THE MARS SOCIETY OF CALTECH HUMAN EXPLORATION OF MARS ENDEAVOR

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Abstract

The Mars Society of Caltech Human Exploration of Mars Endeavor (Mars SCHEME) is a detailed description of robotic and human missions necessary to establish a permanent human presence on the surface of Mars. The sequence begins in 2009 with a robotic Mars sample return mission on a larger scale than that currently planned. This is followed in 2011 by a pair of HEDS landers designed to test in-situ propellant production and other necessary technologies. Cargo for the human crews is sent in 2016 and in 2018, with the first five-member crew traveling to Mars during the 2020 opportunity. The Mars SCHEME features design redundancy; for example, the capsules for Earth ascent, Mars ascent, and Earth arrival are based upon a common design. Systems redundancy is also included to provide multiple habitats on Mars and in interplanetary space. The plan uses only chemical propulsion, starting with the Z-5 launch vehicle that can deliver up to 112,000 kg to low Earth orbit. Costs of human missions are comparable with those of the NASA Design Reference Mission 3.0. Human missions have low recurring costs, high reliability, and high scientific return. Extensive computer simulations were used to develop launch vehicles and trajectories. Further details are available at *http://mars.caltech.edu/*.

1. Overview

1.1 Statement of Design Problem

Our design problem is the creation of the safest, most cost-effective, and most easily achievable human Mars mission architecture possible. This architecture must also lead to a permanent human presence on Mars and elsewhere.

Our decision in favor of this design problem is first based on what technological decisions are currently most crucial to getting a human Mars mission off the ground. Because such a mission is still in the earliest of design stages and lacks funding, the broader mission architecture decisions are currently more important than the detailed design of individual components. Our design problem is therefore centered on fundamental mission architecture decisions that will shape details later in the design process.

Second in our selection of a design problem was context. We will be relating future human Martian exploration to current robotic Martian exploration and human spaceflight efforts, easing the transition between the two. For this reason, our primary design problem of human Mars mission design will also encompass robotic missions to occur before the first human mission, additional applications of mission hardware, evolution of hardware needed for long-term exploration and settlement, and the fiscal and political pressures that NASA and its potential partners will face in their attempts to send humans to Mars.

1.2 Robotic Predecessors to a Human Mission

At present, NASA's Mars Surveyor program sends robotic spacecraft to Mars in order to accumulate valuable scientific data. Robotic Mars spacecraft can help us send humans to the Red Planet in many ways. Those identified in this study are:

1.2.1. Communications and Navigation Infrastructure. A human Mars mission will require near-constant communication with Earth. Automated communication satellites near Mars will be necessary for occasions when direct radio contact with Earth is impossible. In addition, the mission will require good navigation, both for precision landing of vehicles and for surface rover guidance, so navigation satellites will also be needed. Dual-purpose satellites could fulfill both functions.

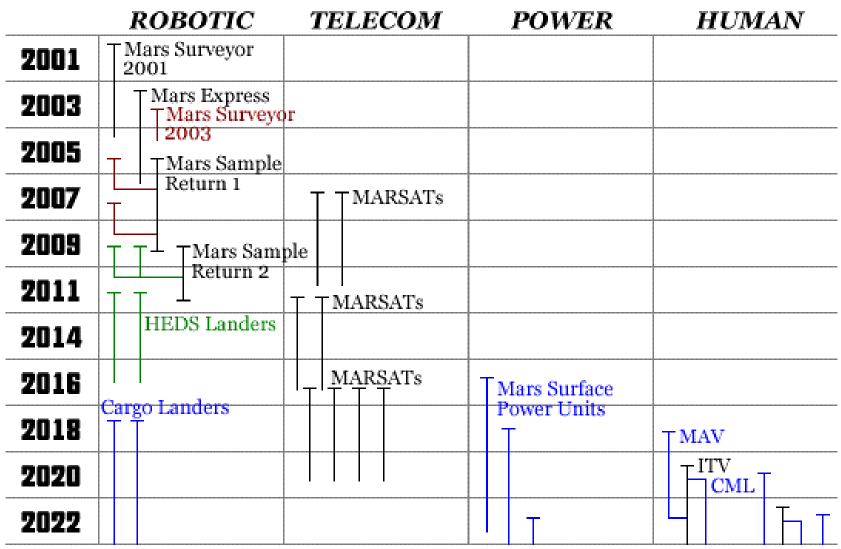
1.2.2. Testing Technologies in the Martian Environment. Though far more expensive than Earth-based tests, operating a technology on the surface of Mars provides the most useful data on how it functions in the Martian environment. Technologies such as in-situ propellant production, precision landings, and aerocapture should be tested with robots if they are to be included in a human mission.

1.2.3. Characterization of the Martian Environment. Before sending humans to Mars, we must better understand the environment that awaits them. Radiation levels, soil oxidants, dust damage to surfaces, and other potential hazards can be studied by robotic landers. Studies of more complex interactions between Martian soil and humans may require returning Martian samples to Earth for analysis. A network of surface meteorological stations and orbiters could survey the pressure, temperature, and wind conditions at potential landing sites.

1.2.4. Scientific Study of Mars. In addition to laying the technological groundwork for human exploration of Mars, scientific instruments on robotic Mars missions will increase our knowledge of the Red Planet. This will place the astronauts' observations in context. More important, it allows us to send the astronauts with the proper tools to answer the most intriguing questions raised by the discoveries of the next generation of robotic probes.

These objectives will be met by three classes of Mars missions. The micromissions, of which the (failed) Deep Space 2 probes were the first, are already part of the Mars exploration program. These are fast and cheap, so many can be flown. The Small Mars Landers (SMLs), such as Mars Pathfinder and Mars Polar Lander (several hundred kilograms), can carry more instruments but are still small. The largest of the robotic probes, the Intermediate-Sized Mars Landers (ISMLs), will have masses in the thousands of kilograms and will fill the technological gap between the current SMLs and the heavy machinery needed for human missions. The ISMLs will also be able to operate high-power experiments and return large samples to Earth for analysis. A possible sequence of robotic missions to Mars (excluding micromissions) leading up to a human mission is shown in Figure 1.2.1 and discussed in more detail in §4.

Figure 1.2.1. Possible Future Mars Missions

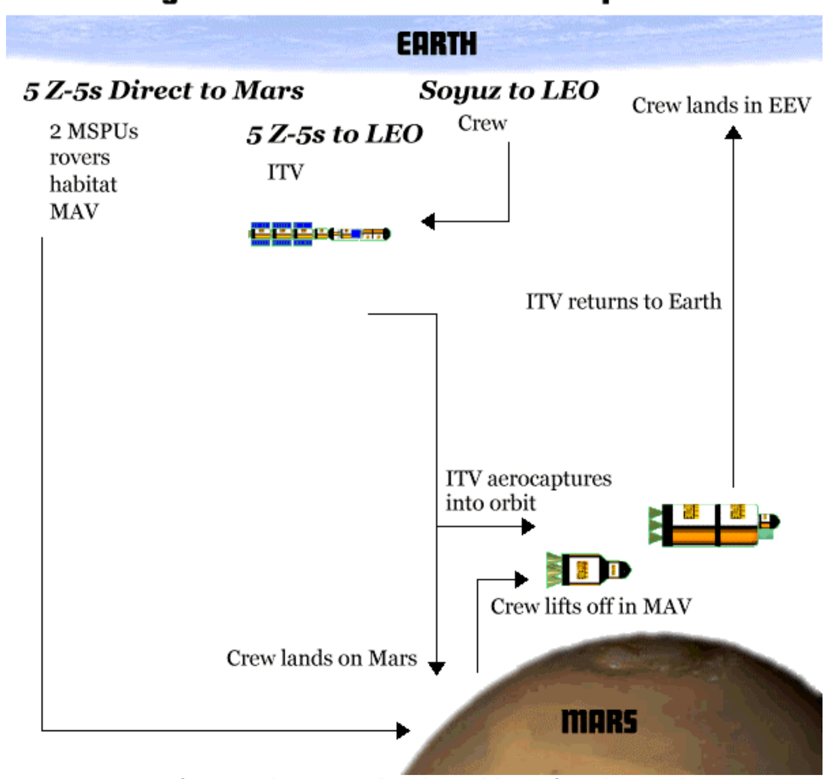


1.3 First Human Mars Mission

The first human Mars mission begins in 2016, when an unmanned Z-5 rocket lifts off from Earth. The Z-5, which can place up to 112 metric tons (112,000 kilograms) into orbit, is roughly the size of the Saturn V that once took astronauts to the Moon. Its first payload is a Mars Surface Power Unit (MSPU), launched directly to Mars. It descends to the surface using parachutes and rockets. In 2018, four more Z-5s launch directly to Mars with Large Mars Landers (LMLs) carrying a Mars Ascent Vehicle (MAV), a crew habitat, a cargo lander with two rovers and other science equipment, and a second MSPU.

When the first MSPU lands, it generates power and deploys small rovers that will connect power cables to the habitat and MAV when they arrive. The MAV contains a small capsule on a liquid hydrogen (LH_2)/liquid oxygen (LOX) powered rocket stage. It arrives on Mars with its LH_2 tank full but its LOX tank empty. Using MSPU power, the MAV draws carbon dioxide (CO_2) from the Martian air. The MAV's array of electrolysis cells pulls an oxygen atom from each CO_2 molecule and liquefies the resulting oxygen, storing it in its LOX tank. By early 2020, the MAV propellant tanks are full.

Figure 1.3.1. Human Mission Sequence



In 2020, another set of five Z-5 launches assembles an Interplanetary Transfer Vehicle (ITV) in low Earth orbit (LEO). The ITV consists of the crew Mars lander (CML), a habitat, the Earth entry vehicle (EEV), a truss, and four LH_2/LOX rocket stages. When the ITV is complete, five astronauts travel to it in an Earth ascent vehicle (EAV) launched by a Soyuz booster.

The first three of the ITV's four LH_2/LOX stages fire to raise the ITV orbit to near Earth escape. Finally, the fourth and final stage fires, sending the ITV on a 146-day trajectory to Mars. During the trip to Mars, the ITV spins at four revolutions per minute to provide about 1/3 of normal Earth gravity in the habitat. Upon arrival at Mars, the crew enters the Mars lander and separates from the habitat, descending to the surface. The truss is jettisoned, and the remainder of the ITV aerocaptures into Mars orbit.

