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An Unmanned Mission to Mars with Sample Collection and In-Situ Resource Utilization

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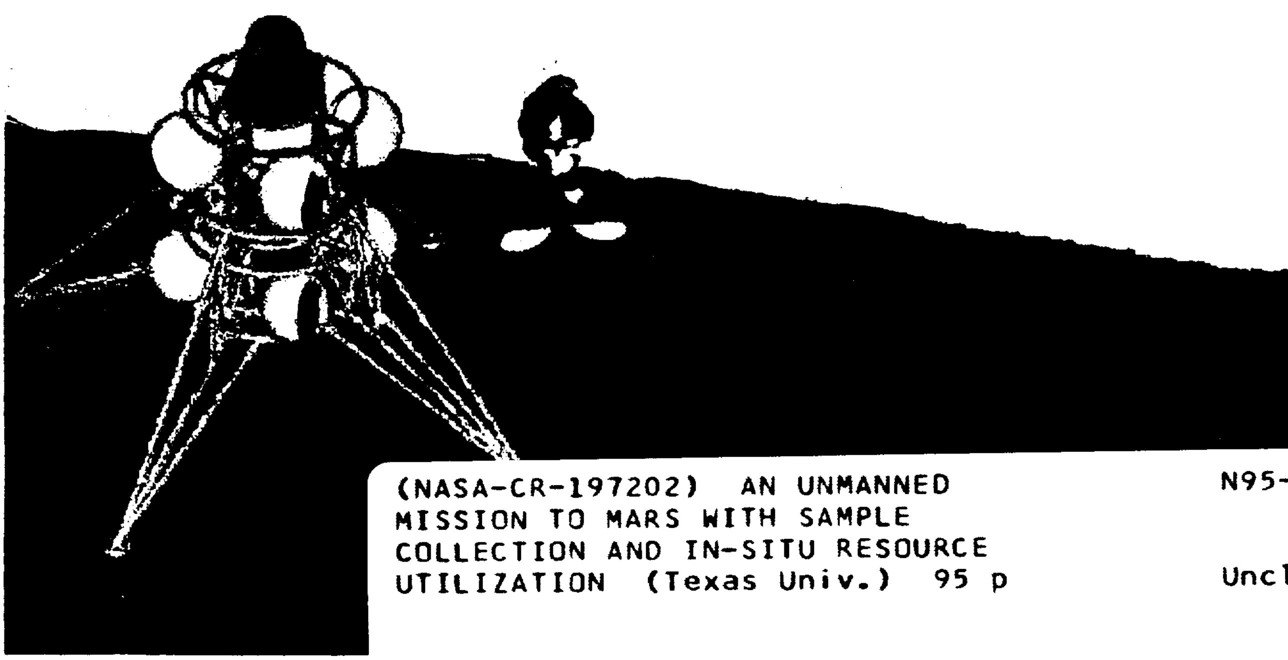
Project

MARVIN



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An Unmanned Mission to Mars with Sample Return and In-Situ Resource Utilization

PDR-2 Report

Project MARVIN
Response to RFP Number ASE274L.0194

Submitted to:
Dr. Wallace Fowler
The University of Texas at Austin
Department of Aerospace Engineering and Engineering Mechanics
Austin, Texas 78712

Submitted by:
Exploratory Technologies

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Abstract

In response to USRA/NASA request for proposal number ASE274L.0194, Exploratory Technologies submits this report, which outlines its design of the Mars Analysis and Return Vehicle with *In Situ* Resource Utilization (MARVIN) project. The MARVIN mission is designed to collect samples of the Martian environment; to produce fuel from local Martian resources; and to use the fuel produced to return the samples to Earth. It uses only existing technologies. Exploratory Technologies' mission-design efforts have focused on methods of orbit determination, sample collection, fuel production, propulsion, power, communications, control, and structural design.

Lambert Targeting provided ΔVs, launch dates, and travel times. The landing site is the Tharsis Plateau; to the southeast of Olympus Mons, chosen for its substantial scientific value. Samples of soil, dust, and atmosphere are collected with lander-based collection devices: the soil sample, with a robotic arm similar to those used in the Viking missions; the atmospheric sample, from a bleed line to the compressor in the fuel-production facility; a dust sample, from the dust-collection container in the fuel-production facility; and a redundant dust sample, with a passive filter system, which relies upon neither a power source nor other collection methods. The Sample-Return Capsule (SRC) houses these samples, which are triply contained to prevent contamination.

Proven technology can be used to produce methane and oxygen for fuel with relative ease at the landing site: the Sabatier reactor produces methane and water by combining carbon dioxide and hydrogen (brought from Earth); the Reverse Water-Gas Shift unit combines carbon dioxide and hydrogen to form carbon monoxide and water; a water-electrolysis unit splits the water into hydrogen and oxygen. The Mars-Lander Vehicle (MLV) transports the equipment from Earth to Mars. The Mars-Ascent Vehicle (MAV) contains the SRC and the engine, which is the same for both the MLV and the MAV. All equipment that is unnecessary for the Mars-Earth trajectory remains on Mars. This report presents detailed sizing information, for which a spreadsheet has been developed. The trends suggest possibilities for expansion, and suggestions for future work in these areas are offered.

Acknowledgments

Exploratory Technologies owes a debt of gratitude to the many students, faculty, professional engineers, and scientists who have contributed their time and expertise to the MARVIN Project. In particular, we wish to thank Tom Sullivan and Dr. David Kaplan, both of whom provided invaluable assistance in the field of fuel production; Dr. James Gooding, whose advice on sample-return requirements has lent this report his substantial credibility; and Robert Cataldo, our source for otherwise elusive information on the Dynamic Isotope Power System. We are also grateful to the faculty and students in the Department of Aerospace Engineering and Engineering Mechanics at the University of Texas at Austin who attended the conceptual and preliminary design reviews and made many helpful suggestions: Dr. Hans Mark, Mark Fisher, Elfego Piñon, Calina Seybold, and Ralf Huber, to name a few. But most of all, Dr. Wallace Fowler deserves our deepest thanks for inspiring us, with his inexhaustible good will and enthusiasm for space exploration, to aim higher.

