frac{0.492}{Npr})^{0.563})^{-0.444}$$

$$= 0.31 \text{ W/m}^2\text{-°K}$$

$$L = \text{MAV height} = 2 \text{ m}$$

$$Ra_L = Gr_L\,Pr$$

$$h_{forced}\ (\text{vert. cyl.}) = \frac{k}{D} * \left(0.3 + \frac{0.6Re_D^{0.5}Pr^{0.33}}{(1+(0.4/Pr)^{0.667})^{0.25}}\right)$$

$$= 0.39\ W/m^2\text{-}^\circ K$$

$$D = \text{average MAV diameter} = 2.1\ m$$

Therefore, the total convection constant for the propellant tanks is 0.846 W/m^2-°K, and the estimated MAV cabin convection constant (modeling the MAV as a vertical cylinder) is 0.80 W/m^2-°K.

Utilizing these quantities, the total required MAV cabin heat is calculated. Figure 10-15 illustrates the daily variation in cabin required heat during the expected warmest and coldest days at the Martian surface site. From these data, the MAV cabin average heat requirement of 438 W is calculated.

Two separate propellant tank heat calculations were performed, one with bulk insulation protected tanks, and the other with MLI insulated tanks. For the foam insulation case, 0.0762 m (3 in) of insulation is placed on the oxidizer tanks, while 0.0381 m (1.5 in) of foam is sprayed on the propellant tanks.

Figure 10-16 shows the propellant required heat for the same temperature extremes as the MAV cabin. This results in an average of 377 W of required thermal power throughout the Martian year for the bulk insulation system.

For the MLI insulated system, 13.6 W of total power are required, including 4.7 W of NTO heat and 2.1 W of MMH heat. Each hydrazine tank requires an additional 3.1 W of heat. These requirements are all less than the space heat requirements, and thus don't drive the power system design.

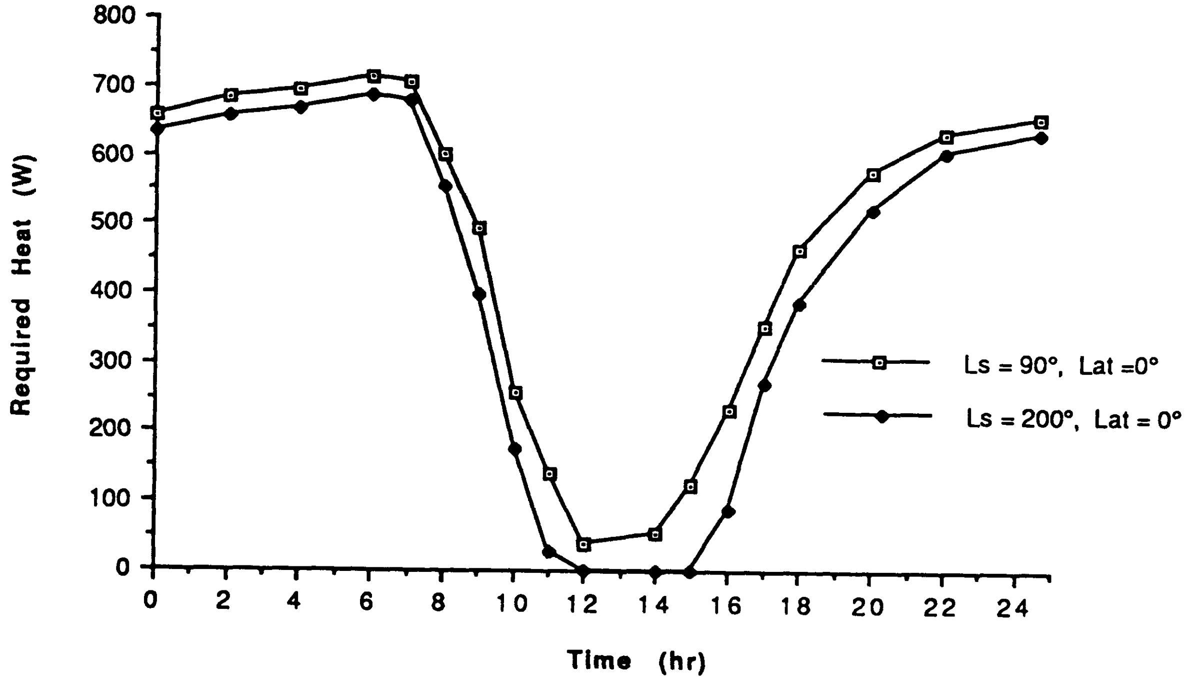

Figure 10-15: Required Heat in MAV Capsule

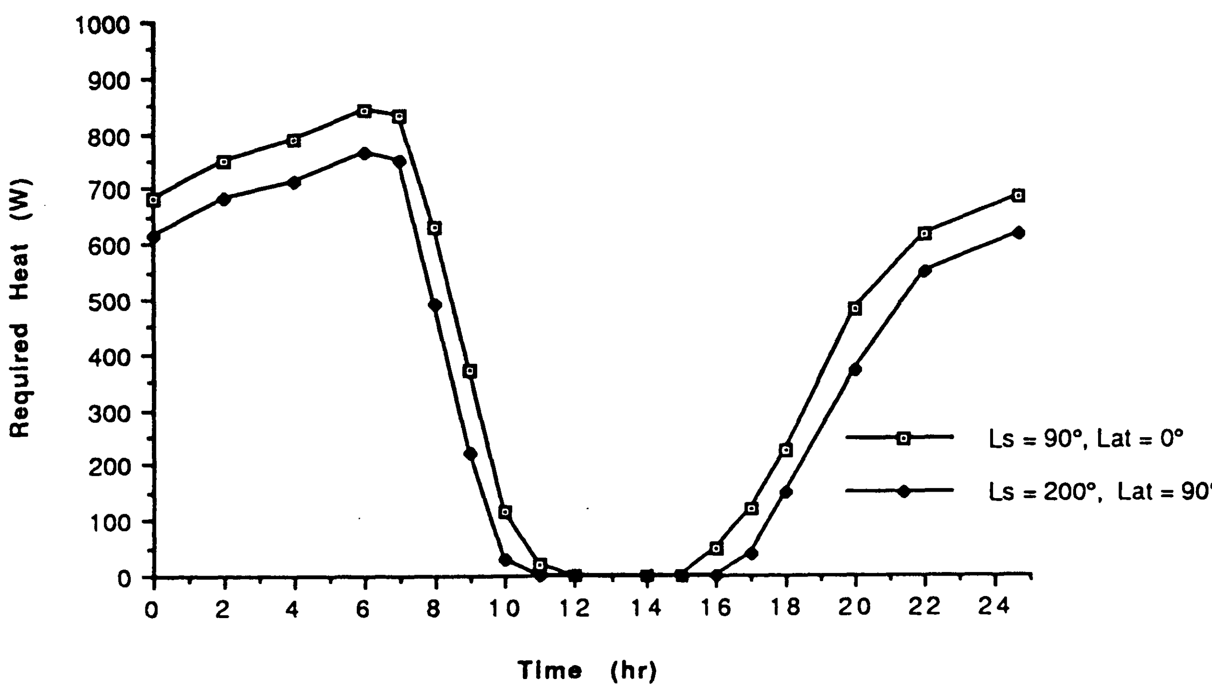

Figure 10-16: Required Heat for Foam Insulated Propellant Tanks

10.5.2.3 Martian Ascent

SINDA was also utilized to determine the internal heat loads from the ascent heating described in Figure 9-7. By assuming the heat distribution to be as in Figure 10-5, the internal heat loads are calculated and shown in Figure 10-17 utilizing temperature and heat load profiles from Section 9.0.

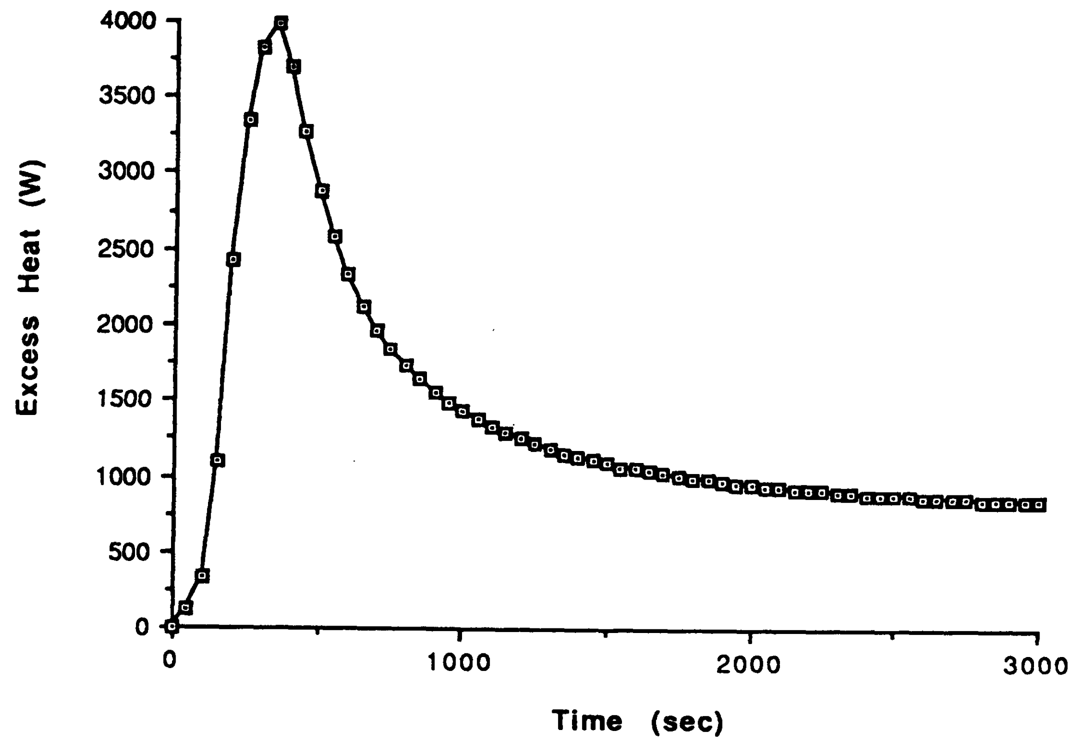

Figure 10-17: Required Ascent Heat Rejection

The total external energy that must be absorbed by the ablator and radiator water system is 24.3 MJ. The radiator must absorb 8.7 MJ, while the ablator absorbs the remaining 15.6 MJ of heat.

10.5.3 Summary

Table 10-3 summarizes the MAV thermal control system mass, power, and volume requirements. The vacuum pumps are not included in the totals, since they are assumed to be left behind during ascent. Their power requirements are included, however, since the pumps may may be operating during a MDV system failure. Heat pipes are sized for 0.3 kg/m of length.

Table 10-3: Thermal Control System Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
heat pipes	36	16.3	0	0.016
working fluid	36	8.0	0	0.008
heat exchanger	2	10.0	1785.0	0.010
radiator	1	20.0	0	0.007
radiator attachments	8	4.0	0	0.003
water		5.4	0	0.005
water storage	1	1.3	0	0.001
water pipes		8.0	0	0.010
pumps	2	5.0	5.0	0.005
ablative material		5.6	0	0.003
capsule insulation		70.5	0	2.070
hyd. insulation		9.5	0	0.278
MLI		13.0	0	0.560
Gr/Ep shell	4	64.0	0	0.043
vacuum pump	(2)	(20.0)	441.0	(0.040)
TOTAL	88	240.6	2231.0	3.018

10.5.4 Reliability

Required system reliability of 0.995 is obtained through heat pipe redundancy. Adequate heat rejection is obtained with only half of the avionics heat pipes, and with only one of the circulation systems operating. This, combined with the high reliability of the passive heat pipes, is expected to sustain the required reliability.

10.6 Conclusions and Recommendations

Even with the MLI propellant tanks, the thermal power requirements for the MAV are fairly significant. One possible way to reduce these requirements would be to let the MAV capsule and propellant tanks cool to Martian surface temperatures. Before launch occurs, the MAV cabin would then be heated with MDV power, which causes a delay in an abort-to-orbit scenario.

The problem with this method of power reduction is that the propellants would need to melt back to the liquid state. Chunks of solid propellant could cause plumbing problems and possible engine failure. If a safe method of assuring that all of the propellant is in the liquid state can be devised, this solution would offer great overall power savings.

Further research and development must also take place to determine the feasibility of utilizing a vacuum system on the Martian surface.

11.0 Docking System

11.1 Introduction

Once the MAV accomplishes rendezvous with the MOV in Martian orbit, the surface astronauts must transfer back to the main spacecraft. A docking module is provided for shirt-sleeve transfer of the MAV crew.