The crew explores Mars for over 500 days, living in either their lander or in the habitat. At the end of their stay on Mars, the crewmembers enter the MAV and blast into Mars orbit. There they dock to the ITV and transfer into its habitat. The ITV propulsion system fires its engines to send the crew back toward Earth, where they arrive 177 days later. The crew performs a direct entry at Earth in the EEV, splashing down in the Pacific Ocean in 2023.

1.4 Subsequent Missions

A crew of five can be sent to the same site on Mars every 2.14 years using this architecture if, in each launch opportunity, six Z-5s and a Soyuz launch a MAV, an ITV, and an EEV. In addition, MSPU systems, consumables, and science payloads are replaced whenever they are consumed, break, or wear out. However, additional possibilities may be opened after the first mission. Since there will be a significant infrastructure on Mars after the first mission, it makes sense to make this primitive Mars base less dependent on Earth to reduce the cost of the Mars missions. For example, finding usable in-situ water would reduce the costs of resupply from Earth.

1.5. Methods and Validation

Three computer simulations were designed using the C programming language to calculate interplanetary trajectories and launch capability (from Earth and Mars).

1.5.1. Trajectory Program. The trajectory program analyzed mean Keplerian orbital elements of Earth and Mars and assumed a heliocentric conic section transfer orbit. Within these approximations, trajectories were calculated exactly. The program was validated by comparison to previous interplanetary probes.[1,2] Table 9.2.1 displays this validation, using C_3 as the benchmark trajectory feature. After establishing this small absolute error in C_3 for recent Mars trajectories, the program was deemed valid for calculations used in designing this mission. (Note that because the 200 km error in the Mars Climate Orbiter trajectory is small compared to the distance scale of the inner planets, Climate Orbiter was considered an acceptable reference against which to validate the trajectory program.)

Table 1.5.1.1. Validation of Trajectory Program

Probe	*Predicted* C_3 *[km²/s²]*	*Actual* C_3 *[km²/s²]*	*Absolute Error [km²/s²]*
Mars Global Surveyor	9.9846	10.0194	0.0348
Mars Climate Orbiter	10.93	11.19	0.26

It was initially desired to run the trajectory program in a faster, two-dimensional mode in which the inclination of the Mars orbit was neglected. A quick check, however, indicates that this is not a good idea; compare the parameters of the 2022 Mars mission trajectory as shown in the table below. In particular, we note that the two-dimensional assumption is optimistic, as it is in nearly all cases. (Earth departure on SAT 17 SEP 2022 and Mars arrival on SUN 26 MAR 2023 were assumed for 190 day transit time.)

Table 1.5.1.2. Comparison of 3D and 2D Trajectory Simulations in 2022 Opportunity

Simulation Type	*Earth departure* C_3 *[km²/s²]*	*Mars entry velocity [km/s]*	*Launch declination*
Three-dimensional	19.9	6.27	39°N
Two-dimensional	18.0	6.23	23°N

1.5.2. Launch Vehicle Program. The launch vehicle program assumed a gravity turn trajectory, thrust, and a simple model for air drag. Within these approximations, the payload capacity to low-Earth orbit (LEO) was calculated exactly. The Space Shuttle was used as a test case for the launch vehicle program, which predicted a payload capacity of 28.442 MT to LEO, as opposed to an actual 29.5 MT,[3] an error of 3.59%. Given that the error is expected to be greatest for vehicles on which the payload is a small fraction of the mass at burnout (such as the Shuttle, unlike the Z-5 launch vehicles described in §2.4), this program was considered valid for use in designing the mission architecture.

1.5.3. Aerocapture Program. The simulation program used for Mars aerocapture numerically integrates the trajectory of a spacecraft in the Martian atmosphere. A drag force proportional to atmospheric density and the square of the spacecraft velocity was assumed, as was a constant lift-to-drag ratio and an exponential atmosphere with a scale height of 11 km.

1.5.4. Cryogenic Systems. These were sized using the model of Kittel *et al*[4] with a 25% mass margin and 100% heat load margin.

2. Launch Systems

2.1. Launch Needs for Mars Exploration

Neither robots nor humans can get to Mars without a launch vehicle for Earth-to-orbit (ETO) transportation. For the current generation of Mars spacecraft, vehicles that can send roughly one metric ton of payload to Mars are sufficient, but future missions such as Mars Sample Return will need to send an order of magnitude more payload to Mars. Eventually, human missions will require at least an order of magnitude more payload still; the ITV is projected to have a mass up to 421 MT upon departure from LEO. Even if such a large spacecraft is launched in several pieces, a large launch vehicle becomes a necessity. The human Mars mission plan we have outlined requires a launch vehicle with 111 MT to LEO capacity; a smaller vehicle could be used at the expense of reduced efficiency, but current launch vehicles under the 25 MT to LEO regime would require nearly twenty launches for the ITV alone, clearly not reasonable if we wish to travel to Mars on a regular basis. Thus for the near term current launch vehicles are sufficient, whereas a human mission will require something larger.

2.2. Current Launch Vehicles

The near-term Mars missions are likely to fly on Delta II vehicles, including the upcoming 2001 Mars orbiter. Future robotic missions may require payload capacities as great as that of the Titan IVB Centaur or an Evolved Expendable Launch Vehicle. Additionally, the Ariane 5 will be upgraded in the early years of the 21st century; a cryogenic upper stage, currently under consideration,[5] would increase its payload capacity substantially. Thus for the next decade, the approximate maximum that can be delivered to Mars in a single launch is 7,500 kg. For this reason, the ISML is designed for this size.

2.3. Need for a New Launch Vehicle

There are several reasons why current launch vehicles, despite their applicability to the robotic Mars missions of the next decade, are inadequate for human missions to Mars.

2.3.1. Payload Fairing Diameter. Current launch vehicles typically have payload fairings no wider than five meters. Packaging the Mars Ascent Vehicle, for example, into such a narrow fairing is nearly impossible given the wide hydrogen tanks and rocket engines. A mission has two options for avoiding this difficulty: extensive on-orbit assembly, or a larger fairing. The latter is simpler and probably much cheaper and better in the long run; it would be expensive and dangerous for astronauts to assemble Mars landers or aeroshells on orbit.

2.3.2. Number of Launches. A 25 MT to LEO vehicle, probably typical of the heaviest rockets that would be built for commercial, military, and scientific missions, would require at least 17 launches to build the ITV in orbit. Operationally, the prospect of 17 launches just for this part of the Mars mission presents difficulties. For example, there is a high probability that one launch would fail. Additionally, some components, in particular the ITV's large cryogenic stages, are not split easily into smaller pieces because the dry mass fraction of cryogenic systems increases as they become smaller (higher surface area to volume ratio).

2.3.3. Earth Orbit Rendezvous. Rendezvous in Earth orbit is a well-tested technology, but sixteen rendezvouses add significantly to the number of failure points in the mission.

A new launch vehicle is clearly needed. It must be a large launch vehicle with a wide fairing. A compromise must be made between the capacity of the launch vehicle and its associated development costs; a good choice is probably a vehicle about equal in size to the Space Shuttle or the Saturn V, as this is the largest size with which there is operational experience.

2.4. The Z-5 Launch Vehicle

The Z-5 expendable launch vehicle consists of three stages. The third is used only on direct-to-Mars missions, not on LEO missions. The Z-5 will be launched from Kennedy Space Center. The stages are summarized in Table 2.4.1.

Table 2.4.1. Characteristics of Z-5 Booster

	First stage	*Second stage*	*Third stage*
Propellant	LOX/RP1	LOX/LH_2	LOX/LH_2
Engines	5 RD-170	4 Vulcain 2	5 RL-10D
Thrust	39.45 MN (*vac*) 36.30 MN (*sl*)	5.40 MN	1.11 MN
Specific impulse	337 s (*vac*) 309 s (*sl*)	433 s	472 s
Burn time [min:s]	02:14	04:22	06:15
Dry mass	150 MT	35 MT	10 MT
Propellant mass	1795 MT	350 MT	90 MT

Length	30 m	22 m	15 m
Diameter	11.7 m	10.5 m	8.0 m

Vacuum performance data for the RD-170 and Vulcain 2 engines are from Andrews Space and Technology.[6,7] The RL-10D is a derivative of the existing Pratt and Whitney RL-10 engines used on the Centaur and Delta III vehicles.[8]

The overall height of the Z-5, including a 10.5 × 45 m payload fairing that encloses the third stage and payload, is 97 m (318 ft), taller than the Space Shuttle but somewhat shorter than the Saturn V. The total liftoff mass for a direct-to-Mars mission is 2.48 million kg (5.48 million lb), and the liftoff thrust is 36.30 MN (8.16 million lb). The Z-5 payload capacities are calculated for a 51.6° inclination orbit. Launch trajectories were determined both for ITV assembly missions and for direct-to-Mars missions. Acceleration is kept under 6 *g* at every point in the trajectory. Table 2.4.2 shows the launch trajectory for the Z-5 on a direct-to-Mars mission at 51.6° inclination. The three-stage Z-5 can carry 44 MT to C_3 = +14.9 km²/s². The third stage burns for 80 seconds after second stage separation. It then coasts to the proper TMI point and burns for an additional 295 seconds to place its payload en route to Mars.

Table 2.4.2. Z-5 Direct to Mars: Sequence of Launch Events

Event	*Time [min:s]*	*h [km]*	*d [km]*	*v [km/s]*	*Notes*
Liftoff	T+00:00	0	0	0	1.47 *g* T/W
Mach 1	T+00:50	7.1	1.7	0.33	
First stage separates	T+02:30	75	99	2.72	Peak acceleration from first stage 5.5 *g*
Payload fairing separates	T+03:00	107	177	2.89	
Second stage separates	T+07:04	226	1,250	6.82	Peak acceleration from second stage 3.0 *g*
Third stage shutdown	T+08:24	228	1,820	7.48	79% propellant remains in third stage tanks

h: altitude above surface; *d*: downrange distance; *v*: ground-relative velocity

For a LEO mission, such as the ITV propulsion stages, the third stage is replaced with a Star 48/TE-M-711-8 solid motor. The payload capacity to a 360 km orbit is 112 MT. After second stage separation at T+07:04, the spacecraft coasts for 45 minutes until apogee at 360 km altitude. There the Star 48 fires for 88 seconds, providing 48 m/s of ΔV. (The Star 48 has a dry mass of 116 kg, 2,000 kg of propellant, and an I_{sp} of 292.9 seconds.[9]) Alternatively, the first stage of the Z-5 can be throttled down to 60% two minutes into launch. This ensures that acceleration remains below 4 *g* but reduces payload capacity to 109 MT.


