

# Reaching Mars for Less: The Reference Mission Design of the MarsDrive Consortium

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The reference mission design of the MarsDrive Consortium is discussed, which has been created to facilitate exploration of the red planet through methods that are both realizable and cost-sustainable with existing technology. This mission plan—known as *Mars for Less*—is predicated on the use of existing medium-lift launch vehicles. In this architecture, approximately 25-tonne propulsion stages are placed individually in low-Earth orbit, where they are mated to Mars-bound payloads and ignited at successive perigees to execute trans-Mars injection. Spacecraft follow conjunction-class trajectories to the red planet and utilize aerodynamic methods for orbital capture and descent. Return vehicles are fuelled with methane/oxygen bipropellant synthesized primarily from Martian resources. Dispatching expeditions from orbit with individual, high energy stages—rather than directly from the Earth’s surface—allows for the division of mission mass into more manageable components, which can be launched by vehicles that exist today. This plan does not require the development of heavy-lift launch technology: an effective yet costly proposition that may otherwise hinder current space exploration initiatives. Without the need for heavy-lift boosters, piloted missions to Mars may be undertaken presently, and within the capabilities of private initiatives. It is argued that the mission design herein represents a more viable method of conducting early human Mars exploration than proposals which require heavy-lift launch vehicles—an alternative method by which the red planet can be opened to humanity.

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## 1.0 Introduction

Heavy-lift launch vehicles (HLLVs) are regarded by many as the key technology for an aggressive, cost-sustainable program of human spaceflight beyond low-Earth orbit. Indeed, decades of both moon and Mars mission analysis have been dominated by plans calling for the development of at least Saturn V-class launch technology, capable of delivering 100 tonnes or better to low-Earth orbit (LEO)—and on the surface, this would appear quite prudent. A human expedition to either destination will necessarily require significant mass initially launched to orbit; and because bigger boosters can carry larger amounts of payload in far fewer launches, they have been championed by a majority of astronautical engineers as the most cost-effective technology to lift the future of manned spaceflight. Correspondingly, NASA has adopted the design of a shuttle-derived heavy-lift launch vehicle (recently named the *Ares V*) to support the first priority of the agency’s new mandate: a series of lunar return missions to commence within 15 years.

Yet there remain both significant drawbacks and significant uncertainties in the development and use of a heavy-lift booster. A shuttle-derived HLLV—while potentially more economical than building a comparable booster “from scratch”—is nevertheless projected to cost upwards of \$10 billion USD: one of the largest expenses in a program that is becoming increasingly infamous for fluctuating cost projections, mission objectives and capabilities. With an increasing number of technical obstacles being encountered in developing the predecessor to Ares V—the solid rocket booster-derived Crew Launch Vehicle—the possibility of delayed HLLV development is quickly growing as well. Furthermore, as the most significant pending expense of the current Vision for Space Exploration, plans for developing the *Ares V* may prove impossible to sustain through the imminent presidential administration change in 2009. While it may be possible to accelerate the current timeline of the HLLV program, this would certainly require significant increases in NASA spending—and consequently, expediting its development may adversely impact other NASA priorities, such as developing the shuttle-replacement Crew Exploration Vehicle (CEV). Creating a new heavy-lift launch system

will necessarily have significant ripple effects throughout the entire pre-existing agency—effects which may not be fully appreciated by those who most aggressively advocate the rapid development of a major new booster.

Yet perhaps more significant than the uncertainty of HLLV development are its implications to the private sector. If NASA succeeds in developing its shuttle-derived heavy-lift vehicle, then a program of human spaceflight beyond low-Earth orbit will evolve in the next few decades that only NASA itself will be capable of undertaking. Because NASA spacecraft will be optimized for launch on heavy-lift boosters, there will be limited potential at best for either the private sector or other countries to become involved in human exploration in a significant way. No other nation in the world has the necessary resources to build a HLLV, and no private initiative will be capable of developing comparable technology anytime in the foreseeable future. Consequently, spaceflight beyond Earth orbit will remain within the realm of NASA exclusively; and its future will hinge upon a single vehicle's success.

## 2.0 Mars for Less

The *Mars for Less* architecture [1], [2] was designed to circumvent the need for heavy-lift vehicles entirely. Rather than basing plans for human Mars expeditions on as-yet undeveloped technology, *Mars for Less* is predicated on the use of more modest, more realizable medium-lift launch vehicles (also referred to as Evolved-Expendable Launch Vehicles), with approximately 25-tonne to low orbit payload capacities. By dividing Mars-bound spacecraft and propulsion systems into smaller components, and launching them on smaller boosters as opposed to heavy-lift vehicles, the need for significant new launch technology can be eliminated; consequently, the only vehicles requiring development “from scratch” would be those actually used for Mars exploration, resulting in a program that is fundamentally easier to initiate.

### 2.1 Mission Overview

*Mars for Less* is essentially Robert Zubrin’s *Mars Direct* mission architecture, divided into smaller components and delivered to orbit by smaller launch vehicles. Comprehensively discussed in references [3] and [4], the *Mars Direct* plan represents a global minimum-energy solution for any given launch opportunity, and leverages surface resources and long-duration stays to maximize both surface time and capability on Mars, while simultaneously permitting reasonably expeditious piloted flight times without the need for advanced (i.e. nuclear) in-space propulsion systems.

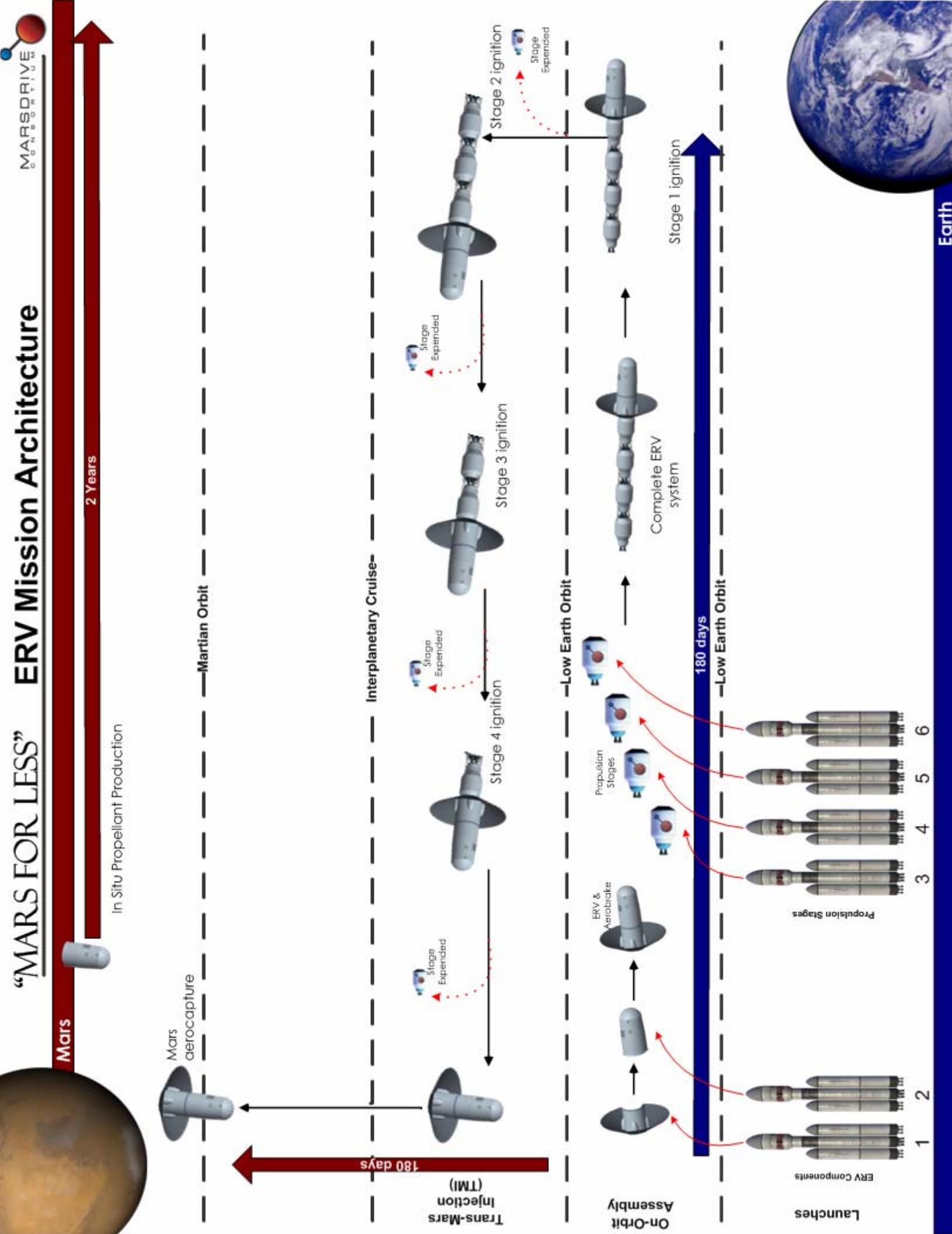
Each complete mission in the *Mars Direct* architecture requires two spacecraft, launched approximately 26 months apart:

- An *Earth Return Vehicle* (ERV), which flies to Mars unmanned and fuels itself automatically with methane/oxygen bipropellant manufactured primarily from indigenous Martian resources; and
- A crewed *Mars Transfer and Surface Vehicle* (MTSV), which houses a crew of 4 astronauts during the outbound leg of each mission and for the majority of each surface stay.