11.2 Requirements

The MAV docking system must be:

- compatible with MOV
- easily accessible by all crew members
- able to withstand same loads as stringers
- minimize impact to reduce structural weight
- allow passage of fully suited astronaut with safety margin
- allow shirt-sleeve passage
- reliability = .995
- shelf-life = 2 years

Other docking system requirements include:

- approach velocity < 0.1 m/s
- capability for a single crewman to execute the docking operation
- direct visibility of the target
- capability to perform docking maneuver in pressure suit

11.3 Options

Figure 11-1 illustrates possible docking mechanisms for this MAV/MOV crew transfer.

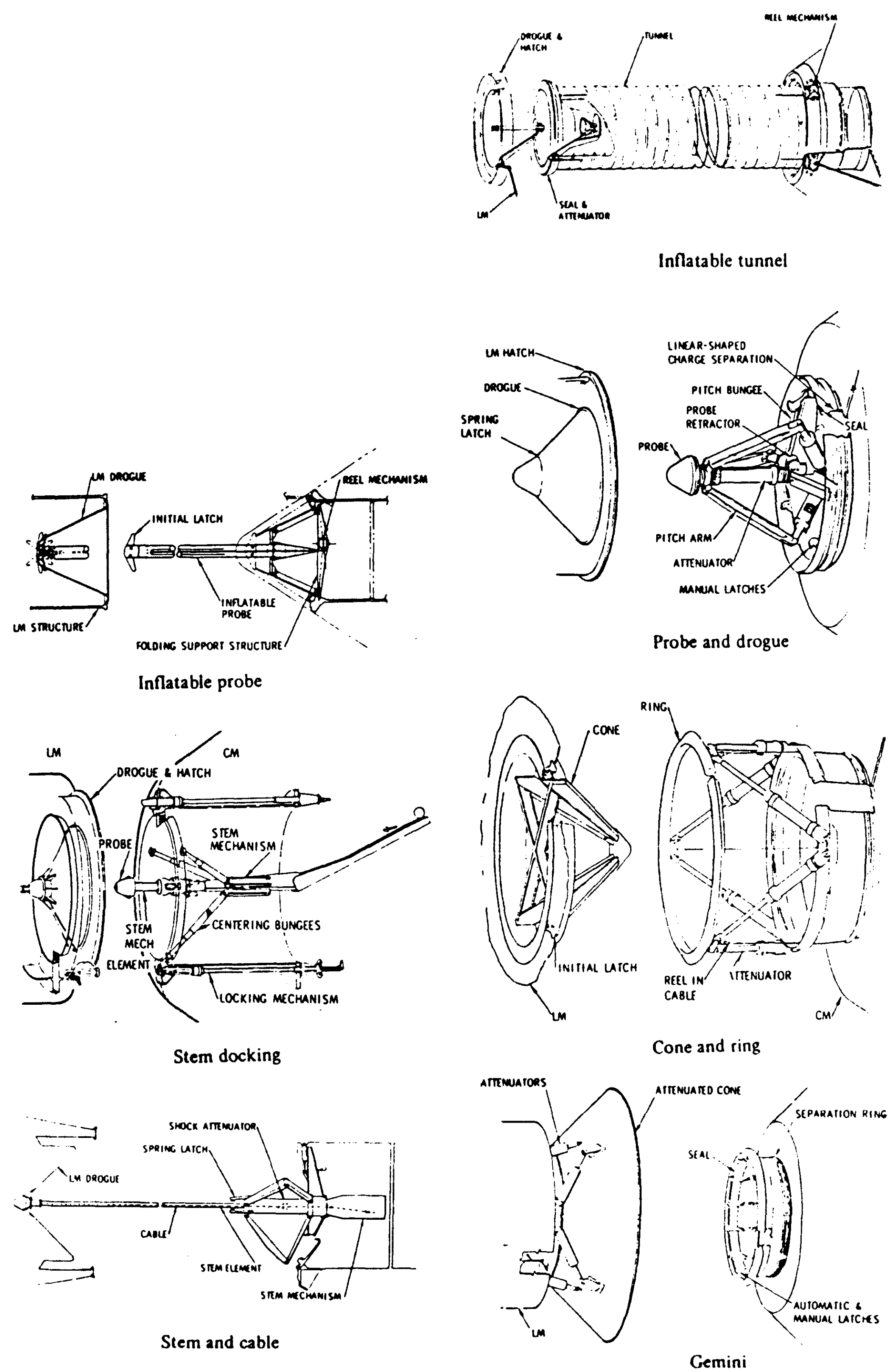

Figure 11-1: Possible Docking Mechanisms (Bloom, 1969)

11.4 Point Design

For 5 out of the 7 possible docking mechanisms, the only required MAV system is a lightweight inverted cone structure . The docking system associated with the MAV therefore has a very low mass. Because of its thorough testing and design, the Apollo probe and drogue docking method is baselined for this mission.

11.4.1 System Description

In the probe and drogue method of docking, the MAV drogue impacts with the MOV docking probe. The latches on the probe then grab the MAV and an electrical reeling mechanism tows in the spacecraft. The docking mechanism is manually removed, permitting shirt-sleeve astronaut passage.

11.4.1.1 Probe

The docking probe located on the MOV consists of an attenuator, a spring latch, and an electrical retractor (see Figure 11-2). The attenuator is filled with high pressure gas, which is compressed upon MAV impact.

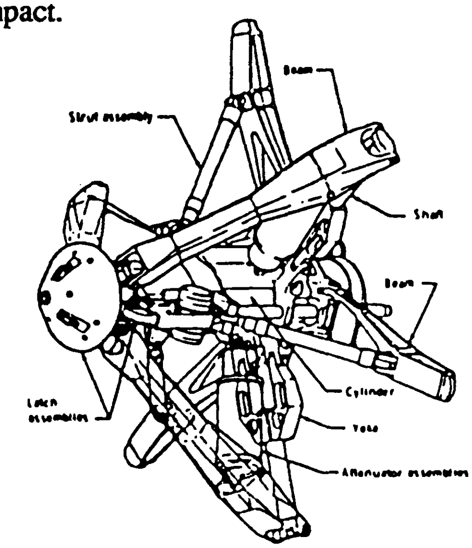

Figure 11-2: Apollo Docking Probe (Clark, M., 1987)

11.4.1.2 Drogue

The drogue located on the front of the MAV is simply an aluminum cone and a seal, and is expected to have a mass of approximately 15 kg.

To install the probe and drogue system, the surface astronauts first enter the MAV. The probe and drogue system is then installed in the intermediate tunnel. After the MAV detaches from the MOV, the nose cone (which is required for Mars ascent heating) swings into place over the drogue.

11.4.2 Docking Scenario

11.4.2.1 Main Scenario

In the main mission scenario, the crew transfer occurs in a shirt-sleeve environment. The nose cone is first removed from the tip of the MAV. After the impact between the MAV and MOV occurs, the probe reels in the MAV, sealing the two spacecraft together. The surface astronauts (or the MOV astronauts, if necessary) then remove the probe and drogue structure manually, opening up the tunnel from the MAV for crew transfer.

11.4.2.2 Abort Scenario

If the docking system fails, an EVA backup is provided. The MAV astronauts exit from the MAV rear hatch in their pressure suits, transferring to the MAV through an alternate MOV entrance.

12.0 Other Systems

12.1 Payload

100 kg of samples are returned by the MAV from the Martian surface. Figure 12-1 illustrates the sample locations, as well as the layout of the other elements described in this section.

12.1.1 Requirements

Martian sample requirements:

- 100 kg samples
- samples must be kept below Martian temperatures on ascent
- human acceleration levels (see Section 5.0) are acceptable for samples
- no volatile or toxic cargo unless crew is protected

12.2.2 Point Design

12.2.2.1 Sample Container Design

The sealed sample containers design is based on MRSR cannisters (see Figure 12-2). Both atmospheric and soil samples are stored in aluminum tubes, which are then sorted and labeled according to sample acquisition locations. Samples are stored in the MAV containers throughout the surface stay in case of emergency abort-to-orbit.

12.2.2.2 Location

Samples are stored external to the spacecraft in order to maintain them at Martian surface temperatures throughout the ascent. They are stored inside an uninsulated container on the bottom of the MAV. A door to the main MAV capsule is utilized to transfer the samples to the MOV once rendezvous has occured.

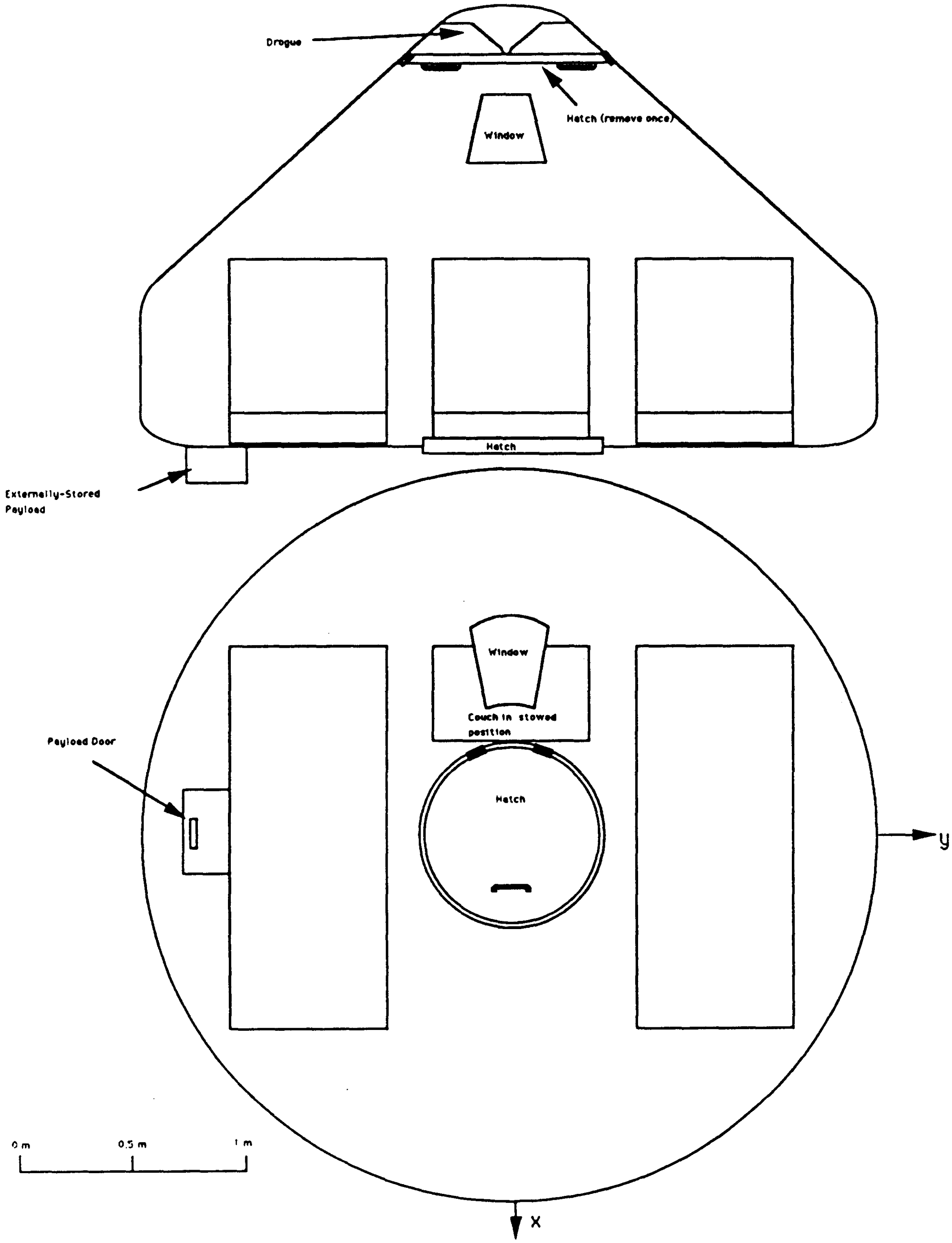

Figure 12-1: Sample, Hatches, and Window Layout

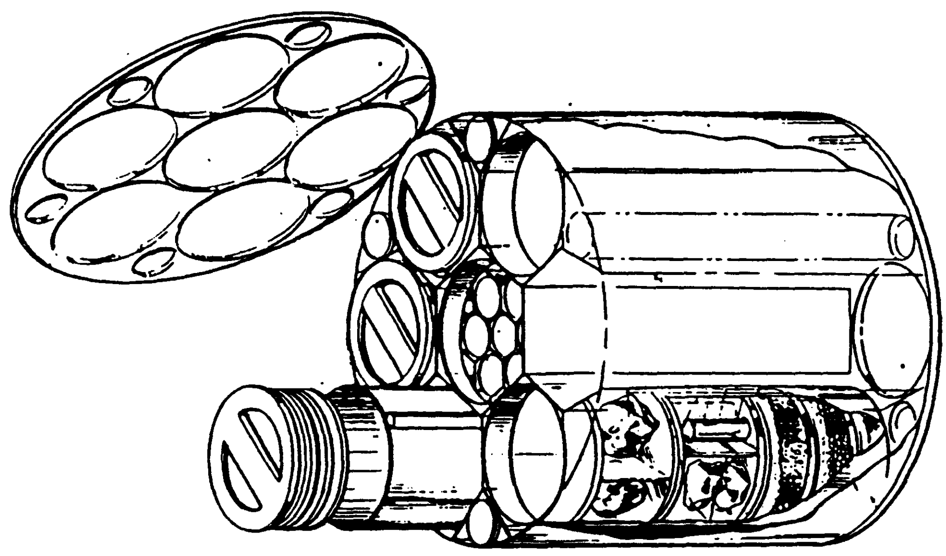

Figure 12-2: Sample Return Container (MRSR, 1989)

12.2 Hatches

12.2.1 Requirements

The MAV hatches must provide:

- attachment to MDV habitation module
- attachment to MOV/docking mechanism
- passage of fully suited astronaut: 95 cm diameter (NASA-STD-3000, 1987; Figures 8.8.1.1-1 and 14.3.4.1-1)
- manual as well as electromechanical releases for safety

Other requirements:

- must be accessible by all crew members
- must withstand same loads as main cabin structure

12.2.2 Point Design

12.2.2.1 Locations

Hatches are required to permit astronaut passage to the MOV and the MDV habitation module. One hatch is located at the front of the MAV, and allows astronaut entrance to the MAV before surface descent. It also permits transfer between the MAV and the MOV after rendezvous and docking occurs.