Figure 2.4.1. Z-5 Launcher

2. 5. Z-5 Design Considerations

A number of tradeoffs were considered for the Z-5 launch vehicle. It could be expendable, reusable, or mixed (like the Space Shuttle); it could be parallel or sequentially staged; and each stage could use any of several propellants.

2.5.1. Expendable or Reusable Vehicle. A large RLV would be a significantly costlier development program than a large expendable due to the complexity of recovering and refurbishing a rocket. Furthermore, reusability only pays off for systems that fly often, and these missions will only need several Z-5 flights per year. It is also possible to envision a partially reusable launch vehicle such as “Magnum” which would have liquid fly back boosters as its first stage and an expendable core vehicle as its second. This strategy was not chosen due to the potential high development costs associated with liquid fly back boosters. Thus an expendable vehicle was chosen.

2.5.2. Propellant. Hydrogen/oxygen is undoubtedly the best choice for the upper stages of the launch vehicle; it is the only current propellant that achieves I_{sp} in excess of 400 s. This prevents the heavy-lift vehicle from becoming unreasonably large. The lower stages should use a low-energy, high-density, high-thrust propellant: solid propellant (Al/NH_4ClO_4), storables (N_2H_4/N_2O_4 and derivatives), or LOX/RP1. Of these, LOX/RP1 has the highest I_{sp}, with the relatively high-thrust RD-170 rocket engine providing 337 s I_{sp} in vacuum. Although it has the operational difficulties associated with cryogenic oxygen, the other choices have worse difficulties. Since solid propellant cannot be loaded on the launch pad, explosive propellant is present during much of the launch processing, and the exhaust has a high concentration of acidic HCl, an environmental concern. Storable propellants are highly toxic and require special precautions to handle. Given these drawbacks and the relatively advanced state of LOX/RP1 propulsion technology in Russia, LOX/RP1 was selected for the Z-5 first stage.

2.5.3. Parallel or Sequential Staging. Parallel staging, in which the first stage is composed of booster rockets strapped to the core stage, has advantages in that the core stage can be ignited on the ground, allowing a simpler ignition system and verification before launch commit. However, the core stage spends part of its burn pushing not only its own mass but that of the boosters as well, inefficient from the ΔV perspective. This can be solved with a cryogenic upper stage, as in the Mars Direct Ares or CMSM2 Janus. [10] To avoid this additional stage, a sequentially staged configuration was chosen for the Z-5.

2.6. Soyuz/EAV Launch

The crew is launched in an Earth Ascent Vehicle on the Soyuz booster from Baikonur, Kazakhstan. The Soyuz has four boosters burning kerosene and oxygen in RD-107 engines and a two-stage core. The first core stage burns kerosene and oxygen in the RD-108, very similar to the RD-107; the second stage burns kerosene and oxygen in an RD0110 engine.[11] With an escape tower, the Soyuz booster can lift the 6,850 kg mass of the Soyuz-T to LEO.[12]

The Soyuz was chosen to launch the crew because of its long history of reliable transportation to orbit and because of its launch facilities, which can already handle a crew. A capsule capable of carrying five humans to orbit within the Soyuz launch capacity will be built anyway for the MAV; modifying this capsule for Earth ascent is expected to be a minor part of total mission cost. Some modifications to the Soyuz launch system might be necessary to accommodate the EAV.

2.7. Selection of the Z-5

The selection of a launch vehicle remains a major issue for a human mission to Mars, but it probably will not be resolved until the time of program approval. This mission is baselined with the Z-5 as the launch vehicle.

3. Trajectories

3.1. Orbital Mechanics of Mars Missions

A human Mars mission will require a selection of trajectories, both for cargo vehicles (one-way) and humans (two-way). Most cargo vehicles will use either a Type I (6-9 months) or Type II (8-12 months) trajectory, each of which has a departure C_3 of 12 km²/s² and an entry velocity at Mars of 6 km/s. For human missions, several mission profiles could be considered. There are three major options to be considered: fast missions, opposition missions, and conjunction missions. These are compared in Table 3.1.2.

Table 3.1.2. Possible Profiles for Human Crews to Mars

Mission profile	*Assessment*
Fast 3 months to Mars 1 month on Mars 3 months to Earth	Short mission is desirable for initial mission, but ΔV over 50 km/s results in absurdly massive mission with present technology. **Is not currently feasible.**
Opposition 7 months to Mars 2 months on Mars 11 months to Earth	Again, short mission is desirable for initial mission, but is not much shorter than conjunction mission. ΔV is about 1.5 km/s higher than for conjunction mission. VGA is needed, so trajectory is highly variable from one launch opportunity to the next. Surface stay is a small fraction of the total mission. **Feasible, but difficult and only moderately rewarding.**
Conjunction 7 months to Mars 16 months on Mars 7 months to Earth	Long length of mission is a drawback, but most time is spent on Mars at the (relative) safety of the base and under the protection of Martian atmosphere (radiation shielding) and in Martian gravity field. **Feasible, technically easiest mission, and most rewarding.**

3.2. Interplanetary Trajectories for Humans

Since the fast mission's high ΔV prevents it from being performed with present or near-term technology, a human Mars mission must use either the opposition or conjunction profile. An opposition mission is somewhat shorter in total but requires a larger ΔV. Additionally, most of the conjunction mission is spent on Mars, whereas most of the opposition mission is in interplanetary space. The opposition mission might be appropriate for a "flags-and-footprints" mission, but it defeats the purpose of an overall Mars exploration program. For these reasons, a conjunction trajectory was chosen for the first missions.

Within the conjunction class missions, there is a choice of slower versus faster trajectories between Earth and Mars. Faster transit times reduce deep space radiation and microgravity exposure at the expense of higher ΔV, requiring a smaller spacecraft, better propulsion, or more fuel; in addition, fast trajectories raise entry velocities, making aerocapture more difficult. With an Earth departure C_3 of 20.25 km²/s², a transit time of 220 days or less can be achieved with Mars hyperbolic approach velocities no greater than 3.9 km/s in all launch opportunities from 2020 to 2033. This trajectory requires a ΔV of 1,300 m/s greater than that of the optimal Hohmann transfer. (See §1.5 for further details on the program used to compute interplanetary trajectories.) Table 3.2.1 lists trajectories for human Mars missions between 2020 and 2033.

Table 3.2.1. Details of Human Trajectories, 2020–2033

Launch	*Leg*	*Departure*	*Arrival*	*Transit time*	*Orbital elements*
2020	E→M	MON 10 AUG 2020 C_3=20.2; δ=+9°	SUN 03 JAN 2021 v=6.28	146 (568:553)	e=0.26; p=1.00; a=1.72 T=1.59; i=0.6°
	M→E	MON 25 JUL 2022 C_3=17.5; δ=–10°	WED 18 JAN 2023 v=12.67	177	e=0.23; p=0.88; a=1.39 T=1.21; i=1.7°
2022	E→M	SAT 17 SEP 2022 C_3=19.9; δ=+39°	SUN 26 MAR 2023 v=6.27	190 (526:537)	e=0.28; p=1.00; a=1.78 T=1.64; i=2.5°
	M→E	MON 02 SEP 2024 C_3=17.2; δ=+1°	FRI 07 MAR 2025 v=13.05	186	e=0.25; p=0.88; a=1.47 T=1.27; i=0.1°
2024	E→M	THU 24 OCT 2024 C_3=20.0; δ=+43°	MON 26 MAY 2025 v=6.29	214 (512:499)	e=0.27; p=0.99; a=1.70 T=1.56; i=2.5°
	M→E	SAT 10 OCT 2026 C_3=17.5; δ=+15°	MON 19 APR 2027 v=13.00	191	e=0.26; p=0.92; a=1.57 T=1.39; i=1.4°
2026	E→M	WED 02 DEC 2026 C_3=19.8; δ=+28°	SAT 10 JUL 2027 v=6.27	220 (499:486)	e=0.25; p=0.96; a=1.60 T=1.45; i=0.9°
	M→E	MON 20 NOV 2028 C_3=17.5; δ=+27°	THU 24 MAY 2029 v=13.00	185	e=0.29; p=0.94; a=1.68 T=1.50; i=2.0°
2029	E→M	THU 11 JAN 2029 C_3=19.9; δ=+4°	SUN 19 AUG 2029 v=6.20	220 (678:660)	e=0.22; p=0.94; a=1.49 T=1.34; i=1.5°
	M→E	FRI 17 JAN 2031 C_3=17.4; δ=+28°	SAT 28 JUN 2031 v=13.03	162	e=0.30; p=0.94; a=1.76 T=1.57; i=1.6°
2031	E→M	TUE 04 MAR 2031 C_3=19.8; δ=–19°	SUN 14 SEP 2031 v=6.29	194 (565:550)	e=0.20; p=0.93; a=1.41 T=1.27; i=2.4°
	M→E	FRI 01 APR 2033 C_3=16.8; δ=+7°	MON 08 AUG 2033 v=13.04	129	e=0.30; p=0.92; a=1.40 T=1.51; i=0.0°
2033	E→M	THU 12 MAY 2033 C_3=19.4; δ=–36°	TUE 11 OCT 2033 v=6.27	152 (602:586)	e=0.19; p=0.95; a=1.40 T=1.27; i=1.8°
	M→E	TUE 05 JUN 2035 C_3=17.6; δ=–19°	SUN 07 OCT 2035 v=13.03	124	e=0.26; p=0.85; a=1.46 T=1.24; i=1.6°

C_3: Jacobi constant for departure orbit [km²/s²]
δ: injection declination with respect to planet's equator
v: entry velocity [km/s] (assuming 125 km entry interface altitude and 4.93 km/s escape velocity at Mars; 122 km entry interface altitude and 11.07 km/s escape velocity at Earth)
Transit time in days; surface stays in parentheses (Earth days:Martian sols)
e: orbital eccentricity; p: perihelion [AU]; a: aphelion [AU]
T: orbital period [yr]; i: orbital inclination

3.3. Mars orbits

In this mission, the ITV travels from Earth to Mars, inserts into Mars orbit, and then returns to Earth. The Mars orbit must be accessible from the Earth-to-Mars trajectory and must bring the ITV to the proper point for trans-Earth injection.