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ΔV	Delta V
3-D	Three Dimensional
CCD	Charged Couple Device
CH_4	Methane
CMG	Control-Moment Gyroscope
CO	Carbon Monoxide
CO_2	Carbon Dioxide
DIPS	Dynamic Isotope Power System
DSN	Deep Space Network
GNC	Guidance, Navigation, and Control
g_o	Gravitational Acceleration at the Surface of the Earth
H_2	Hydrogen
H_2O	Water
I_{sp}	Specific Impulse
ISRU	*In Situ* Resource Utilization
JPL	Jet Propulsion Laboratory
kg	Kilogram
km	Kilometer
kW	Kilowatt
LEO	Low Earth Orbit
m	Meter
m_f	Dry Mass of the Spacecraft
m_i	Dry Mass of the Spacecraft + Mass of the Fuel
m_{prop}	Fuel Mass
μ	Gravitational Parameter of a Planet
μ_{earth}	398600 km^2/s^2
μ_{mars}	42830 km^2/s^2
MARVIN	Mars Analysis and Return Vehicle with *In situ* resource utilization
MAV	Mars-Ascent Vehicle
MLV	Mars-Lander Vehicle
Mpa	Megapascal
MPL	Mission Planning Library
N	Newton
NASA	National Aeronautics and Space Administration
O_2	Oxygen
^{238}Pu	Plutonium 238 (a radioactive isotope)
r	Radius of the Spacecraft in a Circular Orbit
r_a	Radius of the Spacecraft's Orbit at Apoapsis
r_p	Radius of the Spacecraft's Orbit at Periapsis
RFP	Request For Proposal
RIG	Rate-Integrated Gyroscope
RTG	Radioisotope Thermoelectric Generator

RWGS	Reverse Water-Gas Shift
s	Second
SCA	Sample-Container Assembly
SRC	Sample-Return Capsule
W	Watt

Executive Summary

Overview

The MARVIN project is a design for an unmanned mission to Mars focusing on sample return and *in situ* fuel production. It is likely to serve as a precursor to any manned Mars missions. The MARVIN spacecraft lands on Mars, collects samples of the Martian environment, produces fuel from local Martian resources, and uses this fuel to return the samples to Earth. Producing fuel on Mars reduces the amount of fuel the spacecraft needs to carry from Earth. Martian atmosphere can also be used to help provide water and oxygen for life support. However, before astronauts can rely on these *in situ* systems, the systems must be tested on Mars.

Mission Objectives

The MARVIN mission has three objectives:

1) Collect and return samples of the Martian environment.
2) Produce fuel from local Martian resources.
3) Use the fuel produced on Mars for the return voyage.

Design Specifications

Exploratory Technologies incorporated proven components and existing technologies in its design for the MARVIN mission. Exploratory Technologies' design encompasses aspects of trajectory analysis, fuel production, sample collection, propulsion, power, structures, communications, and control.

Public Support

The support and involvement of the public in any space activity is paramount for mission success. Even if the mission fails to produce fuel, to collect samples, or even to return to Earth, it is a success if the public recognizes the challenges that the mission represents. Furthermore, students can learn the importance of space exploration and how the information we collect can help us better understand Earth if they are sent the information scientists gather from these first samples from another planet. Teachers can use this first-ever information to lend excitement to their lesson plans, in courses as diverse as history, mathematics, geography, physics, chemistry, and biology.

Mission Scenario

The MARVIN spacecraft is launched from Earth on a Titan IV/Centaur booster/upper stage configuration. When the Mars-Lander Vehicle (MLV) separates from the booster/upper stage, a Pratt & Whitney RL-10 rocket engine continues to provide ΔV for midcourse corrections, Mars orbit insertion, and aerobraking maneuvers. After aerobraking is complete, the aerobrake is jettisoned, and the parachute and landing gear are deployed. Once on the surface, lander-based systems collect samples of soil, dust, and atmosphere while fuel is being produced. The samples are stored in the Sample-Return Capsule (SRC). The RL-10 engine is used to launch the Mars Ascent Vehicle (MAV) into a low Mars orbit. The engine then propels the MAV to Earth, where the SRC separates from the MAV. The SRC is retrieved after atmospheric entry.

Advantages of ISRU

In an *In Situ* Resource Utilization (ISRU) mission, local resources are used to produce materials that are critical to the mission. This principle has several advantages. Producing fuel and life-support substances during the mission (e.g. on Mars) can reduce the initial launch weight of the vehicle. These reductions translate directly into cost savings. ISRU is even more advantageous for larger, manned missions. The table below illustrates the savings when ISRU is implemented in the MARVIN mission. Since this mission uses the same engine for both the Earth-Mars and Mars-Earth segments of the mission, methane is burned during both orbital transfers. If ISRU were not implemented on Mars, another more efficient fuel, such as hydrogen-oxygen, would likely be used for both mission phases. All other components of the two designs are assumed to be identical.

Comparison of Mars Missions: Mass Launched from Earth

With ISRU methane fuel	Without ISRU hydrogen-oxygen fuel
4670 kg	7800 kg

These savings in launch mass of the ISRU misson are the result of lower fuel mass, and they translate into lower cost because a less expensive booster can be used.

Trajectory Analysis

For the Earth-Mars transfer, the Titan IV/Centaur provides most of the ΔV required. The remaining 3.7 km/s ΔV required for orbit insertion at Mars is provided by a Pratt & Whitney RL-10 rocket engine. At Mars, aerobraking maneuvers require a total ΔV of .3 km/s. The transfer time from Earth to Mars is 201 days.

For Mars ascent, a ΔV of 3.5 km/s places the spacecraft into a low Mars orbit of 200 km. An additional 2.14 km/s inserts the spacecraft into the Mars-Earth trajectory. The travel time on this trajectory is 205 days. At Earth, the relatively small SRC uses direct entry to enter the atmosphere.

Sample Collection

The MARVIN spacecraft lands at the Tharsis Plateau, where it collects dust, soil, and atmosphere samples from the Martian environment with lander-based collection devices. The samples are then stored in the Sample-Return Capsule, where they are triply contained to prevent contamination. Each sample is collected by a dedicated, unique system, which provides redundancy in the sample-collection process.

The Martian wind-blown dust circulates over the entire planet. Therefore, a sample of it can provide planetary geologists with data on the average composition of Mars, making it the most important sample. Its importance demands that two such samples be collected by separate methods. One sample is collected from the cyclone filter in the fuel-production facility. The filter separates the dust from the atmosphere, and the dust drops into a container. A robotic arm removes the container from the fuel-production facility and places it in the SRC. Dust is also collected with a windsock/funnel configuration attached to the SRC, which catches airborne dust carried by the wind. The dust falls through the funnel into a storage container in the SRC itself.

A robotic arm collects rocks and soil samples from the ground near the lander. The robotic arm has seven degrees of freedom and a 3 meter reach. These samples are also stored in containers in the SRC.

An atmospheric sample benefits scientists who are developing fuel-production and life support systems. The low density of the Martian atmosphere makes an active collection device necessary. A bleed line from the compressor in the fuel-production facility leads to a storage container in the SRC. The atmospheric sample is collected at 1.3 MPa.

A maximum of 5 kg of samples is collected. This total is apportioned as follows: 1.5 kg of primary dust, .5 kg of redundant dust, 2.5 kg of soil, and .5 kg of atmosphere. The individual sample containers are vacuum sealed and are placed in the Sample-Container Assembly (SCA) in the SRC. The SCA provides two redundant seals for sample containment.