A second hatch, located on the bottom of the MAV, is utilized to permit transfer of the astronauts from the MAV to the MDV habitation module (see Figure 12-1).

12.2.2.2 Summary

Because the top hatch is put in place after astronaut entrance, it is only removed once (after docking). Thus, it can have a fairly simple, and therefore low mass, design. The bottom hatch, on the other hand, must allow repeated astronaut transfer. Both hatch structures are constructed of aluminum.

12.3 Windows

12.3.1 Location

A window is required only for manual docking. It is therefore located such that the middle MAV astronaut has a forward-oriented view (see Figure 12-1).

12.3.2 Structure

As mentioned before (see Section 10.0), two panes of glass are utilized in order to minimize heat losses through the window pane. Because the spacecraft must absorb external ascent heat loads, quartz glass is utilized.

The window gap is filled with cabin pressure air. Therefore, the outer pane need only withstand a 34.5 kPa (5 psia) pressure differential:

$$t = (\beta \frac{Pb^2}{\sigma_t})^{0.5} \quad \text{(Rourk, 1954)}$$

β = constant = 0.66 (a/b = 1.5)

b = panel width = 0.3048 m (12 in)

P = pressure load = 34.5 kPa (5 psia)

σ_t = quartz ult. tensile stress = 27.6 MN/m^2 (4000 psi)

Therefore, the outer pane window thickness needs (with FOS) to be 18 mm (0.69 in) thick, translating to a 5.4 kg pane (density = 2200 kg/m^3). The inner pane is sized for the same load to protect against an outer pane leak.

12.4 Pyrotechnics

Pyrotechnics are required to separate the MAV from the MDV habitation module before Martian ascent and to remove the nose cone to allow docking to occur.

12.4.1 Requirements

The MAV pyrotechnics requirements:

- provide release of MAV from MDV before ascent
- release MAV nose cone

12.4.2 Point Design

Pyrotechnic devices are utilized at the junction between the MAV and the MDV support structure (see Figure 2-2). The devices disconnect the MAV after ignition and engine power-up have occured. Additional pyrotechnic devices are located around the MAV nose cone.

12.5 Tunnel to MDV

For astronaut transfer between the MAV and the MDV habitation module, a tunnel is provided.

12.5.1 Requirements

Tunnel requirements:

- allow passage of fully suited astronaut (95 cm diameter)
- withstand 5 psia internal pressure
- detachable before ascent from inside MAV

12.5.2 Point Design

To allow astronaut transfer, the tunnel connects the MAV lower hatch to the habitation module upper hatch (see Figure 2-2). Utilizing aluminum for this tunnel, the required thickness can be estimated by analyzing hoop stress:

$$t = \frac{P\,r}{\sigma_t}$$

Therefore, 0.063 mm (0.0025 in) is the required thickness. As in Section 4.0, this is too thin to avoid a possible accidental puncture. Utilizing 1.27 mm (0.05 in) aluminum translates to 32 kg of tunnel shell mass. In addition to the shell, a ladder must be placed in the tunnel to permit astronaut transfer.

Latches are provided inside the MAV cabin to manually release the tunnel seal before Mars ascent occurs.

12.6 Summary

Table 12-1 summarizes the subsystem elements described in this section. The tunnel and MDV pyrotechnics do not ascend with the MAV, and are not included in the total mass.

Table 12-1: Misc. Systems Summary

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
payload:				
sealed samples		30.0	0	0.010
sealed containers	2	10.0	0	0.004
non-sealed samples		70.0	0	0.023
non-sealed containers	1	6.0	0	0.002
atmosphere samples	2	0.0001	0	0.006
atmosphere container	2	6.0	0	0.002
film		1.0	0	0.001
film containers	2	2.0	0	0.001
misc. structure		15.0	0	0.005
top hatch:				
structure	1	8.0	0	0.003
seals	1	2.0	0	0.002
latches, hinges	4	4.0	0	0.002
handles	1	1.0	0	0.001
bottom hatch:				
structure	1	16.0	0	0.006
seals	1	4.0	0	0.004
latches, hinges	4	4.0	0	0.002
handles	1	1.0	0	0.001
window:				
panes	2	10.8	0	0.005
seals	2	2.0	0	0.002

Table 12-1: Misc. Systems Summary (cont.)

Element	Number	Total mass (kg)	Energy (W-hr)	Volume (m^3)
support structure	1	2.0	0	0.001
nose cone pyros	2	2.0	0.1	0.002
pyrotechnic devices	(4)	(4.0)	0.1	(0.004)
tunnel:				
shell	(1)	(32.0)	(0)	(0.011)
ladder	(1)	(10.0)	(0)	(0.004)
attachments	(2)	(4.0)	(0)	(0.001)
TOTAL	30	196.8	0.2	0.083

13.0 System Integration

13.1 Configuration

13.1.1 Subsystem Locations

Subsystem locations are illustrated in each of the technical subsections (see Sections 4.0-12.0). In general, subsystems were placed to maintain mass symmetry about the x- and y- axes of the MAV.

A single level MAV was chosen over a smaller diameter bi-level MAV due to control and instrumentation accessibility and for structural simplicity. If two astronauts were placed on the bottom level, and a third astronaut were located above them, there would be less space for the spacecraft controls. Additionally, the top docking hatch would not be as easily accessible.

Furthermore, the MAV structure would be more complex. One acceleration couch would need to be suspended above the MAV floor, either by attachment to a secondary floor, or by being suspended from the top of the MAV. The spacecraft would also need to be taller, and wider at the top to accommodate the third MAV astronaut.

Finally, since drag does not strongly influence ΔV requirements (see Section 9.0), decreasing the MAV diameter is not very mass-beneficial.

13.1.2 Moments of Inertia

The spacecraft moments of inertia are utilized by the GN & C system to determine the OMS thrust rates. They are also needed to determine the rotation effect of engine-out on the spacecraft (see Section 6.0).

Spacecraft moments of inertia were determined by placing all of the subsystem element locations in a spreadsheet, and utilizing the following formula:

$I_{total} = mR^2 + I_{body}$

m = mass of object

R = x, y, or z distance of object from center of gravity

I_{body} = moment of inertia of object about its own center of gravity axis

The spreadsheet first calculates the MAV center of gravity location. After subtracting this from the subsystem location, the moment of inertias of the MAV about all three body axes are calculated. Spacecraft moments of inertia are summarized in Figures 13-1 through 13-3 over the entire mission length.

13.1.3 Center of Gravity

Elements of the subsystems were placed such that the MAV is mass symmetric about the x- and y-axes. Therefore, the center of gravity distance from the defined x- and y- axes (see Figure 6-8) is zero. The distance of the center of gravity from the z-axis is described in Figure 13-4.