To a first approximation, orbits around Mars follow the familiar Keplerian orbital mechanics laws. However, Mars has a gravitational quadrupole moment J_2=0.001959 [see [13]] due primarily to its equatorial bulge, causing a gradual precession of orbits. Essentially, this precession leaves the period, eccentricity, and inclination with respect to the Martian equator fixed but perturbs the nodal and apsidal axes. It is undoubtedly significant for any spacecraft that lingers in Mars orbit for an extended period of time; a spacecraft in a low-inclination, low-altitude Mars orbit would have a precession rate of about 12° per day.

The ITV will fly in a near-polar, circular orbit around Mars at 250 km altitude. The nodes regress at the rate of 11.9° per day times the cosine of the orbital inclination i. In 450 days, an orbit of inclination i=90° does not precess at all, while an inclination of i=88.07° is sufficient to cause a one-half orbit precession. By varying the inclination between 88.07° and 90°, we can adjust the "final" ITV orbit plane (that is, the ITV orbit plane after 450 days or more of Mars orbiting) to be within 2° of any direction we choose. This is useful for the trans-Earth injection maneuver.

4. *Robotic Mars Missions*

4.1. Mars Sample Return and ISMLs

Current plans call for a Mars Sample Return (MSR) mission, returning about a kilogram of Mars rocks, sometime in the next decade. While this mission would be a useful step in the exploration of Mars, it is not sufficient for the needs of a human Mars mission that will spend up to 600 days on the surface. A human Mars mission will require at least 10 kg from the site of the first Mars base. There are several ways to increase sample size, such as changing ascent propellants (solid, storable, or in-situ produced propellants). ISPP was rejected since cryogenic systems do not scale well to small vehicles. A single-stage storable rocket was chosen because it is a better analogue to the MAV that will carry the astronauts. (Specifically, it allows the ascent vehicle to play the active role in the rendezvous rather than the Earth return vehicle.) The general architecture (two landers which launch samples into orbit where an orbiter grabs them and returns to Earth) is very similar to that of the first MSR mission, except that all the vehicles launch in the 2009 opportunity.

The Mars sample collection systems and ascent vehicle are to be landed on Mars by an Intermediate-Sized Mars Lander (ISML), which has a mass at TMI of 7,000 kg. Upon approach to Mars, the ISML separates from its cruise stage and enters the atmosphere of Mars, protected by an aeroshell of L/D = 0.4. Parachutes and three hydrazine/nitrogen tetroxide rockets slow the ISML to a touchdown on the surface of Mars, with a useful landed payload of 2,000 kg. The ISML mass allocation is shown in Table 4.1.1. (Margins are included in the individual items.)

Table 4.1.1. Intermediate-Sized Mars Lander mass budget

Component	*Mass [kg]*
Cruise stage	800
Aeroshell, heat shield, and reaction control system for descent section	1,550
Descent parachute (20 m diameter)	175
Descent propellant (for 700 m/s ΔV with 8% residuals)	1,030
Descent propulsion system (engines, propellant tanks and feed systems; 300 s I_{sp}, 35 kN thrust at full throttle)	250
Power supply (dynamic isotope)	445
Landing system structure, communications, and information management systems	750
Payload	2,000
Total	**7,000**

The ISML would be launched on a large vehicle [see §2.2] such as an EELV or an upgraded Ariane 5. The ISML power supply will be a dynamic isotope power supply (DIPS) using Stirling power conversion technology. The mass of a 2.5 kWe DIPS using 27 kg $^{238}PuO_2$ is estimated at 350 kg;[14] here 445 kg was budgeted, some of the increase necessary to move the DIPS system some 50 m away from the ISML. (When the Mars ascent vehicle launches, the ISML and any equipment remaining on it will be destroyed. We wish to conserve the Martian ^{238}Pu isotope inventory for future use.)

The MSR lander payload consists of a sample acquisition system, spacecraft utilities, and an ascent vehicle. The ascent vehicle is capable of producing 4.8 km/s of ΔV using a 50/50 mixture of hydrazine and dimethylhydrazine ("aerozine-50") and nitrogen tetroxide oxidizer, which can yield 320 s vacuum I_{sp} (used on the Delta II second stage). It can lift an 80 kg capsule containing 10 kg of Martian samples into low Mars orbit; its total liftoff mass is 1,500 kg and it has an inert mass of 245 kg excluding the capsule but including residual propellants. The ascent vehicle has a single engine providing 9 kN thrust.

The sample acquisition system features a robotic arm of length 4 m that obtains samples of Martian regolith and rocks and loads them into the ascent vehicle. This design is simpler than a sample-collecting rover, which would require a robotic arm anyway to raise the samples to the ascent vehicle. The lack of a rover will bring a scientific loss, since only the most accessible Martian material can be acquired; however, the first MSR mission is primarily scientific, whereas the second MSR mission is intended to acquire Martian material in bulk for compatibility analysis and testing with humans and their spacecraft. This objective is met just as well by typical Martian dirt as by any specially selected sample.

4.2. HEDS Lander

Before humans travel to Mars, it will be necessary to test out the Mars surface technologies needed for human exploration on a scale larger than, for example, the currently planned MIPP. Also, certain data on the proposed base site will be needed that the second MSR mission cannot return. An oxygen generator that produces at least 500 kg of oxygen in 400 days must be tested. (The human mission will require three generators to produce 17,710 kg O_2 in the same time frame.) Radiation levels of all varieties (neutrons, gamma rays, ultraviolet, and charged particles) must be measured, since radiation levels can vary significantly with site due to sunlight, altitude, and soil composition. Weather patterns at the landing site must be monitored for a full Martian year or more to show that diurnal thermal cycling (for example) will not damage critical systems.

Concentrations of CO_2 and H_2O in the air should also be monitored since ice or dry ice may condense on a vehicle. Water content of surface and subsurface material must be measured to as great a depth as possible.

The HEDS lander will serve these needs. It uses the ISML landing system, common with the second MSR vehicle, to save development costs. Its most massive payload element is the oxygen generator and storage system. The oxygen tank has 1 m^3 volume to store 1,100 kg of liquid oxygen at 92 K. The internal temperature is maintained by 100 layers of MLI and one of four cryocoolers capable of extracting 2.5 W of heat from the interior of the tank. The tank radius is 70 cm including insulation and a vacuum jacket necessary for the MLI to work on the surface of Mars. The oxygen generation system is 399 kg and draws 2,115 W, scaled from [DRM1/p. 3–106].

Another major mass item on the HEDS lander is a 10 m surface drill with a mass of 260 kg [DRM1/p. 3–52]. Due to power constraints, this drill cannot operate when oxygen is being generated. The mass of the sample analyzer, radiation monitoring instruments, and the spacecraft material exposure system is estimated under 100 kg. These instruments, the drill, and the oxygen generator easily fit within the ISML mass budget.

5. Cargo Vehicles and the Large Mars Lander

5.1. Selection of a Permanent Base Site

After the second MSR mission and the twin HEDS landers, a site for a permanent Mars base may be selected. The following considerations are key in landing site selection:

5.1.1. Availability of Water. It is generally believed that water concentration increases toward the poles and decreases toward the equator due to the temperature gradient. Water, of course, is a key resource for a Mars base.

5.1.2. Sunlight. Some daylight during each Martian sol is probably desirable, for psychological and operational reasons; EVAs may be difficult at night.

5.1.3. Elevation. Lower elevations are desirable due to increased atmospheric density. This results in an easier task for the atmospheric compressor of an ISRU system and greater protection from radiation.

5.1.4. Temperature. To simplify vehicle design, it is desirable to choose a landing site at which the spacecraft is always operating at a temperature significantly greater than that of its surroundings. If the temperature at the base site reached 280 K, for example, a very large radiator or active cooling system (both undesirable) would be necessary to prevent spacecraft overheating. Overheating is likely to be at least as great a danger as cooling to Mars base spacecraft because of the high power consumption compared with current robotic Mars missions.

5.1.5. Agricultural Potential. Crops may be grown on Mars using either natural sunlight or artificial light. The former will obviously be easiest at the equator due to greater sunlight; the latter will be easiest in polar regions due to the colder temperatures, which reduce radiator size. Artificial lighting, which is more dependable, can operate in a limited volume (such as an inflatable habitat style module), and avoids water condensation on the roof of an inflatable greenhouse, may be desirable. In this case, the polar regions may be favored.

Once a base site is selected, it is time to deliver cargo and humans there. This will require a larger lander, the Large Mars Lander (LML), and its cargo payloads.

5.2. Large Mars Lander

The LML is a circular shelf of 3.4 m radius on three 2.7 m tall landing legs. On its underside are four 1.48 m diameter descent propellant tanks, two containing hydrazine and the other two containing nitrogen tetroxide. A pressure-fed engine consumes this propellant, providing 300 kN thrust at 300 s I_{sp}. The mass allocation for the LML is shown in Table 5.2.1.

Table 5.2.1. Large Mars Lander mass budget

Component	*Mass [kg]*
Payload	19,267
Lander structure	2,000
Descent stage propulsion system dry mass (incl. engine, propellant tanks, and feed system)	1,396
Propellant load (incl. 8% residuals)	7,511
Parachutes (2 main parachutes, 45 meter diameter, plus 2 drogues)	625
Aeroshell and reaction control system	13,001
Interplanetary power supply (inflatable solar arrays, 20 kWe at 1 AU, 7 kWe at Mars aphelion)	200
Total	**44,000**

Here the propellant tanks and the helium tanks for the propellant feed system are scaled from the Space Shuttle orbital maneuvering system[15] and the parachutes and lander structure are scaled from DRM3. The aeroshell and reaction control system (RCS) are allocated 29.7% of the total entry mass since the RCS must provide up to 300 m/s of ΔV in orbital maneuvers using hydrazine/nitrogen tetroxide bipropellant with 315 s I_{sp}.