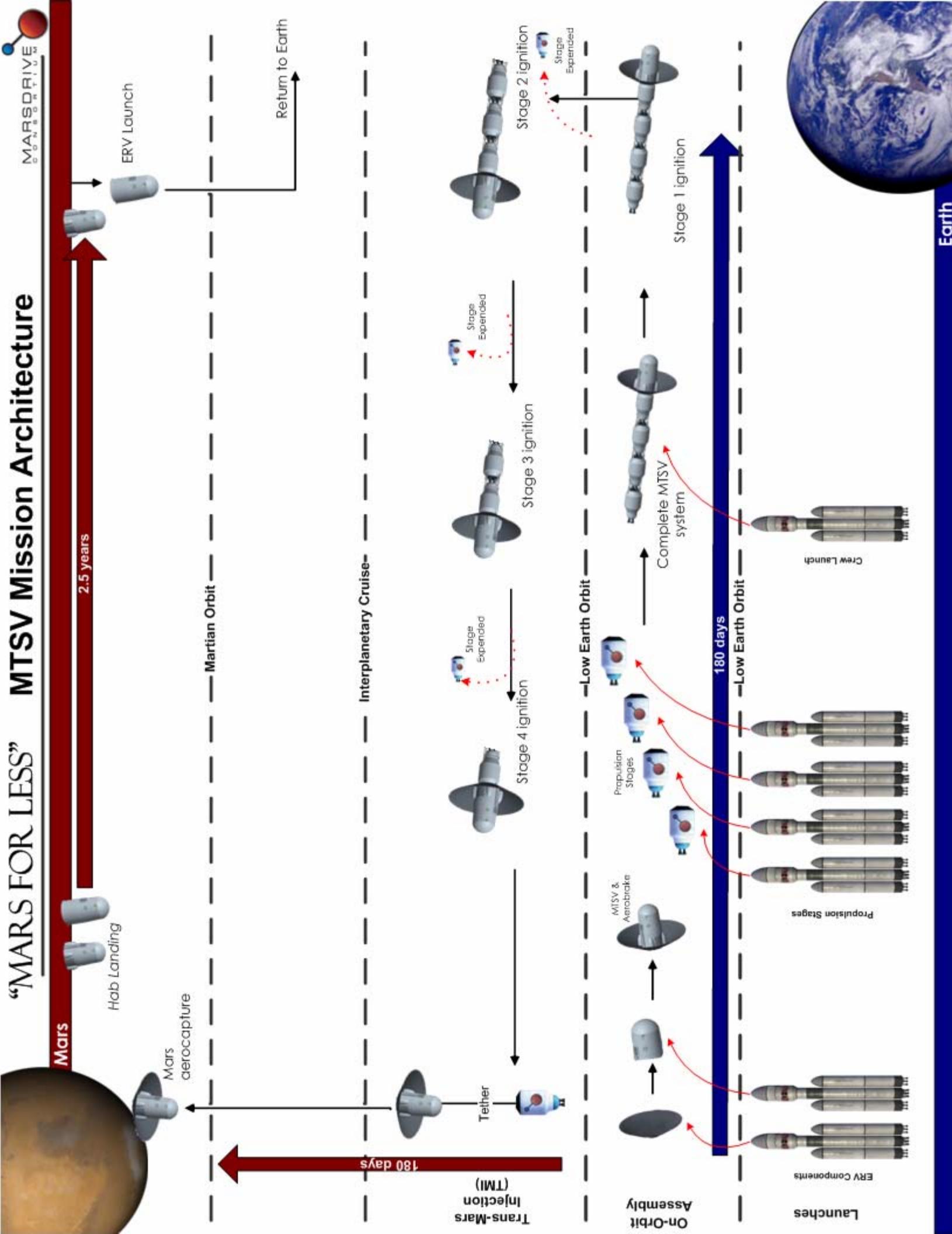
In the *Mars for Less* mission design, each of these spacecraft are launched and assembled in 6 components, delivered separately to orbit using medium-lift vehicles. For each spacecraft, the first launch delivers a two-level common habitat element, to which the second mates a lander, aeroshield and mission-specific modules, consisting of empty propulsion stages, hydrogen feedstock and a chemical reactor for the ERV; and a garage/storage module for the MTSV. Subsequent launches would then deliver 4 high-energy hydrogen/oxygen propulsion stages to each payload, which would be mated aft like train cars and ignited at successive perigees, widening the orbit of each spacecraft until final stages impart sufficient energy for trans-Mars injection.

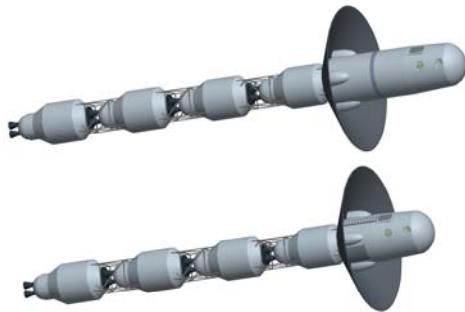
The unmanned ERV proceeds to Mars on a near-minimum energy transfer with a flight time of approximately 8 months, while the crewed MTSV is assembled about 2 years later and follows on a slightly expedited flight lasting 4-6 months, depending on the specific opportunity. Piloted spacecraft have the option of generating artificial gravity using tethers and burnt-out propulsion stages as counterweights. Each spacecraft would aerobrake into Mars Orbital Capture (MOC) and land with the aid of parachutes and rockets at a common site. As in *Mars Direct*, each crew remains on the surface for approximately 1.5 years, during which broad exploration can be performed. Astronauts would be afforded the natural protection of the Martian atmosphere from the hazards of interplanetary space for the duration of their stay, at the conclusion of which they would board their fully-fuelled ERV and launch directly from the surface of Mars onto an Earth-bound trajectory. Upon return, the crewed ERV aerobrakes into Earth orbital capture, and recovery is possible using either orbital rendezvous or by targeting the ERV capsule for a direct Earth reentry and Apollo-style ocean splashdown.

The rationale for selecting a crew of 4 is discussed at length in references [3] and [4]. An iconographic illustration of the the *Mars for Less* mission is presented in Figure 1, and both ERV and MTSV reference designs are shown in Figure 2.



**MTSV Mission Architecture**





**Figure 2:** Earth Return Vehicle (ERV, top) and Mars Transfer and Surface Vehicle (MTSV, Bottom)

While mating spacecraft and propulsion stages in orbit is a less robust approach than launching each Mars-bound spacecraft directly from the Earth's surface using a heavy-lift vehicle, it does represent an easier architecture to *develop*, having been designed around existing medium-lift launch vehicles, which would require little (if any) modification. The approximately 25-tonne components of both spacecraft described above can be placed in LEO using either a combination of existing launch vehicles (such as the Ariane V or Delta IV-H), or alternatively, by emerging launch vehicles with wider payload fairings (such as the NASA Crew Launch Vehicle, SpaceX Falcon 9-29, or Chinese Long March 5).

While the need for orbital assembly is frequently cited by detractors as detrimental to any space mission design, it is noteworthy that—of all the factors required for a piloted Mars mission to succeed (trans-Mars injection, aerobraking into MOC, successful surface landing and rendezvous, and making rocket fuel from Martian air), orbital rendezvous and assembly are the two operations that today's space agencies actually have the *most* experience with. The spacecraft and propulsion stages outlined above would be much easier to assemble than either space stations or candidate spacecraft from past

studies, since there would be no need to extend a complex life support system across the entire series of modules. With discrete, compartmentalized propulsion stages, little more than truss sections would be required to react and re-transmit propulsion loads; and with only structural considerations being primary, the entire assembly and integration process for spacecraft in *Mars for Less* would be much simpler than for projects such as Mir or the International Space Station.

Using smaller launch vehicles to assemble and launch *Mars Direct*-style missions allows for a dramatic reduction in development costs, making this method of undertaking piloted expeditions much more practical for either non-American or privately-sponsored initiatives.

### 3.0 Trajectories and Energy Requirements

Low-energy, conjunction-class transfers are specified for both cargo and crewed spacecraft in this architecture. Propulsion systems for unmanned payloads are designed to provide a C3 (specific departure energy, or departure velocity squared) of  $15 \text{ km}^2/\text{s}^2$ , ensuring that trans-Mars injections can be performed for all minimum-energy launch opportunities. This performance requirement dictates the need for 4 high energy propulsion stages per cargo flight, massing 25 tonnes each (with 22 tonnes propellant).

For crewed vehicles, a higher-energy trajectory with  $C3 = 25$  is assumed. This transfer both significantly reduces flight time (to between 4 and 6 months, depending on the specific opportunity) and also corresponds to an Earth-resonant trajectory with an orbital period of 2 years, thus providing crewed spacecraft the option of free Earth return in the event that landing on Mars is impossible.

Designing for the above trajectories, the total  $\Delta v$  requirements for spacecraft departing low-Earth orbit, including 5% gravity losses, are 4.50 km/s for piloted vehicles and between 3.87 and 4.08 km/s for Earth Return Vehicles. (These values are conservative, as much smaller losses can likely be achieved contingent on specific TMI propulsion characteristics.)

**Table 1:** Summary of Major Propulsive Maneuvers Assumed for Piloted Mars Expeditions

Maneuver	$\Delta v$ (m/s)
Trans-Mars Injection (cargo, $C3 = 15 \text{ km}^2/\text{s}^2$ )	4080
Trans-Mars Injection (piloted, $C3 = 25 \text{ km}^2/\text{s}^2$ )	4522
Earth-Mars Midcourse Correction	100
Post-Aerocapture Maneuvers	100
Descent and Landing	625
Mars Ascent and Orbit Insertion	5340
Trans-Earth Injection (Depart from $250 \times 33,000 \text{ km}$ HEMO, $C3 = 16 \text{ km}^2/\text{s}^2$ )	1740
Mars-Earth Mid-Course Correction	100
Deterministic Maneuvers (each way)	50

Plane changes, mid-course corrections, post-aerocapture maneuvers and landing are accomplished using spacecraft reaction control systems and landing stages. The velocity change values used for spacecraft and propulsion sizing are summarized in Table 1.