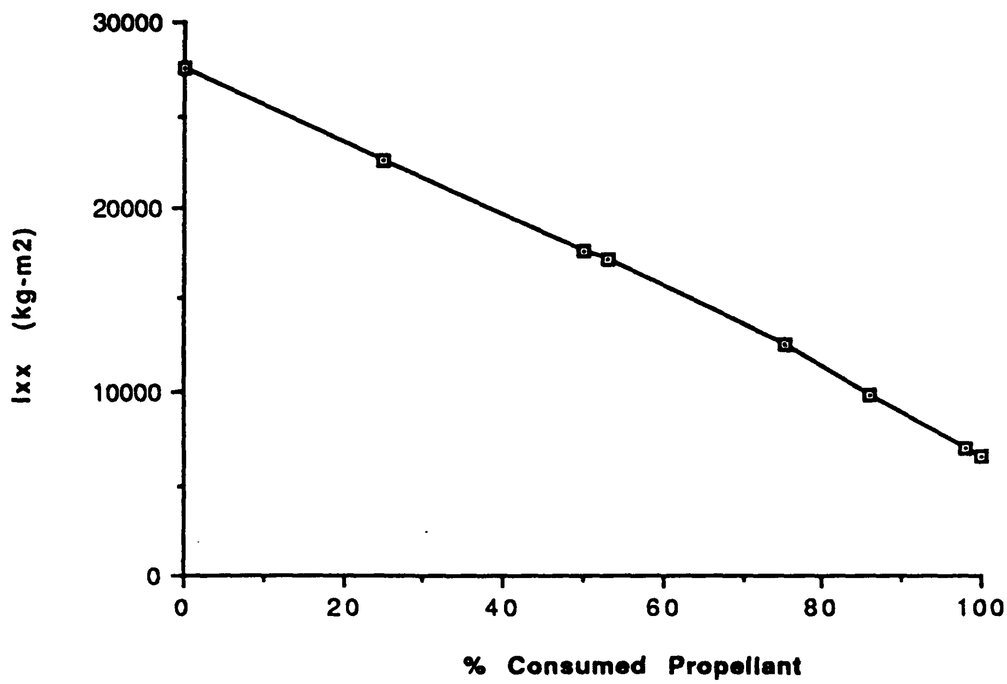

Figure 13-1: MAV I_{xx} vs. Propellant Consumed

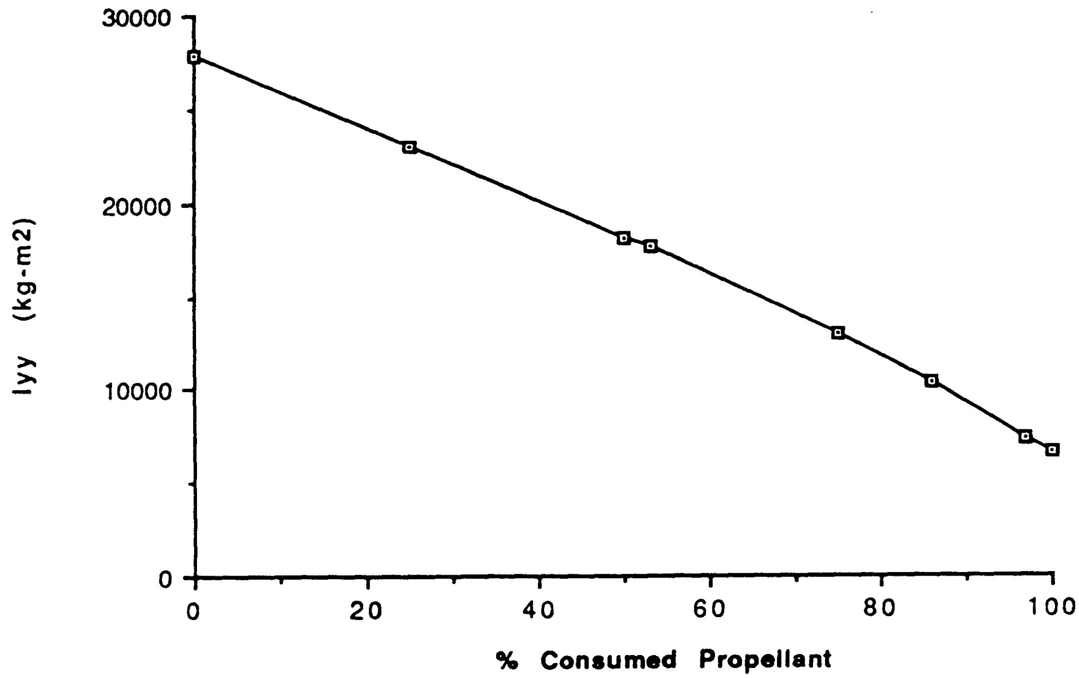

Figure 13-2: MAV I_{yy} vs. Propellant Consumed

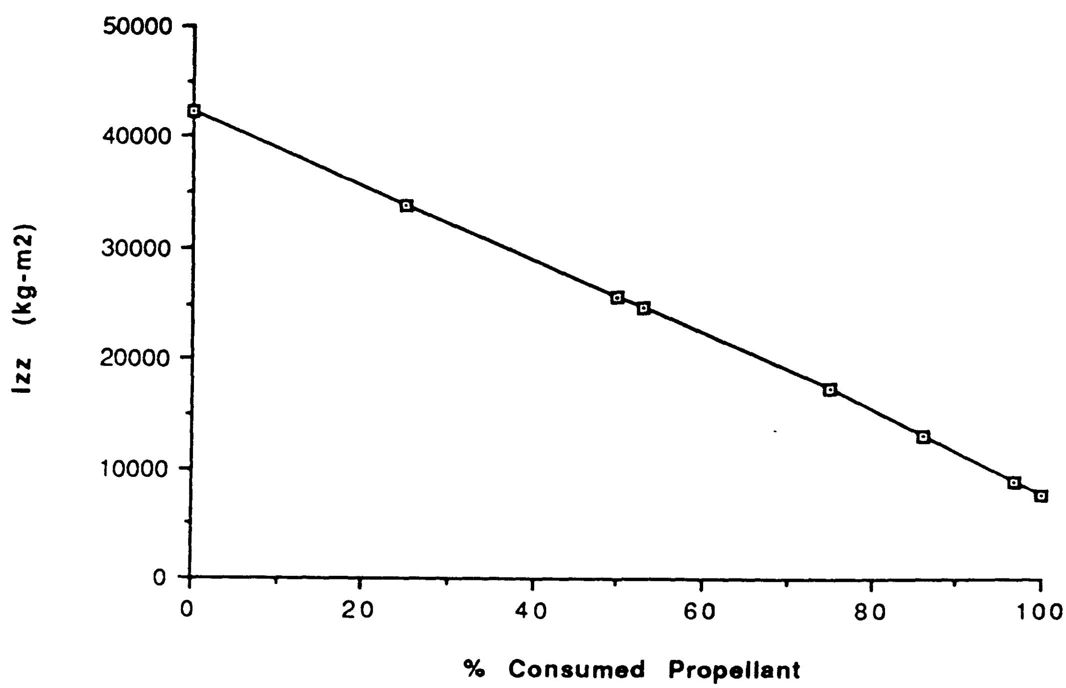

Figure 13-3: MAV I_{zz} vs. Propellant Consumed

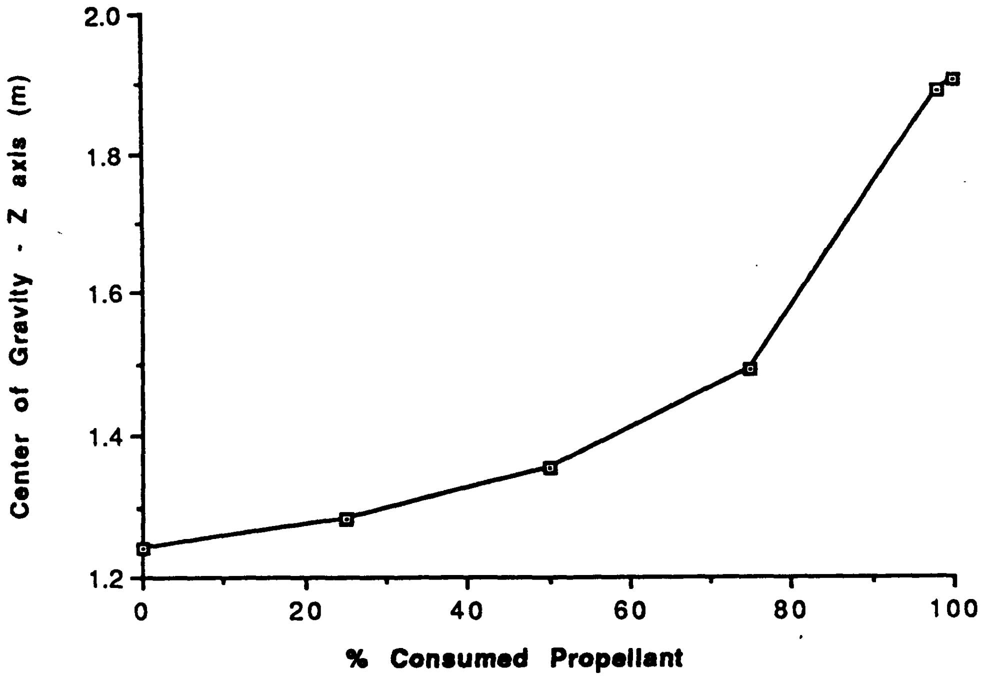

Figure 13-4: Z-axis Center of Gravity vs. Propellant Consumed

13.2 Comparison to Past Systems

Comparison of the MAV to previously flown systems is difficult because of the unique mission requirements of the MAV. The most similar mission to the MAV ascent of past space flights is the Apollo LEM ascent stage. Table 13-1 summarizes these two vehicles.

Table 13-1: LEM vs. MAV

Subsystem	MAV mass (kg)	LEM (ascent stage) mass (kg)
structure	507.45	630.98
LSS (w/out crew & suits)	126.569	239.23 (w/TCS)
main prop. (dry)	874.134	213.06
OMS (dry)	111.076	120.45
power	113.1	332.34

Table 13-1: LEM vs. MAV (cont.)

Subsystem	MAV mass (kg)	LEM (ascent stage) mass (kg)
avionics	213.67	285.94
thermal control	240.55	(included in LSS)
docking	15	(included in structure)
hatches	40	(included in structure)
window	15	(included in structure)
payload	140	279
TOTAL (dry)	2396.549	2101

Comparing the two spacecraft reveals several mass deviations. The difference in propulsion system mass is understandable; it is due to a much larger MAV propulsion mass requirement and higher engine thrust needs. The thermal control mass dissimilarity is due to the MAV insulation requirements on the Martian surface.

The power system mass improvement is influenced by a number of different factors. Most of the LEM power system mass consists of wiring and circuit breakers. However, in the MAV design, the electronic wiring is distributed over the systems which utilize the power, including the avionics system, main propulsion system, and the OMS. Therefore, some of what is considered power system mass in the LEM breakdown is distributed among several MAV subsystems.

An additional factor which influences the lower system mass is the fact that the MAV does not utilize an extensive manual circuit breaker system. Power is distributed and rerouted by the power controller and the central computer. In the LEM design, there was not enough computer power available for this purpose.

13.3 Mass, Power, Volume Summaries

Table 13-2 summarizes the MAV system masses, energy, and volume requirements. Data is drawn from Sections 4.0 to 12.0. In addition to the total mass, the fraction of the subsystem to the total MAV dry mass is tabulated.

Table 13-2: MAV Summary

Subsystem	Mass, kg (% of tot.)	Energy (W-hr)	Volume (m^3)
structure	507.5 (18.4)	0.0	0.185
LSS (w/crew)	417.6 (15.1)	3590.0	4.784
main prop. (dry)	874.1 (31.7)	529.6	15.224
OMS	183.1 (6.6)	0.3	.234
power	113.2 (4.1)	2007.9	.154
avionics	213.7 (7.7)	14142.0	.204
thermal control	240.6 (8.7)	2231.0	3.018
docking	15.0 (.5)	0.0	.015
hatches	40.0 (1.4)	0.0	.018
pyrotechnics	2.0 (.007)	0.1	.002
window	13.8 (.5)	0.0	.007
payload	140.0 (5.1)	0.0	.055
main propellant	11333.0	0.0	9.701
TOTAL (dry)	2760.5	22500.9	23.901
TOTAL (wet)	14093.5	22500.9	33.601

13.4 Mission Stages

13.4.1 Main Scenario

The main mission scenario precedes as follows:

1) astronauts buckle down in MAV	-15 min	
2) pre-ignition check	-14 min	
3) engine-startup	-3 sec	
4) MDV/MAV separation	0 sec	(at node alignment)
5) pitchover	0.90 sec	
6) throttle down	151.24 sec	
7) gravity turn	337.57 sec	
8) MOV orbit synchronization	1520.15 sec	
9) rendezvous	1602.18 sec	
10) docking	30 min	
11) initiate crew transfer	40 min	
12) discard MAV	before Earth return	

This gives an approximate 55 minute main scenario mission length.

13.4.2 Abort Scenarios

The worst-case abort mission scenario occurs as follows:

1) astronauts enter MAV	-12.3 hr	
2) pre-ignition check	-14 min	
3) engine-startup	-3 sec	
4) MDV/MAV separation	0 sec	(at node alignment)
5) pitchover	0.90 sec	
6) throttle down	189.05 sec	

7) gravity turn	385.27 sec
8) MOV orbit synchronization	t_S = 26.5 hr
9) rendezvous	t_S + 80 sec
10) docking	t_S +3 min
11) initiate crew transfer	t_S + 13 min
12) discard MAV	before Earth return

The worst-case abort scenario has an approximate 38.8 hour mission length.

13.5 Sensitivity Analysis

13.5.1 Crew Size

Three astronauts on a single level, as is baselined in this design, is probably the maximum number that could be utilized for a single-floor MAV. If more astronauts were needed, a two-level spacecraft design would be required.

Figure 13-1 illustrates the calculated spacecraft masses for additional crew members on this mission.

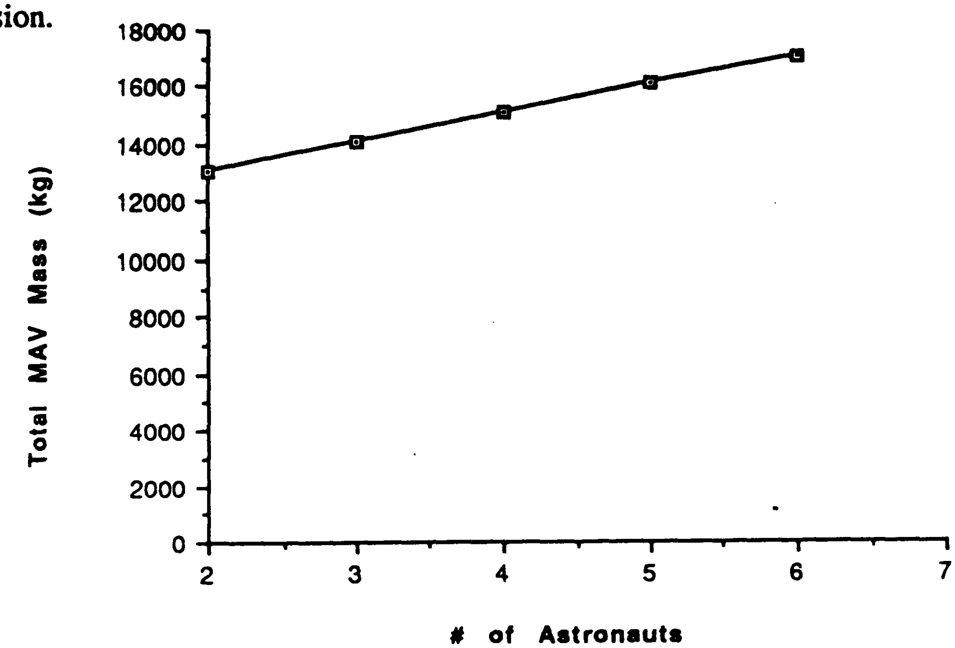

Figure 13-5: MAV Mass vs. Number of Crew Members

13.5.2 Payload Ratio

If 1 kg of payload mass is added, an additional 3.8 kg of propellant are required to lift the MAV to Mars orbit. If a significant payload increase were specified, additional structural and insulation mass would also be required.

14.0 Conclusions

The purpose of this study was to determine the approximate mass, power, and volume of a three astronaut Mars Ascent Vehicle. Emphasis was placed on utilizing state-of-the-art technologies to reduce the overall vehicle mass. Additionally, this study was performed to identify necessary areas of further research, to both allow the Mars mission to occur and to reduce the overall mass.

During the design process, one mission enabling technology was identified. For the MAV power needs, current technology levels are not sufficient. Previously, silver-zinc batteries have been utilized for space-based manned mission power needs, but these batteries have an insufficient shelf-life for this two-year mission. Therefore, either this battery storage life must be extended, or alternate power sources must be utilized.

There are two possibilities for alternate power sources, both of which also require further research. One option is the utilization of lithium-based batteries. These batteries, which have an 8-year life-span, would need to be flight-tested before they could be employed on this mission. Another possibility, which has the added benefit of reducing MAV mass, is to utilize MMH/NTO propellant fuel cells. The propellants are filtered off of the main tanks, and reacted together to obtain power. These fuels have not been previously used for fuel cells, but adapting a fuel cell to these fuels is not expected to be a problem.