The LML descent sequence is as follows:

5.2.1. Mars Aerocapture. Several minutes before Mars arrival, the LML separates from its interplanetary power supply. The LML enters the Martian atmosphere at 125 km altitude at up to 6,300 m/s. The entry flight path angle must be between 9.8° and 12.4° to capture into orbit around Mars without exceeding the 3.2 *g* deceleration limit. To capture into a 160 km circular orbit, the LML would first capture into an elliptical orbit and then aerobrake until the apoares fell to 160 km. Then an engine firing of under 100 m/s, depending on the entry parameters, would place it into this low Mars orbit. The entry corridor is 48 km deep. The LML aeroshell is based upon an ellipsled configuration specified by Lockheed Martin.[16] It has a lift-to-drag ratio of 0.4 and (with this heavy payload) has an estimated ballistic coefficient of 300 kg/m².

5.2.2. Mars Descent. After a final checkout of LML systems, its reaction control system retrofires to send the lander down toward the surface of Mars. At 8 km altitude, terminal velocity is 650 m/s, and the parachute deployment sequence begins. At 1 km altitude, the LML's velocity has dropped to 100 m/s if one of the parachutes has opened and 70 m/s if both have opened. In the former case, the LML separates from its parachute and ignites the single bipropellant engine on its underside, slowing itself to a halt 30 m above the Martian surface. It may then hover for up to 150 seconds in some cases before it must touch down on a smooth landing site. On some missions the LML will need a larger payload than the 19,267 kg listed above. This can be achieved by removing propellant from the descent stage and accepting a reduced hover time, leaving the lander's total entry mass unchanged. For example, the 22,696 kg MAV can be landed on Mars with 45 s of hover time.

5.3. MSPU Lander

The minimum power requirement will be 100 kW for 600 days, the duration of a crew surface stay. Three usable energy sources can be imported from Earth: wind, solar, and nuclear fission. Wind power has many moving parts at risk of malfunctioning in the Martian environment, and the extremely limited flux of sunlight on Mars is prohibitive to solar power.

A 160 kWe MSPU requires a nuclear reactor that can run for seven years, a radiation shield, a power conversion system, a radiator, and power conditioning equipment, for a total mass of 9,738 kg.[17] The radiation shield leaves an acceptable radiation dose (below 5 rem/yr) at a distance of 2.8 km. The MSPU will land roughly this distance from the proposed Mars base site and will be targeted into a crater for additional shielding, making the crew's exposure to MSPU radiation negligible.

After separation from the LML interplanetary power supply, power is provided by fuel cells. The mass and performance are taken from the STS fuel cells,[18] but some changes may be necessary to improve their lifetime. The total mass budget for the fuel cell system, including the five fuel cells generating at least 6 kWe each, is 1,844 kg dry. 2,000 kg reactants can run one fuel cell for 40 days, which will keep the MSPU alive during approach, landing, and deployment.

Once the LML/MSPU payload has landed, its radiators are deployed, and power can be produced. Five small rovers connect it to other payloads; each rover has a total mass of 1,200 kg, of which 704 kg [see [19]] is devoted to the power cable that is rolled off a spool on the back of the rover. Thus the total payload mass of the MSPU's LML (including 320 kg for communications, as in [DRM3/§A4.0], and a 500 kg power distribution system) is 20,402 kg plus rover deployment ramps.

In the dusty Martian environment, a direct metal surface contact like a conventional electrical outlet is not a good way to connect the power rover to a base element. Two schemes avoid the need for a metal surface contact: the metallic connector, which uses a heater to solder two connectors together, and the inductive connector, which uses neighboring coils to transfer alternating current by magnetic induction. The final selection must await a thorough engineering analysis of both options.

5.4. Cargo Payloads

Two cargo payloads will be sent directly to Mars on Z-5s in 2018. One of these payloads will be a backup habitat derived from the CML [see §6.3] for the first crew. In addition, the scientific exploration of Mars will require a long range mobility capability on the surface. Two 5 MT rovers [DRM3/§A2.2.1] are allocated in the first cargo payload, with a total mass of 10 MT. The remainder of the capacity on this first LML will be dedicated to scientific equipment. There is also the capability for additional cargo landers in 2020 and in subsequent opportunities.

5.5. Mars Ascent Vehicle

The mass of the MAV payload is 22,506 kg, broken down in Table 5.5.1.

Table 5.5.1. MAV payload mass budget

Component	*Mass [kg]*
Food for 5 people, 600 days	6,600
MAV capsule [§6.8]	3,100
Dry MAV stage	5,338
Liquid hydrogen in MAV (kept cold by MAV stage cryocoolers, powered by interplanetary	3,152

power supply in transit to Mars)	
Oxygen generators (4)	4,156
Dynamic isotope power supply and radiator	350
Total	**22,696**

The food is intended to be the primary food supply for the crewmembers during their stay on the Martian surface. The dry MAV stage includes a 4.45 m diameter liquid hydrogen tank, a 3.10 m diameter liquid oxygen tank, and four RL-10D engines burning hydrogen/oxygen bipropellant. The RL-10D is a possible future variant of the current RL-10 Pratt-Whitney rocket engines providing 222 kN thrust and 472 s I_{sp} with a mass of 378 kg.[20] The dynamic isotope power supply is the same as that used on the ISML [see §4.1], except that it needs no deployment system. At Mars landing, the MAV hydrogen tank is full but the oxygen tank is empty. The MAV oxygen generators use the same basic process as the HEDS lander oxygen generator but are larger, draw more power, and have a higher output rate.

Table 5.5.2. MAV oxygen generator and storage systems

Component	*Mass [kg]*	*Power [W]*
Atmospheric compressor/CO_2 extractor	115	658
Zirconia cell electrolyzer ($2\ CO_2 \rightarrow 2CO + O_2$)	600	17,838
Oxygen liquefier	116	1,120
Margin (25% mass, 25% power)	208	4,904
Total	**1,039**	**24,520**

The output rate of a MAV oxygen generator is 15 kg O_2 per day, more than sufficient to produce the required 17,710 kg of liquid oxygen in a 400 day time span. The system masses and powers have been scaled from [DRM1/p. 3–106].

Upon separation from the interplanetary power supply, the hydrogen begins to boil off. With a heat of vaporization of 445 J/g, and a capability to boil off up to 200 kg of liquid hydrogen without loss of functionality of the ascent stage, the MAV can handle up to 89 MJ of heat transfer into the hydrogen tank. Since the expected heating rate is 50 W, the MAV will be able to sit on Mars without power for up to 20 days before hydrogen boiloff becomes problematic.

The MAV is then plugged into the MSPU lander, providing power to run the high-power MAV systems, specifically the oxygen generators and the cryocoolers in the hydrogen tank. Three of the four oxygen generators must operate for sufficient oxygen to be produced. Together, they consume 73,560 W power. After 400 days, the MAV is fully fueled for ascent into orbit. This information is transmitted to Earth, allowing the next phase of the Mars mission to begin.

Figure 5.5.1. Mars Ascent Vehicle and LTL system

6. ITV and First Human Mission Design

6.1. Interplanetary Transfer Vehicle Habitat

The ITV is the vehicle in which the crew travels from low Earth orbit to the vicinity of Mars and in which the crew makes the return transit to Earth. Although unneeded items can be jettisoned in the EEV, there is no capability for extensive EVA in interplanetary space. Such a capability was deemed unnecessary because there are no spacecraft systems outside the pressurized compartments that the astronauts could conceivably repair.

6.1.1. Structural and Thermal Systems. The basic structure of the ITV is a rounded cylindrical inflatable habitat 8 meters in diameter and 8.5 meters long, with 300 m³ of internal volume. The mass estimate for this component was taken from the TransHab derived habitat proposed for Mars exploration [DRM3/§A3.1] at 1,039 kg structure and 500 kg thermal control systems. Here we increase the thermal control system's mass budget to 1,000 kg, allowing redundancy for many components, and introduce an additional 25% margin on structural and thermal systems, bringing the total mass budget for this item to 2,549 kg.

6.1.2. Life Support Systems. A life support system for six people (this mission would have five) is estimated as having a mass of 3,796 kg, a power requirement of 5,831 W, and a volume of 19.13 m³.[21] This includes complete recycling of oxygen and water, meaning that only food and power are required as inputs to run the life support system indefinitely. While this mass budget does include spares, it may still be prudent to send two such life support systems due to the lack of in-space experience with extensive recycling of consumables. The 3,796 kg figure was taken unadjusted; the performance reduction (five astronauts versus six) serves as margin. Additionally, food should be sent with the astronauts; the mass of food needed is taken from a TransHab derived module mass budget [DRM3/§A3.1] as 2.2 kg per person per day, or 11 kg/day for a crew of five. The ITV (including the crew Mars lander) will carry 1,000 days of food (enough for the entire mission) through launch and TMI; 800 days of food at MOI; and 200 days at TEI. Adequate consumables are present at each mission phase. To make the ITV lighter during launch, five metric tons of food are launched with the crew Mars lander.

6.1.3. Crew Accommodations. This item includes health care equipment and other crew systems. A mass of 2,356 kg is estimated from a TransHab derived module [DRM3/§A3.1] crew accommodations item, removing food (which we include in the life support category) and adding a 25% margin.

6.1.4. Communications and Information Management. This item is again taken from [DRM3/§A3.1] with a 25% margin, yielding a total mass of 400 kg.

6.1.5. Electrical Power. The power requirement for an ITV-type habitat was estimated at 29,400 W [DRM1/p. 3–93], but the life support power requirement has been reduced in more recent studies from 12 to 6 kW, so a power supply need only provide 24 kWe to the habitat. During the return trip to Earth, therefore, a 30 kWe power supply has been baselined to provide margin. In addition, the cryogenic tanks in the TEI stage must remain chilled, yielding a power requirement of 40 kWe prior to TEI. The most promising power source is a combination of solar and battery power, avoiding the complications of a nuclear reactor or the large quantity of ^{238}Pu (roughly 400 kg) for an isotope power supply. The solar arrays provide the power except during eclipse periods, when the battery is used. It is later recharged from the solar arrays.

The solar arrays will be of the type projected for the (now cancelled) Space Technology 4/Champollion mission. These would provide a power density of 100 W/kg and about 55 W/m² at 1 AU from the Sun,[22] or 35 W/kg at Mars aphelion. The post-TEI array system would have a mass of 857 kg and an area of 1,500 m². The array for Mars orbit should provide 80 kWe at Mars aphelion to adequately charge the batteries during the sunlit portion (at least 62%) of the ITV's orbit; it would have an area of 4,100 m² and a mass of 2,286 kg. Before MOI, power is to be provided from the ITV-CML truss; see §6.4.

