

MarsDrive

Design Reference Mission 2.5

June 20, 2008

I – Introduction

MarsDrive is an international organization dedicated to the exploration and settlement of space. Our Vision is of humanity becoming a space faring civilization. Our Mission is to enable and in every way support the rapid expansion of humanity into space by providing funds to research and develop the infrastructure necessary to establish a sustainable space economy which includes settlements and research facilities in places like the Moon and Mars.

Where Marsdrive is significantly different from the many other space advocacy groups can be summed up in the words of our CEO and Founder, Frank Stratford: "*The future of our expansion into space and onto Mars is really up to each of us. If there is no generation that steps up to make the decision to go, who will?*" MarsDrive is not relying on any outside organization to pick up our work and implement it. Rather, our goal is to put together the *Mars Consortium* – a joint venture of individuals, private investors, and nation-states who desire to be a part of this greatest of all human adventures – to fund, develop, and fly the first human mission to Mars.

The goal of this Design Reference Mission is to serve as a catalyst around which the *Mars Consortium* can coalesce. It is not intended to produce yet another how-to-get-to-Mars plan which ends up gathering dust on various bookshelves. This Design Reference Mission represents a clear departure from other Mars proposals. Some of the more notable points of departure are:

- The DRM does not require any major technological breakthroughs such as nuclear propulsion. All technologies are either in use today or are relatively modest engineering outgrowths of existing technologies.
- The capability for in-flight artificial gravity has been designed into the mission architecture for the outbound cruise stage only. The desire is to keep the crew physically fit to minimize their recovery time when they reach Mars. The inbound cruise stage will be in microgravity.
- The DRM takes full advantage of known Martian resources for the production of rocket propellants for ascent from the Martian surface, for environmental oxygen, for environmental water, and for backup power. The design does not require the existence of useable water resources for the first mission to be successful. However it does allow the mission to switch to 100% *in situ* resource utilization as soon as water resources can be accessed.
- The DRM does not require any expensive and long-lead launch vehicle development efforts. The largest propulsion system development will be a liquid oxygen / liquid methane restartable engine capable of producing 25,000 kgf of vacuum thrust.
- The DRM requires relatively modest launch capabilities; each crewed mission requires three heavy-lift launches (80mT payload to LEO each). Each uncrewed mission requires two heavy-lift launches.

- The DRM does not require support or buy-in by NASA. Whereas many other manned Mars mission designs are predicated on the assumption that NASA will adopt the plan and implement it, the *Mars Consortium* is based on the assumption that if we do not do this ourselves, it will never happen.

II – Mission Summary

This is a so-called *Opposition-class* mission. Each cruise stage is approximately 7 months in duration, with a stay at Mars of approximately 18 months. This restricts the choice of departure windows to approximately once every 26 months, hereinafter referred to as “a *cycle*.” A fundamental mission requirement is that all mission-critical assets – those whose loss would mean the loss of the crew – will be either pre-positioned and validated prior to the crew’s departure from Earth, or will fly with the crew.

The MarsDrive Mission to Mars will be flown as three separate Earth departures over two successive cycles. The first crewed mission will be supported by two uncrewed missions flown in the previous cycle which will pre-position various assets in Mars orbit and on the surface.

Mission C-1 pre-positions a Cargo Lander at the most likely candidate landing site. The *MarsDrive Site Selection Committee* will draw upon available information from then-existing Mars orbiting and landed assets to select this site. The top criteria for site selection will be the probability of available water, altitude¹ compatible with the *Mars Common Lander*, availability of year-round solar power, and minimal orbital plane-change requirements. This lander carries two teleoperable explorer vehicles which will be used to prepare the landing site for the two following missions, and which can be used by the crew for short-range transportation. It also carries four metric tons of hydrogen plus equipment to produce 39 metric tons of water ice using oxygen extracted from the Martian atmosphere. This water will later be used by the crew’s lander to produce its ascent propellants. C-1 will also pre-position a communications satellite in synchronous orbit.

Mission H-1 pre-positions the crew’s Surface Habitat at the chosen landing site. H-1 will also pre-position a second communications satellite in synchronous orbit. The vehicle aerocaptures into Mars orbit where the Habitat Lander loiters until the final site selection decision is made. The Habitat Lander then deorbits and lands.

Mission M-1 is the crewed mission. The Earth-departure stack consists of the crewed Mars Lander, the in-space habitat, and a Service Module, which performs all in-space propulsive maneuvers after Trans-Mars Injection. The stack aerocaptures into Mars orbit. The Crewed Lander then deorbits and lands at the selected landing site. Upon reaching Mars orbit at the end of the mission, the lander’s ascent stage performs a rendezvous with the loitering Service Module and in-space habitat. The Earth-return stack is assembled, consisting of the in-space habitat and the Service Module, which performs the Trans-Earth Injection maneuver. Upon arrival back at the Earth, the stack aerocaptures into Earth orbit. The crew returns to Earth in a crew vehicle which is launched to rendezvous with the orbiting habitat.

II.1 – Mission Goals

The overall goal of this mission is to prove the technology and begin building the infrastructure to support the permanent human habitation of the planet Mars. To accomplish this, three specific goals are needed:

- **Find a reliable source of water.** While the first mission carries sufficient seed hydrogen to produce the ascent propellants and the crew’s drinking water, this is not a long-term strategy. The first goal of this mission is to locate, test, and validate a reliable, accessible source of water.
- **Establish a robust solar power grid.** There are only two practical means of generating electricity on Mars – solar power and nuclear power. Because of the political issues raised by launching nuclear material from the Earth, we have elected to rely on solar power, at least in the early missions. Should the political climate for nuclear power improve, this option will be reconsidered.
- **Construct a greenhouse to grow food.** As with water, long-term habitation of Mars is only practical if foodstuffs can be produced locally. The main scientific experiment of the first mission will be to investigate various options for growing food with the goal of establishing a greenhouse.

II.2 – The Crew

A crew size of six (6) has been chosen for this mission. The crew size was driven by two primary factors – the number of skills required to carry out the mission and the “buddy system.” The “buddy system” is a key element of crew safety. No one crew member goes anywhere outside the main surface habitat alone. This dictated an even number of crew members, which could be divided into two-person sub-crews.

The skills required to carry out the mission were determined to be:

- Mission Commander
- Pilot (need 2)
- Physician (2 or 1 + EMT)
- Engineer (Propulsion)
- Engineer (Electrical / C3)
- Engineer (ECLSS)
- Engineer (ISRU Plants)
- Geologist
- Microbiologist (Planetary Protection Officer)
- Botanist (Greenhouse)

III – The Earth Launch Vehicle

The MarsDrive DRM has been designed to use a single candidate heavy-lift launch vehicle. Four candidates were considered.

NASA’s Ares-V was not selected because the probability of its availability is considered very low. It is likely that the full production quota will be consumed with NASA missions for the foreseeable future. Even if vehicles were available, it is expected that the cost per vehicle will far exceed the other options.

The Russian Energia was not selected because the vehicle has been mothballed for twenty years and the possibility of resurrecting it is considered unlikely. However, if Russia joins the *Mars Consortium* as a partner whose contribution is the required heavy-lift launches, this option becomes more feasible.

A new Heavy-Lift Launch Vehicle developed or contracted for by MarsDrive was not selected because of the inherent expense and time required for such a program. It was felt that the benefits of such a selection were only marginal compared to the selected option.

The Lockheed-Martin Atlas Evolved Phase II Launch Vehicle was selected as the design

target for this DRM. Based on data published by Lockheed-Martin², the growth path for the Atlas family involves an “Evolved Booster” capable of holding 1.7 times the propellant of the Atlas V and using 2 RD-180 engines, and an “Evolved Centaur” capable of holding 6.5 times the propellant of the current Centaur and using up to 4 RL-10 engines. A three-core booster, consisting of a dual RD-180 core and two dual RD-180 liquid-fueled strap-ons, is claimed by Lockheed-Martin to be capable of placing up to 86 metric tons of payload in Low Earth Orbit. To be conservative, this design keeps all LEO payloads below 80mT and does not rely on the more speculative *Atlas Evolved Phase III* concept.

The **Atlas Evolved Phase II Launch Vehicle** (hereinafter *Atlas E-II*) is shown schematically in Figure 3.1. The diagram was taken from the *Mission Planner’s Guide* (Section 8.3) and was annotated with dimensional information provided by Lockheed-Martin.

The **Evolved Phase II Centaur** (hereinafter *Centaur E-II*) can be sized to satisfy a wide range of mission requirements. A Centaur E-II

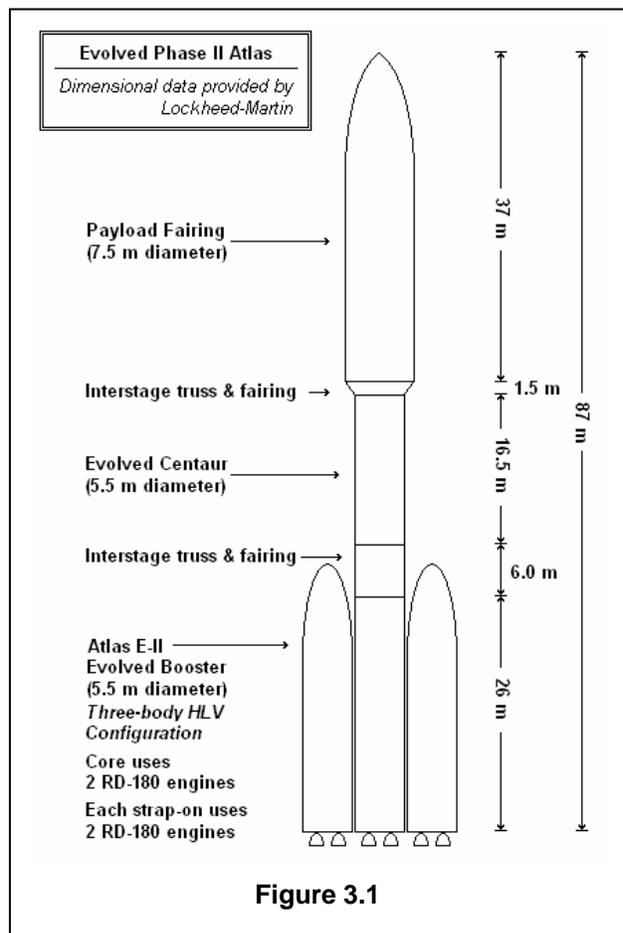


Figure 3.1

stage with propellant capacity of 73 metric tons and a dry mass of 7 mT fits within our chosen target of 80 metric tons payload. Thus the Atlas E-II launch vehicle may be used to place a fully-fueled Centaur E-II stage in low earth orbit. This feature gives us the ability to use the Centaur E-II stage as a Trans-Mars Injection (TMI) stage, thereby obviating the time and expense for MarsDrive to develop its own TMI stage.

Two serially-launched TMI stages will be used for each of the three MarsDrive mission vehicles. The first stage will loiter in LEO for the estimated four weeks required to turn around the Atlas launch facility to launch the second TMI stage. The four-week time between launches is based on Lockheed-Martin's estimate of the turn-around time required for a single Atlas HLLV launch facility such as now exists at the Kennedy Space Center. Should multiple launch facilities become available in the future, this time may be adjusted accordingly.

According to the *Mission Planner's Guide*², "Through the addition of a mission-peculiar kit consisting of solar power and passive thermal shielding, mission durations [for the Centaur E-II] in excess of one year are possible." This claim opens up the possibility of using an appropriately-equipped Centaur E-II derivative as the deep-space propulsive stage for the various missions in this DRM.

IV – The Mars Common Lander

The Mars Common Lander design is used in all MarsDrive missions. It serves as the crew's ascent/descent vehicle in Mission M-1. It also serves as the Mars Surface Habitat lander in mission H-1 and the Cargo Lander in mission C-1. Flying the Common Lander twice prior to the crewed mission provides the opportunity to validate the vehicle's entry, descent, and landing (EDL) characteristics at Mars without placing a crew at risk in an untried vehicle. It also cuts the project's total development cost by requiring the design, development, and certification of only one EDL vehicle.

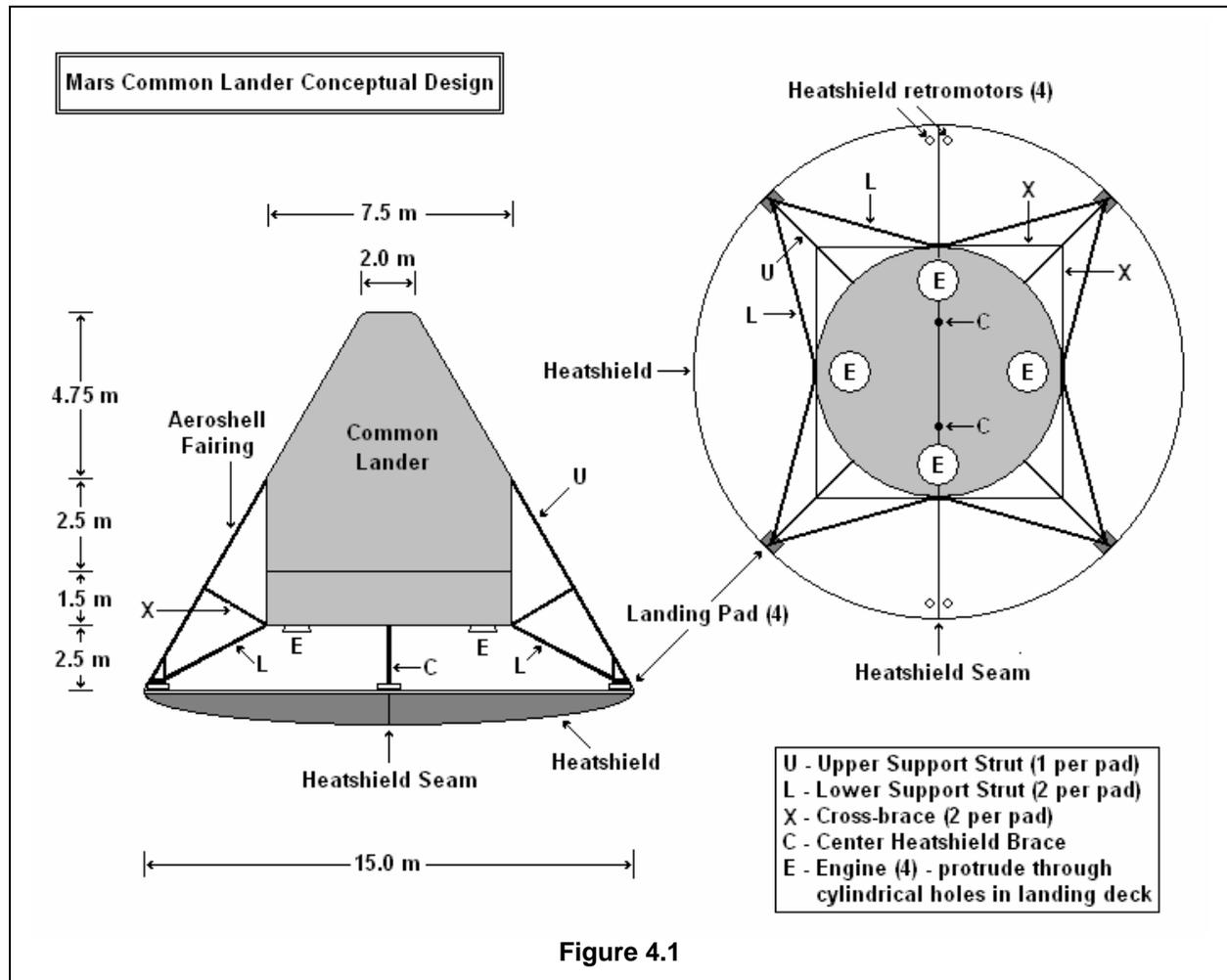
The Common Lander uses an Apollo-legacy heatshield-plus-truncated-cone profile. The vehicle enters the Martian atmosphere ballistically and is slowed by aerobraking. As the vehicle approaches Mach 1, the heatshield is discarded and the vehicle maneuvers to a landing propulsively. The use of supersonic parachutes to slow the vehicle between the aerobraking phase and the propulsive phase was considered, but was eventually discarded. Investigations by Braun and others^{3,4} have shown that the advantages of an intermediate parachute stage diminishes as the entry mass increases, vanishing altogether around 60 metric tons of entry mass. At the entry mass of the Common Lander, the small advantages of adding a parachute stage are offset by the increased risk of adding yet another element to what is already the riskiest part of the entire mission. Should further investigations reveal a significant advantage, this option can be revisited with little or no affect on the overall mass budget.

The Common Lander must fit within the 7.5 meter payload diameter of the Atlas E-II launch vehicle. However, this presents three significant problems. First, a 7.5-meter diameter heatshield proved to be insufficient to aerobrake the lander to the vicinity of Mach 1 during descent. Second, in order to ignite the landing engines, the heatshield, which covers the engines, must be discarded. Given the relative ballistic properties of the heatshield and the lander, it is difficult to

see how this can be accomplished without the addition of a supersonic parachute. And third, igniting the landing engines while facing into a supersonic airstream is problematic.

IV.1 – The Mars Common Lander – Heatshield Design

The MarsDrive DRM proposes a new ballistic aerobraking design that stays within the proven Apollo-legacy geometry, allows a 15 meter diameter heatshield, and fits within the 7.5 meter diameter payload limit of the Atlas E-II. The design is shown schematically in Figure 4.1.



The vehicle is supported by a four-leg landing gear, each leg consisting of a foot-pad and a conventional 5-element support structure – one upper strut, two lower struts, and two cross-braces to provide lateral rigidity. The landing gear is stowed for Earth launch, giving the lander a fairing diameter of 7.5 meters. The spread of the deployed landing gear is 15 meters.

The primary aerobraking device is a 15-meter diameter carbon-carbon heatshield backed by fiber-form insulation. The heatshield and its support structure rest against the four landing pads as shown in Figure 4.1. The lander essentially “sits” on the heatshield. Two center braces (items ‘C’ in Figure 4.1) are provided to support the center of the heatshield. The heatshield is split

diagonally and joined with a 15-meter long piano hinge, allowing the heatshield to fold in half, shown as item '3' in Figure 4.2. The resulting folded heatshield will then fit inside a 7.5-meter diameter fairing for Earth launch.