#### 4.0 Propulsion System Design

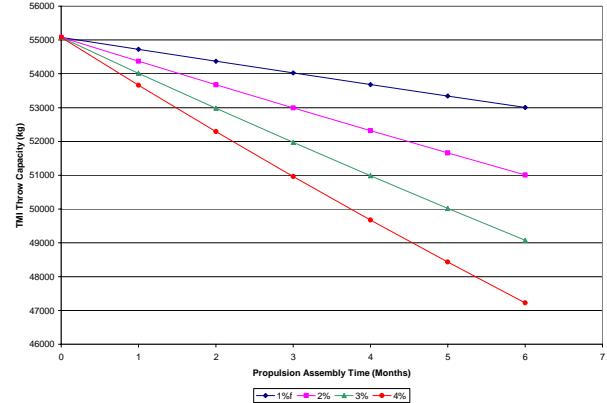
Dividing spacecraft propulsion into stages allows a vehicle to discard otherwise parasitic mass incrementally. Specific to the focus mission, staged trans-Mars injections are used to further enhance propulsion performance by restricting engine operation to short intervals deep in Earth's gravity well, allowing spacecraft to draw greater kinetic energy from each successive burn. Energy is also saved by circumventing the need to accelerate a single heavy upper stage through the entire TMI maneuver.

High performance hydrogen/oxygen stages are baselined for all TMI propulsion in this mission design. The most efficient propellant in use today, current engines burning H<sub>2</sub>/O<sub>2</sub> can yield vacuum specific impulses between 440 and 465 seconds. The individual stages assumed in this architecture have dry masses of 3 tonnes (2 tonnes structural and engines, 1 tonne reserved for docking systems and insulation), propellant masses of 22 tonnes, 220 kN total thrust and *Isp* = 465 seconds (i.e. 2 Pratt and Whitney RL-10B-2 engines).

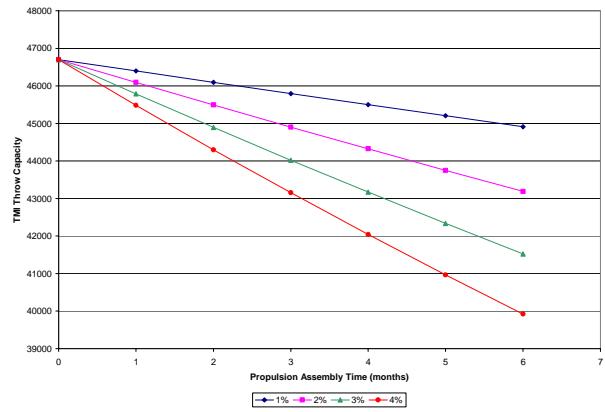
#### 4.1 Propellant Boiloff

Hydrogen/oxygen bipropellants are very cryogenic, and will tend to boil off when stored in space for appreciable amounts of time. Fortunately, this effect can be largely mitigated using multi-layer insulation. 50 sheets of MLI (specifically, insulation consisting of aluminized Kapton with Dacron net separations) can slow the rate of H<sub>2</sub>/O<sub>2</sub> propellant loss to approximately 1% of the total mass per month [3]. Furthermore, additional studies have suggested that H<sub>2</sub>/O<sub>2</sub> boiloff can be reduced even further through active prevention methods (such as chilling oxygen tanks with hydrogen gas as it boils away), and tests conducted almost 10 years ago at the NASA Lewis (now Glenn) Research Center's supplemental MLI research facility have demonstrated that a hybrid thermal control system could allow for the complete elimination of cryogenic propellant boiloff [5].

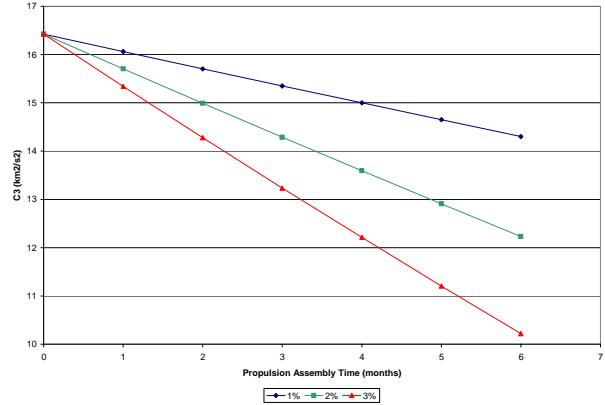
An analysis of performance sensitivity to varying assembly times and propellant boiloff rates was performed for the candidate 4-stage propulsion system outlined above, and results are presented in Figures 3-6. Figures 3 and 4 show the trans-Mars throw capacity for the baselined propulsion system for both cargo (C3=15) and crewed (C3 = 25) opportunities, while Figures 5 and 6 show the effects of boiloff and assembly time on C3 for launching fixed spacecraft masses (Section 5). In all



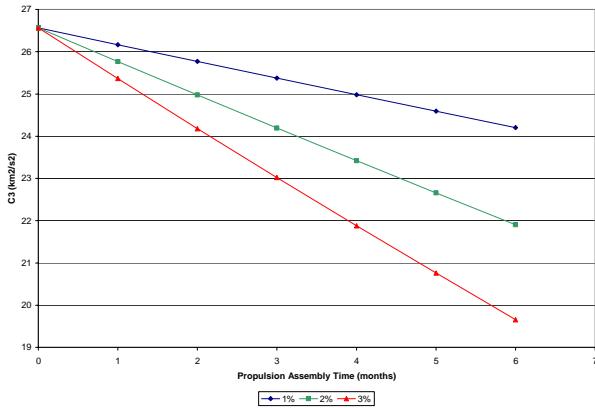
**Figure 3:** TMI Cargo Throw (C3=15) as Function of Boiloff and Assembly Time



**Figure 4:** TMI Crewed Throw (C3=25) as Function of Boiloff and Assembly Time



**Figure 5:** Launch Energy for Cargo Missions as Function of Boiloff and Assembly Time



**Figure 6:** Launch Energy for Crewed Missions as Function of Boiloff and Assembly Time

cases, boiloff impact was analyzed assuming that each propulsion stage is delivered at an equal interval over the total assembly time, and with the assembly time itself taken as the maximum loiter time of the first stage. Thus, in the case of a 4 month assembly, the upper-most stage is assumed to spend 4 months on orbit, whereas the final stage delivered is assumed to spend one month on orbit. It is noteworthy that, within these time constraints, a launch delay for a given stage would actually have the effect of *increasing* its performance, since the delayed stage would spend a reduced amount of time in space prior to use.

From Figures 3-6, it is clear that boiloff rates exceeding 2% are unacceptable without significantly expedited assembly times. In this study, it is assumed that propulsion system assembly requires 4 months in total at a boiloff rate of 1% per month, corresponding to maximum TMI throw capacities of approximately 54 tonnes for cargo and 46 tonnes for crewed opportunities. It is noteworthy that the available launch energy in Figures 2 and 3 increases by approximately  $1.1 \text{ km}^{2/\text{s}^2}$  in all situations for each 1% reduction in gravity loss assumptions.

## 5.0 Spacecraft Design and Assembly

### 5.1 Design Overview

The principal design consideration for both spacecraft in the *Mars for Less* mission design is ensuring that all components are compatible with existing launch systems—both MTSV and ERV modules must fit within the payload dimensions and mass limits of candidate medium-lift boosters, and both must be deliverable to orbit with as few launches as possible.

The recommended spacecraft in this mission design would have exterior diameters not exceeding 4.5 meters, and either folding or rigid-conical aeroshields that can be adapted to the payload fairing of candidate launch

systems. A common crew cabin for both transfer and return vehicles is assumed in all reference designs, with additional sections dedicated to storage or propulsion systems respectively.

The common habitat for both spacecraft is a predominantly cylindrical, slightly tapered structure with a maximum base diameter of 4.5 meters. The basic reference cabin design is two levels, with additional vehicle-specific units consisting of a storage compartment for the MTSV and 2 methane/oxygen propulsion stages for the ERV; habitable sections in this design are connected by a double layered half-cylinder 1.4 meters in diameter running through the centre of the spacecraft. Deck 1 contains a galley, washroom and two sleeping quarters, while deck 2 encloses a laboratory, exercise sections, and 2 additional sleeping quarters, as well as command systems and the primary cabin airlock. This design has three central access points per deck: swinging hatches open into common areas, and sliding doors on the flat section of the hub would permit access to individual crew quarters. The curved portion of the hub would be filled with water, and this combined with materials having high radiation attenuation coefficients would serve as a protective shelter in the event of a solar particle event. Sleeping quarters in this design are small but functional, permitting each astronaut approximately 3.8 square meters of floor space and 2.5 meters of overhead (Figure 7).

Both spacecraft would be optimized primarily for vertical flight and landing profiles. With only one major load path, structural reinforcement can be restricted to one principal axis, thus sparing otherwise significant mass penalties. Mars Orbital Capture (MOC) and atmospheric entry profiles would be ballistic in nature to minimize the amount of thermal protection required; while spacecraft such as the shuttle fly on lifting trajectories rather than ballistic, such profiles actually maximize the surface area subjected to heating as well as exposure time. The failure of the shuttle Columbia unfortunately serves to illustrate this point – the less ballistic the trajectory, the longer the spacecraft is exposed to thermal stressors, which increase failure potential for heat shields and protection systems.