The battery will have to endure at least 10,000 charging cycles; nickel-metal hydride batteries with a specific energy of 55 W-hr/kg can survive only 3,000 cycles.[23] However, some improvement in battery technology can be expected; for this study, a 40 W-hr/kg rechargeable battery that can survive 10,000 cycles is assumed. The batteries must provide up to 42 minutes of power at 40 kWe, corresponding to a mass of 700 kg. Dividing into fourteen units of 50 kg (2 kW-hr energy storage each) and adding two spare units, the total battery mass is 800 kg. Half of these are to be jettisoned just before TEI along with the Mars orbit solar arrays; this reduces the TEI mass but still leaves a backup power source during the trans-Earth cruise.

Additionally, power will be needed for up to several days after MOI while the ITV aerobrakes. It is not desirable to drag a solar array through the Martian atmosphere, so a non-regenerative fuel cell consuming hydrogen and oxygen was selected to power this phase. It is comprised of nine fuel cells using the existing STS fuel cells as a mass and performance estimate,[24] although they would need an improved in-space lifetime. Six fuel cells are needed to produce 37 kWe power; together they consume 300 kg/day of hydrogen and oxygen reactant. If reactants for five days are supplied, the total mass of the fuel cell reactants is 1,500 kg, and their tankage has a mass of 422 kg. The fuel cells themselves total 1,041 kg.

Power must be distributed to the components and radiated away after it is used. Power distribution mass was taken as 550 kg, double the value given in [DRM3/§A3.1] since there is a higher power requirement along with a TEI stage that also requires power; the radiator was scaled from [DRM1/p. 3–96] to a 93 m² area and a 507 kg mass. In total, the power system has a mass of 2,314 kg at TEI, with an additional mass of 5,649 kg to be inserted into Mars orbit.

6.1.6. Earth Entry Vehicle and Return Payload. The Earth entry vehicle is described in greater detail in §6.9; it has an unloaded mass of 3500 kg. The crew has a mass of 500 kg (80 kg per crew member and five 20 kg pressure suits). The samples returned to Earth from Mars have a mass of 500 kg, and there is an additional 100 kg of scientific equipment.

6.1.7. Reaction Control System. This system was designed to provide 80 m/s of ΔV during each of the three legs of the mission (trans-Mars, Mars orbital, and trans-Earth.) It uses hydrazine resistojets with 320 s I_{sp} and thus requires 8,000 kg propellant with a 2,000 kg dry mass. Of this dry mass, 1,000 kg holds propellants which will be used prior to TEI, so this part of the RCS is to be jettisoned along with the Mars orbital power systems just before TEI.

6.1.8. Solar Storm Shelter. This device protects the crew from radiation from solar particle events. It must provide 10 g/cm² of shielding to the crew during a major flare in addition to that available from onboard equipment. This is described further in §8.1, but here we merely note that a 2.6 m diameter sphere using LiH shielding is 2,346 kg.

6.1.9. Atmospheric Repressurization System. The EEV may need to be repressurized several times, for example, if the EEV is used as an airlock through which to jettison unneeded items. If this is to be done three times, 100 kg of air will be expended. The mass of this air and its cryogenic storage systems was estimated at 250 kg. Additionally, the ITV/EEV complex contains 300 kg of air budgeted under this mass item.

Table 6.1.1. ITV mass budgets at major mission stages

	Mass [kg]			
Component	*at launch*	*at TMI*	*at MOI*	*at TEI*
Structural and thermal systems	2,549	2,549	2,549	2,549

Life support (includes consumables)	13,592	13,592	16,392	9,792
Crew accommodations	2,356	2,356	2,356	2,356
Communications and info management	400	400	400	400
Electrical power supply	7,963	7,963	7,963	2,314
EEV and return payload	3,600	4,100	4,100	4,600
Reaction control system	10,000	10,000	4,449	1,816
Solar storm shelter	2,346	2,346	2,346	2,346
Atmospheric repressurization system	550	550	550	550
Total	**43,356**	**43,856**	**41,105**	**26,723**

6.2. ITV Main Propulsion System

The ITV is propelled by a hydrogen/oxygen stage that is required to deliver a ΔV of 300 m/s post-MOI and at least 3,080 m/s at TEI. The stage contains 5,450 kg of usable hydrogen and 32,700 kg of usable oxygen; its dry mass is 6,739 kg distributed according to Table 6.2.1.

Table 6.2.1. Inert mass of ITV main propulsion

Component	*Mass [kg]*
Residual propellants (2%)	763
Propulsion engines (3 RL-10D, common engine with MAV, 222 kN thrust, 472 s I_{sp} [25])	1,134
Liquid hydrogen tank (100 layers MLI plus 4 cryocoolers; 5.49 m diameter including insulation; draws 3,563 W power)	1,870
Liquid oxygen tank (with 35 layers MLI plus 4 cryocoolers; 3.84 m diameter including insulation; draws 1,246 W power)	435
Propellant feeds and stage structure	1,189
Margin (25%)	1,348
Total	**6,739**

If a 2% margin is applied to this stage's I_{sp} (that is, I_{sp} = 462.6 s is assumed rather than the specified 472 s), the propulsion system burns 5,503 kg of its 38,150 kg propellant providing the 300 m/s post-MOI ΔV to the ITV. This leaves up to 3,089 m/s for the TEI burn. The ITV main propulsion stage, including the rocket engines, is 7 m in diameter and 14 m long.

6.3. Crew Mars Lander

The Crew Mars Lander (CML) is the vehicle in which the astronauts will descend to the surface of Mars. On the surface, they will live in either this CML or in the habitat (derived from the CML) that landed in 2018. They move the food landed in the MAV into this habitat and connect it to the MSPU's power grid. It is then the analogue of the habitat module in the Mars Direct plan. It is essentially a Large Mars Lander (LML), similar to those described in §5, but with a different payload and without the solar arrays in interplanetary space. Thus the Mars entry mass is 43,800 kg. The CML payload contains the following elements:

6.3.1. Structural and Thermal Systems. The estimate in §6.1.1 applies; the TransHab derivatives from which the ITV structural/thermal unit was scaled [DRM3/§A3.1] operate on Mars as well as in interplanetary space. Thus we retain the 2,549 kg mass estimate.

6.3.2. Life Support. The crew Mars lander should be able to land on Mars and keep the crew alive for 30 days; in addition, it should operate much longer if power and consumables are available. Thus we provide the 3,796 kg life support system from the ITV, which requires only food as input [see §6.1.2]. As a backup, an open-loop life support system, of mass 1,000 kg, is included in the crew Mars lander as well as 420 kg of hydrogen peroxide for oxygen generation, sufficient for 30 days. Additionally, food sufficient for 45 days (495 kg) is provided, so the crew will be able to survive in the CML for up to 30 days in all cases. (Water is produced in sufficient quantity for crew survival by the fuel cells; see §6.3.5.) This yields a total mass of 5,711 kg. An additional 5 MT of food is present in the CML at launch, since there was not room in the ITV.

6.3.3. Crew Accommodations. 2,356 kg; see §6.1.3.

6.3.4. Communications and Information Management. 400 kg; see §6.1.4.

6.3.5. Electrical Power Supply. The electrical power supply for the crew Mars lander is the fuel cell system from the MSPU [see §5.3]. It has a mass of 1,844 kg dry, with 4,507 kg reactants. The reactants can supply three fuel cells (18 kWe power generation) for 30 days. Loss of any one of the cryogenic reactant tanks still allows the crew to survive for 20 days. The production rate of water from the fuel cells is 150 kg/day, sufficient to meet the crew's needs. A 500 kg power distribution and rejection system has been budgeted.

6.3.6. Crew and EVA Systems. The crew has a mass of 500 kg, including pressure suits [see §6.1.6]. The Reference Mission allotment for EVA systems [DRM3/§A3.2.4] was used here: 195 kg for the airlock and 940 kg for the EVA suits (including one spare since we have a crew of five). Thus the total mass for this item is 1,635 kg. (At launch and TMI, the crew is not in the CML, leaving 1,135 kg.)

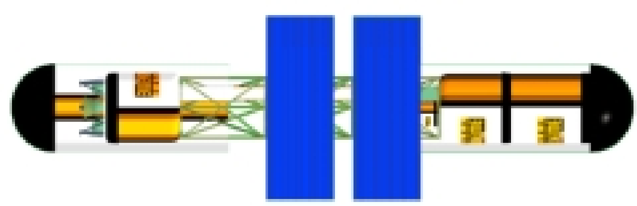
6.3.7. Atmospheric Repressurization System. Like the ITV [§6.1.9], the crew Mars lander contains 300 kg air. It also carries two 1/3 scale versions of the MAV oxygen generators, each with a mass of 346 kg, drawing 8,173 W power, and producing 5 kg/day of O_2 (sufficient to make up for the CML's lack of oxygen recycling). It is not feasible to run the oxygen generators until the crew Mars lander is electrically connected to the MSPU. Also, a buffer gas generation system will be needed on the crew Mars lander if it is to serve as a long-term habitation unit on Mars. The buffer gas is a mixture of nitrogen and argon, minor constituents of the Martian atmosphere; it is added to oxygen in habitation modules to reduce the danger of fire. The two generators each produce 1 kg of buffer gases per day. Mass and power are scaled from [DRM1/p. 3–105] with a 50% mass and power margin for a total mass of 219 kg and power consumption of 699 W. Additionally, there are four 63 kg / 44 W cryogenic tanks for storing up to 0.78 m³ of buffer gas or oxygen each. These are identical to those from the HEDS lander [see §4.5]. This brings the total mass allocation for atmospheric pressurization to 1,682 kg.

Table 6.3.1. CML mass budgets at major mission stages

	Mass [kg]	
Component	*at TMI*	*at Mars arrival*
Structural and thermal systems	2,549	2,549
Life support (including consumables except for H_2O)	5,711	5,711
Crew accommodations	2,356	2,356
Communications and info management	400	400
Electrical power supply (including H_2O production)	6,851	6,851
Crew and EVA systems	1,135	1,635
Atmospheric repressurization system	1,682	1,682
Total	**20,684**	**21,184**

6.4. ITV-CML Tunnel and Truss

Figure 6.4.1. ITV with Solar Arrays Deployed



The ITV and CML will be connected in transit to Mars by a 30 m tunnel encased in an equilateral-triangle shaped truss with 7 m side length and three segments. During launch and TMI, the truss is collapsed to a 10 meter length. One of the three segments supports compression of the truss during these events. After TMI, the other two 10 m segments will deploy. Figure 6.4.1 shows the appearance of the ITV on the way to Mars; the CML is shown at left inside its aeroshell, with the truss and solar panels to the right, followed by the EEV within the truss, and the ITV habitat and propulsion system within their aeroshell.