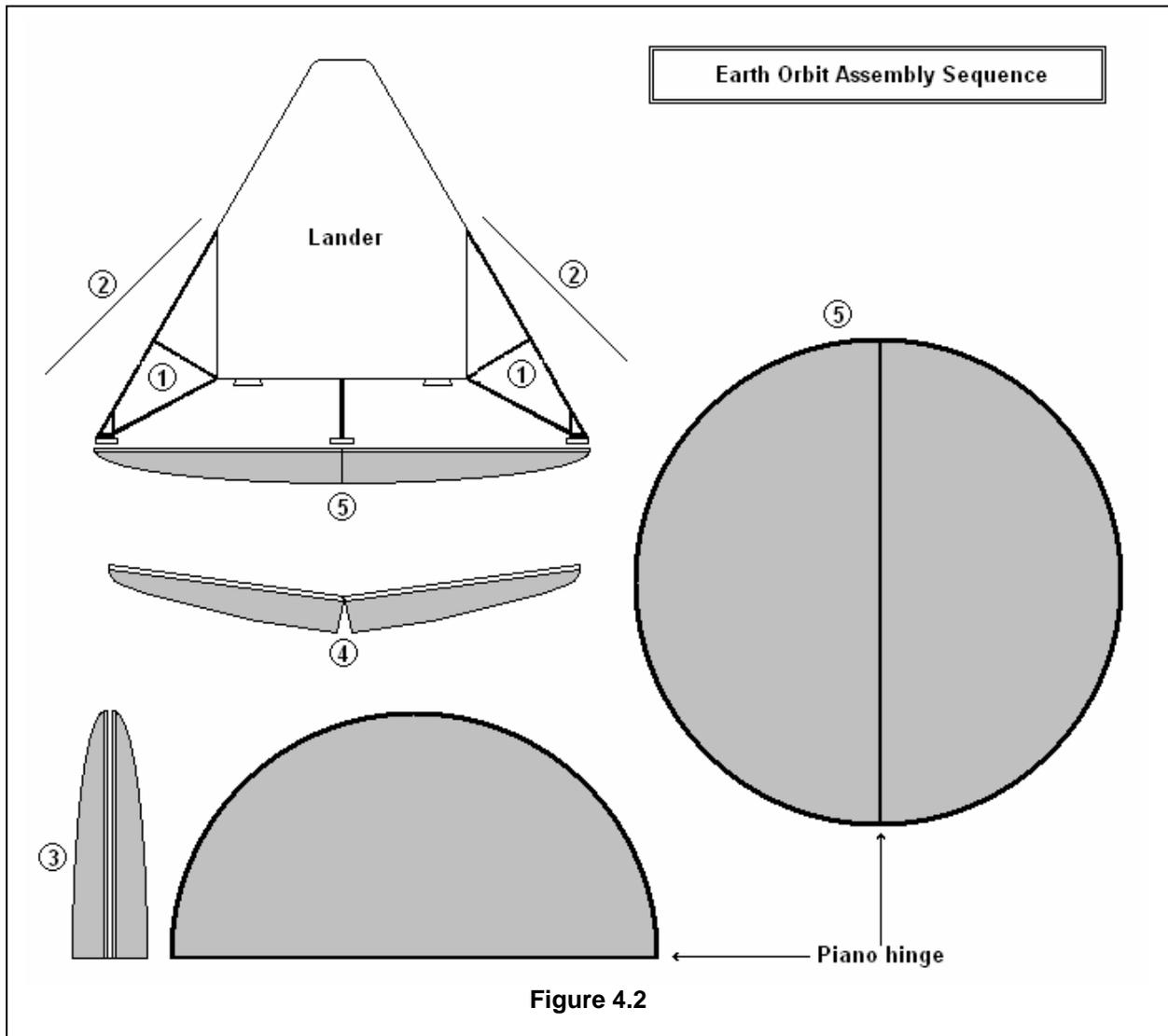


Figure 4.2

The area between the heatshield and the conic section of the lander is covered by an aeroshell (labeled "Aeroshell Fairing" in Figure 4.1 and item "2" in Figure 4.2), which provides protection during both aerocapture and aeroentry. The lander with the aeroshell in place presents a clean 30° truncated cone to the airstream leaving the heatshield.

Referring to Figure 4.2, after the Lander enters Low Earth Orbit, the landing gear (1) is deployed and locked in position. The aeroshell panels (2) are attached to the landing legs. The heatshield (3) is unfolded (4) into a complete disk (5) and attached to the landing pads and the center brace. During aerobraking, the windward stream will compress the join where the heatshield folds, preventing plasma from encroaching into the join itself.

IV.2 – Ejecting the Heatshield

The transition from aerobraking to powered flight begins as the lander approaches Mach 1. At this point the lander assumes a 0° angle of attack. The numbered sequence is depicted in Figure 4.3.

1. The engines ignite. The sheltered space between the aft bulkhead, the heatshield, and the side-wall aeroshell is roughly 450 cubic meters, more than enough volume to absorb the ignition plumes of the four engines.
2. As soon as engine ignition has been confirmed, the aeroshell panels are ejected. The engines come to full throttle.
3. The heatshield clamps are released and the heatshield retromotors (shown in Figure 4.1) are ignited. These four solid motors serve two purposes. They begin to drive the heatshield away from the lander and they cause the heatshield to begin to fold along the piano hinge.
4. The hinged heatshield is in a state of unstable equilibrium. Once it begins to fold, airstream pressure will force the outer edges inward, continuing the folding.
5. The heatshield quickly folds into its collapsed state. This dramatically lowers the drag on the heatshield, which causes it to fall away from the decelerating lander.

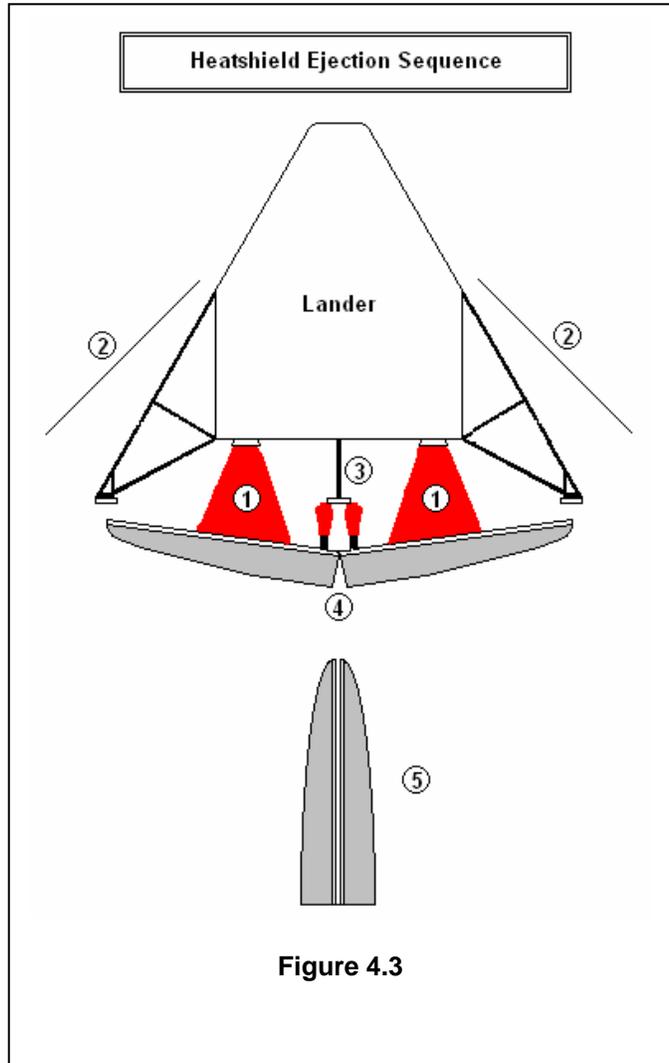


Figure 4.3

Under full power the lander decelerates to a hover in 20 seconds. It then has approximately 300 seconds of propellant remaining to maneuver to its landing site under reduced power.

IV.3 – The Common Lander Structure

The Common Lander, used for all three missions, is a truncated 30° conic section sitting on a 7.5 meter diameter cylindrical section. The total volume of both sections is 270 m³, not all of which is pressurized. The lander’s pressure vessel structure⁵ is an aluminum honeycomb sandwich using Al 2024 for the face sheets and Al 5052 for the honeycomb core. The pressure vessel structure mass was designed to withstand 9.5 psia nominal internal pressure. The assumed density for such a pressure vessel is 14 kg/m³. Key elements of the common lander geometry are given in Table 4.1.

| Lander Geometry (truncated cone over cylinder): | |
|--|----------------------|
| Base Diameter: | 7.5 m |
| Conic Angle: | 30.0 deg |
| Diameter of truncation: | 2.0 m |
| Height of truncated conic section: | 4.76 m |
| Volume of truncated conic section: | 93.8 m ³ |
| Ground clearance: | 2.5 m |
| Height of cylindrical section: | 4.0 m |
| Volume of cylindrical section: | 176.5 m ³ |
| Total enclosed volume of lander: | 270.3 m ³ |
| Heatshield diameter: | 15.0 m |
| Effective ballistic area: | 176.7 m ² |
| Aeroshell surface area: | 265.1 m ² |

Table 4.1

The primary aerobraking device is a 15-meter diameter carbon-carbon heatshield backed by fiber-form insulation. The aerodynamic area of the heatshield is 177 m². Because carbon-carbon experiences no recession over the temperatures experienced during Mars aerocapture and aeroentry², only 1 mm of carbon-carbon is required. The fiber-form backing was sized at 10 cm to limit the bondline temperature to 250° C. The heatshield plus its support structure masses 5015 kg.

The conic section of the lander and the area between the heatshield and the conic section of the lander is covered by an aeroshell which provides protection during both aerocapture and aeroentry. This aeroshell³ is composed of LI-2200, LI-900, Advanced Flexible Reusable Surface Insulation (AFRSI), and Flexible Reusable Surface Insulation (FRSI) at equal thicknesses over a support structure of graphite epoxy/Bismaleimide (BMI) composite skin panels. The aeroshell surface area, including the conical section of the common lander, is 265 m². The aeroshell masses 4085 kg.

The Common Lander has one set of four engines and oxygen/methane propellant tankage which are used for descent and, in the case of the crewed lander, ascent. This gives the vehicle single-engine-out performance at all stages of both ascent and descent. As there is presently no engine that meets the Common Lander’s requirements, design and development of this engine will be a major task facing MarsDrive. Place-holder dimensions and mass of this engine were patterned after the RL-60 oxygen/hydrogen engine.

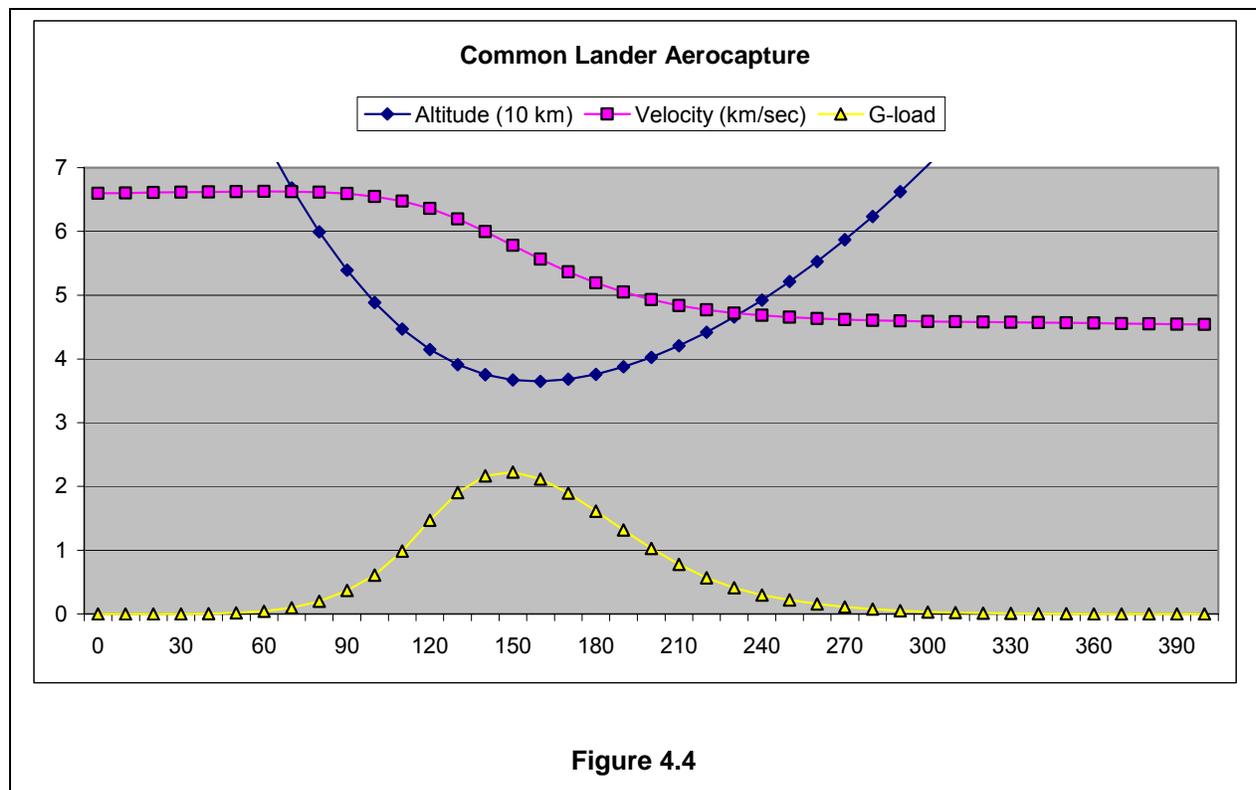
The engines are attached to the thrust structure at a point two meters above the bottom of the cylindrical section just above where the ascent stage separates from the landing deck on the crewed lander. The engine nozzles protrude through cut-outs in the lower 2 meters of the cylindrical section. Differences between the various mission-specific landers will be discussed in detail with each mission.

IV.4 – Common Lander Aerocapture Simulation

In order to evaluate the Common Lander’s aerocapture, entry, descent, and landing characteristics, a simulation tool was developed to accurately predict the behavior of a spacecraft during entry into the Martian atmosphere. This tool added an atmospheric model and lift/drag calculations to a pre-existing tool simulating spacecraft flight dynamics in the Sun-Earth-Mars gravitational environment. The aeroentry simulator is discussed in detail in Section IX of this report.

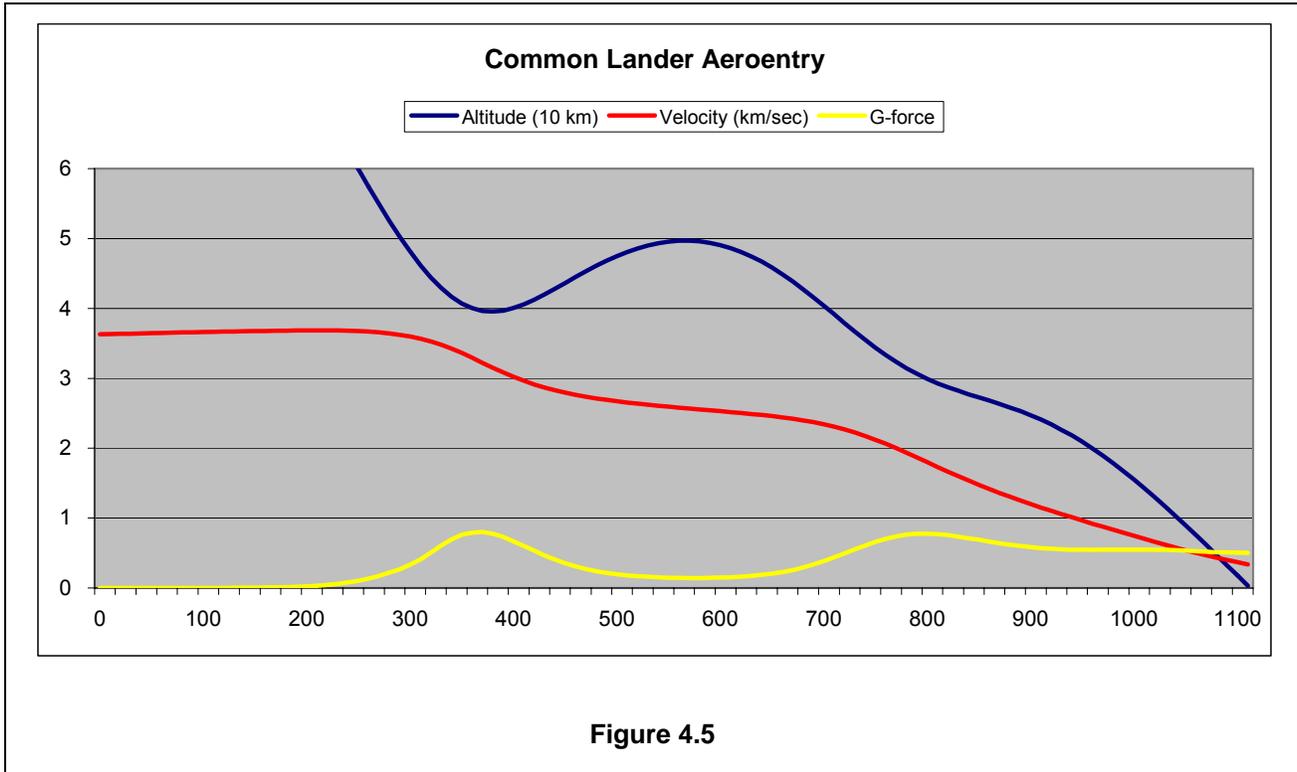
The two uncrewed missions, C-1 and H-1, share a common aerocapture profile. The crewed mission, M-1, has a higher approach mass and requires a second aerocapture pass in order to place the in-space habitat and service module in a more suitable orbit for rendezvous with the returning ascent vehicle. This maneuver will be discussed as part of the M-1 mission profile.

The uncrewed missions aerocapture into highly-elliptical Mars orbits. Based on the simulated Earth-Mars transfer trajectory and a reasonable aerocapture alignment maneuver, aerocapture will place the stack in an orbit with an apoapsis of 3,171 km. An apoapsis ΔV of +30 m/s will raise the orbital periapsis to 181 km. The aerocapture pass is shown in Figure 4.4.



IV.5 – Common Lander Aeroentry Simulation

All three missions share a common aeroentry (EDL) profile. After the completion of their respective mission’s orbital operations, the Landers deorbit by means of a 50 m/s ΔV burn. The ballistic aeroentry profile for a constant Lift/Drag ratio of 0.30 is shown in Figure 4.5. More detailed studies using time-varying angle-of-attack should produce an improved entry profile.



The lander will exit the plasma phase of its descent around 450 seconds after entry interface during the “roller-coaster” maneuver. At this point the landing zone beacons should be acquired, allowing the guidance computers to determine the optimal attitude to minimize the terminal cross-range maneuvering.

At 1100 seconds after entry interface, the lander has slowed to Mach 1.3 at 2 km MOLA. At this point the landing engines ignite and the heatshield is jettisoned as described in Section IV.2. Under full power the lander decelerates to a hover in 20 seconds at 0.5 km MOLA. It then has 300 seconds of propellant remaining to maneuver to the landing site and land.

V – The Cargo Mission (C-1)

Mission C-1 pre-positions two teleoperable rover vehicles, an ISRU plant capable of producing 40 mT of water ice, and miscellaneous cargo at the designated landing site one cycle before the crewed mission. The mass summary of the C-1 mission is shown in Table 5.1.