$\text{CH}_4/\text{O}_2$  reaction control systems and descent stages with steady-state specific impulses of 380 seconds are assumed for both spacecraft. RCS rockets would control spacecraft attitude during orbital assembly, and additional fuel modules that fit within the excess LEO capacity of a final “checkout” launch would be delivered to supplement propellant expended during assembly. For both spacecraft, these thrusters would perform the assumed 100 m/s mid-course correction required in transit, and would aid with orbit and descent maneuvers at Mars. For the MTSV, the RCS system and lander combined would be more than capable of returning the spacecraft quickly from a stranded position between Earth-escape and minimum-energy trans-Mars injection in the event of TMI failure, and post-abort maneuvering performance in this scenario could be even



**Figure 7: Common Cabin Layout**

further increased by jettisoning the spacecraft's cargo module and aeroshield prior to any retro engine burn.

## 5.2 Spacecraft Sizing

The first several iterations of mass estimates for candidate spacecraft were based on values derived from references [3], [4] and [6], each of which differed somewhat in various component estimates and allocations. Certain values—in particular, those for ERV propulsion stages—were increased by several tonnes to account for additional subsystems required for long-duration propellant storage on the Martian surface. Complete mass allocations are presented in Appendix A, Table A-1 by spacecraft launch; total MTSV and ERV TMI masses were estimated to be approximately 46 and 50 tonnes, respectively.

Based on the mass estimates in Table A-1, the 2-launch assembly profile of each spacecraft would proceed as follows: for the Earth Return Vehicle, launch 1 would deliver the upper TEI propulsion stage, ERV cabin, propellant production plant and hydrogen feedstock, a total mass of 24.8 tonnes. The second launch would contain the lower TEI stage, nuclear reactor, aeroshield and lander, also massing 24.8 tonnes for a combined total mass of 49.6 tonnes. The first MTSV launch would carry a 24.2 tonne habitat module and its supplies, to which the second would mate a storage module, aeroshield and lander for a total of 46 tonnes. Once each vehicle is assembled, TMI stages would be launched and mated individually, restricted to assembly time periods discussed in Section 4.1.

A second spacecraft sizing iteration was subsequently performed based on statistical data and known masses for off-the-shelf (OTS) components [7]. Component masses for which there are no established values were estimated based on an initial dry mass prediction of the total spacecraft [7] calculated from statistical data, including margins. Total mass allocations from the second sizing

iteration are presented in Tables A-2 and A-3 in Appendix A, and total spacecraft masses converged closely to initial estimates. By this method, the MTSV was estimated to mass approximately 47 tonnes, and the ERV approximately 54 tonnes. While these figures are slightly in excess ( $\sim 400$ - $700$  kg) of the assumed TMI throw capacities of the proposed 4 stage propulsion system, these payloads could still be accommodated by any of:

- Reducing the rate of propellant boil-off to  $< 1\%/\text{month}$ ;
- Reducing total assembly time to 3 months for propulsion stages;
- Launching shortly after the final stage is delivered to orbit (instead of waiting an additional month); and/or
- Reduced burn times for propulsion stages (corresponding to reduced gravity losses)

Additionally, mass estimates for the ERV in the second iteration exceed the two-launch capacity of most candidate launch vehicles. It is proposed that this excess, should it prove to exist, be mitigated by transferring supplemental equipment to the ERV on a dedicated “checkout” launch prior to trans-Mars injection, which would likely be viewed as a requirement regardless of its necessity for payload transfer. Alternatively, the Boeing Delta-IV Heavy is capable of delivering in excess of 25 tonnes to the candidate assembly orbit, and can be used circumvent a seventh launch.

It is noteworthy that—in both mass estimate methodologies—margins would be significantly increased per spacecraft by the reduction of crew size from 4 members to 3; furthermore, a reduced crew number would also allow for smaller propulsion stages which could be launched on an even larger number of existing boosters. However, in spite of these apparent advantages, a crew of 4 is widely viewed as the minimum number required to accomplish significant surface exploration; thus, no crew size reduction is assumed.

## 5.4 Spacecraft Assembly Orbit

The target spacecraft assembly orbit is assumed to be 200 km circular, inclined 28.5 degrees. This orbit is comparable to those in which NASA has been docking spacecraft since the 1960s. Specific orbital elements, however, would nevertheless be subject to variation, depending on the particular launch system chosen to deliver each component. (For example, the performance of the Ariane V is degraded somewhat by launching heavier payloads to this inclination, but would still be able to deliver a lighter component such as the MTSV cargo module). It is noteworthy that the specific inclination of the assembly orbit is only of consequence to the launch vehicles used to deliver spacecraft components; the orbital plane of either the MTSV or ERV can easily be changed for negligible propulsive  $\Delta v$  later in the mission, at the apogee of the pre-TMI Earth orbit.

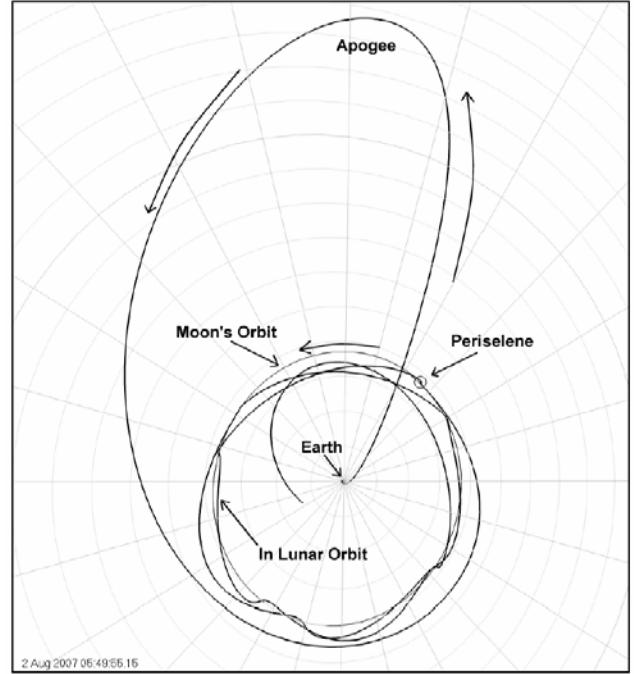
Perturbations to the candidate assembly orbit were simulated the High-Precision Orbit Propagator (HPOP) in Satellite Tool Kit (STK) 6.0. The most significant orbital perturbations to the target orbit were found to be due to aerodynamic drag effects, dictating the need for small re-boost maneuvers over the course of vehicle assembly. Orbital maintenance can be accomplished using either integral spacecraft reaction control systems (which are over-budgeted in terms of total impulse capability), or alternatively, by a dedicated module delivered by the “checkout” launch.

## 6.0 Lunar Mission Implementation

A subset of the vehicles and systems proposed for the *Mars for Less* mission design can also be used to execute high leverage lunar expeditions, either concurrently with a program of Mars exploration, or—in a more likely scenario—as a series of precursor missions. A lunar implementation of the candidate architecture would require 2 spacecraft per complete mission:

- A *Lunar Transfer Vehicle* (LTV), which would carry the crew to and from the lunar surface on 3-day transfers; and
- A *Lunar Surface Vehicle* (LSV), which would be dispatched to the moon ahead of the LTV on a 90-day Weak Stability Boundary (WSB) transfer, and would subsequently house astronauts for the duration of their surface stay.

The complete LSV would be identical to the MTSV (minus aeroshield and with an augmented lander/descent stage), and would be stocked with enough supplies and provisions to support three 4-person expeditions for 150 days each. The LTV would similarly be adapted from the Mars ERV design, and would consist of an ERV cabin,



**Figure 8: Weak Stability Boundary Lunar Transfer in Earth-Centered Inertial Coordinates [8]**

fully-fuelled upper stage and lander/descent stage.

The Lunar Surface Vehicle would be forward-deployed to the moon on a Weak Stability Boundary (WSB) transfer. This trajectory, illustrated in Figure 8 in Earth-Centered Inertial (ECI) coordinates, is designed to take advantage of the lunar weak stability boundary region, where sensitive chaotic dynamics prevail. Arriving in this region, a spacecraft can ballistically capture into lunar orbit with no propulsive maneuvers whatsoever—this was the technique used by the Japanese spacecraft Hiten to successfully reach the moon in 1991 [8]. This transfer requires only 75% of a conventional Hohmann transfer’s total  $\Delta v$ , and would allow the LSV to be dispatched to the lunar surface using only three propulsion stages leaving low-Earth orbit (as opposed to the 4 stages required in Mars expeditions). Upon Lunar Orbital Capture (LOC), the LSV would be de-orbited and subsequently landed by its descent stage.

**Table 2: Summary of Major Propulsive Maneuvers for Lunar Expeditions**

Maneuver	$\Delta v$ (m/s)
WSB Trans-Lunar Injection	3180
WSB Lunar Orbit Insertion	0
Standard Trans-Lunar Injection	3133
Standard Earth-Moon Midcourse Correction	50
Standard Lunar Orbit Insertion	920
Deorbit	18.6
Descent and Landing	1862
Ascent	1815
Lunar Orbit Circularization	19
Trans-Earth Injection	879
Moon-Earth Mid-Course Correction	50
Deterministic Maneuvers (each way)	50

Once the LSV was successfully deployed to the lunar surface, a crewed expedition would be launched in the Lunar Transfer Vehicle on a more conventional, Apollo-style 3-day free return trajectory. This vehicle, stocked to support 4 astronauts for approximately 2 weeks, would also be dispatched from LEO using only three of the outlined propulsion stages.

Each expedition to both the moon and Mars in this architecture leaves the primary crew habitat behind on the surface at the mission's conclusion, leading to an eventual build-up of habitable volumes at each destination. Furthermore, since each lunar mission is accomplished with a subset of the hardware and systems required for Mars expeditions, undertaking return flights to the moon would, in this approach, represent a guaranteed step towards developing a Mars exploration capability.