Fuel Production

The landing site affects the choice of fuel produced. Further constraints on the type of system to be used include the mass and the reliability of the components. Furthermore, the components must use only proven technology. For these reasons, a methane-oxygen plant was chosen. The major components of the fuel-production facility include a Sabatier reactor, a Reverse Water-Gas Shift unit, and an electrolyzer. Other components include filters, a compressor, pumps, gravity separators, heat exchangers, and refrigeration devices. The Sabatier reactor combines hydrogen imported from Earth with carbon dioxide extracted from the Martian atmosphere to produce methane and water. The electrolyzer splits the water into oxygen (which is stored) and hydrogen (which is fed back into the system). The RWGS unit combines carbon dioxide with hydrogen to produce carbon monoxide and water. The methane and oxygen produced are liquefied and are stored for use in the rocket engine.

During the approximately 540 day stay on Mars, the fuel-production facility must produce 2,241 kg of oxygen and 560 kg of methane. This corresponds to a total production rate of 5.19 kg/day.

Propulsion

A Titan IV with a Centaur upper stage was chosen to launch the MARVIN spacecraft from Earth on the basis of its launch capacity and its fairing size. Once the Titan IV and Centaur separate from the spacecraft, the spacecraft still requires a propulsion system to carry it through the last stages of its Earth-Mars transfer. A Pratt & Whitney RL-10 engine is used here, and it is used again for the launch from Mars and the Mars-Earth transfer. This engine burns methane and oxygen (I_{sp}=370 s) for all mission phases, using 2,384 kg of fuel from Earth to Mars and 2,801 kg from Mars to Earth. The engine has a thrust capacity of 30,000 N - 98,000 N and includes a gimbaled nozzle for controllability. The engine has been tested with methane and has performed well under conditions where multiple restarts were required. Multiple-restart capability is important in this mission because periodic firings of the engine may be necessary to maintain the engine's performance over the long stay on Mars and because the engine is used repeatedly on orbit.

The fuel-production facility, the robotic arm, and various communications and control equipment require power. The largest consumer of power is the fuel-production facility (1900 W) which is only in operation on the surface of Mars. Therefore, two separate power sources can be used: one for on orbit operations, and one for surface operations. For surface operations, a Dynamic Isotope Power Source (DIPS) provides power to the fuel-production facility, the robotic arm, and communications devices. The DIPS can provide up to 2500 W of power and weighs 352 kg. The DIPS is left on Mars to reduce the weight of the return spacecraft. Since on-orbit operations do not require much power, the DIPS can be left on Mars, and a smaller, lighter-weight system can be used. On orbit, solar cells and batteries are the most practical sources of power: they are lightweight and require little space.

Communications, Navigation, and Control

Control of the spacecraft must be automated since real-time remote control is impossible. Furthermore, the control should also be reprogrammable from earth. The spacecraft is primarily three-axis stabilized. However, during thrusting procedures, spin stabilization is necessary. Rate-integrated gyros track the angular orientation of the spacecraft. An accelerometer integrates thrust to provide ΔV calculations. Monopropellant hydrazine control jets are used to correct the attitude of the spacecraft. One set of eight control-jet triads produces one Newton of thrust for each jet.

On the spacecraft, star mappers provide data for automatic position determination. Simultaneously, Earth-based computers use NASA's Deep Space Network to determine the position and orbit of the spacecraft.

Structures

The MARVIN spacecraft consists of three configurations: the Mars Lander Vehicle (MLV), the Mars Ascent Vehicle (MAV), and the Sample-Return Capsule (SRC). The MLV is a truss structure that supports the MAV, the fuel-production facility, the robotic arm, and the landing gear. At Mars ascent, the MAV and SRC separate from the MLV. The MAV consists of the fuel tanks and rocket engine. The SRC sits atop the MAV. At Earth, the SRC separates from the MAV and is retrieved after atmospheric entry.

The truss structure is made of aluminum - lithium 2090-T83. The methane and oxygen tanks are made of titanium. However, since titanium becomes brittle at the temperatures required for cryogenic hydrogen storage, the hydrogen tanks are made of steel. The height of the MLV with the landing gear retracted is 5.0 m, and its diameter is 3.5 m. When the landing gear is fully extended, the MLV is 6.0 m high and 9.0 m in diameter. The MAV is 5.0 m high and 3.5 m in diameter. The SRC is approximately 1 m high by 1 m wide.

1.0 Introduction

Exploratory Technologies submits this report in response to RFP number ASE274L.0194. The report details a preliminary design for an unmanned Mars mission that focuses on *in situ* fuel production and on sample return. This report outlines Exploratory Technologies' approach to the mission objectives in two sections: Mission Elements and Budget.

1.1 Background

Many studies have addressed the design of *in situ* fuel-production facilities and sample-return missions. Some of these studies have been conducted by groups at The University of Washington, NASA Johnson Space Center and the Jet Propulsion Laboratory. Exploratory Technologies has incorporated their research in its efforts to design a simple, lightweight spacecraft that uses current technology.

The MARVIN mission includes both sample return and *in situ* fuel production. However, the sample-return portion of the mission constrains the *in situ* resource utilization (ISRU) fuel production. Scientific requirements for the samples govern the choice of landing site, which in turn determines the types of fuel that can be produced. However, the landing site must also be compatible with available fuel-production methods: the fuel-production system must be capable of producing fuel for the return mission from resources at the science-driven landing site.

This mission is not merely justified—it is essential to future Mars missions. Producing fuel at the destination reduces the amount of fuel the spacecraft needs to carry from Earth. This decrease in fuel reduces both the size and the cost of the mission. The Martian atmosphere is approximately 96% carbon dioxide, 3% nitrogen, and 1% argon. So, for example, methane, oxygen, and water can be formed from atmospheric carbon dioxide and imported hydrogen, and the methane can be used as a propellant. Oxygen serves as the oxidizer. Also, the oxygen and water can provide life support. These concepts suggest that future missions can be made less expensive with ISRU.

Such a mission is likely to serve as a precursor to any manned Mars missions, on which ISRU is most easily justified on the much larger, manned missions. The fuel-production facility can be tested in a Mars-like environment, complete with dust and carbon dioxide, on Earth. However, astronauts cannot rely on such a system until it has in fact been tested and certified on Mars.

Martian samples will not only help answer scientific questions about Mars and its evolution, but they will also help us understand our own planet. A small-scale sample-return mission does not profit much from ISRU. The tradeoff between the mass of the fuel for the return trip and the mass of the fuel-production facility is negligible. However, demonstrating that Mars ISRU is feasible and testing the equipment (including the rocket engine) verifies that the complete system is reliable and that it can be used for future Mars missions.

1.2 Mission Objectives

The MARVIN mission has three objectives:

- Collect and return samples from the Martian environment.
- Produce fuel from local Martian resources.
- Use the fuel produced on Mars for the return voyage.