In the process of the MAV point design, several enhancing technologies were identified. One technology that would significantly reduce the vehicle mass is the utilization of cryogenic fuels in the main propulsion system. To determine whether these propellants can be employed, cryogenic boil-off rates must be determined for the year-long surface stay. Safety issues have to be addressed also, to determine whether catastrophic propellant boil-off is a problem.

If these fuels are utilized, a vacuum system must be employed to reduce boil-off rates. Even if storable propellants are utilized, a vacuum system significantly reduces the power required to maintain the propellants in the liquid phase for the Martian surface stay.

In addition to this technology identification, a number of unexpected results were obtained from this study. In the area of orbital mechanics, the spacecraft shape did not greatly influence the drag losses on Martian ascent. Even with a drag coefficient of 2.0, the ΔV drag penalty was only 3.8% of the total ascent requirement.

Furthermore, expected performance advantages for utilizing a two-stage propulsion system did not materialize. Only a 6% mass advantage was attained by utilizing a two-stage pump-fed propulsion system over a single-stage, less-complex MAV.

Another problem materialized in the MAV thermal system. There were severe penalties for maintaining the MAV at room temperature on the surface of Mars and maintaining the propellants in the liquid state. Even with 200 kg of bulk insulation, the total heating requirements were still over 800 W. Further research and development should be performed to devise methods for reducing this required thermal power.

15.0 References

Aerojet TechSystems Company, "ATC-Storable Pump-Fed Upper Stage Engines," Aerojet, September, 1988.

Agrawal, Brij N., "Design of Geosynchronous Spacecraft," Prentice-Hall Inc., Englewood Cliffs, New Jersey, 1986.

Ash, Robert L., Mark A. Wiseman, Cheryl C. Jackson, "Radiator Design Considerations for Martian Surface Apllications," to be presented at 1987 ASME meeting in Boston, December 1987.

Bloom, Kenneth A., George E. Cambell, "The Apollo Docking System," Proceedings of the Eigth International Symposium on Space Technology and Science, Tokyo, 1969.

Boyce, Clay, Aerojet TechSystems, Personnel Communication, September 13, 1989.

Clark, Ben, "H/O as the Key to Survival for Humans in Deep Space," SAE Technical Paper Series, presented at 18th Intersociety Conference on Environmental Systems," San Francisco, California, July 11-13, 1988.

Clark, Mark H., "The Design of Orbital Docking Mechanisms At NASA: A History," unpublished thesis, May 1987.

Clifford, Stephen M., Cheryl J. Bartels, "The Mars Thermal Model: A FORTRAN 77 Finite-Difference Program Designed for General Distribution," University of Illinois, 1986.

Edwards, D. K., V. E. Denny, A. F. Mills, "Transfer Processes, an Introduction to Diffusion, Convection, and Radiation," McGraw-Hill Book Company, New York City, New York, 1973.

Harris Corporation, "Mars Rover Sample Return Mission Processor Analysis," prepared by M. Iacoponi, R. Gogan, for Martin Marietta Astronautics Group, July, 1989.

Honeywell Inc., "Engineering Information for Application of Selected Honeywell Technologies to Mars Rover/Sample Return Mission: Mars Lander and Ascent Vehicle," Honeywell Inc., Glendale, Arizona, May 26, 1989.

JSC-32025, "Contingency Return Vehicle for Space Station," Lyndon B. Johnson Space Center, Houston, Texas, January, 1987.

Linden, David, "Handbook of Batteries and Fuel Cells," McGraw-Hill Book Company, New York City, New York, 1984.

MMSS, "First Quarterly Report," Martin Marietta Astronautics Group, August 1987.

MMSS, "FY89 Case Studies Overview," Martin Marietta Astronautics Group, June 1989.

MRSR, NAS9-18140, "Mars Delivery and Return Study: Midterm Review for MRSR," Martin Marietta Astronautics Group Presentation, September, 1989.

NASA R 17076, "Gemini Program, Spacecraft Weight and OLV Payload Capability," 1966.

NASA-STD-3000, "Man-System Integration Standards," NASA, 1987.

NASA TM 4075, "OEXP Exploration Studies: a Technical Report," NASA, December 1988.

NASA TM 88174, "NASA Space Systems Technology Model," NASA Headquarters, June 1985.

NAS1-18147, MCR-87-583, "Program to Optimize Simulated Trajectories (POST), Final Report," Volume II, Martin Marietta Corporation, 1987.

NAS9-1100, "Lunar Excursion Module, Familiarization Manual," Grumman Aircraft Engineering Corporation, 1965.

NAS9-1100, LSR-490-75, "LM Monthly Mass Properties Status Report,: prepared by H. Zwick, December 1, 1969.

NAS9-17448, MCR-86-594, "SINDA '85/FLUINT Systems Improved Numerical Differencing Analyzer and Fluid Integrator, User's Manual," Martin Marietta Corporation, 1987.

OSHA, "Occupational Hazard and Safety Administration," 1989.

Peterson, Fred, Personal Communication, August, 1989.

Purser, Paul E., Maxime A. Faget, Norman F. Smith, "Manned Spacecraft Engineering Design and Operation," Fairchild Publications Inc., New York City, New York, 1964.

Putnam, Les, Personal Communication, Phone # 818-354-3485, December 11, 1989.

Redd, Larry, Personal Communication, July 19, 1989.

Roark, Raymond J., "Formulas for Stress and Strain," McGraw-Hill Book Company Inc., New York City, New York, 1954.

Schmidt, Eckart, "Hydrazine and Its Derivatives," Wiley-Interscience Publication, New York City, New York, 1984.

Simonsen, Lisa C., Marc J. Debarro, Jeffery T. Farmer, Carolyn C. Thomas, "Conceptual Design of a Lunar Base Thermal Control System," presented Houston, Texas, April 5-7, 1988.

Wolf, Helmut, "Heat Transfer," Harper & Row, New York City, New York, 1983.

Zocher, Roy, Personal Communication, Phone # 505-667-4731, December 12, 1989.