The Earth-departure stack consists of the Cargo Lander, a Communications relay satellite, and the Service Module. The stack aerocaptures into Mars orbit using the lander’s heatshield. Mission C-1’s Earth-departure stack is shown schematically in Figure 5.1.

| Mission C-1 Launch Vehicle Configuration | | |
|--|-------------|-----------|
| Cargo Lander: | 55.0 | mT |
| ComSat: | 3.7 | mT |
| Service Module: | 21.0 | mT |
| Launch C-1-a Payload Mass: | 79.7 | mT |
| Launch C-1-b -- TMI Stage: | 79.9 | mT |

Table 5.1

The ComSat contains communications and solar power arrays for use both as the primary power and communications server during the outbound cruise, and as a Mars-Earth and Mars-Mars communications relay satellite. After aerocapture, the Service Module/ComSat separates from the Lander and flies to synchronous orbit. The ComSat is positioned 45° longitude west of the landing site. From this altitude, approximately 17,100 km above the surface, the satellite should be above the horizon within a 160° cone. Thus virtually all locations on the planet from 80° North latitude to 80° South latitude, and ± 80° of longitude from the landing site should be covered.

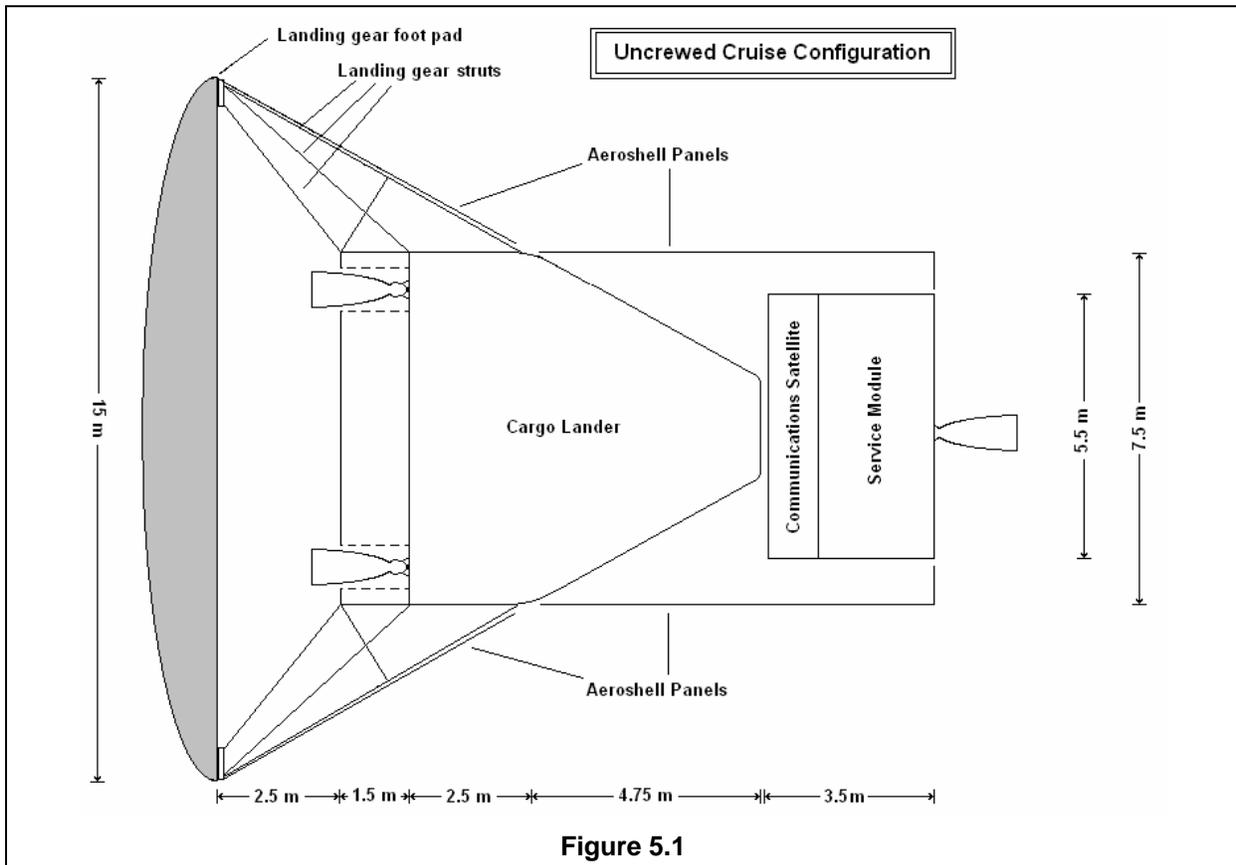


Figure 5.1

Because Mars' synchronous orbit is relatively close to Deimos' orbit (~ 3,000 km radial difference) and therefore subject to perturbations, significant station-keeping propellants have been provided in the ComSat. The actual degree of perturbations by both Phobos (~ 7,700 km radial difference) and Deimos require further study.

Once the communications satellite is positioned in synchronous orbit, the lander deorbits and lands. Most of its entry, descent and landing will be in line-of-sight with the communications satellite to facilitate tracking and data relay to Earth.

V.1 – The Cargo Lander

The Cargo version of the Common Lander carries two teleoperable robot exploration vehicles plus enough seed hydrogen to produce the crew's ascent propellants and the crew's potable water while on Mars. The mass budget for the Cargo lander is detailed in Table 5.2.

The cargo lander carries 4 metric tons of seed hydrogen on board plus processing equipment to extract oxygen from the Martian atmosphere using a YSZ (Yttria-stabilized zirconia) solid oxide electrolysis system. The oxygen thus produced and seed hydrogen react in a fuel cell to produce 36mT of water and 83,000 kwh of electricity over two years. The electricity supplements the lander's primary solar power grid. The water is stored as ice until the crewed lander arrives, at which point it becomes feedstock to the ascent vehicle's propellant manufacturing plant. Excess water is used by the crew for environmental purposes. The ISRU system is further detailed in Section VIII.

The teleoperable rover vehicles are used prior to final site selection to explore the chosen landing site for resources and for obstacles to further landings. The rovers are capable of clearing small obstacles from the landing site and, with their built-in radiolocation transponders, can help guide subsequent landers to a pinpoint landing.

After the crew arrives on mission M-1, the rovers can be converted for use by the crew. In this configuration, the rovers resemble open-cockpit four-wheel All-Terrain Vehicles (ATVs) common in use on Earth.

| Cargo Lander Mass Budget | |
|---------------------------------------|------------------|
| Volume of lander: | 270.3 m3 |
| Less Propellant Tank Volume: | (17.0) m3 |
| Less 5% for structure: | (13.5) m3 |
| Interior volume: | 239.8 m3 |
| Structure: | 3,592 kg |
| Landing gear: | 1,500 kg |
| Control + Avionics: | 500 kg |
| Power: | 200 kg |
| 20% Growth: | 1,058 kg |
| Engines: | 2,000 kg |
| Propellant Tanks: | 1,020 kg |
| Total dry mass: | 9,870 kg |
| Seed hydrogen: | 4,351 kg |
| Hydrogen tanks + plumbing: | 3,677 kg |
| CO2 electrolysis + fuel cells: | 355 kg |
| 20% Growth: | 806 kg |
| Cargo + Scientific Equipment: | 12,700 kg |
| Landed mass: | 31,758 kg |
| Propellants: | 14,149 kg |
| Heatshield: | 5,015 kg |
| Aeroshell: | 4,085 kg |
| Total Mass at Entry Interface: | 55,008 kg |

Table 5.2

V.2 – Mission C-1 Earth Orbit Assembly

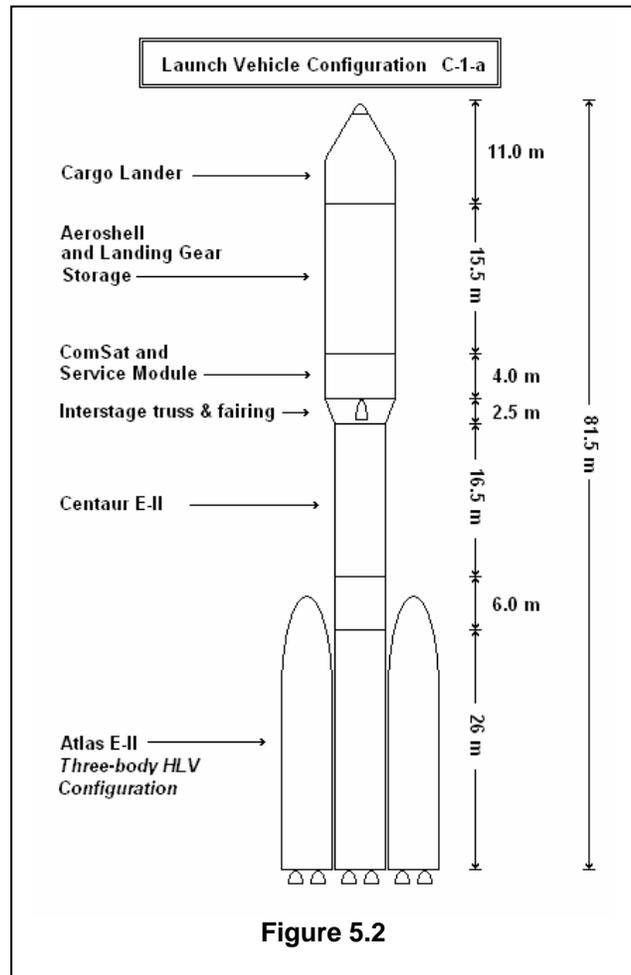
Mission C-1 requires two HLLV launches spread over four weeks. As this assembly is quite similar to that required in the following two missions, this presents us with a good opportunity to train the assembly crew and to prove the assembly procedures.

The first launch, designated **C-1-a**, is scheduled approximately four weeks prior to the Mars launch window. This launch carries the Cargo Lander, the Communications relay satellite, and the Service Module into low Earth orbit. Launch vehicle C-1-a is shown schematically in Figure 5.2.

Once on orbit, the Centaur booster stage separates and deorbits. The interstage truss joining the Centaur and the Service Module is retained on orbit, and will be used to dock the Trans-Mars Injection (TMI) stage to the stack. The lander separates from the stack, opens and locks its landing gear in the deployed position. The 15-meter diameter heat-shield, stowed in its folded attitude within the aeroshell, is unfolded and attached to the landing gear foot-pads as shown in Figure 4.1. Aeroshell panels are placed against the landing gear upper struts to complete a smooth conic surface from the heatshield up to the top of the lander.

The stack is assembled as shown in Figure 5.1. The Cargo Lander is mated to the Service Module with its heatshield forward. This heatshield will be used as the aerobraking device during aerocapture of the stack into Mars orbit.

The second launch of Mission C-1, designated **C-1-b**, carries the Trans-Mars Injection (TMI) stage to orbit. This launch is common to all three missions (referred to as C-1-b, H-1-b, and M-1-c). It is shown schematically in Figure 7.4. Once on orbit, it performs a rendezvous with the mission stack and docks to the Service Module end of the stack. Once this stage is docked at the rear of the Service Module the vehicle is ready for Earth departure.



VI – The Surface Habitat Mission (H-1)

Mission H-1 pre-positions the crew’s Surface Habitat at the designated landing site one cycle before the crewed mission. The Earth-departure stack consists of the Surface Habitat Lander, a communications relay satellite, and the Service Module. The mass summary of the H-1 mission is shown in Table 6.1.

The Mission H-1 Earth-departure stack is identical to the Mission C-1 stack shown in Figure 5.1. The stack aerocaptures into Mars orbit using the surface habitat’s heatshield.

The communications relay satellite contains communications and solar power arrays for use both as the primary power and communications server during the outbound cruise, and as a Mars-Earth and Mars-Mars relay. After aerocapture, the Service Module/ComSat separates from the Lander and flies to synchronous orbit. The ComSat is positioned 45° longitude east of the landing site. This positions the two comsats 90° apart, assuring that one comsat is always in line-of-sight with the Earth except for those few days during Sun/Mars conjunction.

| Mission H-1 Launch Vehicle Configuration | | |
|---|-------------|-----------|
| Habitat Lander: | 55.0 | mT |
| ComSat: | 3.7 | mT |
| Service Module: | 21.0 | mT |
| Launch H-1-a Payload Mass: | 79.7 | mT |
| Launch H-1-b -- TMI Stage I: | 79.9 | mT |

Table 6.1

VI.1 – The Mars Habitat Lander

The Mars Habitat version of the Common Lander flies on Mission H-1. The habitat is the crew’s primary living and working space while on the surface. The mass budget for the Mars Habitat lander is detailed in Table 6.2.

The surface habitat lander carries 6.4 metric tons of cargo and scientific equipment. Included in this cargo is a greenhouse experiment and an inflatable habitat extension. The greenhouse experiment is one of the primary focuses of the first crewed mission. It will establish the baseline requirements for one of MarsDrive’s primary goals – growing human-consumable food on Mars.

The inflatable habitat fits inside the spread of the landing gear under the lander itself. This structure provides the crew an additional 400 cubic meters of pressurized workspace which may be used for storing and maintaining outside equipment such as the crew’s rovers and geological equipment. This extra layer of

| Surface Habitat Lander Mass Budget | | |
|---|---------------|-----------|
| Volume of lander: | 270.3 | m3 |
| Less Propellant Tank Volume: | (17.0) | m3 |
| Less 5% for structure: | (13.5) | m3 |
| Pressurized Volume: | 239.8 | m3 |
| Volume per Crew Member: | 40.0 | m3 |
| Propellant Tanks: | 1,019 | kg |
| Engine Mass: | 2,000 | kg |
| Pressure Shell Mass: | 3,043 | kg |
| Structure: | 7,488 | kg |
| Landing Gear: | 1,500 | kg |
| ISRU + Power: | 500 | kg |
| Control + Avionics: | 200 | kg |
| Environmental: | 300 | kg |
| 20% Growth: | 2,606 | kg |
| Total dry mass: | 18,658 | kg |
| Cargo + Scientific Equipment: | 6,350 | kg |
| Crew Consumables: | 6,750 | kg |
| Landed mass: | 31,758 | kg |
| Propellants: | 14,149 | kg |
| Heatshield: | 5,015 | kg |
| Aeroshell: | 4,085 | kg |
| Total Mass at Entry Interface: | 55,007 | kg |

Table 6.2

protection also serves to reduce the dust problems at the airlock entrance to the habitat itself.

VI.2 – Mission H-1 Earth Orbit Assembly

Mission H-1 requires two HLLV launches spread over four weeks. The first launch, designated **H-1-a**, is scheduled approximately four weeks prior to the Mars launch window. This launch carries the Mars Surface Habitat, the communications relay satellite, and the Service Module into low Earth orbit. Launch vehicle H-1-a is shown schematically in Figure 6.2.

Once on orbit, the Centaur booster stage separates and deorbits. The interstage truss joining the Centaur and the Service Module is retained on orbit, and will be used to dock the Trans-Mars Injection (TMI) stage to the stack.

The lander separates from the stack, opens and locks its landing gear in the deployed position. The 15-meter diameter heat-shield, stowed in its folded attitude within the aeroshell, is unfolded and attached to the landing gear foot-pads as shown in Figure 4.1. Aeroshell panels are placed against the landing gear upper struts to complete a smooth conic surface from the heatshield up to the top of the lander.

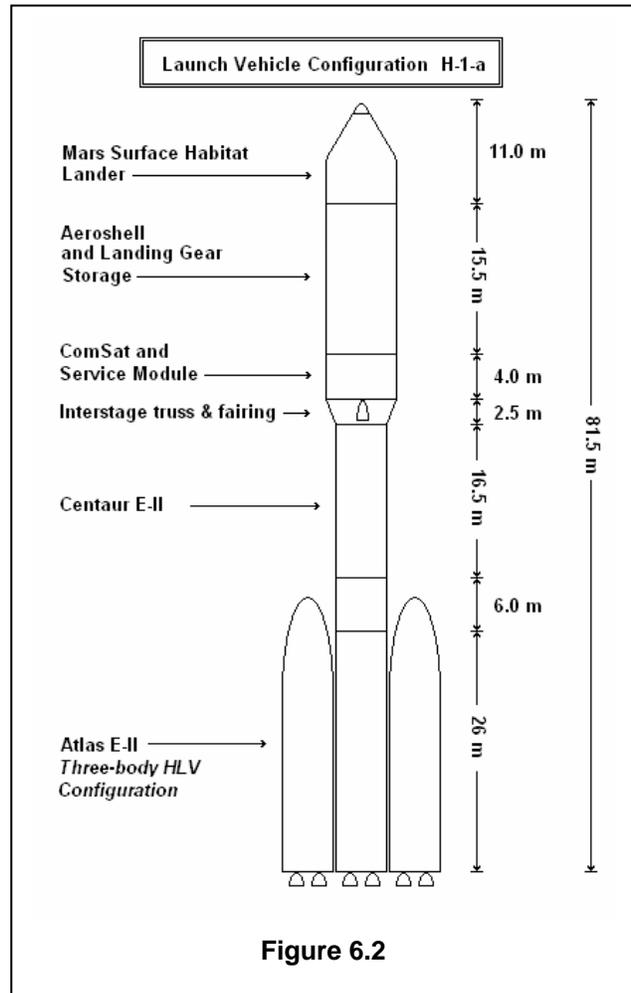


Figure 6.2

The stack is assembled as shown in Figure 5.1. The Surface Habitat Lander is mated to the Service Module with its heatshield forward. This heatshield will be used as the aerobraking device during aerocapture of the stack into Mars orbit.

The second launch of Mission H-1, designated **H-1-b**, carries the Trans-Mars Injection (TMI) stage to orbit. This launch is common to all three missions (referred to as C-1-b, H-1-b, and M-1-c). It is shown schematically in Figure 7.4. Once on orbit, it performs a rendezvous with the mission stack and docks to the Service Module end of the stack. Once this stage is docked at the rear of the Service Module the vehicle is ready for Earth departure.

VII – The Crewed Mission (M-1)

Mission M-1 is the crewed mission. This mission will not leave Earth until the Surface Habitat has successfully landed at the designated landing site and all its systems have been verified. The mass summary of the mission is shown in Table 7.1.