Each mission objective is equally important. Therefore, this report mentions several redundancy options that are designed to ensure mission success. The first option is to launch two identical vehicles. Each vehicle has one fuel-production facility, one set of sample collection devices, and one primary engine system. The second option is to launch one vehicle that carries extra methane and oxygen in the event the fuel-production facility produces no fuel or produces unusable fuel. Also, in the second option the primary engine system includes three or more engines as a provision for a single engine failure. This report presents a design for the vehicle to be used in the first option, which follows in the footsteps of the Viking, Voyager, and Mars Observer missions, all of which used a similar strategy.

1.3 Advantages of ISRU

In an *In Situ* Resource Utilization (ISRU) mission, local resources are used to produce materials that are critical to the mission. This principle has several advantages. Producing fuel and life-support substances during the mission (e.g. on Mars) can reduce the initial launch weight of the vehicle. These reductions translate directly into cost savings. ISRU is even more advantageous for larger, manned missions. Table 1.3.1 illustrates the savings when ISRU is implemented in the MARVIN mission. Since this mission uses the same engine for both the Earth-Mars and Mars-Earth segments of the mission, methane is burned during both orbital transfers. If ISRU were not implemented on Mars, another more efficient fuel, such as hydrogen-oxygen, would likely be used for both mission phases. All other components of the two designs are assumed to be identical.

Table 1.3.1 Comparison of Mars Missions:
Mass Launched from Earth

With ISRU (methane fuel)	Without ISRU (hydrogen-oxygen fuel)
4670 kg	7800 kg

These savings in launch mass of the ISRU mission are the result of lower fuel mass, and they translate into lower cost because a less expensive booster can be used.

1.4 Design Specifications

The RFP specifies that the design of the MARVIN mission must include certain elements and embrace certain design principles:

- a fuel-production facility
- sample-collection device(s)
- an available booster/upper stage for launch from Earth
- a Mars-Earth trajectory using ISRU fuel
- proven components
- existing technologies

1.5 Deliverables

Exploratory Technologies has delivered a design that meets the above specifications. In addition, Exploratory Technologies has built a model of the spacecraft and has created a poster depicting the phases of the mission. Exploratory Technologies' design encompass aspects of trajectory analysis, fuel production, sample collection, propulsion, power, structures, communications, and control:

- The trajectory-analysis design includes ΔVs for the both Earth-Mars trajectory and the Mars-Earth trajectory and the travel time between planets.
- The fuel-production design identifies the type of fuel to be produced; identifies the components of the fuel-production facility; calculates the fuel-production rates; and determines the mass, size, and power requirements of these components.
- The sample-collection design presents landing-site selection; the types and amounts of samples to be collected, the devices to collect the samples; and the requirements for sample-storage containers.
- The propulsion design identifies the booster/upper stage selected for Earth launch, discusses the primary engine and its sizing, and calculates the fuel requirements for the Earth-Mars and Mars-Earth trajectories.
- The power design includes the power-source selection, the mass and size of the selection, and the power-output capability.
- The structural design includes the Mars-Launch Vehicle layout, including size and mass, the Mars-Ascent Vehicle layout, and the Sample-Return Capsule layout.
- The communications and control design identifies essential equipment for this mission and presents DSN antenna selection.

1.6 Mission Scenario

The MARVIN spacecraft is launched from Earth on a Titan IV/Centaur booster/upper stage configuration. When the Mars-Lander Vehicle (MLV) separates from the booster/upper stage, a Pratt & Whitney RL-10 rocket engine continues to provide ΔV for midcourse corrections, Mars orbit insertion, and aerobraking maneuvers. After aerobraking is complete, the aerobrake is jettisoned, and the parachute and landing gear are deployed. Once on the surface, lander-based systems collect samples of soil, dust, and atmosphere while fuel is being produced. The samples are stored in the Sample-Return Capsule (SRC). The RL-10 engine is used to launch the Mars Ascent Vehicle (MAV) into a low Mars orbit. The engine then propels the MAV to Earth, where the SRC separates from the MAV. The SRC is retrieved after atmospheric entry.

1.7 Public Support

Public support of and involvement in any space activity is paramount for mission success. Even if this mission fails to produce fuel, collect samples, or even return to Earth, the mission has not been a complete failure if the public recognizes the challenges that the mission represents. It has been 25 years since we landed on the Moon. No one has forgotten that event because of the public's involvement. The novelty of watching live footage of the astronauts encouraged the public to become involved, and they wholeheartedly supported the Apollo program.

However, since no astronauts are involved in this mission, encouraging public involvement requires a more creative solution. Video images can once again be provided; however, they lose some of their impact when no astronauts are involved. Student involvement is clearly a way to attract public support. Students can learn the importance of space exploration and how the information we collect can help us better understand Earth if they are sent the information scientists gather from these first samples from another planet. Teachers can use this first-ever information to lend excitement to their lesson plans, in courses as diverse as history, mathematics, geography, physics, chemistry, and biology; and they may be able to base homework or classwork on the data from Martian samples.

2.0 Mission Elements

This section discusses details of Exploratory Technologies' design of the MARVIN mission, an introduction to which is presented here. Lambert Targeting is being used to determine ΔVs, launch dates, and travel times. The landing site is the Tharsis Plateau; to the southeast of Olympus Mons, chosen for its substantial scientific value. Samples of soil, dust, and atmosphere are collected with lander-based collection devices: the soil sample, with a robotic arm similar to those used in the Viking missions; the atmospheric sample, from a bleed line to the compressor in the fuel-production facility; a dust sample, from the dust-collection container in the fuel-production facility; and the redundant sample, with a passive filter system, which relies upon neither a power source nor other collection methods. The Sample-Return Capsule (SRC) houses these samples, which are triply contained. Proven technology can be used to produce methane and oxygen as fuel with relative ease at the landing site: the Sabatier reactor produces methane and water by combining carbon dioxide and hydrogen (brought from Earth); the Reverse Water-Gas Shift unit combines carbon dioxide and hydrogen to form carbon monoxide and water; a water-electrolysis unit splits the water into hydrogen and oxygen. The Mars-Lander Vehicle (MLV) transports the equipment from Earth to Mars. The Mars-Ascent Vehicle (MAV) contains the SRC and the engine, which is the same for both the MLV and the MAV. All equipment that is unnecessary for the Mars-Earth trajectory remains on Mars.

2.1 Trajectory Analysis

This section discusses the ΔV requirements for all phases of the MARVIN mission. It also presents the Earth-launch date and the associated dates for arrival at Mars, departure from Mars, and arrival at Earth.