Appendix A: POST Input File

```
;       dt = 9/23/89            by scott geels
;
;....   search and print specifications
ideb = 1,                       /print trial step summaries
srchm = 4,                      /targeting using the projected gradient algorithm
;
ipro = -1,                      /print final trajectory only
maxitr = 0,                     /maximum number of iterations for optimization
;
;....   constants controlling automatic pert selection
npad = 0,                       /disable automatic pert option
;
;....   automatic control weighting factors
modew = 0,                      /use input scaling factors
;....   step size limitations
pctcc = 0.10,                   /max relative change allowed in magnitude of weighted vector
stpmax = 1.,                    /max absolute change allowed in magnitude of weighted vector
;....   iteration convergence tolerances
coneps =        90.0,  0.,      0.,
                0.,    0.,      0.,
p2min  = 1.,                    /value of sum of squares of errors below
;                               /which iteration is considered targeted
;
;....   curve fitting tolerances
consex =        1.0e-05,        1.0e-05,
fiterr =        1.0e-06,        1.0e-03,
;
;....   unit conversion factors
ioflag = 3,                     /metric input and metric output
;
;....   control variable specification
nindv = 4,                      /number of control variables
indxi = 2, 3, 4, 5,             /indices of active controls
;....  launch azimuth (dg)
indvr (1) = 'azl',
indph (1) = 10.,
pert (1) = .0001,               /(dg)
wvu (1) = .1,                   /(1/dg)
;
;....   sg1 inertial kickover pitch rate
indvr (2) = 'pitpc2',
indph (2) = 40.,
pert (2) = .00001,              /(dg)
wvu (2) = 2.,                   /(s/dg)
;
;....   apoapsis radius (m) at booster burnout
indvr (3) = 'critr',
indph (3) = 70.,
pert (3) = 10.,                 /(m)
wvu (3) = .00001,               /(1/m)
```

```
;
;....   true anomoly at bgn of sg2 synchronization burn
 indvr (4) = 'critr',
 indph (4) = 80.,
 pert (4) = .0001,             /(dg)
 wvu (4) = .5,                 /(1/dg)
;
;....apoapsis radius at synchronization burn
 indvr (5) = 'critr',
 indph (5) = 90.,
 pert (5) = 100.,              /(m)
 wvu (5) = .000001,            /(1/m)
;
 u (1) = 48.921,
 u (2) = -4.625,
 u (3) = 3639565.93,
 u (4) = 180,
 u (5) = 37249063.9,
;....   constraint variable specification
 ndepv = 5,
 indxd = 2, 3, 4, 5, 6,        /indices of active constraints
;
;...    altitude at sg1 burnout
 depvr (1) = 'altito',
 depph (1) = 60,
 depval (1) = 65000.,          /(m)
 deptl (1) = 1.,               /(m)
 idepvr (1) = 0.,              /equality
;
;....   apoapsis radius (m) at rendezvous orbit injection
 depvr (2) = 'aporad',
 depph (2) = 90.,
 depval (2) = 37249064.,       /(m)
 deptl (2) = 100.,             /(m)
 idepvr (2) = 0.,              /equality
;
;....   radius (m) at rendezvous-orbit injection
 depvr (3) = 'gcrad',
 depph (3) = 80.,
 depval (3) = 3639565.93,      /(m)
 deptl (3) = 10.,              /(m)
 idepvr (3) = 0.,              /equality
;
;....   flight-path angle (dg) at rendezvous injection
 depvr (4) = 'gammai',
 depph (4) = 90.,
 depval (4) = 0.,              /(dg)
 deptl (4) = .001,             /(dg)
 idepvr (4) = 0.,              /equality
;
;....   final orbit inclination
 depvr (5) = 'inc',
 depph (5) = 90.,
 depval (5) = 37.6,            /(dg)
```

```
 deptl (5) = .001,               /(dg)
 idepvr (5) = 0.,                /equality
;
;.... periapsis altitude at rendezvous-orbit inj.
 depvr (6) = 'altp',
 depph (6) = 90.,
 depval (6) = 250000.,           /(m)
 deptl (6) = 1.,                 /(m)
 idepvr (6) = 0.,                /equality
;
;....   objective function specification
 opt = 1.,                       /maximize objective
 optvar = 'weight',              /(kg)
 optph = 90.,
 wopt = 1.,                      /(1/kg)
 $
p$gendat
 title(1:40)
      = 'manned mars ascent trajectory      ',
 title(41:80)
      = '0-duration phs to align body axis with la',
 title(81:96)
      = 'unch frame (l)',
 event  = 10.,
 mdl = 1,                        /trigger event if and only if critr
;                                /equals value
;....   steering specification
 iguid (1) = 1,                  /inertial euler angle steering
 iguid (2) = 0,                  /use same functional type to
;                                /specify all attitude angles
 iguid (4) = 1,                  /all attitude angles are cubic polynomials
;                                /in time with constant terms input
 rolpc (1) = 0.,                 /inertial roll euler angle (dg)
 yawpc (1) = 0.,                 /inertial yaw euler angle (dg)
 pitpc (1) = 0.,                 /inertial pitch euler angle (dg)
;....   trajectory abort specifications
 fesn = 100.,                    /final event
 altmin = -10000.,               /minimum altitude (m)
 maxtim = 5000.,                 /maximum time (s)
;....   propagation specification
 npc (2) = 1,                    /4th order runge-kutta integration
 dt = 4.,
;....   vehicle mass specification
 npc(30) = 3,                    /use component step and propellant
;                                /mass model
;....   vehicle non-propellant mass data
;
 go = 1.,                        /weight to mass conversion
 nstpl = 1,                      /lowest index of any step
 nstph = 3,                      /highest index of any step
 istepf = 3*1,
;....   mav step 1
 wstpd (1) = 0,
;....   mav step 2
```

```
 wstpd (2) = 1247.,
;....   mav payload
 wstpd (3) = 1513.0,
;....   vehicle propellant mass data
 nengl = 1,                     /lowest index of any engine
 nengh = 2,                     /highest index of any engine
;....   mav engine for step 1
 wprp (1) = 6000.0,
 iengmf (1) = 0,                /turn off engine 1
;....   mav engine for step 2
 wprp (2) = 5333.,
 iengmf (2) = 0,                /turn off engine 2
;....   propulsion specification
 npc (9) = 0,                   /no rocket thrust simulation
;
;....   state vector initialization
 time = -3.,                    /initial trajectory time (sec)
 timeo = -3.,                   /time at which eci frame initialized
 npc (4) = 2,                   /initialize position vector with
;                               /spherical coordinates of pad
 altito = 0.,                   /initial altitude (m) of vehicle cg
;                               /above the reference ellipsoid
 gdlat = 0.,                    /geodetic latitude (dg) of launch pad
 long = 0.,                     /longitude (dg east) of launch pad
;
;....   equation of motion specification
 npc (14) = 1,                  /fix vehicle rigidly to pad
;....   inertial launch frame (l-frame) specification
 azl= 52.812,                   /azimuth (dg) of l-frame z-axis
 latl = 0.,                     /astronomic latitude of launch pad
 lonl = 0.,                     /astronomic longitude of launch pad
;....   jpl mars model parameters
 npc (16) = 0,                  /oblate planet model
 j2 = 196.5e-06,                /2nd zonal harmonic coefficient
 j3 = 0.,                       /3rd zonal harmonic coefficient
 j4 = 0.,                       /4th zonal harmonic coefficient
 mu = 4.28283e13,               /newtonian grav constant (m^3/s^2)
 omega = 0.70882181e-04,        /mars rotation rate
 re = 3397241.5,                /mean mars equitorial radius (m)
 rp = 3375542.5,                /mean mars polar radius (m)
;....   atmospheric specification
 npc (5) = 1,                   /general tables of atemt,cst,denst
;....   specification of variable to be integrated
 npc (24) = 1,                  /compute integral of gderv(i) as ginti
 gderv (1) = 'thrust',          /calculate integral of total thrust as ginti
 gint (1) = 0.,                 /initialize value of thrust integral
 gderv (2) = 'asm',             /calc integral of sensed acceleration as gint2
 gint (2) = 0.,                 /initialize value of sensed acceleration int
;....   specifications of variables to be monitored for extremes
 monx (1) = 'dynp',             /monitor dynamic pressure
 mony (1) = 'time',             /record time at dynamic pressure extremes
 monx (2) = 'qaltot',           /monitor total qalpha
 mony (2) = 'time',             /record time at extremes
;....   range calculation specification
```

```
 npc (12) = 1,                /compute crrng & dwnrng based on
;                             /relative great circles
 altref = 0.,                 /altitude of reference circular orbit for
;                             /calculating crrng & dwnrng
 azref = 52.812,              /azimuth reference for comp crrng & dwnrng
 latref = 0,                  /reference latitude for comp crrng & dwnrng
 lonref = 0.,                 /reference longitude for comp crrng & dwnrng
;....   conic calculation
 npc (1) = 2,                 /calculate and print conic block only at events
;....   profil-file specification
 prnc = -10.,                 /write profil block at each integration
;....   print interval specification
 pinc = 10.,                  /time interval between print blocks
;....   print block specification
; prnt (1)  ='time ', 'altito ','veli ', 'gammai','thr1', 'clr'   ,
; prnt (7)  ='tdurp', 'gcrad', 'azveli','weicon','thr2', 'tvac'   ,
; prnt (13) ='weight ','gdlat', 'gclat', 'long', 'asm',  'asmg'   ,
; prnt (19) ='dprng1','w',     'cs',    'xi',    'vxi',  'axi'    ,
; prnt (25) ='wdot',  'wd1',   'wd2',   'yi',    'vyi',  'ayi'    ,
; prnt (31) ='thrust','gammar','gamad', 'zi',    'vzi',  'azi'    ,
; prnt (37) ='ib11',  'ib12',  'ib13',  'dens',  'mach', 'alpha'  ,
; prnt (43) ='ib21',  'ib22',  'ib23',  'pres',  'dynp', 'beta'   ,
; prnt (49) ='ib31',  'ib32',  'ib33',  'atem',  'qaltot','alptot',
; prnt (55) ='ca',    'roli',  'yawr',  'rolbd', 'velr', 'ahi'    ,
; prnt (61) ='alphai','yawi',  'pitr',  'yawbd', 'azvelr','ahid'  ,
; prnt (67) ='betai', 'piti',  'rolr',  'pitbd', 'dlr',  'dli'    ,
; prnt (73) ='xmax1', 'yxmx1', 'lift',  'videal','tvlr', 'tvli'   ,
; prnt (79) ='xmax2', 'yxmx2', 'drag',  'int1',  'atl',  'atli'   ,
; prnt (85) ='xmin2', 'tmmn2', 'vela',  'int2',  'clr',  'gli'    ,
; prnt (91) ='pstop',
 prnt (1) = 'time ', 'heatrt ', 'tlheat ', 'ahid ' ,'htbt ','htlf',
 prnt (7)= 'htrt ', 'http ', 'veli ', 'altito ', 'pstop',
 $
p$tblmlt
 atemm = 0.555555555,         /(dk/dr)
 presm = 6894.4896,           /(144 in^2/f^2 x 47.879094 psc/(lf/f^2))
 densm = 515.37899,           /((kg/m^3)/(slug/f^3))
 csm = .30480370,             /(f/m)
 $
p$tab
;....   "mars reference atmosphere" jpl 1978 (updated with data
;       from jpl in oct 87)
;....   atmospheric temperature (degr) vs altitude (f)
 table  ='atemt' ,      1.,     'altito',
                129.,   1.,     1.,
                1.,
                                6000.,  382.3,          14000., 356.0,
  -9.99e9,385.2,                7000.,  379.4,          15000., 353.2,
  0.,    385.2,                 8000.,  376.6,          16000., 350.3,
  1000.,  385.0,                9000.,  372.8,          17000., 347.5,
  2000.,  384.8,                10000., 369.0,          18000., 344.6,
  3000.,  384.5,                11000., 365.8,          19000., 341.8,
  4000.,  384.1,                12000., 362.5,          20000., 339.0,
  5000.,  383.2,                13000., 359.3,          21000., 336.2,
```

```
22000., 333.4,
23000., 330.9,
24000., 328.5,
25000., 326.2,
26000., 324.0,
27000., 321.7,
28000., 319.5,
29000., 317.2,
30000., 315.0,
31000., 312.7,
32000., 310.5,
33000., 308.2,
34000., 306.0,
35000., 303.7,
36000., 301.5,
37000., 299.1,
38000., 296.6,
39000., 294.5,
40000., 292.3,
41000., 290.2,
42000., 288.0,
43000., 286.2,
44000., 284.4,
45000., 282.6,
46000., 280.8,
47000., 279.1,
48000., 277.4,
49000., 275.7,
50000., 274.0,
51000., 272.2,
52000., 270.5,
53000., 269.1,
54000., 267.7,
55000., 266.3,
56000., 265.0,
57000., 263.6,
58000., 262.3,
59000., 260.9,
60000., 259.6,
61000., 258.5,
62000., 257.4,
63000., 256.5,
64000., 255.6,
65000., 254.7,
66000., 253.8,
67000., 252.9,
68000., 252.0,
69000., 251.5,
70000., 251.1,
71000., 250.6,
72000., 250.2,
73000., 250.2,
74000., 250.2,
75000., 250.2,
76000., 250.2,
77000., 250.2,
78000., 250.2,
79000., 250.2,
80000., 250.2,
81000., 250.2,
82000., 250.2,
83000., 250.2,
84000., 250.2,
85000., 250.2,
86000., 250.2,
87000., 250.2,
88000., 250.2,
89000., 250.2,
90000., 250.2,
91000., 250.2,
92000., 250.2,
93000., 250.2,
94000., 250.2,
95000., 250.2,
96000., 250.2,
97000., 250.2,
98000., 250.2,
99000., 250.2,
100000.,250.2,
110000.,153.0,
120000.,153.0,
130000.,153.0,
140000.,153.0,
150000.,153.0,
160000.,153.0,
170000.,153.0,
180000.,153.0,
190000.,153.0,
200000.,153.0,
210000.,153.0,
220000.,153.0,
230000.,153.0,
240000.,153.0,
250000.,153.0,
260000.,153.0,
270000.,153.0,
280000.,153.0,
290000.,153.0,
300000.,153.0,
310000.,153.0,
320000.,153.0,
330000.,153.0,
340000.,153.0,
350000.,153.0,
360000.,153.0,
9.99e9, 153.0,
$
p$tab
;....  atmospheric pressure (lbf/i^2) vs altitude (f)
 table  ='prest',        1.,       'altito',
               129.,   1.,       1.,
               1.,
 -9.99e9,9.23e-2,
 0.,     9.23e-2,
 1000.,  8.42e-2,
 2000.,  7.69e-2,
 3000.,  7.02e-2,
 4000.,  6.41e-2,
 5000.,  5.85e-2,
 6000.,  5.34e-2,
 7000.,  4.87e-2,
 8000.,  4.44e-2,
 9000.,  4.04e-2,
 10000., 3.68e-2,
 11000., 3.35e-2,
 12000., 3.04e-2,
 13000., 2.76e-2,
 14000., 2.51e-2,
 15000., 2.27e-2,
 16000., 2.06e-2,
 17000., 1.86e-2,
 18000., 1.68e-2,
 19000., 1.52e-2,
 20000., 1.37e-2,
 21000., 1.24e-2,
 22000., 1.12e-2,
 23000., 1.01e-2,
 24000., 9.07e-3,
 25000., 8.16e-3,
 26000., 7.33e-3,
 27000., 6.59e-3,
 28000., 5.92e-3,
 29000., 5.31e-3,
 30000., 4.76e-3,
 31000., 4.26e-3,
 32000., 3.82e-3,
 33000., 3.41e-3,
 34000., 3.05e-3,
 35000., 2.73e-3,
 36000., 2.43e-3,
 37000., 2.17e-3,
```

```
38000., 1.93e-3,
39000., 1.72e-3,
40000., 1.53e-3,
41000., 1.36e-3,
42000., 1.21e-3,
43000., 1.07e-3,
44000., 9.52e-4,
45000., 8.44e-4,
46000., 7.47e-4,
47000., 6.61e-4,
48000., 5.85e-4,
49000., 5.17e-4,
50000., 4.57e-4,
51000., 4.03e-4,
52000., 3.56e-4,
53000., 3.13e-4,
54000., 2.76e-4,
55000., 2.43e-4,
56000., 2.14e-4,
57000., 1.88e-4,
58000., 1.65e-4,
59000., 1.45e-4,
60000., 1.27e-4,
61000., 1.12e-4,
62000., 9.80e-5,
63000., 8.59e-5,
64000., 7.53e-5,
65000., 6.59e-5,
66000., 5.77e-5,
67000., 5.05e-5,
68000., 4.42e-5,
69000., 3.87e-5,
70000., 3.38e-5,
71000., 2.96e-5,
72000., 2.58e-5,
73000., 2.26e-5,
74000., 1.97e-5,
75000., 1.73e-5,
76000., 1.51e-5,
77000., 1.32e-5,
78000., 1.15e-5,
79000., 1.01e-5,
80000., 8.82e-6,
81000., 7.72e-6,
82000., 6.75e-6,
83000., 5.91e-6,
84000., 5.17e-6,
85000., 4.52e-6,
86000., 3.95e-6,
87000., 3.46e-6,
88000., 3.03e-6,
89000., 2.65e-6,
90000., 2.32e-6,
91000., 2.03e-6,
92000., 1.78e-6,
93000., 1.56e-6,
94000., 1.36e-6,
95000., 1.19e-6,
96000., 1.04e-6,
97000., 9.15e-7,
98000., 8.01e-7,
99000., 7.02e-7,
100000.,6.15e-7,
110000.,1.55e-7,
120000.,3.90e-8,
130000.,1.16e-8,
140000.,4.05e-9,
150000.,1.66e-9,
160000.,7.39e-10,
170000.,3.33e-10,
180000.,1.52e-10,
190000.,7.05e-11,
200000.,3.30e-11,
210000.,1.56e-11,
220000.,7.38e-12,
230000.,3.51e-12,
240000.,1.68e-12,
250000.,8.05e-13,
260000.,3.88e-13,
270000.,1.88e-13,
280000.,9.11e-14,
290000.,4.44e-14,
300000.,2.17e-14,
310000.,1.07e-14,
320000.,5.27e-15,
330000.,2.61e-15,
340000.,1.30e-15,
350000.,6.46e-16,
360000.,3.24e-16,
9.99e9,3.24e-16,
$
p$tab
;....     atmospheric density (slug/f^3) vs altitude (f)
 table  ='denst',        1.,     'altito',
              129.,  1.,      1.,
              1.,
 -9.99e9,3.02e-5,
 0.,     3.02e-5,
 1000.,  2.76e-5,
 2000.,  2.52e-5,
 3000.,  2.30e-5,
 4000.,  2.10e-5,
 5000.,  1.92e-5,
 6000.,  1.76e-5,
 7000.,  1.62e-5,
 8000.,  1.49e-5,
 9000.,  1.37e-5,
 10000., 1.26e-5,
 11000., 1.15e-5,
 12000., 1.06e-5,
 13000., 9.69e-6,
 14000., 8.87e-6,
 15000., 8.11e-6,
 16000., 7.40e-6,
 17000., 6.76e-6,
 18000., 6.16e-6,
 19000., 5.61e-6,
 20000., 5.11e-6,
 21000., 4.65e-6,
 22000., 4.23e-6,
 23000., 3.84e-6,
 24000., 3.48e-6,
 25000., 3.15e-6,
 26000., 2.85e-6,
 27000., 2.58e-6,
 28000., 2.33e-6,
 29000., 2.11e-6,
 30000., 1.90e-6,
 31000., 1.72e-6,
 32000., 1.55e-6,
 33000., 1.40e-6,
 34000., 1.26e-6,
 35000., 1.13e-6,
 36000., 1.02e-6,
 37000., 9.15e-7,
 38000., 8.22e-7,
 39000., 7.37e-7,
 40000., 6.61e-7,
 41000., 5.92e-7,
 42000., 5.29e-7,
 43000., 4.73e-7,
 44000., 4.22e-7,
 45000., 3.76e-7,
 46000., 3.36e-7,
 47000., 2.99e-7,
 48000., 2.66e-7,
 49000., 2.36e-7,
```

```
50000., 2.10e-7,
51000., 1.87e-7,
52000., 1.66e-7,
53000., 1.47e-7,
54000., 1.30e-7,
55000., 1.15e-7,
56000., 1.02e-7,
57000., 8.99e-8,
58000., 7.94e-8,
59000., 7.01e-8,
60000., 6.19e-8,
61000., 5.45e-8,
62000., 4.80e-8,
63000., 4.22e-8,
64000., 3.71e-8,
65000., 3.26e-8,
66000., 2.87e-8,
67000., 2.52e-8,
68000., 2.21e-8,
69000., 1.94e-8,
70000., 1.70e-8,
71000., 1.49e-8,
72000., 1.30e-8,
73000., 1.14e-8,
74000., 9.95e-9,
75000., 8.70e-9,
76000., 7.60e-9,
77000., 6.65e-9,
78000., 5.81e-9,
79000., 5.08e-9,
80000., 4.45e-9,
81000., 3.89e-9,
82000., 3.40e-9,
83000., 2.98e-9,
84000., 2.60e-9,
85000., 2.28e-9,
86000., 1.99e-9,
87000., 1.74e-9,
88000., 1.53e-9,
89000., 1.34e-9,
90000., 1.17e-9,
91000., 1.02e-9,
92000., 8.96e-10,
93000., 7.84e-10,
94000., 6.87e-10,
95000., 6.01e-10,
96000., 5.26e-10,
97000., 4.61e-10,
98000., 4.04e-10,
99000., 3.54e-10,
100000.,3.10e-10,
110000.,8.63e-11,
120000.,1.94e-11,
130000.,5.08e-12,
140000.,1.53e-12,
150000.,5.29e-13,
160000.,2.33e-13,
170000.,1.04e-13,
180000.,4.71e-14,
190000.,2.16e-14,
200000.,1.00e-14,
210000.,4.72e-15,
220000.,2.24e-15,
230000.,1.06e-15,
240000.,5.09e-16,
250000.,2.44e-16,
260000.,1.18e-16,
270000.,5.69e-17,
280000.,2.76e-17,
290000.,1.35e-17,
300000.,6.59e-18,
310000.,3.24e-18,
320000.,1.60e-18,
330000.,7.90e-19,
340000.,3.93e-19,
350000.,1.96e-19,
360000.,9.81e-20,
9.99e9, 9.81e-20,
$
p$tab
;....   atmospheric speed of sound (f/s) vs altitude (f)
 table  ='cst',        1.,      'altito',
               129.,   1.,      1.,
               1.,
-9.99e9,768.,
0.,    768.,
1000.,  768.,
2000.,  768.,
3000.,  768.,
4000.,  767.,
5000.,  766.,
6000.,  766.,
7000.,  763.,
8000.,  760.,
9000.,  757.,
10000., 754.,
11000., 751.,
12000., 748.,
13000., 745.,
14000., 741.,
15000., 739.,
16000., 736.,
17000., 733.,
18000., 730.,
19000., 727.,
20000., 725.,
21000., 722.,
22000., 720.,
23000., 716.,
24000., 714.,
25000., 712.,
26000., 709.,
27000., 707.,
28000., 705.,
29000., 702.,
30000., 700.,
31000., 698.,
32000., 695.,
33000., 693.,
34000., 691.,
35000., 688.,
36000., 686.,
37000., 683.,
38000., 681.,
39000., 678.,
40000., 676.,
41000., 674.,
42000., 671.,
43000., 669.,
44000., 667.,
45000., 665.,
46000., 663.,
47000., 661.,
48000., 659.,
49000., 657.,
50000., 656.,
51000., 654.,
52000., 652.,
53000., 650.,
54000., 648.,
55000., 647.,
56000., 645.,
57000., 644.,
58000., 642.,
59000., 641.,
60000., 639.,
61000., 638.,
62000., 637.,
63000., 635.,
64000., 634.,
```

```
 65000., 633.,          87000., 628.,          190000.,763.,
 66000., 632.,          88000., 628.,          200000.,767.,
 67000., 631.,          89000., 628.,          210000.,767.,
 68000., 630.,          90000., 628.,          220000.,767.,
 69000., 630.,          91000., 628.,          230000.,767.,
 70000., 629.,          92000., 628.,          240000.,767.,
 71000., 629.,          93000., 628.,          250000.,767.,
 72000., 628.,          94000., 628.,          260000.,767.,
 73000., 628.,          95000., 628.,          270000.,767.,
 74000., 628.,          96000., 628.,          280000.,767.,
 75000., 628.,          97000., 628.,          290000.,767.,
 76000., 628.,          98000., 628.,          300000.,767.,
 77000., 628.,          99000., 628.,          310000.,767.,
 78000., 628.,          100000.,628.,          320000.,767.,
 79000., 628.,          110000.,565.,          330000.,767.,
 80000., 628.,          120000.,598.,          340000.,767.,
 81000., 628.,          130000.,638.,          350000.,767.,
 82000., 628.,          140000.,685.,          360000.,767.,
 83000., 628.,          150000.,747.,          9.99e9, 767.,
 84000., 628.,          160000.,751.,          endphs = 1.,
 85000., 628.,          170000.,755.,          $
 86000., 628.,          180000.,759.,
p$gendat
;
 title(41:80)
     = 'fixed-rltv atd phs f go-intertial to sg1',
 title(81:96)
     = ' ignition   ',
 event = 20.,
 critr = 'tdurp',
 value = 0.,                 /duration of previous phase
;....   steering specification
 iguid (1) = 2,              /relative euler angle steering
 iguid (4) = 0,              /all attitude angles are cubic polynomials
;                            /in time with constant terms carried over
;                            /from previous phase
 endphs = 1.,
 $
p$gendat
;
 title(41:80)
     ='fixed inrtl atd phs f sg1 ignition t bgn',
 title(81:96)
     =' kickover pr  ',
 event = 30.,
 critr = 'tdurp',
 value = 3.,                 /duration of previous phase (sec)
;....   steering specification
 iguid (1) = 1,              /inertial euler angle steering
 iguid (4) = 0,              /same as before
; heat rate calculation
 npc (15) = 1,               /chapman heating
 rn = .5,
;....   propagation specification
```

```
 dt = .0625,
;....    propulsion specification
 npc (9) = 1,                   /rocket thrust
 iengmf (1) =1,                 /turn on sg1 engine
;....    equation of motion specification
 npc (14) = 0,                  /remove rigid connection to mdv
;
;....    aerodynamic-force specification
 npc (8) = 1,                   /input drag coefficients
;
 sref = 8.8,                    /forward area (m^2)
;....    atmospheric specifications
 npc (6) = 0,                   /no wind simulation
;....    aeroheating specification
 npc (26) = 2,                  /compute time integral of product of dynamic
;                               /pressure and velocity relative to atmosphere (ahi)
; heatk (1) = 1,
; heatk (2) = 17600,
; heatk (3) = 26000,
;....    velocity loss calculations
 npc (25) = 2,                  /calculate velocity losses
;
 $
p$tblmlt
 tvc1m = 4.44822161,            /(n/lbf)
 wd1m = .45359702,              /(kg/lm)
 ae1m = .092905299,             /(m^2/f^2)
 cam  = 1.0,
 $
p$tab
;....    sg1 vacuum thrust (lf) versus time (s) from ignition
 table ='tvc1t',        0,      30000.,
 $
p$tab
;....    sg1 mass flowrate (lm/s) versus time (s) from ignition
;        343 isp
 table ='wd1t',         0.,     87.46,
 $
p$tab
;....    sg1 engine exit area (f^2)
 table ='ae1t',         0.,     106.78,
 $
p$tab
;....    stage 1 & 2 aerodynamic axial force coefficient vs mach
 table  ='cat', 1,      'mach',
                2,      1,      1,
                1,
        0.00, 0.80,    0.10, 0.80,    0.25, 0.70,    0.50, 0.65,
        0.75, 0.50,    0.95, 0.62,    1.04, 0.90,    1.50, 1.00,
        2.00, 0.90,    3.00, 0.73,    4.00, 0.64,    5.00, 0.60,
        10.0, 0.48,    15.0, 0.46,    20.0,0.455,    30.0, 0.45,
 endphs = 1.,
 $
p$gendat
```

```
;
 title(41:80)
      ='fixed inrtl rate phs f bgn t end of kick',
 title(81:96)
      ='over maneuver',
 event = 40.,
 critr = 'time',
 value = 0.9,                   /end time (s) of vertical flight
;                               /and start time of open-loop prl
;....   propagation specification
 dt = 0.125,
 endphs = 1.,
 $
p$gendat
;
 title(41:80)
      ='0 rltv atck angle sg1 phs to minimize po',
 title(81:96)
      ='inting velocity lss',
 event = 50.,
 critr = 'tdurp',
 value = 4.34,
;....   steering specification
 iguid (1) = 0,                 /relative aerodynamic angle steering
 iguid (2) = 0,                 /all attitude angle are same functional type
 iguid (3) = 1,                 /all angle of attack, sideslip, and bank are
;                               /cubic polynomials in tdurp with
;                               /constants input
 alppc (1) = 0.,                /(dg)
 betpc (1) = 0.,                /(dg)
 bnkpc (1) = 0.,                /(dg)
 alppc (2) = 0.,                /(dg/s)
 betpc (2) = 0.,                /(dg/s)
 bnkpc (2) = 0.,                /(dg/s)
;....   propagation specification
 dt = 0.5,
 endphs = 1.,
 $
p$gendat
 title(41:80)
      ='0 inrt atck angle sg2 thrusty phs t achiev',
 title(81:96)
      ='e xfer ellipse',
 event = 60.,
 critr = 'wprp1',
 value = 0.,
;....   steering specification
 iguid (1) = 3,                 /inertial aerodynamic angles
;....   vehicle mass specifications
 nstpl = 2,                     /lowest index of any step in current vehicle
;
 weicon = 0.,
 nengl = 2,                     /lowest index of any engine in current vehicle
;....   propulsion specification
```

```
 npc (9) = 1,                 /rocket thrust
 iengmf (1) = 0,              /turn off sp1 engine
 iengmf (2) = 1,              /turn on sp2 engine
 $
p$tblmlt
 tvc2m = 4.44822161,          /(n/lbf)
 wd2m = .45359702,            /(kg/lm)
 ae2m = .092905299,           /(m^2/f^2)
 $
p$tab
;....   sg2 vacuum thrust (lf) vs time (s) from ignition
 table ='tvc2t',      0.,     14000.,
 $
p$tab
;....   sg2 mass flow rate (lm/s) vs time (s) from ignition
;       isp = 343 sec
 table ='wd2t',       0.,     40.82,
 $
p$tab
;....   sg2 engine exit area (f^2)
 table ='ae2t',       0.,     53.39,
 endphs =1.,
 $
p$gendat
;
 title(41:80)
      ='0 inrt atck angle sg2 coast phs t aapsis',
 title(81:96)
      =' on xfr ellipse',
 event = 70.,
 critr = 'aporad',
 value = 3639565.93,          /apoapsis radius (ft) for 250 km orbit
;....   steering specification
 iguid (1) = 3,               /inertial aerodynamic angles
;....   propagation specification
 dt = 1.,
;....   propulsion specification
 npc (9) = 0,                 /rocket thrust
 iengmf (1) = 0,              /turn off sp1 engine
 iengmf (2) = 0,              /turn off sp2 engine
 endphs = 1.,
 $
p$gendat
;
 title(41:80)
      ='0 inrt atck angle sg2 thrst phs t crclrz',
 title(81:96)
      =' rndvs orbit  ',
 event = 80.,
 critr = 'truan',
 value = 180,
;....   steering specification
 iguid (1) = 3,               /inertial aerodynamic angles
;....   propagation specification
```

```
 dt = 0.5,
;....   mass specification
 weicon = 0.,
;....   propulsion specification
 npc (9) = 1,                  /rocket thrust
 iengmf (1) = 0,               /turn off sp1 engine
 iengmf (2) = 1,               /turn on sp2 engine
 endphs = 1.,
 $
p$gendat
;
 title(41:80)
      ='0 duration phase to show component masse',
 title(81:96)
      ='s at mro inject ',
 event = 90.,
 critr = 'aporad',
 value = 37249064.,            /apoapsis radious (m) for 33852 km
 tol = 10,
;....   vehicle mass specification
 weicon = 0.,                  /reset propellant consumed
;....   propulsion specification
 iengmf (2)= 0,                /turn off sp2 engine
 npc (9) = 0,                  /no rocket thrust
 endphs = 1.,
 $
p$gendat
;
 event = 100.,
 critr = 'tdurp',
 value = 0.,
 tol = 1.e-6,
 endphs = 1.,
 endprb = 1.,
 endjob = 1.,
 $
```