The Mission M-1 Earth-departure stack consists of the crewed Mars Lander, the in-space habitat, and the Service Module. Mission M-1 Outbound Cruise Configuration is shown schematically in Figure 7.1. The Aeroshell panels shown form a smooth surface for aerocapture. The lower aeroshell panels, attached to the landing gear struts, form a continuous conic section from the heatshield forward during the lander's entry into the Martian atmosphere.

| Mission M-1 Launch Vehicle Configuration | | |
|--|-------------|-----------|
| Crewed Lander: | 48.6 | mT |
| Cargo: | 1.6 | mT |
| Consumables: | 2.6 | mT |
| Service Module: | 27.1 | mT |
| Launch M-1-a Payload Mass: | 79.8 | mT |
| In-space Habitat + Earth heatshield: | 11.0 | mT |
| Consumables: | 3.8 | mT |
| TMI Stage II: | 65.1 | mT |
| Launch M-1-b Payload Mass: | 79.9 | mT |
| Launch M-1-c -- TMI Stage I: | 79.9 | mT |

Table 7.1

The Service Module provides the stack with all propulsive maneuvers after Trans-Mars Injection (TMI), including both outbound and inbound mid-course corrections, Mars entry alignment, Mars post-aerocapture orbital maneuvering, Trans-Earth Injection (TEI), and Earth-approach alignment. The Service Module and In-space Habitat loiter in Mars orbit until the crew arrives back from their sortie to the surface. The Service Module's solar panels provide power to

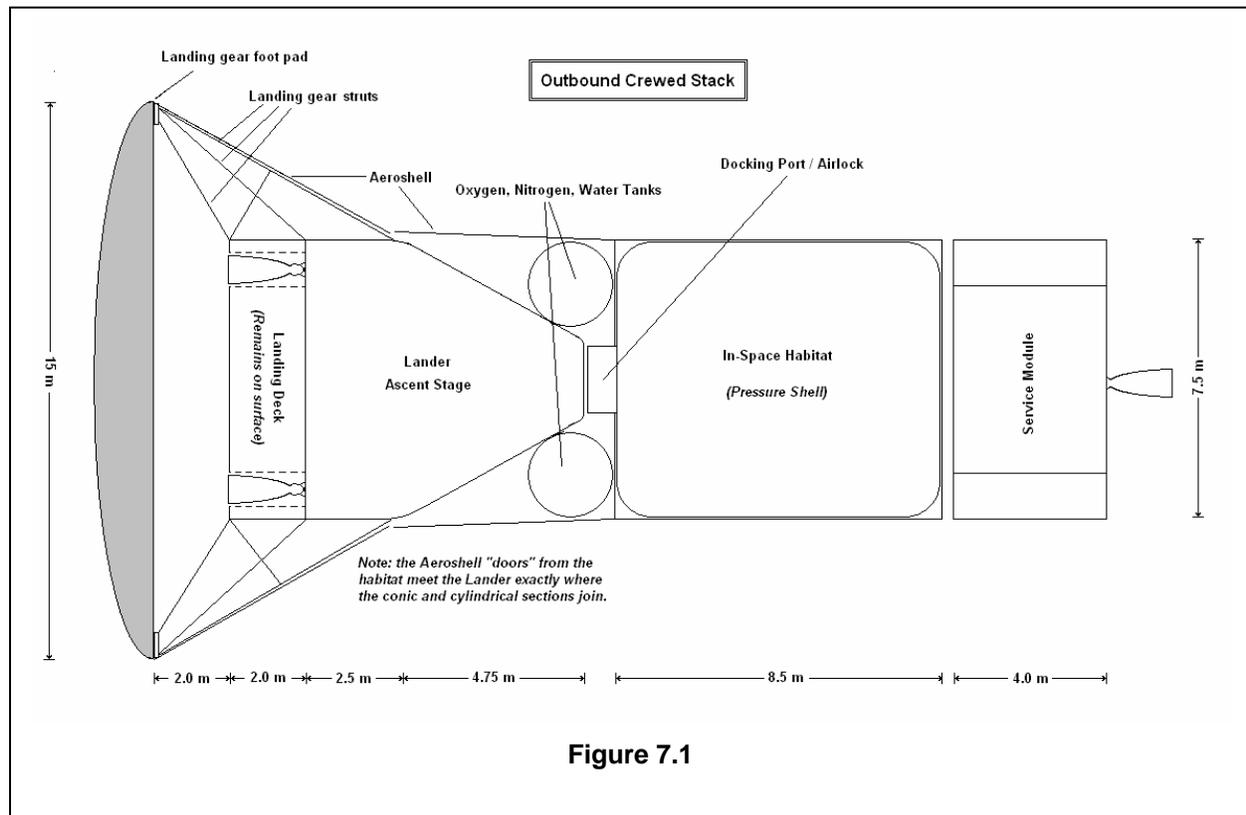


Figure 7.1

the entire stack except during the spun part of the outbound cruise. The Service Module also contains the primary wide-band communications system between the vehicle and Earth.

VII.1 –The Crewed Lander

The Crewed version of the Common Lander flies on Mission M-1. The crewed lander is composed of an **Ascent Vehicle** which carries the crew from the surface back to Mars orbit and a **Landing Deck** which remains on the surface. The two combined serve as the crew’s **Descent Stage**. The landing deck contains all equipment not required during ascent, orbital rendezvous, and docking, including the propellant *in situ* resource utilization (PISRU) plant. It also serves as the attachment point for the landing gear and as a stable platform for the launch of the ascent vehicle.

The ascent stage is the truncated 30° conic section and the top 2.5 meters of the cylindrical section. The total volume of both sections is 204 m³. Of this, 85 m³ is tankage to hold the ascent propellants. Prior to landing these tanks contain 12.7 mT of descent and landing propellants. While on the surface they are filled with 70.9 mT of oxygen and methane propellants by the on-board PISRU plant. The mass budget for the crewed lander is detailed in Table 7.2.

The descent stage consists of the ascent stage plus the landing deck. The ascent stage engines, fuel tanks, crew acceleration couches, controls and avionics are used for both the ascent and descent phases of the flight, thus saving the mass of duplicating these functions which are identical to both phases. This is a marked departure from other Mars mission designs which use separate ascent and descent vehicles and results in significant mass savings.

Once on the surface, the crewed lander augments the Surface Habitat with additional crew living and working space. It can also serve as an emergency shelter should the surface habitat experience a systems failure which renders it uninhabitable.

| Crewed Lander Mass Budget | |
|---------------------------------------|------------------|
| Volume of ascent vehicle: | 204.3 m3 |
| Less Propellant Tank Volume: | (85.1) m3 |
| Less 5% for structure: | (10.2) m3 |
| Pressurized Volume: | 109.0 m3 |
| Volume per Crew Member: | 18.2 m3 |
| Ascent Stage: | |
| Internal structure: | 5,659 kg |
| Pressure shell mass: | 3,043 kg |
| LIDS: | 395 kg |
| Power: | 500 kg |
| Control + Avionics: | 200 kg |
| Environmental: | 300 kg |
| 20% Growth: | 2,020 kg |
| Engines: | 2,000 kg |
| Propellant Tanks: | 5,105 kg |
| Crew + Cargo: | 8,800 kg |
| Propellants: | 70,855 kg |
| Lift-off Mass: | 98,877 kg |
| Descent Stage: | |
| Landing Deck structure: | 477 kg |
| Landing gear: | 1,500 kg |
| PISRU + power: | 3,000 kg |
| 20% Growth: | 995 kg |
| Ascent Stage Dry Mass: | 19,222 kg |
| Crew + Cargo: | 3,250 kg |
| Propellants: | 12,673 kg |
| Aeroshell: | 4,085 kg |
| Heatshield: | 5,015 kg |
| Total Mass at Entry Interface: | 50,219 kg |

Table 7.2

VII.2 – Mission M-1 Earth Orbit Assembly

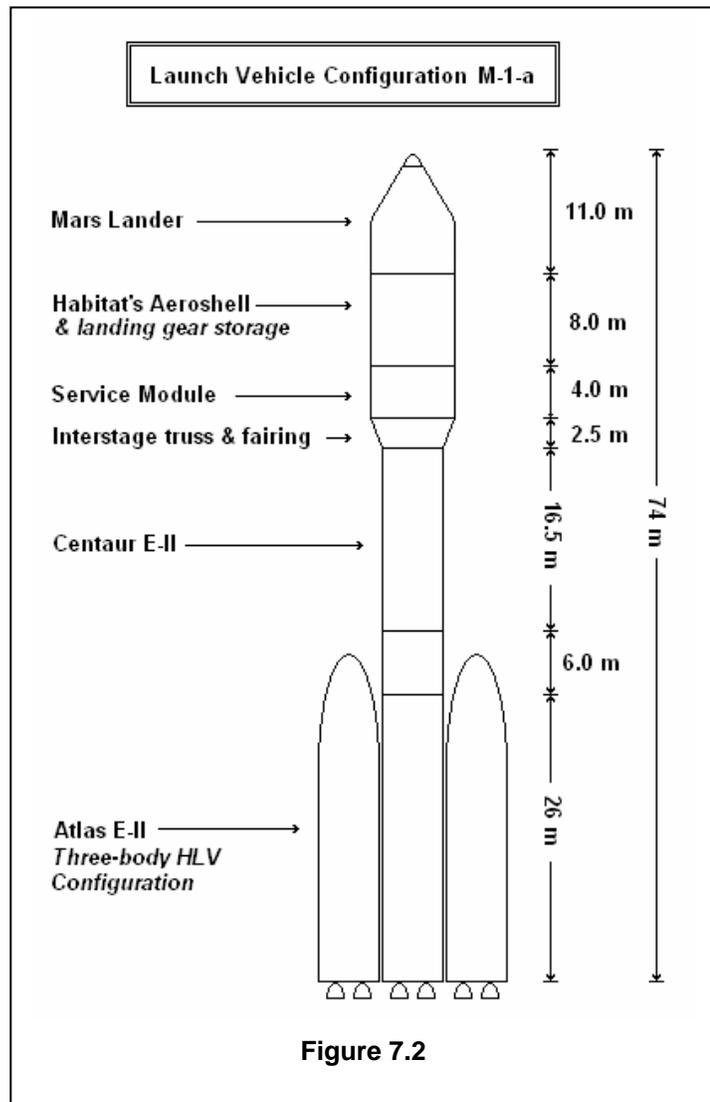
Mission M-1 requires three HLLV launches spread over eight weeks and one crew-launch vehicle. The SpaceX Falcon/Dragon vehicle or its equivalent has been selected as a place-holder crew-launch vehicle.

The first launch, designated **M-1-a**, approximately nine weeks prior to the Mars launch window, carries the Mars Lander and the Service Module into low Earth orbit. Launch vehicle M-1-a is shown schematically in Figure 7.2.

Once on orbit, the Centaur booster stage separates and deorbits. The interstage truss joining the Centaur and the Service Module is retained on orbit, and will be used to dock the Trans-Mars Injection (TMI) stages to the stack.

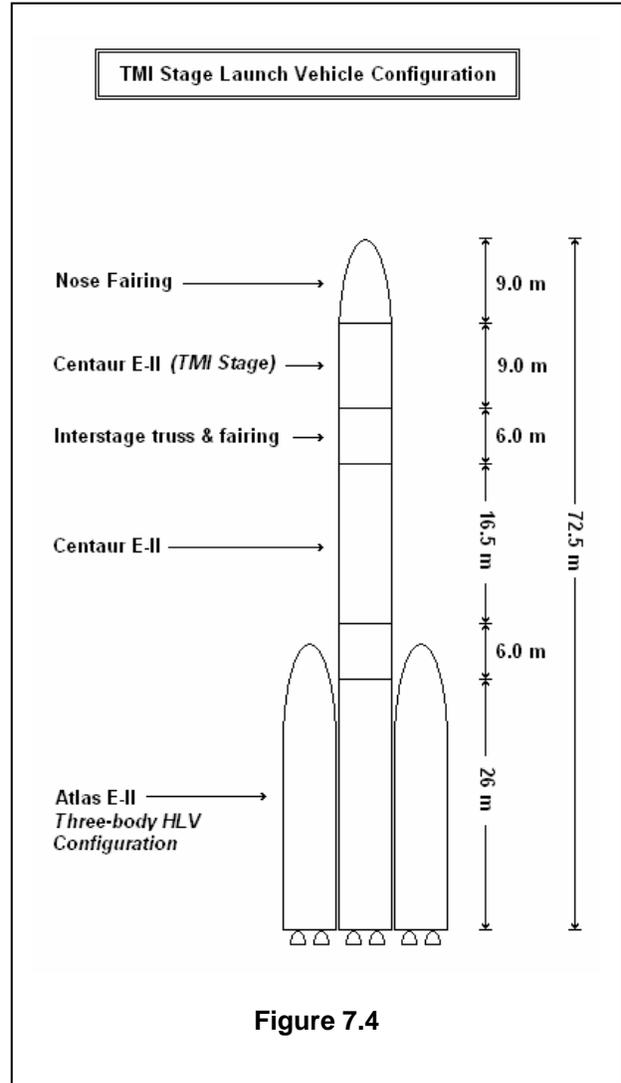
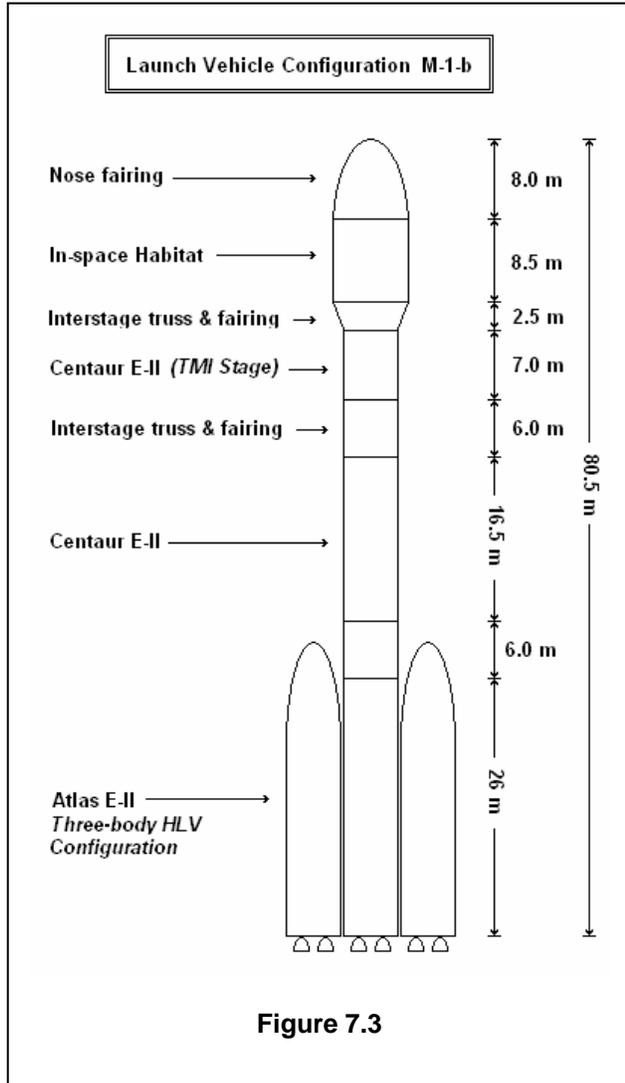
The second launch of Mission M-1, designated **M-1-b**, carries the In-space Habitat and the second stage of the Trans-Mars Injection (TMI) booster. Launch vehicle M-1-b is shown schematically in Figure 7.3.

The crew arrives after the successful launch of M-1-b and its rendezvous with the M-1-a payload. The lander is separated from the stack. Its landing gear is opened and locked in the deployed position. The 15-meter diameter heat-shield, stowed in its folded attitude within the aeroshell, is unfolded and attached to the landing gear foot-pads as shown in Figure 4.1. Stowed aeroshell panels are placed against the landing gear upper struts to complete a smooth conic surface from the heatshield up to the top of the lander.



The stack is assembled as shown in Figure 5.1. The Crewed Lander is mated to the in-space habitat with its heatshield forward. This heatshield will be used as the aerobraking device during aerocapture of the stack into Mars orbit. The TMI Stage II booster is mated to the end of the Service Module.

The third launch of Mission M-1, designated **M-1-c**, carries the first stage of the Trans-Mars Injection (TMI) booster. Launch vehicle M-1-c is shown schematically in Figure 7.4. This launch is common to all three missions (referred to as C-1-b, H-1-b, and M-1-c). Once on orbit, it performs a rendezvous with the mission stack and docks to the rear of the TMI Stage II booster. Once this stage is docked, the vehicle is ready for Earth departure



VII. 3 – The In-Space Habitat

The In-space Habitat houses and sustains the crew during the transits from Earth orbit to Mars orbit and from Mars orbit back to the vicinity of Earth. The habitat is parked in an elliptical orbit around Mars while the crew sorties to the surface. The In-space habitat’s mass budget is shown in Table 7.3.

The habitat is a cylinder 7.5 meters in diameter and 8.5 meters in length. This provides 350 cubic meters of habitable volume, 60 cubic meters per crew member. The cylinder consists of a 5mm aluminum pressure shell, 1.5cm of polyethylene sheathing for additional radiation protection, and an outer shell of graphite epoxy/BMI composite to provide hull integrity, micrometeoroid protection, and additional radiation protection. Internal equipment is attached to the inside of the pressure hull to provide additional protection. All together this should provide an equivalent 20g/cm² of protection, at the top end of the 15–20g/cm² range recommended by NASA.

A Low-Impact Docking System (LIDS) is provided at the forward end of the cylinder to allow docking with the Mars Lander, the crew Earth-launch vehicle, and the Earth Re-entry Vehicle.

The environmental control system consists of air and water management and food storage and preparation systems. The crew’s food, water, and oxygen needs are not accounted for in the Habitat’s mass budget. They are accounted for in the crew’s mass budget.

Because the total oxygen demand for a crew of six for a 260-day transit is only 1800 kg, it was decided to carry the crew’s entire oxygen needs, rather than relying on a complex oxygen recycling system. This part of the *Environmental* line item in the mass budget consists of the air handlers and CO₂ scrubber. Carrying the entire oxygen supply also greatly reduces the overall risk of this phase of the mission.

Water requirements are an entirely different matter. The water requirement for a crew of six for a 260-day transit is 44 mT, making it impractical to carry. However water recycling is a well-understood off-the-shelf technology. This design assumes a water recycling system based on a reverse osmosis process which is 95% efficient. While this is above current efficiency levels, ongoing research is expected to produce 98% efficient units in as little as five years, so assuming 95% in 20 years seems prudent.

| In-Space Habitat Mass Budget | |
|-------------------------------------|---------------------|
| Diameter: | 7.5 m |
| Length: | 8.5 m |
| Surface Area: | 200 m ² |
| Total Pressurized Volume: | 357 m ³ |
| Pressurized Volume per crew: | 59.5 m ³ |
| Furnishings per crew member: | 150 kg |
| Shell Mass: | 2,766 kg |
| Internal Structure: | 2,000 kg |
| Airlock: | 250 kg |
| LIDS: | 395 kg |
| Furnishings: | 900 kg |
| Power: | 200 kg |
| Control + Avionics: | 200 kg |
| Environmental: | 500 kg |
| 20% Growth: | 1,442 kg |
| Earth Aerocapture Heatshield mass: | 2,379 kg |
| Total Inbound Habitat Mass: | 11,032 kg |
| Artificial Gravity Assembly: | 3,234 kg |
| Total Outbound Habitat Mass: | 14,266 kg |

Table 7.3

For this study it was assumed the habitat will be spun to produce 0.38 standard gravities (1 Mars surface gravity) for most of the outbound cruise phase of the trip. Approximately ten days after Trans-Mars Insertion, after the initial mid-course correction propulsive maneuver, the habitat and the Lander will be undocked and separated by a 118-meter long tether consisting of four 7 x 19 SSAC steel cables, each sized so any two cables can bear the tensile load of the spinning assembly. The cables are spaced two meters apart and are tethered to each other along their length by spreaders. This is designed to prevent a sheared cable from whiplashing and penetrating either the habitat or the lander.

The Service Module is undocked and flown to a position several hundred meters to the solar north of the habitat. The habitat and lander are spun up to 2 rpm, providing the desired 0.38 gees to the habitat. The Service Module functions as a stabilized platform for the high-gain Earth communications antenna. High-resolution television cameras can provide an external view of all parts of the spun stack. Communications between the Service Module and the Habitat are by omnidirectional antennas.