A summary of the major propulsive maneuvers required for lunar missions is presented in Table 2, and mass allocations for candidate spacecraft are presented in Appendix B, Tables B-1 and B-2. For all spacecraft, midcourse corrections, post-capture maneuvers and landing are performed using methane/oxygen propellants.

## 7.0 Artificial Gravity

Exposure to long periods of zero-gravity such as those inherent to manned Mars missions will inevitably result in detrimental health effects, including but not limited to cardiovascular deconditioning, demineralization of bones, and general muscular atrophy. While these issues are often cited as reasons to postpone manned space exploration beyond LEO in favor of continuing zero-g health research, in 40 years the only means by which many such risks can truly be circumvented is the only one not yet investigated in space: specifically, the generation of simulated gravity via centrifugal force.

Specified here as an optional component of the focus mission, artificial gravity for crewed Mars-bound spacecraft may be generated after TMI by deploying the final burnt-out propulsion stage on a 1500 meter tether, then using it as a counterweight and rotating the crewed vehicle about a centre of gravity 100 meters from the cabin at 2 RPM; this would result in the crew experiencing an approximately Mars-equivalent artificial gravity of 0.38g. While perhaps not sufficient to completely eliminate the adverse physiological effects of non-Earth gravity levels, this technique would nevertheless circumvent the need for what may otherwise prove a difficult acclimatization period for astronauts once on the Martian surface.

While neither the complete dynamics nor physiological effects of such an artificial gravity system have yet been ascertained, it is nevertheless noteworthy that, with the

set of hardware specified for the focus mission, flight testing of this system could be achieved with only one 20-tonne launch to low orbit (stripped down crew cabin plus empty upper stage) – a simple and informative test, the costs of which would be “in the noise” of overall mission development. Since all Mars-bound spacecraft in this architecture must be designed to operate in both zero- and Mars equivalent-gravity regardless of whether this option is implemented, flight testing this system is highly recommended, even if not employed in the overall mission architecture.

## 8.0 Discussion

The principal benefits of the *Mars for Less* mission design are in its anticipated reduction of development costs: with no need for new launch technology, the resources required to initiate this architecture are essentially already at hand—and thus, the price of beginning manned Mars exploration can possibly be reduced by an order of magnitude or more over prevailing HLLV-based proposals.

However, there nevertheless remain apparent drawbacks to this approach, which must be addressed. The key disadvantages of the *Mars for Less* approach surround the need for multiple launches: a need which carries with it the associated risks of multiple delays and multiple failures in both spacecraft assembly and operation. While deeper consideration of these issues permits different interpretations as to their overall significance, it is nevertheless acknowledged that this mission plan represents a generally more complex method of undertaking piloted Mars expeditions than the single- or dual-launch approaches which have dominated the majority of relevant analyses to date.

### 8.1 Multiple Launches

In terms of failure probability alone, the need for multiple launches is greatly to the detriment of the focus mission. Assuming a reasonable success rate of 98% for each candidate medium-lift launch system, the probability of losing a component of each Mars-bound payload due to launch failure would be approximately 11% per spacecraft, and about 22% per complete mission.

Yet perhaps the issue of failure in multi-launch mission designs is not, in and of itself, as critical as detractors contend. Indeed, in a *political* context, dividing mission mass into multiple components and launches may actually be regarded as a great strength: while multiple launches certainly increase the likelihood of losing a component because of launch failure, any such loss would represent only a small percentage of the total mission – and so, with the exception of the booster delivering the crew to orbit (which would only be one out of every 12 launches in the *Mars for Less* mission design), failure in any single launch could be offset by delivering a replacement component to

orbit; and while a failing launch system may be shut down for investigation, a program that uses not just multiple launches—but also multiple *types of launchers*—needn’t suffer the same fate. Conversely, a program of Mars exploration predicated on the use of a single, dedicated heavy lift booster may find itself indefinitely delayed—or even permanently grounded—as a result of only *one* failed launch, which is all but certain to happen at some point during the operation of any vehicle. For a program based on deploying entire missions with a single booster, losing that booster would mean losing:

- The crew;
- The spacecraft;
- The mission; and
- The possibility of near-term follow-on missions

Literally and figuratively, a heavy-lift booster keeps all the mission eggs in one basket; but launch failure in a program using smaller boosters is, in short, less likely to be either *mission or program critical*—an important level of flexibility for a program of manned Mars exploration to have, and something that heavy-lift launch vehicles simply can’t offer.

An additional benefit of using a larger number of smaller launch vehicles is the increased viability of undertaking international missions. The use of existing rockets flying discrete, compartmentalized payloads would permit simplified, more predictable costs per launch, mitigating to some extent the financial uncertainty for potentially interested but reticent nations. Using today’s launch vehicles—while cumulatively more risk intensive—avoids the otherwise nebulous development cost of new launch technology, and may thus prove to be a more viable method of undertaking an internationally cooperative mission.

## 8.2 Propulsion Stage Failure

The issue of TMI stage failure is in a sense more complicated than launch failure, frankly because it is far less decisive. Several different, entirely circumstantial scenarios for backup or abort exist for spacecraft experiencing this variety of malfunction, depending on which particular stage fails and at what point in the overall trans-Mars injection maneuver. While piloted spacecraft in this architecture are designed with the ability for Earth return in the event of stage failure between escape and minimum-energy TMI (from where the mission most likely cannot be recovered), the issue of backup versus abort is less absolute while still in orbit, and requires further analysis.

However, depending on the specific design of each TMI stage, the probability of injection failure would represent a much lesser portion of mission risk than initial launch failure. Assuming each propulsion stage would have a single engine reliability on the order of 99% for required burn durations, the likelihood of TMI failure would be approximately 4% only—and this in turn assumes that the loss of one engine would be sufficient to spoil the entire maneuver. Reference propulsion stages are designed with three parallel-configured engines, 2 outboard and 1 inboard; consequently, 2 of the motors would need to fail *per stage* to constitute a mission critical, irreparable malfunction. Such 2-engine failure is highly improbable (on the order of 1 chance in 10,000) and is thus considered a negligible component of overall mission risk.

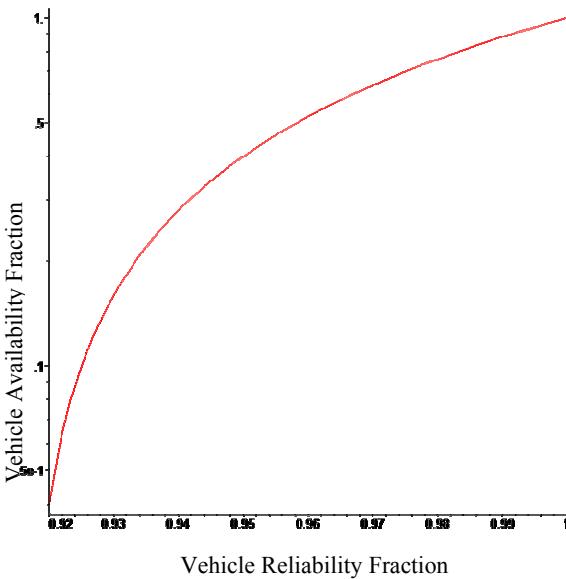
## 8.3 Launch Vehicle Availability

The availability of any given launch system (as a percentage of time throughout a year) can be estimated from its predicted reliability; production capacity; the ability of the launch operations system to support the desired launch rate; existing launch commitments; and its demonstrated stand-down time following failure. The relationship:

$$A = 1 - L \left( \frac{(1-R)t_d}{1 - \frac{1}{S}} \right)$$

can be used to estimate the *expected launch availability* ( $A$ ) based on the vehicle’s reliability  $R$ , nominal launch rate  $L$  (which, for the *Mars for Less* mission would be 6 per year), the estimated or demonstrated stand-down time  $t_d$  following launch failure, and the surge rate capacity  $S$ , representing the ability of a given launch vehicle to achieve a higher flight rate than planned [9]. For the *Mars for Less* mission design,  $A$  can be plotted as a function of  $R$ ; letting  $S=1.33$  (i.e. a vehicle which can handle 8 launches per year, or 2 more than planned) and  $t_d = 0.5$  years (estimated), the curve shown in Figure 9 can be produced.

While it is important to exercise caution with this method of analysis (the availability equation encounters a singularity for surge rates approaching 0), it nevertheless allows one to narrow the field of potential launch vehicles. For example, for a launch vehicle with a reliability of approximately 98% as discussed in Section 8.1, the vehicle can expected to be available approximately 77% of the time, suggesting that it would be desirable to have at least two individual systems for launching candidate payloads.



**Figure 9: Launch Vehicle Availability as a Function of Reliability**

#### 8.4 Launch Delays and LEO Wait Time

Delays in this or any other mission design are both inevitable and problematic, particularly when orbital assembly of a cryogenic propulsion system is required. Lengthy delays result in longer wait times in low-Earth orbit, which increase the amount of propellant lost from boil-off and reduce overall performance. Yet it was shown in Section 4.1 that, for both spacecraft in this architecture, optimal trans-Mars injections can still be accomplished for propulsion assembly times up to 4 months in duration (which does not even include the time needed for assembling initial, non-cryogenic components) at a realistic propellant boiloff rate of 1% per month.

In fact, closer examination of the impact which launch delays would have on reference propulsion systems shows that, for a fixed interval of time, the net performance available for trans-Mars injection actually increases with launch delays, since less total time is spent by the *entire* propulsion assembly in LEO. While this assumes a fixed total assembly time, exceeding any such allowance by a substantial margin would introduce the more serious problem of a missed launch window to Mars. Budgeting extra time overall to account for launch and assembly delays is thus recommended, and would not, as shown, be critically detrimental to candidate propulsion systems.