Appendix B: SINDA Input File

```
C *********************************************************************
HEADER OPTIONS DATA
TITLE  MARS ASCENT
       MODEL = MAV
       OUTPUT = /users/sgeels/thermal/ascent/ascent.out
       USER1= /users/sgeels/thermal/ascent/ascent.usr
       QMAP = /users/sgeels/thermal/ascent/qmap
C *********************************************************************
C
HEADER CONTROL DATA,GLOBAL
C
       UID = SI
       NLOOPS = 3000 ,            ABSZRO = -273.16
       SIGMA  = 1.0 ,             TIMEO = 0.0
       TIMEND =  5000.0
C
HEADER CONTROL DATA,RAD
C
       DRLXCA =  0.1 ,            ARLXCA = 0.1
       EBALSA =  0.1,             EBALNA = 0.1
       EXTLIM =  10.0 ,           ITEROT = 2000
       NLOOPT =  50 ,             DTIMEI = 0.5
       OUTPUT = 50.0
C
C *********************************************************************
HEADER USER DATA,GLOBAL
C
C
       FOO = 0. , FOO2 = 0., FOO3 = 0.
C *********************************************************************
HEADER ARRAY DATA,RAD
C
   10= 0., 10., 20., 30., 40., 50., 60., 70., 80., 90., 100., 110.
       120., 130., 140., 150., 160., 170., 180., 190., 200., 210.
       220., 230., 240., 250.
       260., 270., 280., 290., 300., 310., 320., 330., 340., 350.
       360., 370.
       380., 390., 400., 410., 420., 430., 440., 450., 460., 480.
       500., 520.
       540., 560., 580., 600., 620., 640., 660., 680., 700., 750.
       800., 850.
       900., 950., 1000., 8000.
   11= 0., 3.15, 31.6,  124.,  328.,  687.,  1248.,  2044.,  3101.
       4420.,  5993.,  7797.,  9754., 11841.,  13980., 16093.
       15508.,  14634.
       13830.,   13083.,  12438.,  11840.,  11281., 10792., 10345.
       9952.
       9589.,  9247.,  8936.,  8647.,  8387.,  8125.,  7869.
       7600.,  7102.
```

```
     6000.,  5070.,  4295.,  3640.,  3217.,  2879.,  2501.
     2063., 1701.
     1517.,  1312., 1072.,  742.,  560.,  393.,  314.,  224.
     187.,  143.
     120., 100.,   84., 73., 60., 42., 29., 21., 16., 11.,
     0., 0.
  12= -59.,  -59., -59., -59., -60., -61., -64., -69., -73., -79.
     -84., -88., -93.
     -97.,-102.,-106.,-110.,-115.,-118.,-121.,-123.,-126.,-128.
     -130.,-132.,-131.
     -134.,-134.,-134.,-134.,-134.,-134.,-134.,-134.,-134.
     -134.,-134.,-134.,-134.
     -142.,-155.,-167.,-181.,-188.,-188.,-188.,-188.,-188.
     -188.,-188.
     -188.,-188.,-188.,-188.,-188.,-188.,-188.,-188.,-188.
     -188.,-188.,-188.,-188.,-188.,-188.,-188.
  14= -200., .00849,  -100., .01612,    0., .02379
     100., .03139,  200., .0391,    300., .04665
     500., .06192,  7000., .5581
  16= -273., 627.6,   -240., 627.6,   -128., 648.5
     -73., 669.4
     -18., 753.1,     38., 1171.,    93.,  1318.
     149.,  1359.8,  7000., 1359.8
  17= 0., 10., 90., 180., 270., 360., 450., 7000.
  18= 0.8,  13.54,  14.45,  1.49,  .137
     .0198,   0.,  0.
  20= 0.8,  4.,     2.,  .2,  0., 0., 0., 0.
C
C*******************************************************************
HEADER NODE DATA,RAD
C
          -5,   -69.00,   0.0                      $ EXTERNAL TEMP
  SIV    10,   -69.00,   A16,   .7486              $ WALL NODES
  SIV    15,   -69.00,   A16,   2.246
  SIV    20,   -69.00,   A16,   4.671
  SIV    25,   -69.00,   A16,   6.828
  SIV    30,   -69.00,   A16,   8.230
  SIV    35,    20.00,   A16,   .7486
  SIV    40,    20.00,   A16,   2.246
  SIV    45,    20.00,   A16,   4.671
  SIV    50,    20.00,   A16,   6.828
  SIV    55,    20.00,   A16,   8.230
  SIV    60,    20.00,   A16,   5.331
  SIV    65,    20.00,   A16,   3.648
  SIV    70,    20.00,   A16,   1.964
  SIV    75,    20.00,   A16,   .374
  SIV    80,   -69.00,   A16,   5.331
  SIV    85,   -69.00,   A16,   3.648
  SIV    90,   -69.00,   A16,   1.964
  SIV    95,   -69.00,   A16,   .374
        -100,   -69.00,   0.0                      $ EXTERNAL TEMP
         -37,    20.00,   -900. * 1.591
         -42,    20.00,   -900. * 4.773
         -47,    20.00,   -900. * 9.9275
```

```
        -52,    20.00,   -900. * 14.51
        -57,    20.00,   -900. * 17.49
        -62,    20.00,   -900. * 11.33
        -67,    20.00,   -900. * 7.75
        -72,    20.00,   -900. * 4.17
        -77,    20.00,   -900. * 0.79
C ******************************************************************
HEADER SOURCE DATA,RAD
C
  TVS   10,  A10,   A11,  0.5840                  $ NOSE AERO-HEAT RATE
  TVS   15,  A10,   A11,  1.4016
  TVS   20,  A10,   A11,  2.1866
  TVS   25,  A10,   A11,  2.1307
  TVS   30,  A10,   A11,  1.2842
C
   GEN  35,   9,     5,  80.0                     $ INTERNAL HEAT
        10,   583. * 0.46 * .55                   $ SUN RADIATION
        15,   583. * 1.36 * .55                   $ ""
        20,   583. * 2.85 * .55                   $ ""
        25,   583. * 4.16 * .55                   $ ""
        30,   583. * 0.001 * .55                  $ ""
C
C ******************************************************************
HEADER CONDUCTOR DATA,RAD
C

C    CONDUCTORS ARE W/K

     TVS   1,    5,  10, A17,   A18,   0.5840
     TVS   2,    5,  15, A17,   A18,   1.7520
     TVS   3,    5,  20, A17,   A18,   3.6443
     TVS   4,    5,  25, A17,   A18,   5.3267
     TVS   5,    5,  30, A17,   A18,   6.4210
     SIV   6,   10,  15,    A14,   0.0912/.4877
     SIV   7,   15,  20,    A14,   0.1824/.5367
     SIV   8,   20,  25,    A14,   0.2918/.5855
     SIV   9,   25,  30,    A14,   0.4013/.5974
     SIV  10,   10,  35,    A14,   0.5840/.0762
     SIV  11,   15,  40,    A14,   1.7520/.0762
     SIV  12,   20,  45,    A14,   3.6443/.0762
     SIV  13,   25,  50,    A14,   5.3267/.0762
     SIV  14,   30,  55,    A14,   6.4210/.0762
     SIV  15,   35,  40,    A14,   0.0912/.4877
     SIV  16,   40,  45,    A14,   0.1824/.5367
     SIV  17,   45,  50,    A14,   0.2918/.5855
     SIV  18,   50,  55,    A14,   0.4013/.5974
     SIV  19,   55,  60,    A14,   0.4013/.5334
     TVS  20,   80, 100, A17,   A20,   4.1588
     TVS  21,   85, 100, A17,   A20,   2.8463
     TVS  22,   90, 100, A17,   A20,   1.5320
     TVS  23,   95, 100, A17,   A20,   0.2917
     SIV  24,   80,  85,    A14,   6.2832*.0381/.3794
     SIV  25,   85,  90,    A14,   6.2832*.0381/.6191
     SIV  26,   90,  95,    A14,   .13679
```

```
     SIV  27,    60,  80,    A14,    4.1588/.0762
     SIV  28,    65,  85,    A14,    2.8463/.0762
     SIV  29,    70,  90,    A14,    1.532/.0762
     SIV  30,    75,  95,    A14,    .29172/.0762
     SIV  31,    60,  65,    A14,    6.2832*.0381/.3794
     SIV  32,    65,  70,    A14,    6.2832*.0381/.6191
     SIV  33,    70,  75,    A14,    .13679
     SIV  43,    30,  80,    A14,    0.4013/.5334
          -44,   10,   5,   .20*5.667E-08 * 0.5840
          -45,   15,   5,   .20*5.667E-08 * 1.7520
          -46,    5,  20,   .20*5.667E-08 * 3.6443
          -47,    5,  25,   .20*5.667E-08 * 5.3267
          -48,    5,  30,   .20*5.667E-08 * 6.421
          -49,   80, 100,   .20*5.667E-08 * 4.1588
          -50,   85, 100,   .20*5.667E-08 * 2.8463
          -51,   90, 100,   .20*5.667E-08 * 1.532
           52,   95, 100,   .20*5.667E-08 * .29172
           54,   35,  37,   204.*.5840/.00254/100.
           55,   40,  42,   204.*1.752/.00254/100.
           56,   45,  47,   204.*3.644/.00254/100.
           57,   50,  52,   204.*5.327/.00254/100.
           58,   55,  57,   204.*6.421/.00254/100.
           59,   60,  62,   204.*4.159/.00254/100.
           60,   65,  67,   204.*2.846/.00254/100.
           61,   70,  72,   204.*1.532/.00254/100.
           62,   75,  77,   204.*.2917/.00254/100.

C *****************************************************************
HEADER OPERATIONS DATA
BUILD MAV,RAD
C
C
C      CALL STDSTL
C
       CALL QMAP('RAD', 'DAB', 1)
C      QPRINT('RAD')
C      GOTO 103
C
 102   CONTINUE
C
       CALL FWDBCK
C
 103   CONTINUE
C
C *****************************************************************
HEADER VARIABLES 1, RAD
C
C
       CALL D1D1DA(TIMEM, A10, A12, FOO)
       T5 = FOO
       T100 = FOO
C      CALL QMAP('RAD','DAB',1)
C *****************************************************************
HEADER OUTPUT CALLS,RAD
```

```
C
M      CALL TPRINT('RAD')
C
       CALL HNQPNT('RAD')
       FOO3=Q37+Q42+Q47+Q52+Q57+Q62+Q67+Q72+Q77
       WRITE(6,100) FOO3
F 100 FORMAT(/,10X,'HEAT REQUIRED = ', F12.2)
       WRITE(NUSER1,101) TIMEM, FOO3
F 101 FORMAT(2F12.4)
```