The habitat depends on solar cell arrays for its power. A power demand budget for the habitat has not been determined. During the spun phase of the outbound cruise, power is provided by solar arrays stretched along the tethers connecting the habitat and the Lander. During the despun phase of the outbound cruise, the time loitering in Mars orbit, and the entire inbound cruise, habitat power is provided by the Service Module.

VII.4 – The Service Module

The M-1 Service Module provides several functions during the outbound cruise and while the crew is deployed to the surface including propulsion, power, and communications. Its primary function is to perform all propulsive maneuvers after Trans-Mars Injection (TMI) including outbound mid-course corrections, Mars entry alignment, Mars orbit alignment (post aerocapture), Trans-Earth Injection, inbound mid-course corrections, and Earth approach alignment. The Service Module mass budget is shown in Figure 7.4.

The propulsive core of the Service Module is derived from the Centaur E-II stage described in Section III. This paper assumes the stage has been fitted with two RL-10-B-2 engines instead of its usual RL-10-A engines. The 5.5 meter diameter stage is centered in a 7.5 meter diameter fairing which serves as a sun shield to help with propellant thermal management. Propellant management is handled by a solar powered active cryocooler to permit retention of the liquid oxygen and liquid hydrogen propellants over the approximate 3 year mission duration.

| Service Module Mass Budget | | |
|-----------------------------------|--------------|-----------|
| Structure: | 1,743 | kg |
| Avionics + Solar Power: | 1,000 | kg |
| 20% Growth: | 549 | kg |
| Engine Mass: | 604 | kg |
| Tankage: | 3,330 | kg |
| Service Module Dry Mass: | 7,225 | kg |
| Total Propellant: | 19,844 | kg |
| Service Module Wet Mass: | 27,069 | kg |

Figure 7.4

The Service Module is the primary communications link between the M-1 vehicle and Earth. Communications between the Service Module and Earth is via a high-gain full duplex K_u-band system. Communications between the Service Module and the in-space habitat is via UHF (voice & data) and S-band (video) systems. Because of the short separation between them (~200 meters), these links can be low power using omnidirectional antennas.

The Service Module contains furlable solar panels which supply power to the active propellant management system, to the primary Earth communications system, and to the in-space habitat during those phases of the mission when the habitat is spun down and the Service Module is docked. The solar panels and the high-gain X-band antenna are furled and stowed within the Service Module's outer fairing during Earth ascent, during Mars aerocapture, and during Earth aerocapture.

High resolution video cameras provide celestial navigation inputs to the crew. These cameras may be used by the crew and by Mission Control to observe the exterior of the spun stack.

While the crewed habitat and the lander are spun to provide 0.38 standard artificial gravities during the outbound cruise phase of the mission, the Service Module flies separate from but in formation with the spun stack to provide a stable communications platform. At the beginning and end of the outbound cruise, during layover in Mars orbit, and during the entire inbound cruise, the Service Module is docked with the in-space habitat and provides both power and communications. While the crew is deployed to the Martian surface, the Service Module is available as a backup communications system with Earth should both of the communications satellites left in synchronous orbit by the C-1 and H-1 missions fail.

VII.5 – The Outbound Cruise Profile

The Earth-departure stack consists of the In-flight Habitat, the Crewed Lander, and the Service Module. The Mars surface habitat and rover(s) have been pre-positioned. This profile assumes that it has been decided to provide the crew with 0.38 standard gravities for the majority of the outbound cruise by spinning the Habitat with the Lander as its counterweight. If this turns out to be unnecessary, the stack will simply stay joined throughout the cruise phase of the flight.

After the TMI propulsive maneuver, the Earth Departure Stage separates itself from the stack and self-disposes into a solar orbit which will not intersect Earth or Mars for an arbitrarily long time. The Service Module solar wings are opened and supply power to the entire system until separation. The Service Module K_u band high-gain antenna is deployed and the communications link to Earth is activated. While in microgravity, the crew unpacks the stowed cruise materials and installs them in pre-fitted assemblies.

After ten to fourteen days of cruise the vehicle's course has been accurately determined by ground computers based on data from the on-board transponder and the first mid-course propulsive maneuver is determined. This maneuver is performed by the Service Propulsion System, after which the engine is purged and the propellant management system is activated. This system maintains the two cryogenic propellants in a near-zero boil-off state by use of passive radiation and active cryocooling. The radiators line the back of the solar panels.

Four 17X9 high-tensile-strength steel cables are fed from spools on the periphery of the docking adapter and attached to eyes in the Mars Lander. The cables are sized so any two of the four can support the spinning Habitat and Lander. The Lander is undocked from the Habitat and slowly moved to a point 118 meters away, stretching the four cables. Cable spacers are placed every 5 meters.

The Habitat and Lander are positioned so the intended axis of rotation passes through the Sun to maximize solar insolation. The solar array is unfurled and attached to two of the tether cables so it is aimed at the Sun.

The Service Module separates from the stack and flies to a position several hundred meters to the Solar North of the habitat. This positions it just out of the plane of rotation of the spun stack so it can observe all parts of the stack and so neither the Service Module nor the stack shadows each other.

Using their reaction-control thrusters, the vehicle is spun up to 2 RPM to provide the desired artificial gravity. The reaction-control thrusters are used to precess the vehicle as it moves in its trajectory to keep its axis of rotation aimed at the Sun.

A detailed vibration mode analysis of the tether system remains as forward work. Also, other DRMs which have proposed using non-rigid tethers have ignored the problem of the sudden separation of a tether while under spin. Provision must be made for the emergency separation of the tether should an undampable vibration mode occur. It is also possible that a tether will break unexpectedly while under a very strong tensile load. In either case, provisions must be made for dealing with the whiplash such that neither the Habitat nor the Lander are struck. Impact by a whiplashing steel cable would surely penetrate either vehicle's pressure hull.

Fifteen days before the vehicle reaches Mars it is spun down. The Service Module is docked and again powers the Habitat. The Lander is drawn towards the Habitat as the tether cables are re-coiled. Lander is docked to the docking port. The artificial gravity tethers, winches, and solar arrays are discarded. The Mars Approach Alignment maneuver is performed by the Service Module to align the vehicle for the first aerocapture pass.

VII. 6 – Aerocapture of the M-1 Stack

At the onset of aerocapture the crewed stack masses 85.2 mT as detailed in Table 7.5. The Service Module is assumed to have spent 250 m/s of propellants prior to aerocapture for course corrections and most of the crew’s food, water, and oxygen have been consumed and discarded overboard.

| Mission M-1 Mars Approach Mass Budget | | |
|---------------------------------------|-------------|-----------|
| Crewed Lander: | 48.6 | mT |
| In-space Habitat: | 11.0 | mT |
| Crew + Personal Items: | 1.5 | mT |
| Cargo: | 1.6 | mT |
| Consumables: | 0.2 | mT |
| Service Module: | 22.3 | mT |
| Total mass at Mars Approach: | 85.2 | mT |

The initial aerocapture pass is designed to place the stack into a highly-elliptical orbit. Based on the simulated Earth-Mars transfer trajectory and a reasonable aerocapture alignment maneuver, this initial pass will place the stack in an orbit with an apoapsis of 3,171 km. Details of the initial pass are shown in Figure 7.5. An apoapsis ΔV of +30 m/s will raise the orbital periapsis to 181 km, lifting it out of the Martian atmosphere.

Table 7.5

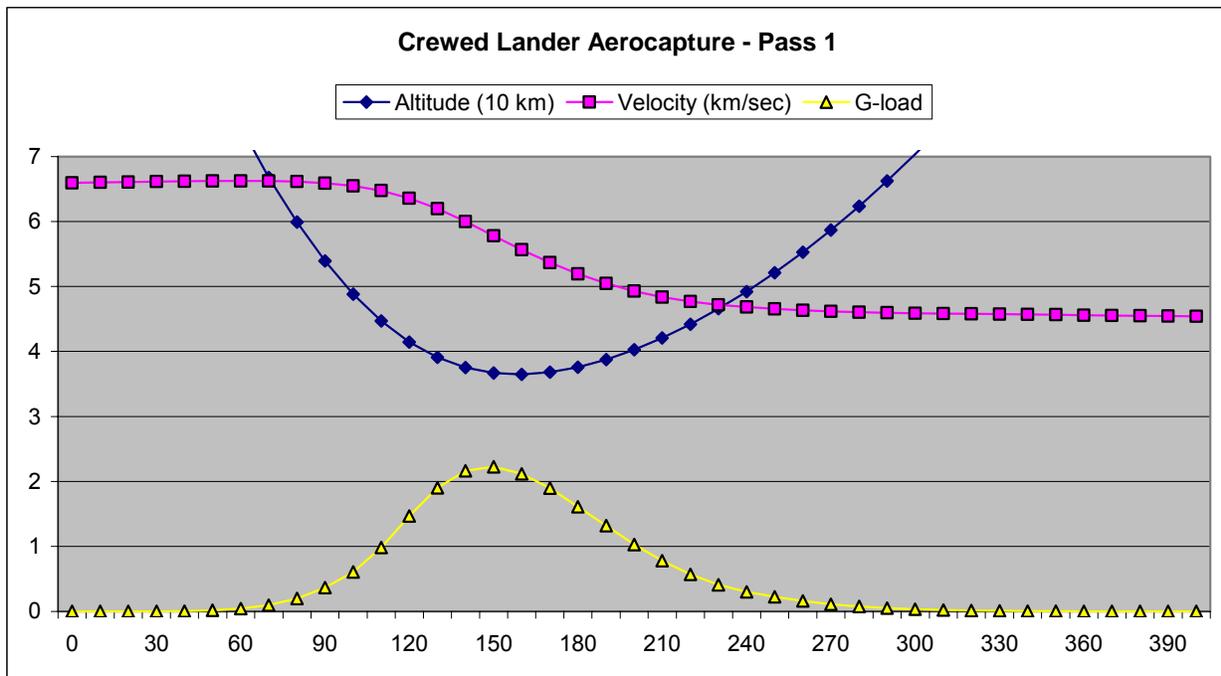


Figure 7.5

The crewed mission will leave the in-space Habitat and Service Module in an orbit which can conveniently be reached by the ascent vehicle at the end of the crew’s surface stay. Achieving this orbit will probably require changes in all orbital parameters – apoapsis, periapsis, phase, and plane. The first three parameters can be accomplished by loitering in the initial highly-elliptical orbit until the orbital phases align, then performing a maneuver to equalize the elliptical

parameters. The plane-change maneuver most likely will require a significant propulsive maneuver. The Service Module budgets 250 m/s ΔV of propellants to accomplish these maneuvers.

The elliptical changes are accomplished with a second aerocapture pass. A retrograde apoapsis ΔV of -24 m/s will drop the periapsis back into the atmosphere for a second aerocapture pass. This pass will reduce the orbital apoapsis to 1015 km. A subsequent apoapsis ΔV of +30 m/s will then raise the orbital periapsis to 184 km. Details of this aerodynamic pass are shown in Figure 7.6.

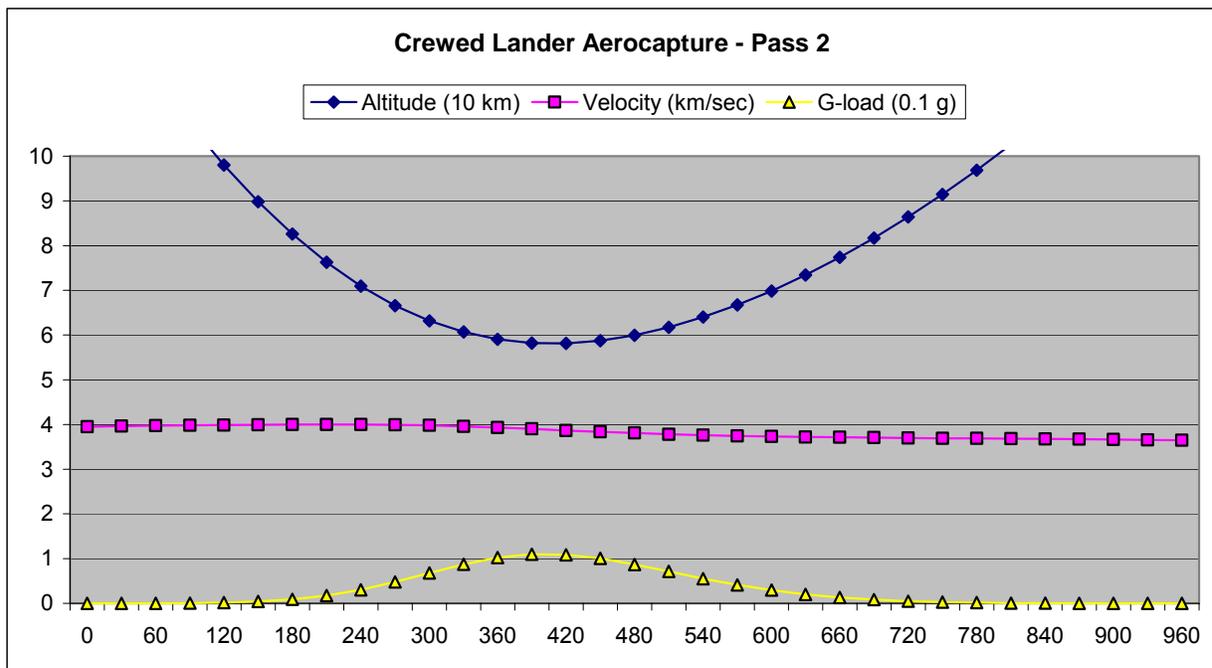


Figure 7.6

VII. 7 – Mars Orbital Operations

Following the second aerocapture maneuver the crew prepares for their descent to the surface. The Lander is activated and checked out. Surface cargo and supplies in the habitat are transferred to the Lander and stowed. The habitat is quiesced. The habitat and service module will remain in orbit while the crew is deployed to the surface.

Finally, the crew transfers to the Lander and undocks from the habitat and Service Module. The Lander deorbits with a retrograde 50 m/s ΔV burn. The Common Lander entry, descent, and landing profile is discussed in Section 4.5.

VII.8 – Mars Surface Operations

Mars surface operations center around the pre-positioned Surface Habitat and rovers. The Mars Surface Habitat is shown schematically in Figure 7.7.

The habitat provides 240 cubic meters of habitable pressurized volume, 40 m³ for each crew member. The habitat's manifest includes an inflatable structure which fits below the lander body in the space bounded by the four landing legs. The four landing legs and the body of the habitat itself support and protect the inflatable. This inflatable provides another 400 m³ of pressurized work space and provides an additional barrier to prevent dust from entering the main body of the habitat. An artist's rendering of the Mars surface habitat with the inflatable in place is shown in Figure 7.8.

The habitat contains equipment for electrolysis of atmospheric carbon dioxide (CO₂) into oxygen (O₂) and carbon monoxide (CO) using the *solid-oxide electrolysis* process discussed in **Section XXX**. The products may be liquefied and stored in the empty descent propellant tanks for use as environmental oxygen and back-up power production using a CO-O₂ fuel cell or internal combustion engine.

Detailed Mars surface operations remain as forward work.