The duration of wait time on orbit also increases the risk of impact from orbital debris; but for time periods of less than 1 year at an altitude below 300 km, the probability of encountering even the smallest particles ( $\sim 1$  mm diameter) is on the order of 0.1% *at worst*, and decreases

exponentially for larger (and thus potentially more dangerous) collisions [8]. While debris impact on orbit is not considered a significant factor in assessing overall mission risk at this stage, such issues would nevertheless require further study with more specific space environment models for candidate spacecraft and assembly orbits.

## 9.0 Conclusions and Recommendations

### 9.1 Summary

The *Mars for Less* architecture represents an alternative method by which manned Mars missions can be initiated presently by either private or non-U.S. initiatives. By circumventing the need for heavy-lift launch vehicles and dividing mission mass into components that can be placed in orbit with boosters flying today, the cost of developing piloted expeditions can possibly be reduced by an order of magnitude or more over prevailing design studies—and the task of opening a new planet to humanity can potentially be realized by a larger number of interested parties.

Minimum-energy interplanetary trajectories are advocated for all cargo launches, while slightly expedited transfers with the option of 2-year free Earth return are recommended for piloted launches. A four-stage hydrogen/oxygen TMI propulsion system is designed and analyzed, and the sensitivity of TMI performance to varying assembly time and rates of on-orbit propellant boil-off time is assessed. Propulsion system performance is found to be sufficient to deliver candidate payloads to Mars after propulsion assembly times of up to four months at boiloff rates of 1%, though decreasing either is found to significantly improve net TMI performance.

Two individual spacecraft sizing iterations are discussed, based on past studies as well as combined OTS components and statistical data, with both methods of mass estimation converging on relatively common values. The largest discrepancy in mass estimation is found in sizing Earth Return Vehicles. While the candidate TMI propulsion system can still deliver both payloads to Mars—even under the most limiting mass estimates—an additional launch (above the nominal 6) may nevertheless be required to deliver supplemental equipment to the Earth Return Vehicle. It is argued that such a “checkout” launch would be prudent to verify any unmanned spacecraft prior to Earth departure, regardless of its need in spacecraft assembly. Outlined spacecraft and propulsion systems are intended as reference designs only, and further analysis is required to completely define and optimize the multiple considerations for Mars-bound vehicles.

A high-leverage lunar mission design using *Mars for Less* spacecraft is also discussed, which utilizes combined short-duration, negative C3 transfers and longer-duration, weak stability boundary transfers to deploy a significant exploration capability on the lunar surface. This is viewed

as a possible prerequisite for Mars mission applications, and itself allows for significantly more payload to be landed on the moon than through architectures currently under development by NASA [10].

The *Mars for Less* mission is advocated as a potentially desirable means by which either non-U.S., private- or international Mars exploration may be undertaken. In any scenario, the compartmentalized nature of required hardware in this mission would allow launch commitments for individual contributors to remain small, with impressive cumulative effects. Without the need for heavy-lift launch vehicle development, the initial investments required to undertake missions to Mars as proposed herein can be significantly reduced, allowing for wider participation across the aerospace community.

## 9.2 Further Work

Major areas requiring further research, which are beyond the scope of this study, include:

- Development of high specific impulse methane/oxygen propulsion technologies;
- Development of long-duration methane storage systems;
- Further research into hybrid thermal control systems, to maintain low rates of propellant boil-off for Earth-orbiting stages;
- Further design and sizing iterations of candidate spacecraft; and
- Further investigation and testing of rotating, tether-based artificial gravity systems.

## 9.3 Final Remarks

While many arguments have been presented in favor of using medium-lift launch systems instead of heavy-lift ones, it is not the objective of this paper to suggest that heavy-lift launch vehicles are either ineffective or unnecessary; however, to a large extent, basing future expeditions to Mars on their use will limit the degree to which entities without comparable launch technology can be involved. For a healthy, cost-sustainable space program to develop the private sector must play a significant role—and the launch industry must be encouraged to grow, particularly in early stages. The use of smaller launch vehicles in initial missions is far more likely to promote such growth than development of a single heavy-lift booster.

Heavy-lift launch vehicles will represent the ideal technology when a demand exists which equals their potential—but such a demand does not yet exist. If the true purpose of President Bush's Vision for Space Exploration is to open the inner solar system to mankind, then undertaking actual missions beyond Earth orbit should be the highest priority. Heavy lift launch

technology doesn't currently exist, and its re-development will require several years—years in which no space *exploration* will occur at all; and with our future in space hinging on the reliability of a single vehicle, the ability of NASA to persevere through setback or tragedy is also called into question.

Launch system redundancy, flexibility, and cost-effectiveness are the real keys to space exploration—especially if such exploration is to last, and parties other than NASA are to play a significant role. For this reason, a mission design using a larger number of more modest, more attainable launch vehicles for human Mars exploration has been made the reference mission design basis for the *MarsDrive Consortium*.

## Acknowledgments

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# Appendix A: Spacecraft Sizing Iterations

**Table A-1:** *Mass Allocations for Mars for Less Mission—First Iteration*

Earth Return Vehicle		Mars Transfer and Surface Vehicle	
Allocation	Mass (tonnes)	Allocation	Mass (tonnes)
<i>Launch 1</i>		<i>Launch 1</i>	
Habitat Structure	3.0	Habitat Structure	3.0
Life Support System	1.0	Life Support System	3.0
Consumables	3.4	Consumables	9.0
Power (5 kWe PVA)	1.0	Power (5 kWe PVA)	1.0
Reaction Control System	0.5	Reaction Control System	0.5
RCS Propellant	3.3	RCS Propellant	3.0
Communications and Data	0.1	Communications and Data	0.2
Interior	0.5	Interior	1.0
EVA Suits (4)	0.4	EVA Suits (4)	0.4
Margin	1.5	Biomedical Package	0.2
Heat Shield	1.8	Crew (4)	0.4
Upper Propulsion Stage	1.5	Margin	2.5
Hydrogen Feedstock	6.3	<b>Launch 1 Total</b>	<b>24.2</b>
Chemical Plant	0.5		
<b>Launch 1 Total</b>	<b>24.8</b>	<i>Launch 2</i>	
<i>Launch 2</i>		Cargo Module Structure	2.0
RTG (100 kWe)	3.0	Science Equipment and Lab	2.0
RTG Truck	0.4	Open Rover	0.5
Lower Propulsion Stage	6.0	Pressurized Rover	2.0
Aeroshield	6.2	Margin	1.0
Lander (fully fuelled)	9.2	Aeroshield	5.8
<b>Launch 2 Total</b>	<b>24.8</b>	Lander (fully fuelled)	8.5
<b>ERV Total</b>	<b>49.6</b>	<b>Launch 2 Total</b>	<b>21.8</b>
		<b>MTSV Total</b>	<b>46.0</b>

**Table A-2: Earth Return Vehicle Sizing—Second Iteration**

Crew Number	4	Legend			
Habitable Volume Fraction	0.65	Input Variables			
Volume/Person [m3]	20	Output			
Total Pressurized Volume [m3]	123				
Flight Duration [d]	180				
Mission Duration [d]	180				
Initial Dry Mass Estimate [kg]	30490				
Average Spacecraft Power Requirements [kW]	5.00				
Solar Constant at Max. Distance [W/m2]	590.00				
EOL Solar Panel Efficiency	0.14				
Cell Packing Factor	0.85				
Required Solar Panel Area	71.21				
Blanket Specific Mass [kg/m2]	1.70				
Blanket Mass [kg]	121.07				
Total Mass [kg]	220.12				
Lander & RCS System Specific Impulse [s]	380.00				
<b>Life Support System</b>					
Subsystem	Allocation	Units	Total [kg]	Energy [W-h]	
<b>Galley &amp; Food Systems</b>					
Food	2.30	kg/p/d	1656	NA	
Freezers (not including food)	400.00	kg	400	16800.00	
Conventional Ovens	50.00	kg	50	3600.00	
Microwave Ovens	70.00	kg	70	1296.00	
Kitchen/Oven Cleaning Supplies	0.25	kg/d	45	NA	
Sink, Spigot for hydration of food and drinking water	15.00	kg	15	NA	
Dishwater	40.00	kg	40	1152.00	
Cooking/Eating Supplies	5.00	kg/p	20	NA	
<b>Waste Collection System</b>					
System (2 Toilets)	90.00	kg	90	27.00	
WCS Supplies	0.05	kg/p/d	36	NA	
Contingency faecal and urine collection bags/mittens	0.23	kg/p/d	165.6	NA	
<b>Personal Hygiene</b>					
Shower	75.00	kg	75	960.00	
Handwash/mouthwash faucet	8.00	kg	8	NA	
Personal hygiene kit	1.80	kg/p	7.2	NA	
Hygiene supplies (consumables)	0.08	kg/p/d	54	NA	
<b>Clothing</b>					
Clothing (6 weeks)	99.00	kg/p	396	NA	
Washing machine	100.00	kg	100	1440.00	
Clothes dryer	60.00	kg	60	2400.00	