Appendix C: Table of Acronyms

AMO	Areosynchronous Mars Orbit
CERV	Contingency Earth Return Vehicle
CPU	Central Processing Unit
EVA	Extravehicular Activity
FLOPS	Floating Operation Per Second
FOS	Factor Of Safety
FPU	Floating Processor Unit
GN&C	Guidance, Navigation, and Control
IR	Infrared
LEM	Lunar Excursion Module
LEO	Lower Earth Orbit
LH_2	Liquid Hydrogen
LMO	Lower Mars Orbit
LOX	Liquid Oxygen
LSS	Life Support System
MAV	Mars Ascent Vehicle
MDV	Mars Descent Vehicle
MIPS	Mega-Instructions Per Second
MLI	Multilayer Insulation
MMH	Monomethyl Hydrazine
MMSS	Manned Mars System Study
MOV	Mars Orbiting Vehicle
MRSR	Mars Rover/Sample Return
MTV	Mars Transfer Vehicle
NASA	National Aeronautics and Space Administration

NTO	Nitrogen Tetroxide
OEXP	Office of Exploration
OME	Orbital Maneuvering Engine
OMS	Orbital Maneuvering System
OSHA	Occupational Hazard and Safety Administration
POST	Program to Optimize Simulated Trajectories
RAM	Random Access Memory
RCS	Reaction Control System
RP	Rocket Propellant
SEU	Single Event Upset
SINDA	Systems Improved Numerical Differencing Analyzer
STS	Space Transportation System
TCS	Thermal Control System
TPS	Thermal Protection System