The tunnel itself will need to be collapsed to 10 m length for launch and deployed to 30 m post-TMI. An inflatable tunnel was suggested by James Cameron, and will be portrayed in his upcoming Mars TV miniseries and IMAX 3D movie; since we have a 10 m initial length to work with, however, the inflatable tunnel will have this initial length.

The three sides of two of the truss segments will be covered with triple-junction ($GaInP_2$/GaAs/Ge) solar cell arrays,[26] which will convert sunlight into power with at least 21% efficiency. After truss deployment, the arrays on the two sides of the truss facing away from the Sun are deployed. Because only 400 m² of solar arrays are needed here, sufficient power can be produced even if one of the four deploying solar arrays fails to open.

If the tunnel cannot be used to connect the CML and the ITV habitat, the mission proceeds nominally but without rotating the ITV as described in §7.2. As a result, the crew will land on Mars after living in microgravity for approximately six months.

The solar panels and their backing/deployment system are estimated at 2,400 kg, three times the panels themselves. The aluminum 2024 T3 alloy primary segment (the one that holds compression during launch and TMI) of the truss will have a mass of 2,045 kg, and the remaining two segments, which do not have to hold nearly this load, will total 2,045 kg. The inflatable tunnel of radius 1 m has a mass of 1,500 kg, scaled from the TransHab-derived module study [DRM3/§A3.1] by surface area with a factor of 3 margin accounting for the differences in configuration between TransHab and the tunnel. A 700 kg docking module for the EAV is located at the CML end of the tube. This gives the ITV-CML tunnel and truss an 8,690 kg mass.

6.5. TMI Stages

The ITV/CML combination will use four stacked hydrogen/oxygen stages to inject to Mars. The first three are modified versions of the Z-5 third stage, fitted with cryocoolers to keep the hydrogen and oxygen propellants cold, solar arrays to power the cryocoolers, a reaction control system providing 150 m/s of ΔV, and insulation to reduce the heat load on the cryocoolers. The fourth stage duplicates the ITV main propulsion system, with a 3 MT adapter that attaches to the CML aeroshell and a 6 MT reaction control system (similar to that on the large TMI stages) for rendezvous with the other components.

Table 6.5.1. Mass budget for first three TMI stages

Component	*Mass [kg]*
Liquid hydrogen tank structure (189 m³, 7.12 m diameter), purge bag, MLI (100 layers/39 W heat leakage rate), and cryocoolers (5; 6,755 W power)	3,457
Liquid oxygen tank structure (70 m³, 5.10 m diameter), foam insulation, MLI (40 layers/96 W heat leakage rate), and cryocoolers (5; 2,477 W power)	831
Primary propulsion system (5 RL-10D engines, common with MAV and ITV main propulsion system)	1,889
Propellant feeds, stage structure, communications and information management system	3,090
Solar arrays (118 m² area, 40 kWe at 1 AU, scaled from DS1/SCARLET)	952
Regenerative fuel cell (for power during eclipse; provides at least 10 kWe continuous in LEO)	694
Radiator	200
Reaction control system dry mass (25% of propellant)	1,327
Margin (25%)	3,110
Total dry mass	**15,550**
Residual hydrogen (2% of tank capacity)	265
Residual oxygen (2% of tank capacity)	1,591
Total mass after firing	**17,406**
Usable hydrogen (12857 kg capacity)	12,755
Usable oxygen (77143 kg capacity)	76,530
Total mass before firing	**106,691**
Reaction control propellant, N_2H_4/N_2O_4 (expended prior to firing; can provide 150 m/s ΔV at 315 s I_{sp} using bipropellant thrusters)	5,309
Total mass at launch	**112,000**

It is frequently suggested that either nuclear thermal rockets (NTR) or solar electric propulsion (SEP) systems be used for TMI instead of conventional chemical rockets. NTR would reduce the number of Z-5 launches needed for the ITV by one and replacement of the Z-5 upper stage with an NTR system would increase the Z-5's trans-Mars delivery capability from 44 to 50 MT. It was determined that this performance gain does not balance out the political difficulties associated with nuclear systems or the need to develop a special new test facility required for NTR engines.

SEP is another possible TMI technology, whether it is used for the entire TMI process or augmented by a chemical "kick" stage.[27] While SEP provides a high specific impulse, a human mission would require an SEP system to be at least two orders of magnitude larger than the kilowatt-scale SEP systems used today on communications satellites and Deep Space One, thus presenting a major development risk. Additionally, since solar arrays do not produce power in Earth's shadow, an SEP spacecraft must either turn its propulsion system on and off once each orbit, discharge/recharge batteries or regenerative fuel cells once each orbit, or fly in an orbit with continuous sunlight. The first option involves running the SEP system and its associated hardware through of order 1,000 on-off cycles, which is undesirable from the standpoint of reliability. The second option greatly increases the mass and unreliability associated with the power system. The third option could involve restricting launches to the solstices, when a 51.6 degree inclination LEO can be in continuous sunlight, or it could involve launching into a (nearly) sun-synchronous orbit. The former idea is operationally undesirable, as it places a large burden on the launch facilities; the latter idea requires launching of the Z-5 from Vandenberg or another site with access to sun-synchronous orbit. Additionally, it adds another 300 m/s to the ΔV required to reach orbit, reducing the booster's payload capacity. Finally, the crew of the Mars mission must either spend months traversing the Van Allen Belts or ride a larger rocket (such as Proton) when their EAV is launched.

The costs of large SEP systems are not known at present, but given their complexity they will undoubtedly be more expensive to produce than chemical stages, although they might cost less to launch. One method of reducing overall costs would be to reuse the SEP system, but depending on the specific configuration, even one reuse may require tens of thousands of hours of thrusting. SEP system lifetimes would have to be extended to make this option feasible. For these reasons, SEP was not used for the TMI scheme in this Mars mission.

6.6. LEO Assembly and the EAV

The complex of the ITV, CML, and four propulsion stages is launched in five pieces.

Table 6.6.1. Sequence of ITV/CML Assembly Launches

Launch	*Payload*	*Mass [MT]*
Z-5 #1	Trans-Mars Injection Stage 4	54
	Crew Mars Lander	49
	ITV-CML tunnel and truss	9
Z-5 #2	ITV habitat	44
	ITV main propulsion system	45
	ITV aeroshell	19
Z-5 #3	Trans-Mars Injection Stage 3	112
Z-5 #4	Trans-Mars Injection Stage 2	112
Z-5 #5	Trans-Mars Injection Stage 1	112

Each component is launched northeast from KSC into 51.6° inclination, 360 km altitude orbits. The orbit nodes precess backward by 5.1° per day, so launch windows to this orbit are 23 hours 36 minutes apart. Because the TMI stages have cryogenic coolers, scheduling of these launches is not critical; they need only occur well in advance of the TMI window.

Finally, several days before departure to Mars, the crew is launched on a Soyuz booster in the EAV. Because of the degrading physiological effects of the space environment, it is desirable not to extend the crew's stay in LEO unnecessarily. Launching the crew on a Z-5 with the CML (for example) would have to occur well before the TMI window because a delay close to the TMI window would force a mission abort. (The Z-5 is a large and complicated launch vehicle, and launch delays are to be expected.)

For this reason, a reliable means of sending the crew to the orbiting ITV/CML/TMI system is necessary. The only system currently capable of carrying a crew of five to the ITV is the Space Shuttle, but experience has shown that it too is susceptible to long delays. Thus a new vehicle will be needed. Fortunately, it does not need all capabilities of the STS; in fact, it should be as simple as possible to reduce the likelihood of a launch delay. An EAV was therefore designed to carry the crew into orbit and to the ITV/CML. It will launch on the Soyuz booster, which at present has been man-rated; humans frequently use it to travel to *Mir* with relatively few delays. The continued production of Soyuz vehicles is considered very likely, both because it has found a commercial role in launching communications satellites and because it will launch Progress and Soyuz spacecraft throughout the ISS program.

Trans-Mars injection uses the three large cryogenic stages and the copy of the ITV main propulsion system for a total ΔV capability of 4342 m/s. The ΔV necessary to reach Mars (C_3=20.25 km^2/s^2) from the 360 km assembly orbit is 4103 m/s.

6.7. Mars Arrival and Landing; Surface Operations

When the ITV arrives at Mars, the ITV/EEV complex separates from the CML, and the two aerocapture into Mars orbit separately. The CML follows the LML aerocapture and landing procedure described in §5.2, while the ITV and EEV capture into low Mars orbit at altitude 250 km. This orbit is nearly polar – its inclination varies between 88.07° and 90° [see §3.3]. The truss and tunnel are jettisoned.

During the crew's surface stay of 553 sols on the first mission [see Table 3.2.1], the crew will have access to the contents of the cargo payload landed in 2018, namely the two rovers and the scientific equipment. Possible examples of scientific equipment include greenhouses to test crop raising in the Martian soil, drills to excavate samples of subsurface materials, and automated rovers to collect samples from nearby locations.

6.8. Mars Ascent and the MAV Capsule

At the conclusion of the crew's surface stay, the MAV lifts off into a 250 km orbit and docks with the ITV and EEV. The MAV's crew capsule stands 4.5 m high and is 3 m wide; its mass budget is detailed in Table 6.8.1.

Table 6.8.1. MAV capsule mass budget

Component	*Mass [kg]*
Structure	700

Thermal control and life support systems	300
Consumables	40
Power, distribution, and rejection systems	600
Communications and information management	200
EVA systems	493
Reaction control system (600 kg propellant, 500 m/s ΔV)	767
Total landed mass	**3,100**
Crew	400
Mars rocks	500
Total mass at Mars ascent	**4,000**

6.9. Trans-Earth Injection, Earth Return, and the Earth Entry Vehicle

After docking of the MAV, the crew transfers its rock samples to the EEV. The MAV is then jettisoned, and the ITV's main propulsion system fires to place the crew on a trans-Earth trajectory. Upon arrival at Earth, the crew enters the EEV, separates from the ITV habitat, and aerobrakes at Earth for a direct splashdown. The Earth Entry Vehicle is derived from the MAV crew capsule; it also measures 3 m in diameter and 4.5 m in height.