<b>Recreational Equipment &amp; Personal Stowage</b>				
Personal stowage	50.00	kg/p	200	336.00
<b>Housekeeping</b>				
Vacuum (prime + 2 spares)	13.00	kg	13	48.00
Disposable wipes for housecleaning	0.00	kg/p/d	0	NA
Trash compactor/trash lock	150.00	kg	150	102.00
Trash bags	0.05	kg/p/d	36	NA
<b>Operational Supplies &amp; Restraints</b>				
Operational supplies (diskettes, ziplocks, tape...)	20.00	kg/p	80	NA
Restraints	100.00	kg	100	NA
<b>Maintenance: All Repairs in Habitable Areas</b>				
Hand tools and accessories	75.00	kg	75	NA
Spare parts and consumables			0	NA
Test equipment (oscilloscopes, gauges, etc.)	250.00	kg	250	12.00
Fixtures, large machine tools, gloveboxes, etc	500.00	kg	500	12.00
<b>Photography</b>				
Equipment (Still and video cameras, lenses, etc.)	120.00	kg	120	384.00
Film (assumes digital approach)	0.00	kg	0	NA
<b>Sleep Accommodations</b>				
Sleep provisions (restraints only)	9.00	kg/p	36	NA
<b>Crew Health Care</b>				
Exercise Equipment	145.00	kg	145	870.00
Medical/Surgical/Dental suite	250.00	kg	250	180.00
Medical/Surgical/Dental consumables	125.00	kg	125	NA
<b>Total Crew System Mass [kg] and Energy/day [kW-h]</b>				<b>5367.8</b>
***Energy Req. Margin = 2***				
<b>Average Power [kW]</b>				<b>2.47</b>

**Definition of Key Terms [7]:**

- ECLSS: Environmental Control and Life Support Systems
- 4BMS: Four-Bed Molecular Sieve
- TCCS: Trace Contaminant Control System
- OGA: Oxygen Generation Assembly
- VCD: Vapour Compression Distillation
- DOD: Depth of Discharge

<b>Earth Return Vehicle—Total Mass Allocations</b>			
<b>System</b>	<b>Sub-System</b>	<b>Description</b>	<b>Mass Allocation [kg]</b>
Structure	Primary	10% of IDME	3049
	Mechanisms	2.0% of IDME	610
ADCS/RCS		2.0% of IDME	610
ECLSS	Air	4BMS, TCCS, OGA, x2 Redundant	680
	Water	VCD x2 Redundant	200
Thermal	Active	2.6% of IDME	793
	Surface Radiator	1.4% of IDME	427
	TPS	3.0% of IDME	915
Crew Systems		See Table	5368
EVA		1 suit per crew member	400
Logistics		2% of IDME	610
Power	Battery	50 kW-h Li-Ion Batteries, 100% DOD	300
	PVA	See Table	220
Heat Shield		10% Earth Reentry Mass	1418
<b>Dry Mass Total</b>			<b>15599</b>
H2 Feedstock	Hydrogen		6778
	Storage System	Use Upper Stage Tank	0
ISPP Plant	Sabatier Reactor	Complete System [???	500
RTG		50 kWe	1500
		RTG Truck	400
ERV Propulsion	Lower Stage (dry)	Delta-V = 4415 m/s; 8% dry mass fraction	7900
	Upper Stage (dry)	Delta-V = 2815 m/s; 8% dry mass fraction	1510
<b>Total Surface Mass</b>			<b>34187</b>
Lander	Dry Mass	10% of Payload Dry Mass	3419
	Propellant Mass	Delta-V for Mars Landing = 525 m/s	5687
Parachute		3x Supersonic Parachutes	600
<b>Total Entry Mass</b>			<b>43893</b>
Aeroshield		15% of Total Entry Mass	6584
Cruise Stage	Dry Mass	2.0% of IDME	610
	Propellant Mass	Delta-V = 200 m/s	2816
<b>Total Spacecraft Mass</b>			<b>53902</b>

**Table A-2: Mars Transfer and Surface Vehicle Sizing—Second Iteration**

	4	Legend		
Crew Number	4	Input Variables		
Habitable Volume Fraction	0.50	Output		
Volume/Person [m3]	20			
Total Pressurized Volume [m3]	160			
Flight Duration [d]	180			
Mission Duration [d]	680			
<b>Initial Dry Mass Estimate [kg]</b>	<b>33388</b>			
Average Spacecraft Power Requirements [kW]	5.00			
Solar Constant at Max. Distance [W/m2]	590.00			
EOL Solar Panel Efficiency	0.14			
Cell Packing Factor	0.85			
Required Solar Panel Area	71.21			
Blanket Specific Mass [kg/m2]	1.70			
Blanket Mass [kg]	121.07			
<b>Total Mass [kg]</b>	<b>220.12</b>			
Lander & RCS System Specific Impulse [s]	380.00			
<b>Life Support System</b>				
Subsystem	Allocation	Units	Total [kg]	Energy [W-h]
<b>Galley &amp; Food Systems</b>				
Food	2.30	kg/p/d	6256	NA
Freezers (not including food)	400.00	kg	400	16800.00
Conventional Ovens	50.00	kg	50	3600.00
Microwave Ovens	70.00	kg	70	1296.00
Kitchen/Oven Cleaning Supplies	0.25	kg/d	170	NA
Sink, Spigot for hydration of food and drinking water	15.00	kg	15	NA
Dishwater	40.00	kg	40	1152.00
Cooking/Eating Supplies	5.00	kg/p	20	NA
<b>Waste Collection System</b>				
System (2 Toilets)	90.00	kg	90	27.00
WCS Supplies	0.05	kg/p/d	136	NA
Contingency faecal and urine collection bags/mittens	0.23	kg/p/d	625.6	NA
<b>Personal Hygiene</b>				
Shower	75.00	kg	75	960.00
Handwash/mouthwash faucet	8.00	kg	8	NA
Personal hygiene kit	1.80	kg/p	7.2	NA
Hygiene supplies (consumables)	0.08	kg/p/d	204	NA
<b>Clothing</b>				
Clothing (6 weeks)	99.00	kg/p	396	NA
Washing machine	100.00	kg	100	1440.00
Clothes dryer	60.00	kg	60	2400.00

<b>Recreational Equipment &amp; Personal Stowage</b>				
Personal stowage	50.00	kg/p	200	336.00
<b>Housekeeping</b>				
Vacuum (prime + 2 spares)	13.00	kg	13	48.00
Disposable wipes for housecleaning	0.00	kg/p/d	0	NA
Trash compactor/trash lock	150.00	kg	150	102.00
Trash bags	0.05	kg/p/d	136	NA
<b>Operational Supplies &amp; Restraints</b>				
Operational supplies (diskettes, ziplocks, tape...)	20.00	kg/p	80	NA
Restraints	100.00	kg	100	NA
<b>Maintenance: All Repairs in Habitable Areas</b>				
Hand tools and accessories	300.00	kg	300	NA
Spare parts and consumables			0	NA
Test equipment (oscilloscopes, gauges, etc.)	500.00	kg	500	12.00
Fixtures, large machine tools, gloveboxes, etc	1000.00	kg	1000	12.00
<b>Photography</b>				
Equipment (Still and video cameras, lenses, etc.)	120.00	kg	120	384.00
Film (assumes digital approach)	0.00	kg	0	NA
<b>Sleep Accommodations</b>				
Sleep provisions (restraints only)	9.00	kg/p	36	NA
<b>Crew Health Care</b>				
Exercise Equipment	145.00	kg	145	870.00
Medical/Surgical/Dental suite	1000.00	kg	1000	180.00
Medical/Surgical/Dental consumables	500.00	kg	500	NA
<b>Total Crew System Mass [kg] and Energy/day [kW-h]</b>				<b>13002.8</b>
***Energy Req. Margin = 2***				
<b>Average Power [kW]</b>				<b>2.47</b>

<b>Mars Transfer and Surface Vehicle—Total Mass Allocations</b>			
<b>System</b>	<b>Sub-System</b>	<b>Description</b>	<b>Mass Allocation [kg]</b>
Structure	Primary	12% of IDME	4007
	Airlock	2 crew, dual egress	1000
	Mechanisms	4.0% of IDME	1336
ADCS/RCS		2.0% of IDME	668
ECLSS	Air	4BMS, TCCS, OGA, x3 Redundant	1020
	Water	VCD x2 Redundant	200
Thermal	Active	2.6% of IDME	868
	Surface Radiator	1.4% of IDME	467
	TPS	6.0% of IDME	2003
Crew Systems		See Table	13003
EVA	Suits	1 suit per crew member, crew/2 spares	600
	Pressurized Rover	Open Cockpit, Pressurized Cabin	1500
Logistics		5.0% of IDME	1669
Power	Battery	50 kW-h Li-Ion Batteries, 100% DOD	300
	PVA	See Table	220
<b>Dry Mass Total</b>			<b>28861</b>
Lander	Dry Mass	10% of Payload Dry Mass	2886
	Propellant Mass	Delta-V for Mars Landing = 625 m/s	5795
Parachute		3x Supersonic Parachutes	600
<b>Total Entry Mass</b>			<b>38142</b>
Aeroshield		15% of Total Entry Mass	5721
Cruise Stage	Dry Mass	2.0% of IDME	668
	Propellant Mass	Delta-V = 200 m/s	2454
<b>Total Spacecraft Mass</b>			<b>46985</b>