2.1.1 Analytical Methods

Lambert Targeting is a flexible technique for designing interplanetary trajectories. It provides a means to calculate an Earth-Mars transfer orbit from the following parameters of the MARVIN mission:

- the ΔV capabilities of the booster/upper stage system
- the amount of time that the spacecraft must remain on Mars
- the amount of fuel that can be produced

Lambert Targeting bases its analysis only on the positions of the planets and a predetermined travel time between them. However, in this mission, travel time is related to fuel-production rates, which depend on the fuel requirements of the trajectory. Therefore, an iterative method, in which these parameters and the possible Lambert solutions are functions of one another, can yield the best transfer trajectory for the mission.

"Pork chop plots", shown in Figures 2.1.1 and 2.1.2, [1] for Earth to Mars and Mars to Earth trajectories provide information on the amounts of C3 energies necessary for many possible scenarios and thus provide dates and transfer times for minimum ΔV. The Mission Planning Library (MPL), a set of FORTRAN subroutines, was also used to calculate more precisely the Earth-launch dates and the C3 energies associated with departure from and arrival at the planets for those launch dates. Excel spreadsheets converted these C3 values to ΔVs and fuel masses. Both the Earth-Mars trajectory and the Mars-Earth trajectory were subject to this analysis. Thus, a timetable for the entire mission has been constructed from fuel-production time and the Lambert dates for return to Earth.