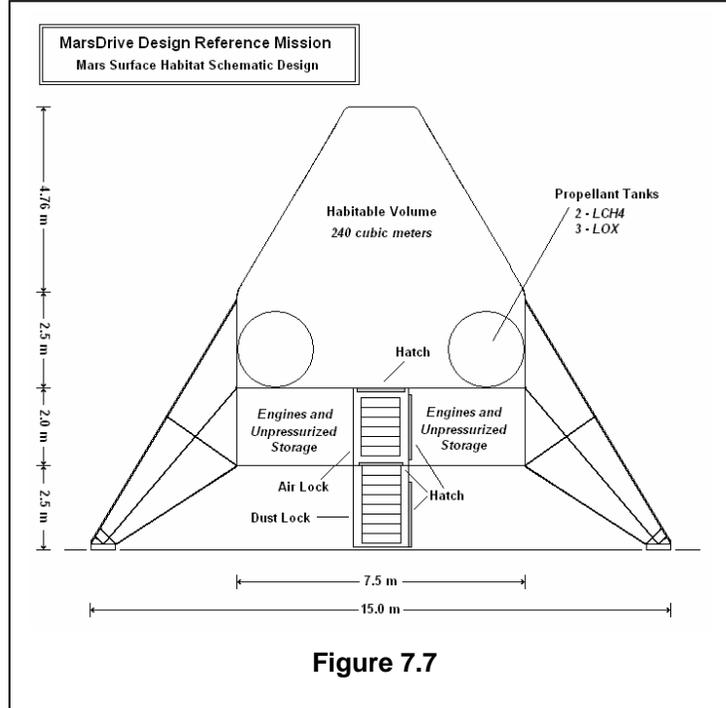


Figure 7.7

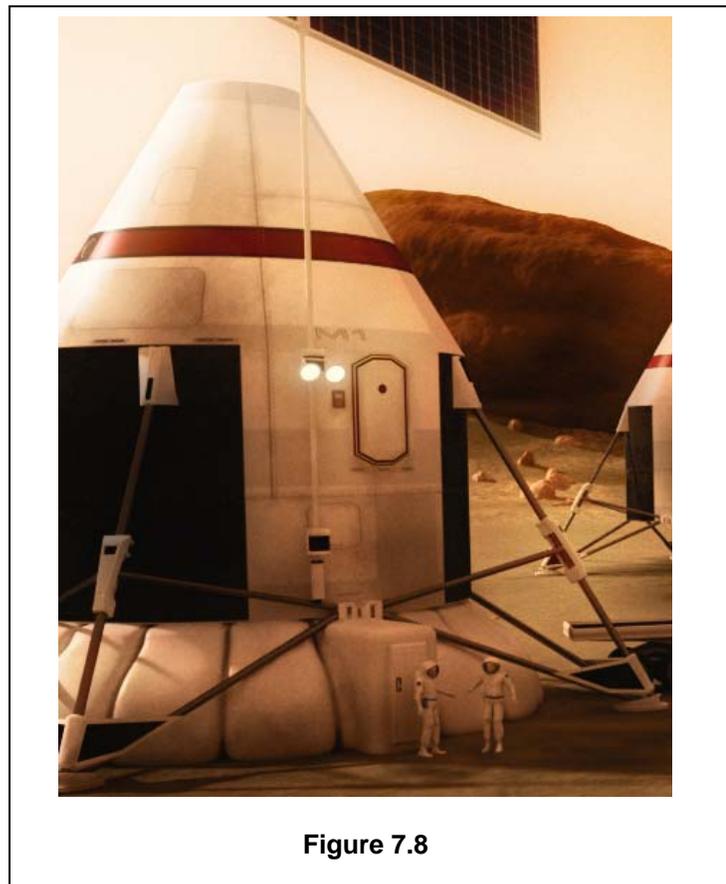


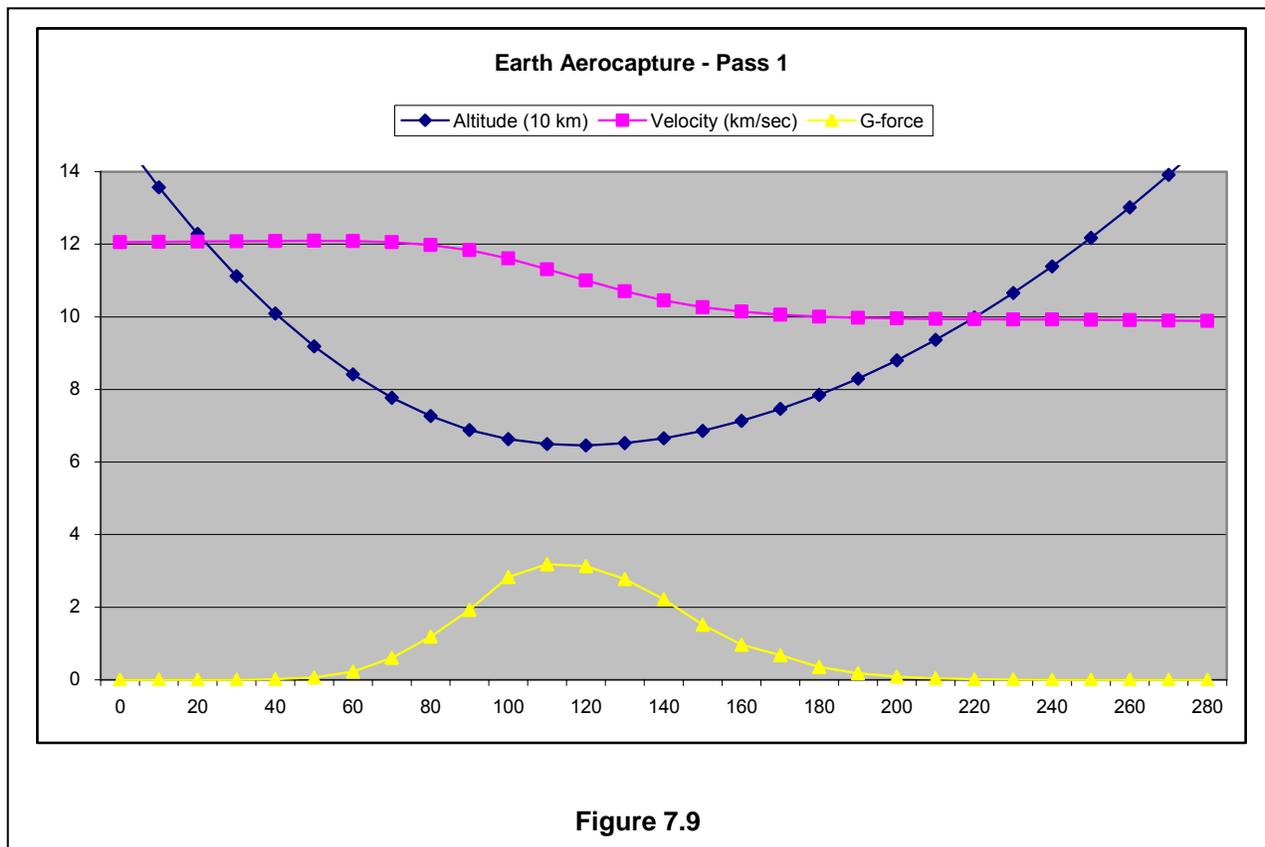
Figure 7.8

VII.9 – The Inbound Cruise Profile

At the end of their stay on the Martian surface, the crew returns to orbit using the Ascent Stage of the Mars Lander as described in Section VII.1. The ascent stage docks with the loitering in-space habitat and service module. The habitat is reactivated and checked out. Supplies for the return trip, surface samples, and cargo are transferred to the in-space habitat. The ascent stage is discarded. The Service Module performs the Trans-Earth Injection burn to put the vehicle into the proper trajectory for the return trip. After ten to fourteen days a mid-course correction maneuver is performed. The service module propulsion system is then purged and the propellant management system re-activated.

Five days from Earth the service module propulsion system performs the Earth Entry Alignment burn to align the vehicle for Earth aerocapture. Two aerocapture passes are assumed to minimize the g-loads on both the crew and the spacecraft. The two aerocapture passes are shown in Figures 7.9 and 7.10.

Following the second aerocapture pass the crew is met by a crew transfer vehicle launched from Earth to rendezvous with the orbiting habitat. The crew return to Earth in the transfer vehicle.



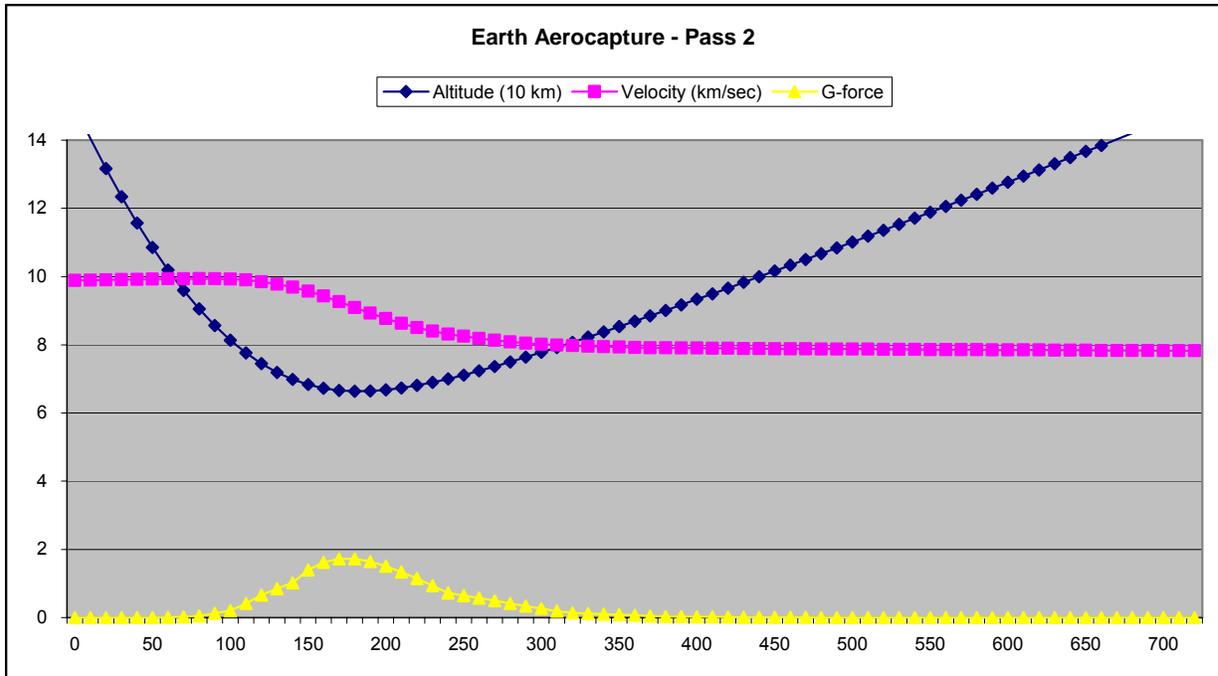
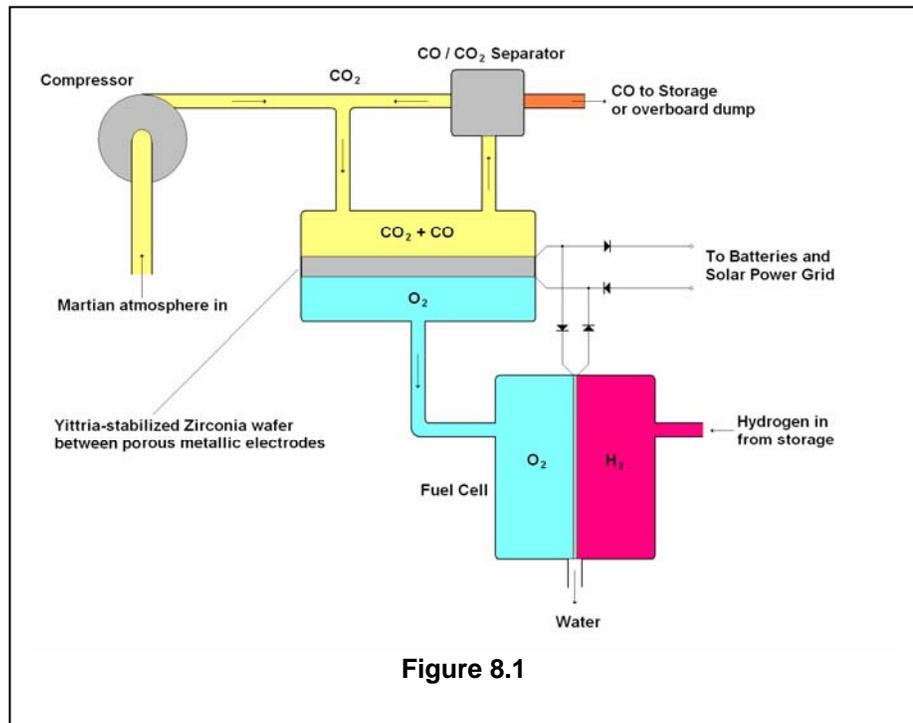


Figure 7.10

VIII – In Situ Resource Utilization

While the first crewed mission carries its entire food supply plus the seed hydrogen necessary to produce both drinking water and the ascent stage propellants, this mission relies completely on being able to use the Martian atmosphere to produce environmental oxygen, ascent oxygen propellant, plus the carbon component of the ascent methane (CH_4) propellant. One of the major goals of this mission is to locate reliable, accessible sources of water so future missions will be free of the need to carry hydrogen from Earth.

The ISRU plant carried on the C-1 mission uses 4.4 mT of stored hydrogen to produce 39 mT of water which is stored as ice until the arrival of the M-1 mission. This ISRU plant is shown schematically in Figure 8.1. The ISRU plant carried on mission H-1 duplicates the first part of this system, storing the O_2 and CO products in the lander's now-empty propellant tanks. The ISRU plant carried in mission M-1 uses the water produced by C-1 plus atmospheric CO_2 to produce 71 mT of ascent propellants. Detailed analysis of the propellant and environmental ISRU plants remain as forward work.



IX – Simulating Aerocapture and Aeroentry

In order to evaluate the Common Lander’s aerocapture, entry, descent, and landing characteristics, a simulation tool was developed to accurately predict the behavior of a spacecraft during entry into the Martian atmosphere. This tool added an atmospheric model and lift/drag calculations to a pre-existing tool simulating spacecraft flight dynamics in the Sun-Earth-Mars gravitational environment.

The NASA Mars Global Reference Atmospheric Model 2005 was used to create a mid-season, mid-latitude density profile of the Martian atmosphere below 140 km altitude. Inputting different seasonal and latitudinal data to Mars-GRAM-05 produce surprisingly different density profiles. This one was chosen to produce a mid-season, low-latitude model which is representative of a reasonable entry profile. The atmospheric density model used is shown logarithmically in Figure 9.1.

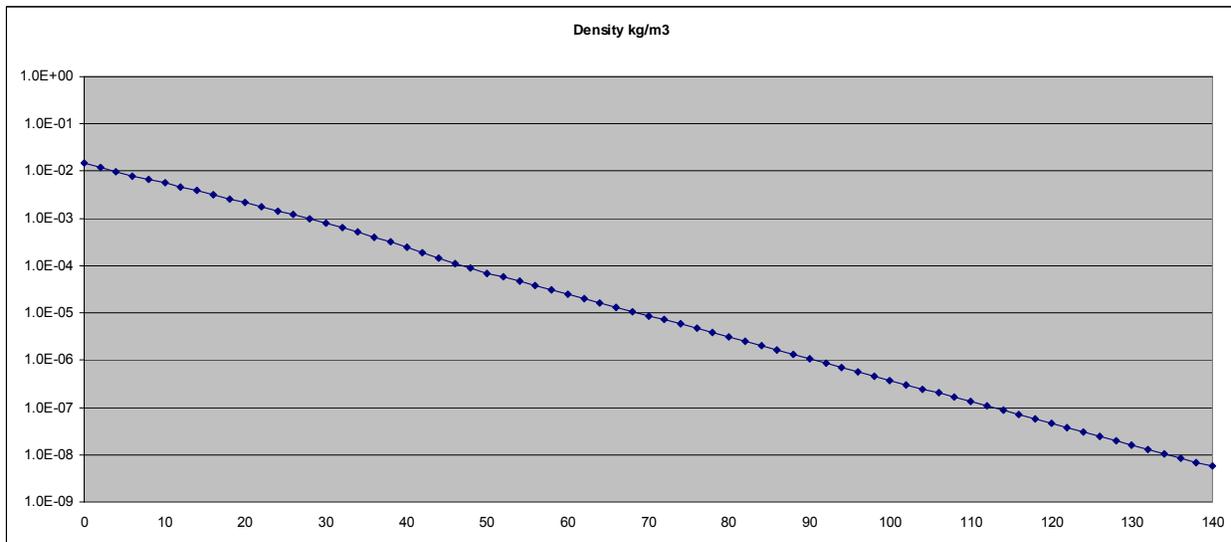


Figure 9.1

The simulator was qualified by reconstructing the known entry profiles of three legacy Mars missions – Viking I, MER-A “Spirit” and Mars Pathfinder.

IX.1 – Simulator Aerocapture Validation

Before any tool can be reliably used its results must be validated. For a simulator, the standard accepted validation methodology involves the simulation of one or more events whose outcome is already known, a process known as *Reconstruction Validation*.

Because there are no actual cases of successful aerocapture at Mars, this proved to be a challenge. As the best alternative available, we chose to reconstruct an aerocapture simulation done by simulation tools available within NASA. While this is not sufficient validation on which to rely for an actual mission, at this early stage in our mission design it should be sufficient.

In its initial design, the Mars Surveyor Program 2001 Orbiter (MSP-01) was to use aerocapture directly from its inbound transfer trajectory (i.e. – without a propulsive capture). This mission was studied carefully by NASA before being discarded in favor of a pure propulsive capture. Why this decision was made is speculative, but evidence points to a lack of faith in the then-existing Martian atmospheric models. Subsequent data obtained from various orbiting observers and from actual Entry, Descent and Landing (EDL) missions such as the two Mars Exploration Rovers have led to more reliable atmospheric models, culminating in the Mars-GRAM-05 model used herein.

The best available data from the NASA MSP-01 aerocapture studies came from Dr. Robert Braun of Georgia Tech⁶. Table 9.1 shows the physical parameters of the vehicle at Entry Interface.

| MSP-01 Entry Interface Parameters | |
|--|---------------------|
| Mass: | 554 kg |
| Drag Coefficient: | 1.68 |
| Ballistic Area: | 5.52 m ² |
| L/D Ratio (Up): | 0.30 |
| Entry Altitude: | 140 km |
| Entry Velocity: | 6600 m/s |

This data was used as the initial conditions for the simulation, which was allowed to run for 300 seconds of mission time. The resulting Altitude, Velocity, and Acceleration curves are shown in Figure 9.2.

Table 9.1

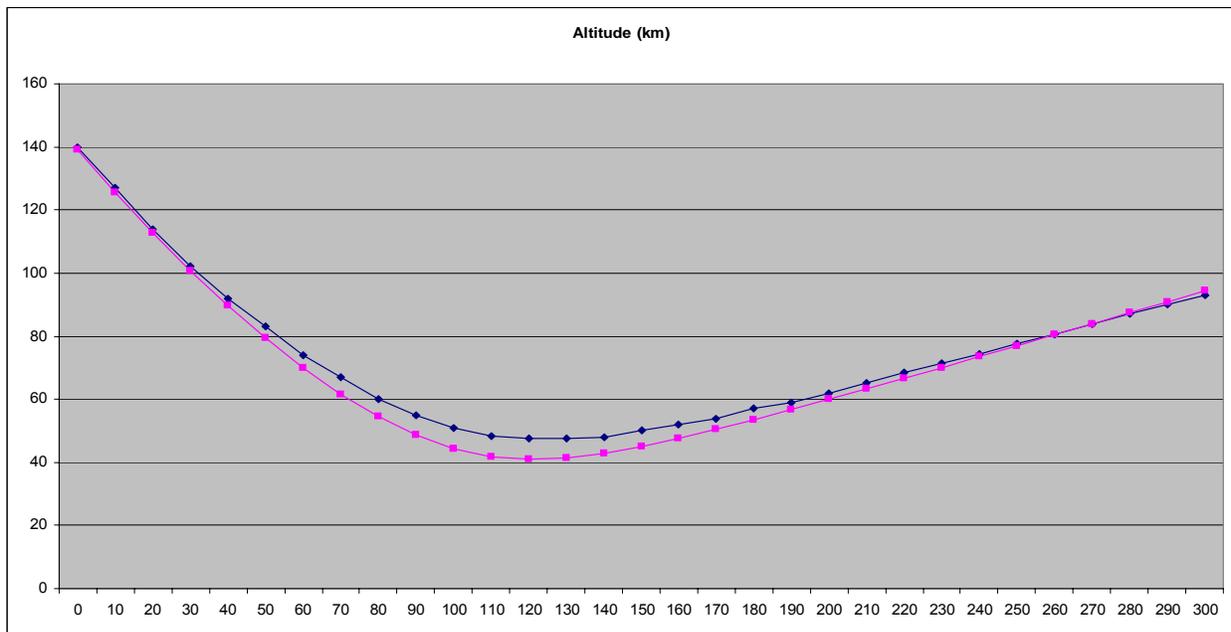


Figure 9.2

Figure 9.2 shows the comparison between the published MSP-01 altitude data and the data calculated by the simulator. In all figures, the MSP-01 data is in black with data points shown as black diamonds. The simulator output data is shown in red with square data points. The RMS of the difference between the two curves is 3.72 km.

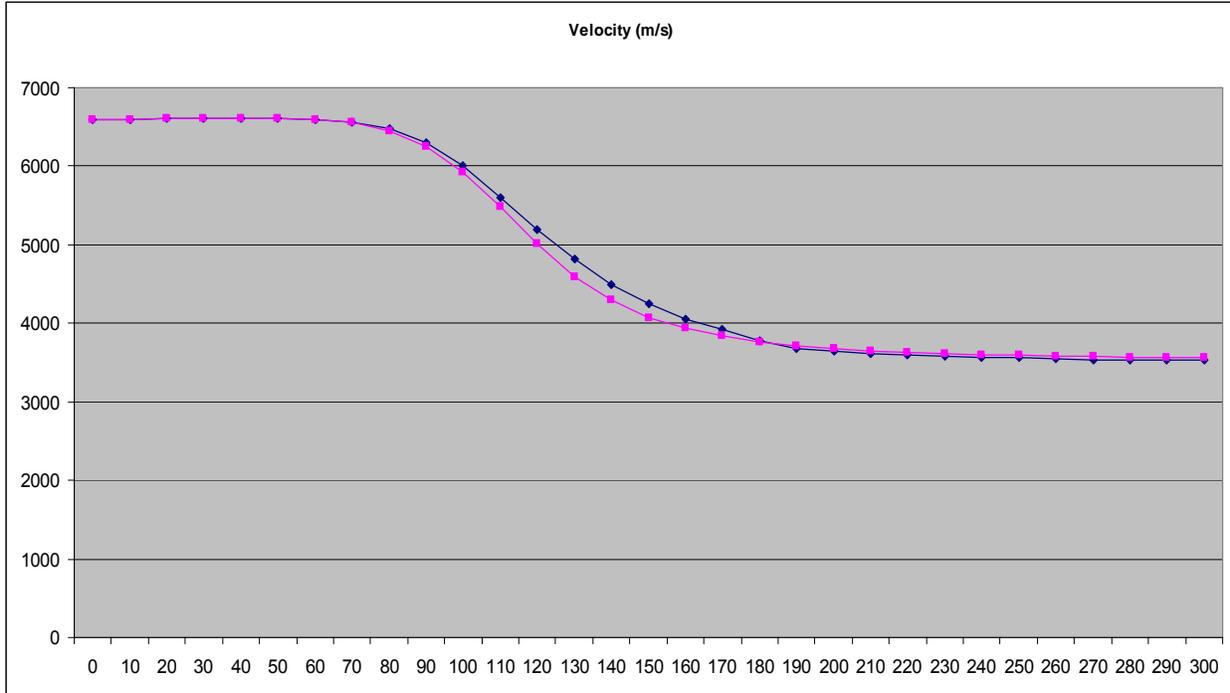


Figure 9.3

Figure 9.3 shows the comparison between the published MSP-01 velocity data and the data calculated by the simulator. The RMS of the difference between the two curves is 83.8 m/s.