## Appendix B: Lunar Mission Mass Allocations

<b>Table B-1: Lunar Transfer Vehicle (LTV) Sizing</b>				
Crew Number	4			<b>Legend</b>
Habitable Volume Fraction	0.50			Input Variables
Volume/Person [m <sup>3</sup> ]	20			Output
Total Pressurized Volume [m <sup>3</sup> ]	160			
Flight Duration [d]	90			
Mission Duration [d]	12			
<b>Initial Dry Mass Estimate [kg]</b>	<b>26268</b>			
Average Spacecraft Power Requirements [kW]	10.00			
Solar Constant at Max. Distance [W/m <sup>2</sup> ]	1360.00			
EOL Solar Panel Efficiency	0.14			
Cell Packing Factor	0.85			
Required Solar Panel Area	61.79			
Blanket Specific Mass [kg/m <sup>2</sup> ]	1.70			
Blanket Mass [kg]	105.04			
<b>Total Mass [kg]</b>	<b>190.99</b>			
Lander & RCS System Specific Impulse [s]	380.00			
<b>Life Support System</b>				
Subsystem	Allocation	Units	Total [kg]	Energy [W-h]
<b>Galley &amp; Food Systems</b>				
Food	2.30	kg/p/d	110.4	NA
Freezers (not including food)	200.00	kg	200	16800.00
Conventional Ovens		kg	0	3600.00
Microwave Ovens		kg	0	1296.00
Kitchen/Oven Cleaning Supplies		kg/d	0	NA
Sink, Spigot for hydration of food and drinking water		kg	0	NA
Dishwater		kg	0	1152.00
Cooking/Eating Supplies	5.00	kg/p	20	NA
<b>Waste Collection System</b>				
System (2 Toilets)	45.00	kg	45	27.00
WCS Supplies	0.05	kg/p/d	2.4	NA
Contingency faecal and urine collection bags/mittens	0.23	kg/p/d	11.04	NA
<b>Personal Hygiene</b>				
Shower	20.00	kg	20	960.00
Handwash/mouthwash faucet	8.00	kg	8	NA
Personal hygiene kit	1.80	kg/p	7.2	NA
Hygiene supplies (consumables)	0.08	kg/p/d	3.6	NA
<b>Clothing</b>				
Clothing (6 weeks)	30.00	kg/p	120	NA

Washing machine	0.00	kg	0	1440.00
Clothes dryer	0.00	kg	0	2400.00
<b>Recreational Equipment &amp; Personal Stowage</b>				
Personal stowage	25.00	kg/p	100	336.00
<b>Housekeeping</b>				
Vacuum (prime + 2 spares)	13.00	kg	13	48.00
Disposable wipes for housecleaning	0.00	kg/p/d	0	NA
Trash compactor/trash lock	50.00	kg	50	102.00
Trash bags	0.05	kg/p/d	2.4	NA
<b>Operational Supplies &amp; Restraints</b>				
Operational supplies (diskettes, ziplocks, tape...)	20.00	kg/p	80	NA
Restraints	100.00	kg	100	NA
<b>Maintenance: All Repairs in Habitable Areas</b>				
Hand tools and accessories	75.00	kg	75	NA
Spare parts and consumables			0	NA
Test equipment (oscilloscopes, gauges, etc.)	100.00	kg	100	12.00
Fixtures, large machine tools, gloveboxes, etc	100.00	kg	100	12.00
<b>Photography</b>				
Equipment (Still and video cameras, lenses, etc.)	40.00	kg	40	384.00
Film (assumes digital approach)	0.00	kg	0	NA
<b>Sleep Accommodations</b>				
Sleep provisions (restraints only)	9.00	kg/p	36	NA
<b>Crew Health Care</b>				
Exercise Equipment	0.00	kg	0	870.00
Medical/Surgical/Dental suite	40.00	kg	40	180.00
Medical/Surgical/Dental consumables	40.00	kg	40	NA
<b>Total Crew System Mass [kg] and Energy/day [kW-h]</b>			<b>1324.04</b>	<b>59.24</b>
***Energy Req. Margin = 2***				
<b>Average Power [kW]</b>				<b>2.47</b>

### Lunar Transfer Vehicle—Total Mass Allocations

System	Sub-System	Description	Mass Allocation [kg]
Structure	Primary	10% of IDME	3049
	Mechanisms	2.0% of IDME	610
ADCS/RCS		2.0% of IDME	610
ECLSS	Air	4BMS, TCCS, OGA, x2 Redundant	680
	Water	VCD x2 Redundant	200
Thermal	Active	2.6% of IDME	793
	Surface Radiator	1.4% of IDME	427
	TPS	3.0% of IDME	915
Crew Systems		See Table	1324
EVA		1 suit per crew member	400
Logistics		2% of IDME	610
Power	Battery	50 kW-h Li-Ion Batteries, 100% DOD	300
	PVA	See Table	191
		10% Earth Reentry Mass	1011
<b>Dry Mass Total</b>			<b>10108</b>
Lunar Ascent Stage	Dry Mass	Common Upper Stage	1510
	Propellant Mass	Total Delta-V = 2815 m/s	11401
<b>Total Surface Payload</b>			<b>23018</b>
Lunar Lander	Dry Mass	10% Total Surface Payload	2302
	Propellant Mass	Total Delta-V = 1900 m/s	16832
<b>Total Spacecraft Mass</b>			<b>42152</b>

Table B-2: Lunar Surface Vehicle (LSV)—Sizing

				Legend
				Input Variables
				Output
Crew Number	4			
Habitable Volume Fraction	0.50			
Volume/Person [m <sup>3</sup> ]	20			
Total Pressurized Volume [m <sup>3</sup> ]	160			
Flight Duration [d]	180			
Mission Duration [d]	450			
<b>Initial Dry Mass Estimate [kg]</b>	<b>33388</b>			
Average Spacecraft Power Requirements [kW]	50.00			
Solar Constant at Max. Distance [W/m <sup>2</sup> ]	1360.00			
EOL Solar Panel Efficiency	0.14			
Cell Packing Factor	0.85			
Required Solar Panel Area	308.95			
Blanket Specific Mass [kg/m <sup>2</sup> ]	1.70			
Blanket Mass [kg]	525.21			
<b>Total Mass [kg]</b>	<b>954.93</b>			
Lander & RCS System Specific Impulse [s]	380.00			
<b>Life Support System</b>				
Subsystem	Allocation	Units	Total [kg]	Energy [W-h]
<b>Galley &amp; Food Systems</b>				
Food	2.30	kg/p/d	4140	NA
Freezers (not including food)	400.00	kg	400	16800.00
Conventional Ovens	50.00	kg	50	3600.00
Microwave Ovens	70.00	kg	70	1296.00
Kitchen/Oven Cleaning Supplies	0.25	kg/d	112.5	NA
Sink, Spigot for hydration of food and drinking water	15.00	kg	15	NA
Dishwater	40.00	kg	40	1152.00
Cooking/Eating Supplies	5.00	kg/p	20	NA
<b>Waste Collection System</b>				
System (2 Toilets)	90.00	kg	90	27.00
WCS Supplies	0.05	kg/p/d	90	NA
Contingency faecal and urine collection bags/mittens	0.23	kg/p/d	414	NA
<b>Personal Hygiene</b>				
Shower	75.00	kg	75	960.00
Handwash/mouthwash faucet	8.00	kg	8	NA
Personal hygiene kit	1.80	kg/p	7.2	NA
Hygiene supplies (consumables)	0.08	kg/p/d	135	NA
<b>Clothing</b>				
Clothing (6 weeks)	99.00	kg/p	396	NA
Washing machine	100.00	kg	100	1440.00
Clothes dryer	60.00	kg	60	2400.00

<b>Recreational Equipment &amp; Personal Stowage</b>				
Personal stowage	50.00	kg/p	200	336.00
<b>Housekeeping</b>				
Vacuum (prime + 2 spares)	13.00	kg	13	48.00
Disposable wipes for housecleaning	0.00	kg/p/d	0	NA
Trash compactor/trash lock	150.00	kg	150	102.00
Trash bags	0.05	kg/p/d	90	NA
<b>Operational Supplies &amp; Restraints</b>				
Operational supplies (diskettes, ziplocks, tape...)	20.00	kg/p	80	NA
Restraints	100.00	kg	100	NA
<b>Maintenance: All Repairs in Habitable Areas</b>				
Hand tools and accessories	300.00	kg	300	NA
Spare parts and consumables			0	NA
Test equipment (oscilloscopes, gauges, etc.)	500.00	kg	500	12.00
Fixtures, large machine tools, gloveboxes, etc	1000.00	kg	1000	12.00
<b>Photography</b>				
Equipment (Still and video cameras, lenses, etc.)	120.00	kg	120	384.00
Film (assumes digital approach)	0.00	kg	0	NA
<b>Sleep Accommodations</b>				
Sleep provisions (restraints only)	9.00	kg/p	36	NA
<b>Crew Health Care</b>				
Exercise Equipment	145.00	kg	145	870.00
Medical/Surgical/Dental suite	1000.00	kg	1000	180.00
Medical/Surgical/Dental consumables	500.00	kg	500	NA
<b>Total Crew System Mass [kg] and Energy/day [kW-h]</b>			<b>10456.7</b>	<b>59.24</b>
***Energy Req. Margin = 2***				
<b>Average Power [kW]</b>				

### Lunar Surface Vehicle—Total Mass Allocations

System	Sub-System	Description	Mass Allocation [kg]
Structure	Primary	12% of IDME	4007
	Airlock	2 crew, dual egress	1000
	Mechanisms	4.0% of IDME	1336
ADCS/RCS		2.0% of IDME	668
ECLSS	Air	4BMS, TCCS, OGA, x3 Redundant	1020
	Water	VCD x2 Redundant	200
Thermal	Active	2.6% of IDME	868
	Surface Radiator	1.4% of IDME	467
	TPS	6.0% of IDME	2003
Crew Systems		See Table	10457
EVA	Suits	1 suit per crew member, crew/2 spares	600
	Pressurized Rover	Open Cockpit, Pressurized Cabin	1500
Logistics		5.0% of IDME	1669
Power	Battery	50 kW-h Li-Ion Batteries, 100% DOD	300
	PVA	See Table	955
<b>Dry Mass Total</b>			<b>27050</b>
Lander	Dry Mass	10% of Payload Dry Mass	2705
	Propellant Mass	Delta-V for Deorbit & Landing = 1900 m/s	19780
<b>Total Lunar Orbit Mass</b>			<b>49534</b>