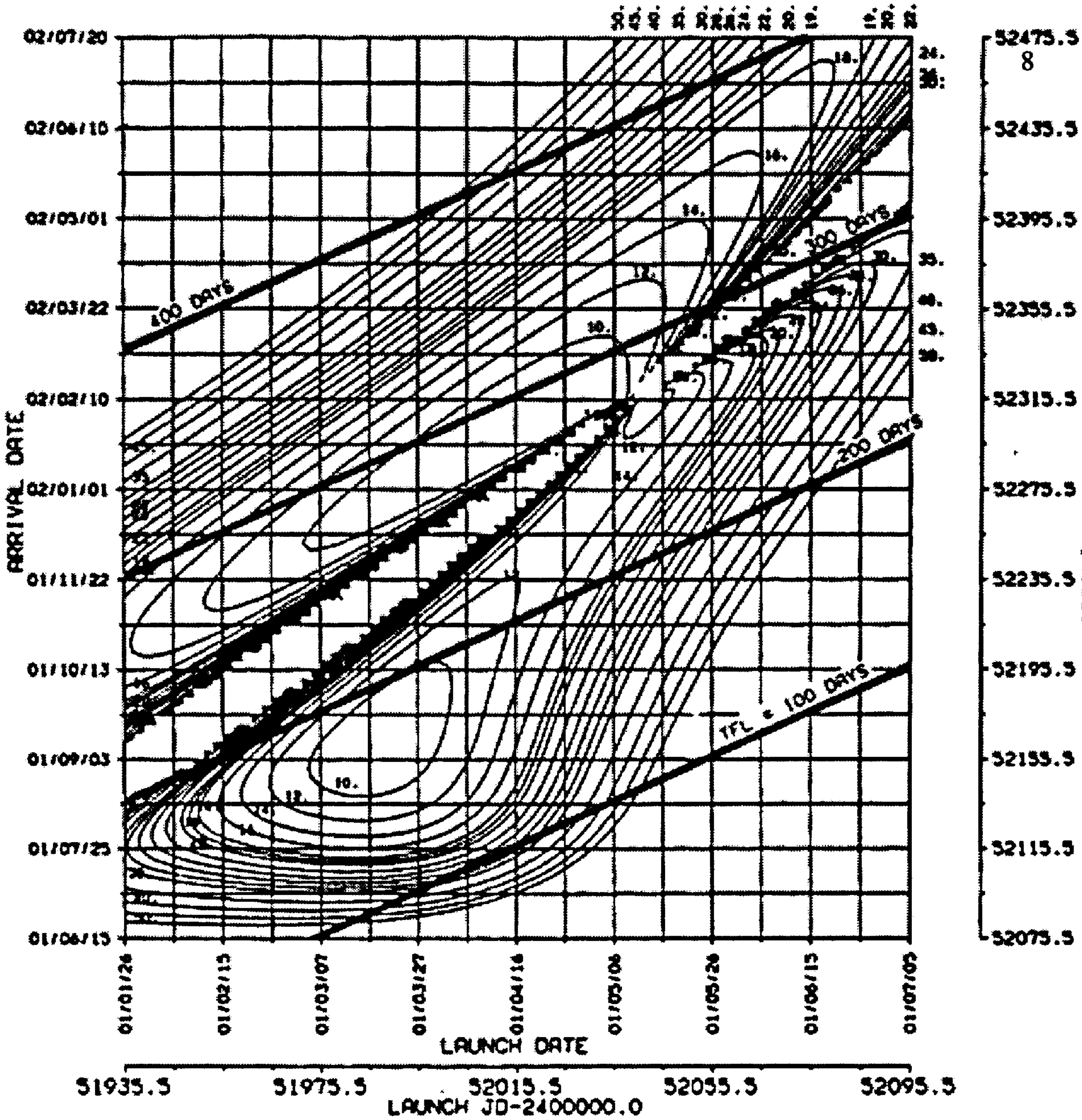


Figure 2.1.1. Pork Chop Plot 1

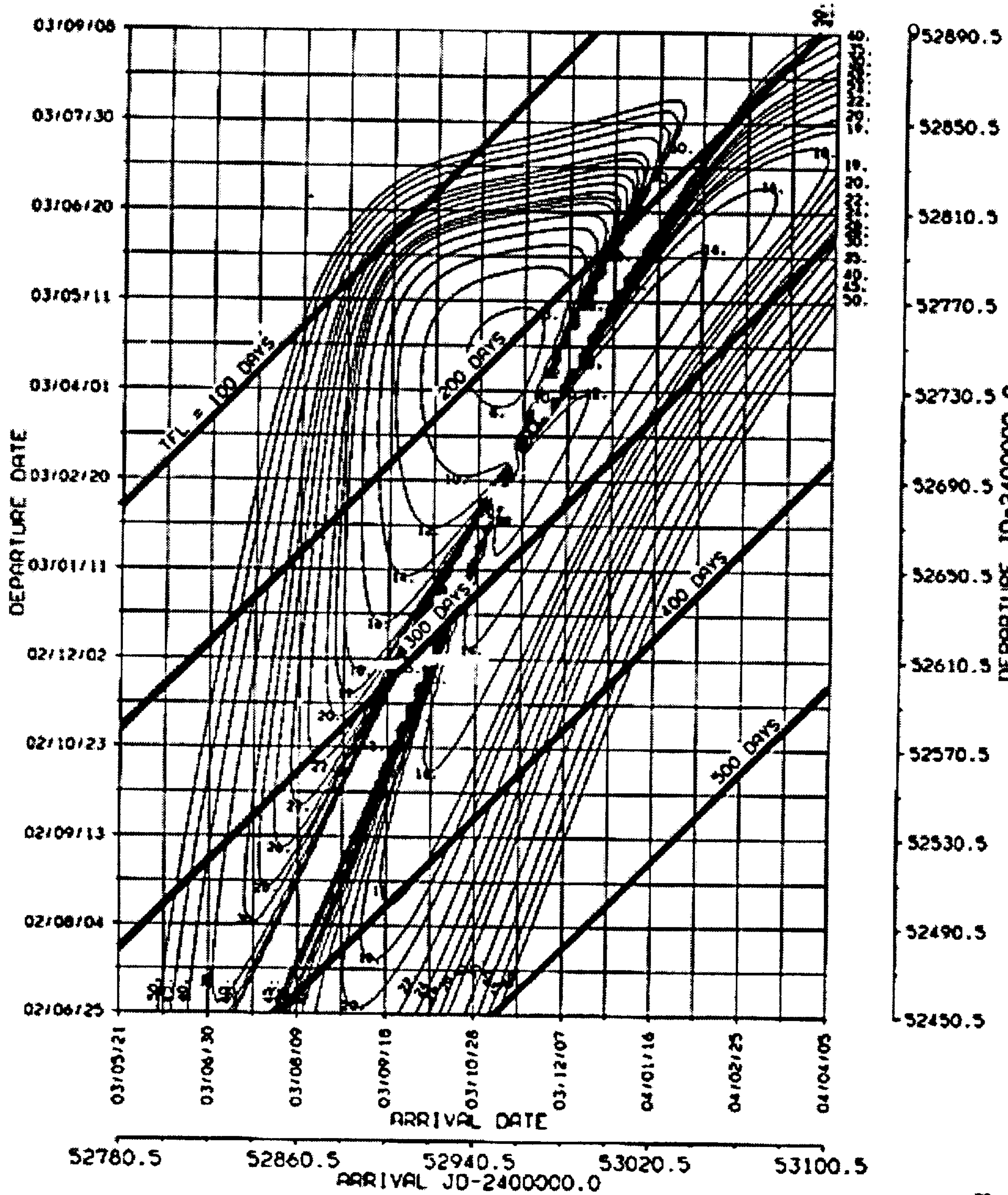


Figure 2.1.2. Pork Chop Plot 2

Also, the ΔV requirements and the payload capabilities of various launch vehicles and booster configurations have been studied with the help of a TK! Solver model. With the dry mass of the spacecraft, this model determines the ΔV that the spacecraft must provide after the booster separates. An iterative process identified the most efficient combination of fuel consumption, spacecraft weight, ΔV, and overall design. Specifying the ΔV from LEO to Mars orbit also sets a maximum payload mass for the spacecraft (as discussed in Section 2.7), which limits the possibilities for changing the design.

2.1.2 Analysis

2.1.2.1 Travel Time

Minimizing the energy required for the return trip reduces the amount of fuel that must be produced on Mars, resulting in a lighter fuel production facility, lighter fuel tanks, and reductions in other system masses. This savings helps minimize not only the fuel requirements for the return trip, but for the outbound trip as well. However, the return date also must be chosen such that it does not require a shorter travel time from Earth or a shorter stay time on Mars, both of which increase fuel mass, as discussed in Section 2.7. A preliminary analysis, using the pork chop plots, identified a window around a return date of April 20, 2003 that minimizes the C3 required for the return trip and provides the mission with roughly a 730 to 750 day interval between Earth launch and Mars launch. Furthermore, the minimized Earth-launch energy associated with this window enables an outbound travel time of roughly 200 days, giving the spacecraft a 530-550 day stay on Mars.

The Mission Planning Library subroutines provided a more thorough analysis of the window. The analysis made use of a FORTRAN program that varied both launch dates and travel times and iterated to find the best solution. The choice of Earth-launch date minimizes energy at Mars arrival since a booster provides the initial injection ΔV. As it turns out, the minimum energy at Mars arrival varies slightly with the minimum energy for Earth departure. The analysis resulted in an Earth departure date of April 8, 2001, a travel time of 201 days, a 540-day stay on Mars, and a Mars departure date of April 19, 2003. The spacecraft arrives at Earth 205 days later.

2.1.2.2 Aerobraking

Aerobraking is a technique in which a spacecraft slows down using a planet's atmosphere to provide drag to lower the spacecraft's energy. A successful maneuver can significantly reduce the ΔV requirements of the propulsion system, thus reducing the initial launch mass. Instead of a direct entry to the landing site, the spacecraft performs multi-pass maneuvers from a parking orbit before landing on the surface. Such a scenario not only reduces fuel use but also allows the rotation of the planet to bring the landing site into proper alignment with the spacecraft. Unfortunately, there are many restrictions on the scenarios that can be used to bring the craft to the surface, and a detailed analysis of the sequence is complex and best studied at the time of the maneuvering. Exploratory Technologies presents aerobraking as a workable solution for mass reduction and offers two multi-pass scenarios for braking to the Martian surface. Details of both scenarios are provided in Tables 2.1.1 and 2.1.2

Table 2.1.1 Aerobraking Scenario 1

	First Pass	Parking Orbit	Second Pass
Periapsis Altitude (km)	-20	250	-97
Apoapsis Altitude (km)	2,470	2,470	600
Apoapsis Burn (km/s)	.0565	.036	-

Table 2.1.2 Aerobraking Scenario 2

	Entry Orbit	First Pass	Second Pass	Third Pass
Periapsis Altitude (km)	200	80	80	10
Apoapsis Altitude (km)	2,000	2,000	500	500
Burn	.023	.261	.015	-

Table 2.1.1 describes a scenario presented in the University of Washington report. This sequence begins with the MLV arriving from Earth in a hyperbolic trajectory and making an immediate atmospheric pass for aerocapture. This pass places the vehicle into an elliptical orbit (2470 x -20 km altitude) about Mars. At apoapsis, a small burn raises the periapsis out of the atmosphere. As soon as the landing site is in proper alignment, the lander performs another burn at apoapsis to lower the periapsis and enter the atmosphere for a second pass. This pass is a "skip maneuver" that lowers the apoapsis to just out of the atmosphere. For the final descent the vehicle aerobrakes to an altitude of 10 km, whereupon parachutes are deployed for continued braking. The burns for this scenario require a ΔV on the order of .1 km/s.

For the second scenario, shown in Table 2.1.2, the spacecraft arrives at Mars in a hyperbolic transfer trajectory and fires its engine at periapsis to enter a highly elliptical orbit about the planet (2,000 x 200 km altitude). At apoapsis the engine fires again to lower the periapsis farther into the atmosphere. Upon arrival at the new periapsis, the spacecraft has lost some velocity and fires once again to lower apoapsis and thus reduce its entry velocity. On the final pass, the spacecraft fires one last time at apoapsis to lower the periapsis altitude deeper into the atmosphere and allow for final braking with the parachute. The three burns for this scenario require a ΔV on the order of only 0.3 km/s.