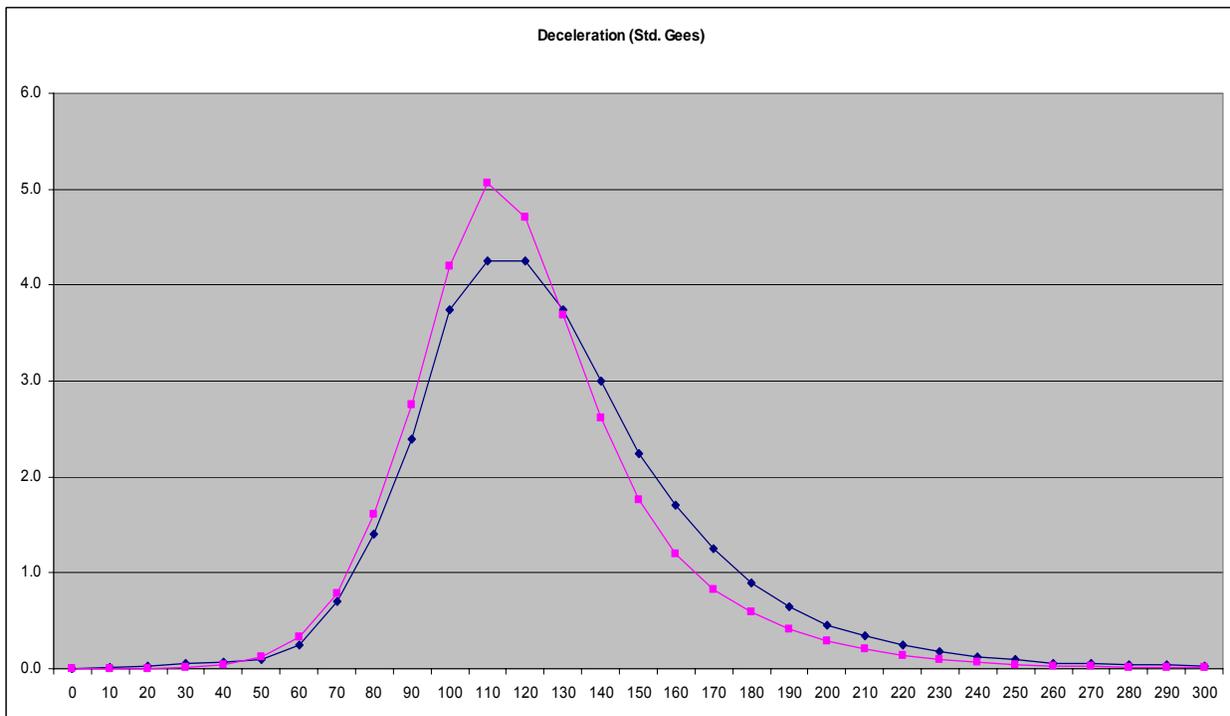


Figure 9.4

Figure 9.4 shows the comparison between the published MSP-01 deceleration data and the data calculated by the simulator. The RMS of the difference between the two curves is 0.273 g's.

It should be noted that in order to achieve the same exit altitude and velocity, the simulated spacecraft had to dive slightly deeper into the atmosphere than did the published MSP-01 vehicle. This discrepancy can be entirely explained by the differences in the atmospheric density models used by the original MSP-01 simulator and by this simulator. As the atmospheric model used by NASA in 1999 to obtain the published results is not available, this assumption is unproven, but seems entirely reasonable.

IX.2 – Simulator Aeroentry Validation

More opportunities exist for reconstructive Aeroentry validation. Braun and Manning [2006] report conditions for the five successful U.S. landings (Viking-1, Viking-2, MPF, MER-A and MER-B) plus simulations for two planned missions (Phoenix and MSL). Based on this data, reconstructions were performed on the MER-A, Viking-1, and Mars Pathfinder missions.

IX.3 – MER-A “Spirit” Aeroentry Reconstruction

Braun and Manning do not publish the altitude or radial distance of the entry threshold. Withers and Smith [Icarus 185 (2006)] listed the entry radial distance as 3522 km from the center of mass (~ 128 km MOLA), but calculated the entry velocity as 5630 m/s, which is inconsistent with other published trajectory data for this mission. It was thus decided to use the Braun and Manning data and assume the entry altitude to be approximately 140 km, roughly the outer edges of the Martian atmosphere. As the velocity curve is relatively flat between 150 and 100 km altitude, this assumption does not materially affect the reconstruction results. It should also be noted that Withers and Smith report a large unknown in the angle of attack of the vehicle, which can lead to a significant variation in the parachute deployment data. Values within their given range of angle of attack were used by trial-and-error to obtain the best fit to the parachute deployment data. The results are summarized in Table 9.2.

| MER-A “Spirit” Aeroentry Reconstruction | | |
|--|------------|------------|
| | Reported | Simulation |
| Mass at entry (kg): | 827 | 827 |
| Aerobrake diameter (m): | 2.65 | 2.65 |
| Ballistic Coefficient (kg/m ²): | 94.00 | 94.01 |
| Altitude at entry (km): | (see text) | 139.89 |
| Relative velocity at entry (m/s): | 5400 | 5399 |
| Flight path angle at entry (deg): | 11.49 | 11.50 |
| Angle of attack (deg): | 0.0 ± 5.0 | 1.88 |
| Altitude at parachute deploy (km): | 7.40 | 7.40 |
| Relative velocity at parachute deploy (m/s): | 411 | 411 |

Table 9.2

Braun and Manning report the vehicle’s velocity at parachute deploy as Mach 1.77. Using data published by Withers and Smith and others, Mach 1 near 0 km MOLA is approximately 232 m/s. The reconstructed altitude, velocity, and deceleration curves are shown in Figure 9.5.

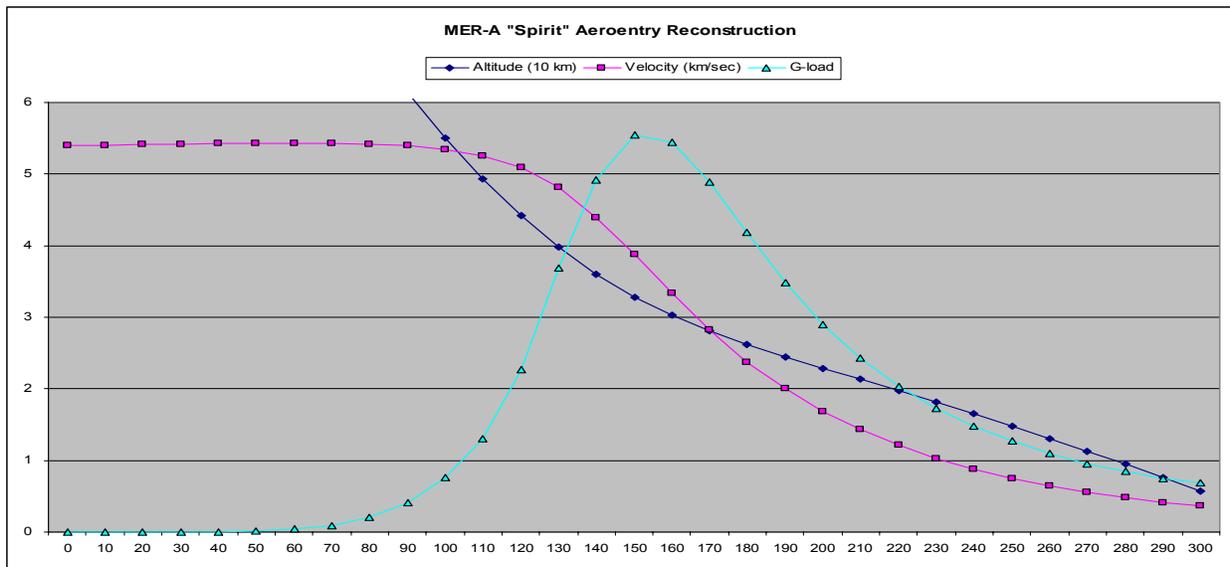


Figure 9.5

IX.4 – Viking-1 Aeroentry Reconstruction

Braun and Manning do not publish the altitude or radial distance of the entry threshold for the Viking-1 mission. Again, it was decided to assume the entry altitude to be approximately 140 km. Parachute deployment occurs at Mach 1.1, corresponding to approximately 275 m/s. Angle-of-attack values were used by trial-and-error to obtain the best fit to the parachute deployment data. The results of the simulation are summarized in Table 9.3.

The reconstructed altitude, velocity, and deceleration curves are shown in Figure 9.6.

| Viking-1 Aeroentry Reconstruction | | |
|--|------------|------------|
| | Reported | Simulation |
| Mass at entry (kg): | 992 | 992 |
| Aerobrake diameter (m): | 3.50 | 3.50 |
| Ballistic Coefficient (kg/m ²): | 64.0 | 64.0 |
| Altitude at entry (km): | (see text) | 139.5 |
| Relative velocity at entry (m/s): | 4700 | 4699 |
| Flight path angle at entry (deg): | 17.00 | 17.00 |
| Angle of attack (deg): | 11.00 | 12.72 |
| Altitude at parachute deploy (km): | 5.79 | 5.79 |
| Relative velocity at parachute deploy (m/s): | 255 | 255 |

Table 9.3

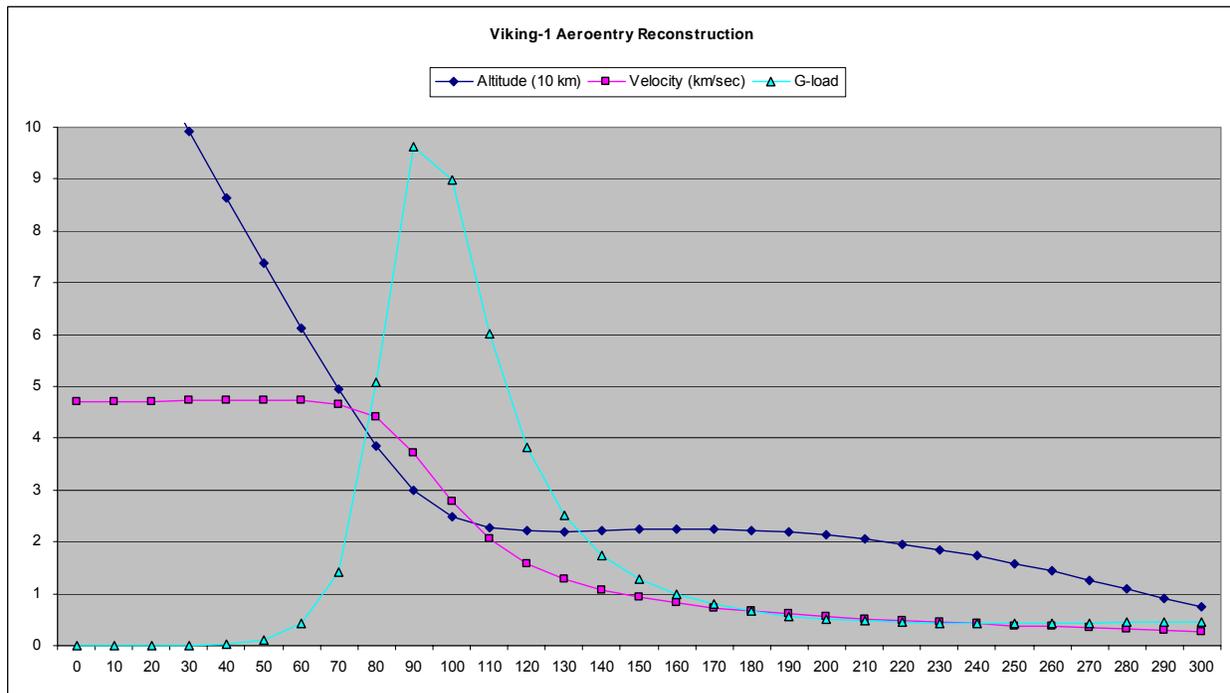


Figure 9.6

IX.5 – Mars Pathfinder Aeroentry Reconstruction

Braun and Manning do not publish the altitude or radial distance of the entry threshold for the 1997 MPF mission. Again, it was decided to assume the entry altitude to be approximately 140 km. Parachute deployment occurs at Mach 1.57, corresponding to approximately 364 m/s. Angle-of-attack values were used by trial-and-error to obtain the best fit to the parachute deployment data. The results of the simulation are summarized in Table 9.4.

The reconstructed altitude, velocity, and deceleration curves are shown in Figure 9.7.

| Mars Pathfinder Aeroentry Reconstruction | | |
|--|------------|------------|
| | Reported | Simulation |
| Mass at entry (kg): | 584 | 584 |
| Aerobrake diameter (m): | 2.65 | 2.65 |
| Ballistic Coefficient (kg/m ²): | 63.00 | 63.03 |
| Altitude at entry (km): | (see text) | 139.1 |
| Relative velocity at entry (m/s): | 7260 | 7258 |
| Flight path angle at entry (deg): | 14.06 | 14.00 |
| Angle of attack (deg): | 0.00 | 1.05 |
| Altitude at parachute deploy (km): | 9.40 | 9.40 |
| Relative velocity at parachute deploy (m/s): | 364 | 364 |

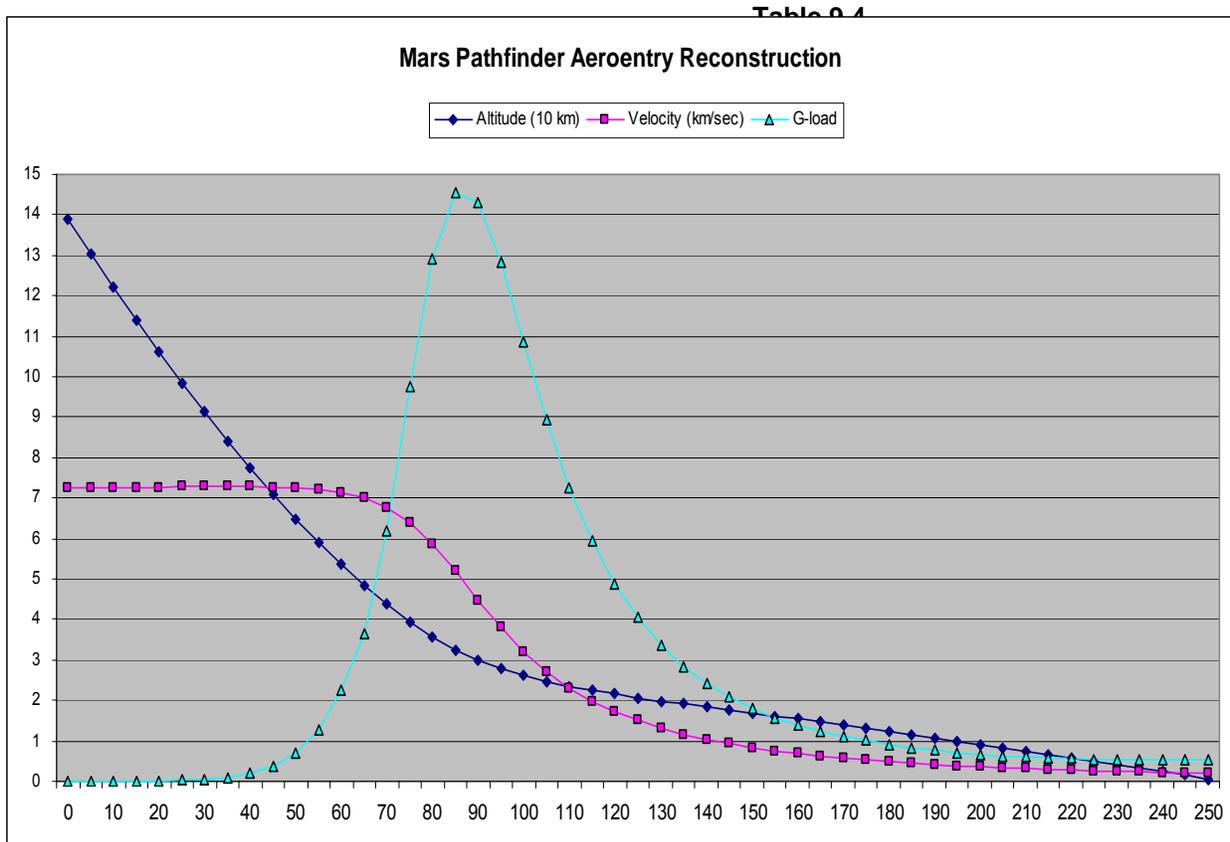


Figure 9.7

Appendix 1

Common Lander Analysis

| Common Lander Mass Budget | |
|---------------------------------------|-------------|
| Number of Engines: | 4 |
| Isp (LOX/CH4): | 360 sec |
| Total thrust: | 100,000 kgf |
| Engine Mass: | 2,000 kg |
| Ascent Stage: | |
| Height of cylindrical section: | 2.5 m |
| Total enclosed volume: | 204 m3 |
| Internal structure: | 5,659 kg |
| Pressure shell mass: | 3,043 kg |
| LIDS: | 395 kg |
| Power: | 500 kg |
| Control + Avionics: | 200 kg |
| Environmental: | 300 kg |
| 20% Growth: | 2,020 kg |
| Ascent stage dry mass less tankage: | 14,117 kg |
| Descent Stage: | |
| Landing deck volume: | 66 m3 |
| Landing Deck structure: | 477 kg |
| Landing gear: | 1,500 kg |
| PISRU + power: | 3,000 kg |
| 20% Growth: | 995 kg |
| Descent stage dry mass less tankage: | 5,973 kg |
| Habitat Lander: | |
| Total enclosed volume: | 270 m3 |
| Internal structure: | 7,488 kg |
| Pressure shell mass: | 3,043 kg |
| Power: | 500 kg |
| Control + Avionics: | 200 kg |
| Environmental: | 300 kg |
| Landing gear: | 1,500 kg |
| 20% Growth: | 2,606 kg |
| Habitat Lander dry mass less tankage: | 17,638 kg |
| Cargo Lander: | |
| Structure: | 3,592 kg |
| Landing gear: | 1,500 kg |
| Power: | 500 kg |
| Control + Avionics: | 200 kg |
| 20% Growth: | 1,058 kg |
| Cargo Lander dry mass less tankage: | 8,850 kg |
| Aeroshell: | |
| Heat Shield Area: | 176.7 m2 |
| Heat Shield mass: | 5,015 kg |
| Aeroshell surface area: | 265.1 m2 |
| Aeroshell mass: | 4,085 kg |
| Drag Coefficient: | 1.68 |