Table 6.9.1. EEV mass budget

Component	*Mass [kg]*
Structure	700
Thermal control and life support systems	300
Consumables	40
Rechargeable batteries and power distribution	300
Communications and information management	100
Reaction control system (245 kg propellant, 170 m/s ΔV)	445
Aeroshell and descent system	1,290
Additional margin	325
Total unloaded mass	**3,500**
Crew and pressure suits	500
Mars rocks and scientific payload	600
Total mass at Earth entry	**4,600**

7. Crew Health Issues

7.1. Radiation

The majority of the crew's radiation dose during the first several human missions will be acquired in interplanetary space. Although the Martian atmosphere is much thinner than Earth's, it still provides reasonable shielding. At altitude 4 km, the atmosphere provides the equivalent of at least 11 g/cm² shielding in the vertical direction. For a surface stay in 2020–2022 similar to that called for in §3.2, Simonsen and Nealy estimate a dose equivalent in the blood forming organs of no more than 19.0 rem from GCR.[28] Solar flares are unlikely considering the fact that this mission occurs shortly after solar minimum, but subsequent missions at solar maximum can expect comparable dose equivalents from solar particle events.

Since GCR is continuous, a shelter for this type of radiation is not feasible. Shielding must be available throughout the entire interplanetary transfer vehicle. A TransHab-derived habitat would have about 5-8 g/cm² with included equipment, with typical atomic number between that of polyethylene and aluminum; the crew's dose equivalent would be about 65 rem/yr at solar minimum.[29] The missions listed in Table 3.2.1 all spend of order 1 year in transit between planets. As a result, the crew's maximum total dose from GCR over the course of the mission is about 100 rem.

On the other hand, a variety of shielding materials may be used for solar particle events; those with high atomic numbers are inadvisable because of the secondary radiation produced when particles collide with these large atomic nuclei. By contrast, when a particle strikes a low-*Z* material such as hydrogen, little of this secondary radiation is produced. Since hydrogen has the lowest atomic number, it would appear to be the logical choice, but logistical difficulties prevent it from being a useful shielding material. Only the liquid form of hydrogen is dense enough to provide appreciable shielding, and although the crew has an ample supply of LH_2 in the ITV's main propulsion system, the rotation of the structure [see §7.2 below] means that this tank cannot shield the crew in the case of a solar flare. But it is relatively easy to create a solar flare shield from polyethylene. A shelter of polyethylene can provide 10 g/cm² of shielding to the crew in addition to the habitat's own shielding. For this reason, the ITV includes a polyethylene sphere 2.6 m in diameter as a solar flare shelter.

An additional concern on the Martian surface is neutron radiation produced from interactions of cosmic and solar radiation with the Martian regolith. The severity of this radiation is not known exactly, but our calculations based on a Langley Research Center model of the neutron flux[30] and our own neutron propagation code (validated by experiments using a ^{252}Cf fast neutron source) suggest that the total dose equivalent to a Mars crew from these neutrons is roughly equal to the direct GCR dose for the surface phase of the mission.

7.2. Artificial Gravity

The ITV-CML complex totals 62 m in length. The center of mass lies 20 m from the lowest level of the ITV habitat, providing a 20 m baseline for artificial gravity. The system rotates around a point in the central section of the truss. At three revolutions per minute, the crew experiences a centripetal acceleration of 2 m/s², somewhat higher than lunar gravity. At 4 rpm, the crew's acceleration is 3.5 m/s², slightly below Martian gravity. Revolution rates greater than 4 rpm will likely have disorienting effects on the astronauts. During this rotation, the complex is oriented so that the truss's solar panels face the Sun at all times. The truss is jettisoned at MOI, so artificial gravity is unavailable for the return trip. The astronauts will travel to Earth in microgravity since medical facilities will be available upon arrival.

8. Summary

8.1. Cost Estimates

Cost estimates of the human missions were constructed using the NASA Spacecraft/Vehicle Level Cost Model.[31] A learning curve of 85% (*ie*, the cost of producing twice as many items is only 1.85 times as much) was assumed. Both optimistic and conservative estimates were applied to the results of the cost model, as listed in Table 8.1.1. All values are in 1999 US dollars.

Table 8.1.1. SVLCM Cost Estimates of Mission Components

	Three missions		*Five missions*	
Component	*Optimistic*	*Conservative*	*Optimistic*	*Conservative*
Z-5 Launchers	$11,442,000,000	$19,616,000,000	$13,559,000,000	$23,244,000,000
Habitats	$17,570,000,000	$43,925,000,000	$19,379,000,000	$48,448,000,000
Aeroshells	$2,575,000,000	$7,724,000,000	$2,776,000,000	$8,328,000,000
Propulsion Stages	$4,319,000,000	$10,798,000,000	$4,726,000,000	$11,815,000,000

The estimates also assumed one pre-landed habitat per mission sequence, one ITV and one MAV per human crew, and two MSPUs and rover landers for the three-mission sequence or three each for the five-mission sequence. Estimates of the MSPU and rover costs were derived from [DRM1/p. 3–128], approximating the cost of two MSPUs and rover landers as 11% of the $55 billion overall cost and scaling linearly upward for the five-mission sequence. The costs of Soyuz launch vehicles for the crew was estimated at $18,000,000 per booster in each set of estimates.[32] Finally, the cost of mission support was estimated using the Mission Operations Cost Model.[33]

Table 8.1.2. Other Cost Estimates of Mission Components

	Three missions		*Five missions*	
Component	*Optimistic*	*Conservative*	*Optimistic*	*Conservative*
Surface Support	$3,025,000,000	$9,075,000,000	$4,538,000,000	$13,613,000,000
Soyuz Launchers	$54,000,000	$54,000,000	$90,000,000	$90,000,000
Mission Support	$738,000,000	$1,549,000,000	$822,000,000	$1,736,000,000

The results of these models are compared with optimistic estimates of the NASA Design Reference Mission 3 [see [34]] in Table 8.1.3.

Table 8.1.3. Total Cost Estimates

	Mars SCHEME Optimistic	*Mars SCHEME Conservative*	*NASA DRM3 Optimistic*
Three Missions	$39,724,000,000	$92,742,000,000	$40,320,000,000
Five Missions	$45,890,000,000	$107,274,000,000	$46,729,000,000

From this we conclude that the total costs are comparable to those of DRM3. We also see that the recurring costs are relatively low; each individual mission has a cost of $3 billion to $7 billion after the initial development of components.

8.2. Risk

A number of factors contribute to the reduced risk of the Mars SCHEME. First, each stage of the mission contains redundant crew life support systems. On the Martian surface, the first crew has available the CML in which it landed, as well as the 2018 habitat. (The MAV was not designed as a long-term habitat.) Subsequent crews will have the same two options as well as the CMLs of previous crews. In interplanetary space, the ITV habitat is equipped with two fully redundant life support systems [see §6.1.2], and the CML is also available on the way to Mars.

Second, new technologies are tested in the Martian environment before they are used by the crew. The HEDS landers [see §4.2] each contain a smaller version of the oxygen generators to be carried by the MAV, as well as radiation and materials exposure experiments. The descent and landing system on the crew Mars lander is identical to that used by the other LMLs, which are first tested in 2016 by the MSPU.

Third, a perfect (error below 1 km) surface rendezvous is not a requirement for crew safety. The CML has sufficient power and consumables [see §6.3.2] to support the crew for up to 30 days, enough time to drive the pressurized rover from the base to the crew's landing site by teleoperation and to return to the base.

Fourth, all LOX/LH_2 propulsion stages have engine-out capabilities. The TMI stages of the ITV each need three of five engines, the ITV's main propulsion stage requires two of three engines, and the MAV requires two of four engines. Although the LML descent stage uses only a single engine, it uses hydrazine/nitrogen tetroxide, a highly reliable bipropellant. In addition, the crew launches on a Soyuz rocket, which is already man-rated and has an extensive history of reliability. An escape tower is provided for the Soyuz launch.

Fifth, electrical power systems were also designed to reduce risk. The Mars SCHEME does not rely on retractable-redeployable solar arrays; retraction of such an array would be a major risk element due to the possibility of a failed retraction just hours before an aerocapture, which would be catastrophic to the mission. Also, there are two MSPUs provided for the crew on the Mars surface, providing redundancy in case of failure of one.

8.3. Conclusion and Recommendations

Clearly, the high reliability and low recurring costs of the Mars SCHEME make it a very feasible sequence of Martian exploratory missions. The plentiful infrastructure it establishes on the surface of Mars also make it a good predecessor to a fully staffed base on the Red Planet.

9. Outreach

As the leaders of the Mars Society of Caltech/JPL, we are at the forefront of space advocacy and public outreach on behalf of Mars exploration. We have had tremendous outreach success in politics, education, and in getting the general public excited about Mars. Highlights of our outreach efforts include:

- Production of an educational activity for 4th-6th grade students in which students pretend to be the first Mars explorers and use math and reasoning skills to save their Mars base's greenhouse. We have visited six schools with the activity and distributed it through our high school science teachers, our web page and via the Mars Society quarterly CD-ROM. We have visited nearly a dozen schools from elementary to college to promote Mars exploration.
- Meetings with major politicians, including face-to-face encounters earlier this year with President Bill Clinton and all four major presidential candidates – Bush, Gore, McCain, and Bradley, before the latter two exited the race. We have met with four members of Congress (Waters, Rohrabacher, Rogan, Calvert) and ten congressional offices (the above and Royce, Sanchez, Pomeroy, Conrad [Senator], Waxman, and Kuykendall).
- Consulting for James Cameron's upcoming Mars movies. Chris calculated his trajectories and we provided feedback on the architecture he will be depicting in his projects during a visit to his office.
- Creation and regular updating of our chapter web site, which features the outreach materials we have designed available for download by other Mars Society chapters or other space advocacy groups, information on our technical projects, Mars, and Mars missions, education materials, and photos from our activities.
- As of this writing, we have gathered 1,998 names and E-mails for our own chapter's mailing list and that of the national Mars Society.

A detailed listing of our events is available at our website, http://mars.caltech.edu/.

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