2.1.2.3 ΔV Estimates

Using a low Earth orbit of 200 km provides numerous injection opportunities in a day. The Excel spreadsheets converted the energy associated with the chosen launch date into a value of 3.70 km/s for ΔV at Earth departure.

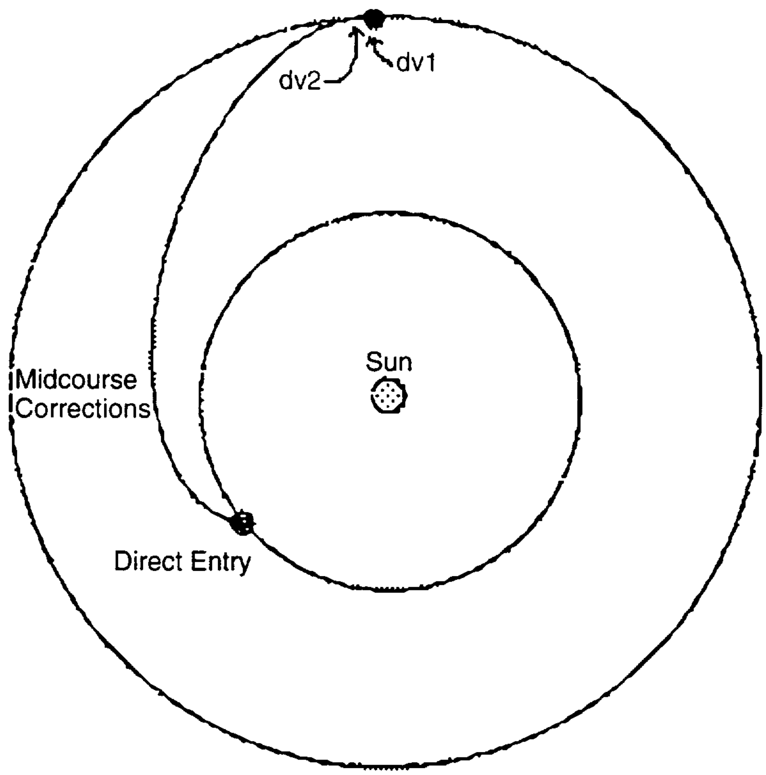
Although a circular orbit around Mars also provides a high number of descent-to-surface opportunities within a given time period, the fuel cost of entering such an orbit is prohibitive. Using aerobraking at Mars for descent, as discussed above, demands very little fuel, and the ΔV for insertion into the 200 km elliptical orbit is 2.15 km/s. The preliminary design presented here uses three passes through the atmosphere, which successively lower the apoapsis and periapsis to lower the final entry velocity. This maneuver, shown in Tables 2.1.1 and 2.1.2, requires a maximum ΔV of .3 km/s to enter the atmosphere and allow for final parachute descent. Thus, for the entire Earth-Mars trajectory, the spacecraft needs to carry only enough fuel for a total ΔV of 2.45 km/s.

A similar analysis can be used to describe the Mars-to-Earth transfer. Because the spacecraft makes a direct entry into Earth's atmosphere upon its return, it requires only enough fuel to escape from the Martian orbit. The spacecraft launches from the surface into a low Mars orbit of 200 km since such an orbit provides frequent injection windows. A parking orbit of this size requires a total ΔV from the surface of Mars to Earth of 5.64 km/s.

Mission Sequence

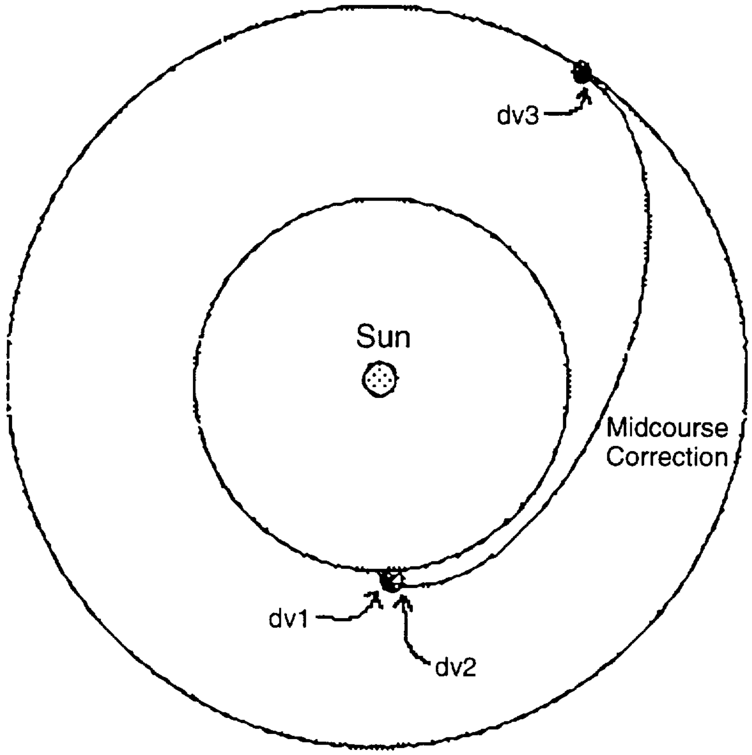
Date	Event	ΔV
8 April 2001*	Earth surface to LEO to transfer trajectory	12.25 km/s
To be determined	Midcourse corrections	~.5 km/s
25 October 2001	Insert into elliptical orbit around Mars	2.15 km/s
	Perform aerobraking burns	>.3 km/s
	Parachute to surface	
	540-day stay	
19 April 2003	Insert into low Mars orbit	3.5 km/s
	Insert into return trajectory	2.14 km/s
19 November 2003	Arrive at Earth; direct entry into atmosphere	

Figures 2.1.3 and 2.1.4 illustrate the two transfer trajectories for this mission.



semimajor axis: 1.870 * 108 km
eccentricity: .21
transfer angle: 151 degrees

Figure 2.1.4. Mars-Earth Trajectory



semimajor axis: 1.769 * 108 km
eccentricity: .17
transfer angle: 147 degrees

Figure 2.1.5. Earth-Mars Trajectory

2.2 Sample Collection

This section identifies the most useful Martian samples to be returned to Earth, describes methods of collecting the samples, and discusses how the samples are protected until they reach Earth. The types of samples have been chosen according to the requirements of planetary geologists, and these types determine the landing site. Furthermore, the methods of collection are tailored to the types and amounts of samples to be collected.

2.2.1 Types of Samples

Three types of samples are collected from Mars and are returned to Earth: wind-blown dust, soil, and atmospheric samples. Each of the samples is collected by a separate, unique collection system, which provides redundancy in the sample-collection process. To provide additional redundancy, a second sample of dust is collected.

Since the Martian wind-blown dust circulates over the entire planet, a sample of it can provide planetary geologists with data on the planet-wide composition of Mars. A sample from the ground near the lander cannot provide such data; it can yield only local geological information. Therefore, the substantial information that a dust sample can provide makes this sample the most important one that MARVIN collects [1].