Appendix 2

Mission M-1 Analysis

| Crewed Stack (M-1) | | |
|---|----------------|-----------|
| Ascent Stage | | |
| Delta-vee to Mars orbit: | 4,448 | m/s |
| Mass Ratio: | 3.529 | |
| Ascent Stage dry mass (less tanks): | 14,117 | kg |
| Crew + Personal items: | 1,500 | kg |
| Consumables: | 6,300 | kg |
| Rocks & Stuff: | 1,000 | kg |
| Total dry liftoff mass (less tanks): | 22,917 | kg |
| Ascent Propellant: | 70,855 | kg |
| Volume: | 85.09 | m3 |
| Tank Mass: | 5,105 | kg |
| Mass at Ignition: | 98,877 | kg |
| <i>check</i> | <i>98,877</i> | <i>kg</i> |
| Acceleration at liftoff (Earth gees): | 1.01 | g |
| Total volume of ascent stage: | 204.3 | m3 |
| Less tank volume: | (85.1) | m3 |
| Less 5% for structure: | (10.2) | m3 |
| Habitable volume: | 109.0 | m3 |
| Volume per crew member: | 18.2 | m3 |
| Descent Stage | | |
| Delta-vee (braking + landing): | 1,300 | m/s |
| Mass Ratio: | 1.446 | |
| Ascent Stage dry mass (less tanks): | 14,117 | kg |
| Tankage: | 5,105 | kg |
| Descent stage dry mass: | 5,973 | kg |
| Crew + Personal items: | 1,500 | kg |
| Consumables: | 150 | kg |
| Cargo: | 1,600 | kg |
| Total Vehicle Mass: | 28,445 | kg |
| Propellant Consumed: | 12,673 | kg |
| Mass at Ignition: | 41,119 | kg |
| <i>check</i> | <i>41,119</i> | <i>kg</i> |
| Acceleration at ignition: | 2.43 | g |
| Aerobraking Mass: | 9,100 | kg |
| Total Mass at Entry: | 50,219 | kg |
| Hypersonic ballistic coefficient: | 169.2 | |
| Crewed Lander Launch Mass Budget | | |
| Ascent Stage dry mass (less tanks): | 14,117 | kg |
| Tankage: | 5,105 | kg |
| Cargo: | 1,600 | kg |
| Descent stage dry mass: | 5,973 | kg |
| Descent Propellants: | 12,673 | kg |
| Aeroshell: | 4,085 | kg |
| Heatshield: | 5,015 | kg |
| Total Crewed Lander mass: | 48,569 | kg |
| Artificial Gravity Assembly | | |
| Habitat Mass: | 18,899 | kg |
| Counterweight (Lander) mass: | 48,569 | kg |
| Angular velocity: | 2.00 | rpm |
| Radius to Habitat: | 85.0 | m |
| Centrifugal Acceleration: | 0.38 | g |
| Radius to Counterweight: | 33.1 | m |
| Length of Tether: | 118.1 | m |
| Tensile force on tether: | 7,190 | kg |
| Tether mass per linear meter: | 8.00 | kg |
| Mass of tether: | 2,834 | kg |
| Mass of solar array / radiator: | 200 | kg |
| Mass of winches: | 200 | kg |
| Total mass of A.G. assembly: | 3,234 | kg |
| In-Space Habitat | | |
| Diameter: | 7.50 | m |
| Length: | 8.50 | m |
| Surface Area: | 200 | m2 |
| Total Pressurized Volume: | 357 | m3 |
| Pressurized Volume per crew: | 59 | m3 |
| Furnishings per crew member: | 150 | kg |
| Shell Mass: | 2,766 | kg |
| Internal Structure: | 2,000 | kg |
| Airlock: | 250 | kg |
| LIDS: | 395 | kg |
| Furnishings: | 900 | kg |
| Power: | 200 | kg |
| Control + Avionics: | 200 | kg |
| Environmental: | 500 | kg |
| 20% Growth: | 1,442 | kg |
| Earth Aerocapture Heatshield mass: | 2,379 | kg |
| Total Inbound Habitat Mass: | 11,032 | kg |
| Artificial Gravity Assembly: | 3,234 | kg |
| Total Outbound Habitat Mass: | 14,266 | kg |
| TMI Payload | | |
| Crewed Lander: | 48,569 | kg |
| Transit Habitat: | 14,266 | kg |
| Service Module: | 27,069 | kg |
| Crew + Personal items: | 1,500 | kg |
| Consumables: | 6,367 | kg |
| Total TMI Payload Mass: | 97,771 | kg |
| Dry mass at aerocapture entry: | 82,050 | kg |
| Propellant for orbital maneuvering: | 4,752 | kg |
| Total mass at aerocapture entry: | 86,802 | kg |
| TMI Stage II (Centaur with RL10-B) | | |
| Delta-vee | 2,059 | m/s |
| Isp | 465.5 | sec |
| Mass Ratio | 1.570 | |
| Total Payload Mass | 97,771 | kg |
| Propellant Consumed: | 59,145 | kg |
| Volume: | 165.42 | m3 |
| Tank length: | 6.96 | m |
| Stage Dry Mass: | 5,914 | kg |
| Stage Wet Mass: | 65,059 | kg |
| Mass at Ignition | 162,830 | kg |
| <i>check</i> | <i>162,830</i> | <i>kg</i> |
| TMI Stage I (Centaur with RL10-B) | | |
| Delta-vee | 1,622 | m/s |
| Isp | 465.5 | sec |
| Mass Ratio | 1.427 | |
| Total Payload Mass | 162,830 | kg |
| Propellant Consumed: | 72,606 | kg |
| Volume: | 203.07 | m3 |
| Tank length: | 8.55 | m |
| Stage Dry Mass: | 7,261 | kg |
| Stage Wet Mass: | 79,866 | kg |
| Mass at Ignition | 242,696 | kg |
| <i>check</i> | <i>242,696</i> | <i>kg</i> |
| Service Module | | |
| Diameter: | 7.50 | m |
| Length: | 4.00 | m |
| Volume: | 176.71 | m3 |
| Structure: | 1,743 | kg |
| Avionics + Solar Power: | 1,000 | kg |
| 20% Growth: | 549 | kg |
| Engines (RL-10-B): | 2 | |
| Specific Impulse: | 465.5 | sec |
| Thrust: | 22,500 | kgf |
| Engine Mass: | 604 | kg |
| Dry Mass less Tankage: | 3,895 | kg |
| ΔV for Earth Approach: | 500 | m/s |
| Mass Ratio: | 1.116 | |
| Mass at Earth Approach: | | |
| SM Dry Mass less tankage: | 3,895 | kg |
| Transit Habitat Dry Mass: | 11,032 | kg |
| Crew + Personal items: | 1,500 | kg |
| Rocks & Stuff: | 1,000 | kg |
| Dry Mass less tankage: | 17,428 | kg |
| Propellant Consumed: | 2,059 | kg |
| Volume: | 5.76 | m3 |
| Tank Mass: | 345 | kg |
| Total mass at Earth Approach: | 19,832 | kg |
| <i>check</i> | <i>19,832</i> | <i>kg</i> |
| ΔV for TEI: | 1,200 | m/s |
| Mass Ratio: | 1.301 | |
| Mass at TEI: | | |
| Mass at Earth Approach: | 19,832 | kg |
| Crew consumables: | 6,300 | kg |
| Dry Mass at TEI less tankage: | 26,132 | kg |
| Propellant Consumed: | 8,281 | kg |
| Volume: | 23.16 | m3 |
| Tank Mass: | 1,390 | kg |
| Total mass at at TEI: | 35,802 | kg |
| <i>check</i> | <i>35,802</i> | <i>kg</i> |
| ΔV for Mars Approach: | 250 | m/s |
| ΔV for orbital corrections: | 250 | m/s |
| Total Mars Approach ΔV: | 500 | m/s |
| Mass Ratio | 1.116 | |
| Mass at Mars Approach: | | |
| SM Dry Mass less tankage: | 3,895 | kg |
| Earth Approach Tanks + Propellant: | 2,404 | kg |
| TEI Tanks + Propellant: | 9,671 | kg |
| In-Space Habitat: | 14,266 | kg |
| Crewed Lander: | 48,569 | kg |
| Crew + Personal items: | 1,500 | kg |
| Consumables: | 150 | kg |
| Total dry mass less tankage: | 80,455 | kg |
| Propellant Consumed: | 9,504 | kg |
| Volume: | 26.58 | m3 |
| Tank Mass: | 1,595 | kg |
| Total mass injected at TMI: | 91,554 | kg |
| <i>check</i> | <i>91,554</i> | <i>kg</i> |
| Service Module Mass Budget | | |
| Dry Mass less tankage: | 3,895 | kg |
| Earth Approach Tankage: | 345 | kg |
| TEI Tankage: | 1,390 | kg |
| Mars Approach Tankage: | 1,595 | kg |
| Total Stage Dry Mass: | 7,225 | kg |
| Earth Approach Propellant: | 2,059 | kg |
| TEI Propellant: | 8,281 | kg |
| Mars Approach Propellant: | 9,504 | kg |
| Total Stage Propellant: | 19,844 | kg |
| Total Stage Wet Mass: | 27,069 | kg |
| Total propellant volume: | 55.50 | m3 |
| Propellant tank length (5.5m di.): | 2.34 | m |

Appendix 3

Mission H-1 Analysis

| Surface Habitat Stack (H-1) | |
|-------------------------------------|-------------------|
| Surface Habitat Lander | |
| Delta-vee (RCS + terminal): | 1,300 m/s |
| Mass Ratio: | 1.446 |
| Habitat Lander dry mass less tanks: | 17,638 kg |
| Crew Consumables: | 6,750 kg |
| Cargo + Scientific Equipment: | 6,350 kg |
| Landed mass less Tankage: | 30,738 kg |
| Propellant Consumed: | 14,149 kg |
| Volume: | 16.99 m3 |
| Tank Mass: | 1,019 kg |
| Mass at Ignition | 45,907 kg |
| <i>check</i> | <i>45,907 kg</i> |
| Heat Shield mass: | 5,015 kg |
| Aeroshell mass: | 4,085 kg |
| Total Mass at Entry: | 55,007 kg |
| Hypersonic ballistic coefficient: | 185.3 |
| Habitat Living Volume | |
| Volume of Lander: | 270.3 m3 |
| Less 5% for structure: | (13.5) m3 |
| Less Propellant Tanks: | (17.0) m3 |
| Pressurized Volume: | 239.8 m3 |
| Volume per crew member: | 40.0 m3 |
| Communications Satellite | |
| Structure: | 100 kg |
| Avionics & Solar Power: | 1,000 kg |
| Station-keeping RCS: | 2,000 kg |
| 20% Growth: | 620 kg |
| Total ComSat Mass: | 3,720 kg |
| TMI Stage (Centaur) | |
| Delta-vee | 2,771 m/s |
| Isp | 466 sec |
| Mass Ratio | 2 |
| Total Payload Mass: | 79,707 kg |
| Propellant Consumed: | 72,672 kg |
| Volume: | 203 m3 |
| Tank length: | 9 m |
| Stage Dry Mass: | 7,267 kg |
| Stage Wet Mass: | 79,940 kg |
| Mass at Ignition | 159,647 kg |
| <i>check</i> | <i>159,647 kg</i> |
| Service Module | |
| Diameter: | 7.50 m |
| Length: | 4.00 m |
| Volume: | 176.71 m3 |
| Structure + Tankage: | 2,486 kg |
| Engines (RL-10-B): | 2 |
| Specific Impulse: | 465.5 sec |
| Thrust: | 22,500 kgf |
| Engine Mass: | 604 kg |
| Dry Mass: | 3,090 kg |
| ΔV to synchronous orbit: | 800 m/s |
| Mass Ratio: | 1.192 |
| Stage dry mass + ComSat: | 6,810 kg |
| Propellant Consumed: | 1,305 kg |
| Mass at Ignition | 8,115 kg |
| <i>check</i> | <i>8,115 kg</i> |
| Total Mars Approach ΔV : | 200 m/s |
| Mass Ratio: | 1.034 |
| Payload Mass: | |
| Service Module + ComSat: | 8,115 kg |
| Surface Habitat Lander: | 55,007 kg |
| Total Payload Mass: | 63,122 kg |
| Propellant Consumed: | 2,171 kg |
| Mass at Ignition | 65,293 kg |
| <i>check</i> | <i>65,293 kg</i> |
| TMI Stage II ΔV : | 910 m/s |
| Mass Ratio: | 1.221 |
| Payload Mass: | 65,293 kg |
| Propellant Consumed: | 14,414 kg |
| Volume: | 40.32 m3 |
| Mass at Ignition | 79,707 kg |
| <i>check</i> | <i>79,707 kg</i> |
| Service Module Mass Budget | |
| Total Stage Dry Mass: | 3,090 kg |
| Synchronous orbit propellant: | 1,305 kg |
| Mars Approach propellant: | 2,171 kg |
| TMI Stage II Propellant: | 14,414 kg |
| Total Stage Propellant: | 17,890 kg |
| Total Stage Wet Mass: | 20,980 kg |
| Total propellant volume: | 50.04 m3 |
| Propellant tank length (5.5m di.): | 2.11 m |

Appendix 4

Mission C-1 Analysis

| Cargo Mission (C-1) | |
|--------------------------------------|-------------------|
| Cargo Lander | |
| Delta-vee (RCS + terminal): | 1,300 m/s |
| Mass Ratio: | 1.446 |
| Cargo Lander dry mass less tanks: | 8,850 kg |
| Seed hydrogen: | 4,351 kg |
| Hydrogen tanks + plumbing: | 3,677 kg |
| CO2 electrolysis + fuel cells: | 355 kg |
| 20% Growth: | 806 kg |
| Cargo + Scientific Equipment: | 12,700 kg |
| Landed mass less Tankage: | 30,738 kg |
| Propellant Consumed: | 14,149 kg |
| Volume: | 16.99 m3 |
| Tank Mass: | 1,020 kg |
| Mass at Ignition | 45,907 kg |
| <i>check</i> | <i>45,907 kg</i> |
| Heat Shield mass: | 5,015 kg |
| Aeroshell mass: | 4,085 kg |
| Total Mass at Entry: | 55,008 kg |
| Hypersonic ballistic coefficient: | 185.3 |
| Seed Hydrogen Calculations: | |
| Total Ascent Propellants: | 70,855 kg |
| Ascent Oxygen: | 55,452 kg |
| Ascent Methane: | 15,403 kg |
| Required Hydrogen: | 3,851 kg |
| Environmental water needs: | 4,500 kg |
| Required Hydrogen: | 500 kg |
| Total required seed hydrogen: | 4,351 kg |
| Volume of liquid hydrogen required: | 61.3 m3 |
| Required Oxygen to produce water: | 34,806 kg |
| Mass of water produced: | 39,157 kg |
| Volume of water produced: | 39.2 m3 |
| Hours in 26 months: | 10,000 hrs |
| Water production per hour: | 3.92 kg |
| Power production: | 10.44 kw |
| Required oxygen production per hour: | 3.48 kg |
| Required ion current: | 10,710 amps |
| Communications Satellite | |
| Structure: | 100 kg |
| Avionics & Solar Power: | 1,000 kg |
| Station-keeping RCS: | 2,000 kg |
| 20% Growth: | 620 kg |
| Total ComSat Mass: | 3,720 kg |
| Service Module | |
| Diameter: | 7.50 m |
| Length: | 4.00 m |
| Volume: | 176.71 m3 |
| Structure + Tankage: | 2,486 kg |
| Engines (RL-10-B): | 2 |
| Specific Impulse: | 465.5 sec |
| Thrust: | 22,500 kgf |
| Engine Mass: | 604 kg |
| Dry Mass: | 3,090 kg |
| ΔV to synchronous orbit: | 800 m/s |
| Mass Ratio: | 1.192 |
| Stage dry mass + ComSat: | 6,810 kg |
| Propellant Consumed: | 1,305 kg |
| Mass at Ignition | 8,115 kg |
| <i>check</i> | <i>8,115 kg</i> |
| Total Mars Approach ΔV : | 200 m/s |
| Mass Ratio: | 1.034 |
| Payload Mass: | |
| Service Module + ComSat: | 8,115 kg |
| Cargo Lander: | 55,008 kg |
| Total Payload Mass: | 63,123 kg |
| Propellant Consumed: | 2,171 kg |
| Mass at Ignition | 65,293 kg |
| <i>check</i> | <i>65,293 kg</i> |
| TMI Stage II ΔV : | 910 m/s |
| Mass Ratio: | 1.221 |
| Payload Mass: | 65,293 kg |
| Propellant Consumed: | 14,415 kg |
| Volume: | 40.32 m3 |
| Mass at Ignition | 79,708 kg |
| <i>check</i> | <i>79,708 kg</i> |
| Service Module Mass Budget | |
| Total Stage Dry Mass: | 3,090 kg |
| Synchronous orbit propellant: | 1,305 kg |
| Mars Approach propellant: | 2,171 kg |
| TMI Stage II Propellant: | 14,415 kg |
| Total Stage Propellant: | 17,890 kg |
| Total Stage Wet Mass: | 20,980 kg |
| Total propellant volume: | 50.04 m3 |
| Propellant tank length (5.5m di.): | 2.11 m |
| TMI Stage (Centaur) | |
| Delta-vee | 2,771 m/s |
| Isp | 465.5 sec |
| Mass Ratio | 1.836 |
| Total Payload Mass | 79,708 kg |
| Propellant Consumed: | 72,673 kg |
| Volume: | 203.26 m3 |
| Tank length: | 8.87 m |
| Stage Dry Mass: | 7,267 kg |
| Stage Wet Mass: | 79,940 kg |
| Mass at Ignition | 159,648 kg |
| <i>check</i> | <i>159,648 kg</i> |

References

¹ All references to altitude herein imply the so-called Mars Observer Laser Altimeter-derived altitudes unless otherwise noted.

² *Atlas Launch System Mission Planner's Guide* – L-M publication CLSB-0409-1109 – January 2007.

³ “Mars Exploration Entry, Descent and Landing Challenges,” R.D. Braun and R.M. Manning, 2006 IEEE Aerospace Conference, March 2006.

⁴ “Sizing of an Entry, Descent and Landing System for Human Mars Exploration,” John A. Christian, Grant Wells, Jarret Lafleur, Kavya Manyapu, Amanda Verges, Charity Lewis and Robert D. Braun, Georgia Institute of Technology, preprint, 2006.

⁵ Structural mass estimation data are from the NASA Exploration Systems Architecture Study, NASA-TM-2005-214062, November 2005.

⁶ AE 8803 BRA 2005, Georgia Institute of Technology.