The soil sample, which may include rocks, is collected from the ground near the lander. Because the landing site is near Olympus Mons, the soil sample ought to provide planetary geologists with more information about Olympus Mons and its history.

An atmospheric sample will benefit scientists who are developing the fuel-production system. However, because of the low density of the Martian atmosphere, an atmospheric sample must use some sort of active collection mechanism if a usable amount of atmosphere is to be collected in a container of manageable size. The atmospheric sample is therefore collected at a pressure greater than the Martian atmospheric sample.

In addition to these samples, the redundant dust sample is collected with a separate collection mechanism. Collecting another dust sample with a different collection device improves sample-collection redundancy. And, if both the primary and redundant dust samples are successfully returned to Earth, more dust will be available for research.

2.2.2 Amount of Samples

The MARVIN spacecraft returns a maximum of 5 kg of Martian samples to Earth: 1.5 kg of dust, 2.5 kg of soil, 0.5 kg of atmosphere, and 0.5 kg of redundant dust. This amount is sufficient for scientific research because the techniques learned in the processing and analysis of the lunar samples returned by the Apollo program have enabled scientists to conduct research on very small quantities of material. In addition, 5 kg has been suggested by several planetary geologists and is believed to be an optimum tradeoff between Mars takeoff mass and scientific utility [2,3].

2.2.3 Landing Site

The landing site chosen for the MARVIN project is the Tharsis plateau, which lies between 0^{o} and 30^{o} north latitude and 90^{o} and 135^{o} longitude and is shown in Figure 2.2.1 [4]. The Tharsis plateau is interesting to planetary geologists because it includes Olympus Mons, the largest volcano on Mars, and is a relatively young region [1]. Although the region contains three volcanoes, any of several relatively flat plains may be suitable landing sites for the MARVIN lander. The most likely landing site is between 10^{o} - 15^{o} north latitude and 130^{o} - 135^{o} longitude. This area is just south of Olympus Mons, in a plain between 2 and 3 kilometers above the mean surface of Mars [4].

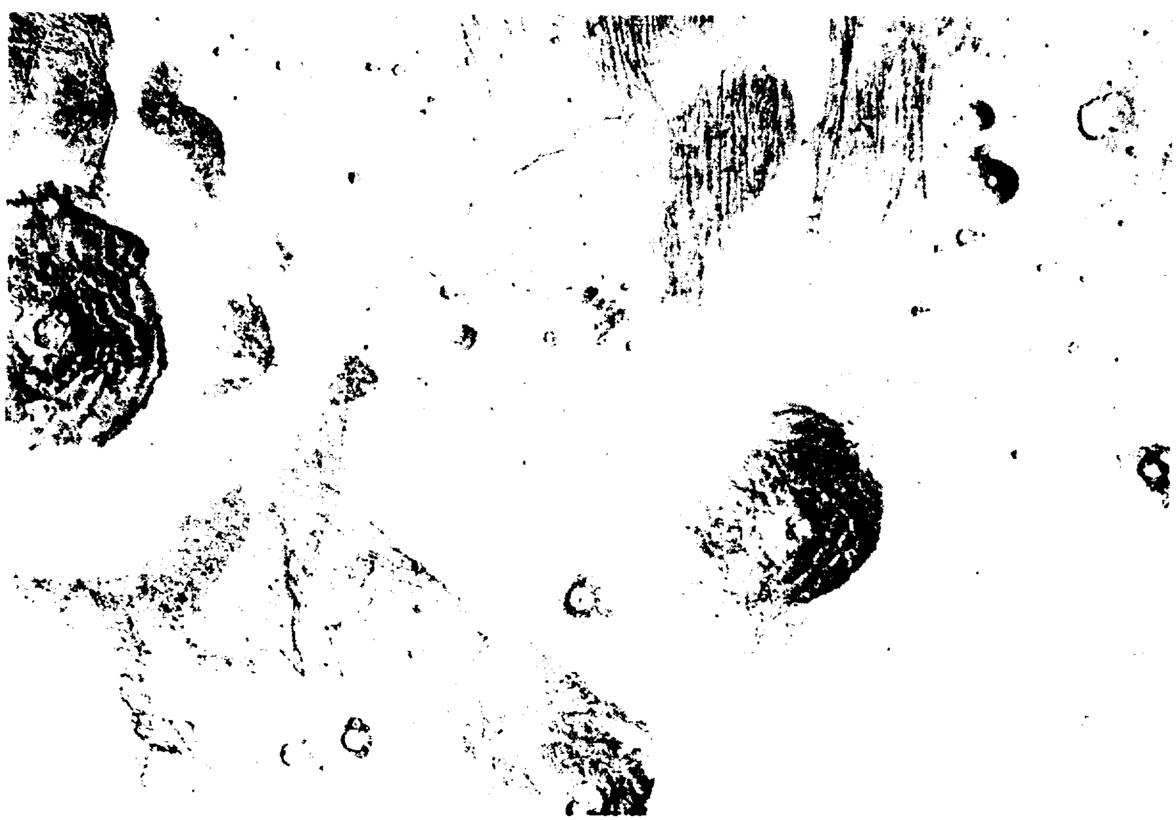


Figure 2.2.1. Tharsis Plateau.

2.2.4 Methods of Collection

To improve redundancy, four separate methods of collection are used to collect the four samples. Each sample has its own sample container, seals, and collection device. However, the robotic manipulator has no redundant counterpart and is required for both the soil and the dust samples. If the manipulator fails, the soil sample cannot be collected, and the dust sample cannot be placed in the Sample-Return Capsule (SRC).

The dust sample is collected from the hydrocyclone filter and pump system in the fuel-production plant. Dust settles into the bottom of the filter, where the dust sample container stores it until the container is full. The robotic manipulator places the dust sample container in the Sample-Return Capsule. Using the fuel-production system to collect the wind-blown dust eliminates the need for a separate filter and pump system, thereby decreasing the number of separate systems on the lander. This decrease reduces total mission mass, increases simplicity, and lowers cost.

The soil sample is collected by a means similar in concept to that used on the Viking missions: a robotic manipulator collects soil and rocks from around the lander and places them in a sample container. However, the manipulator in the MARVIN design is more sophisticated than the Viking spacecrafts': it has 7 degrees of freedom, a reach of 3 or more meters, and it is semi-autonomous. A schematic of the manipulator is shown in Figure 2.2.3. Redundant joints and telescoping members provide the manipulator with operational capability should a failure occur. In addition to collecting the soil samples, the manipulator places the containers with the dust and soil samples in the Sample-Return Capsule. The manipulator is also available for contingency operations and maintenance. Robotic operations requires several cameras, which the lander would not otherwise need, to provide adequate coverage of the worksites.

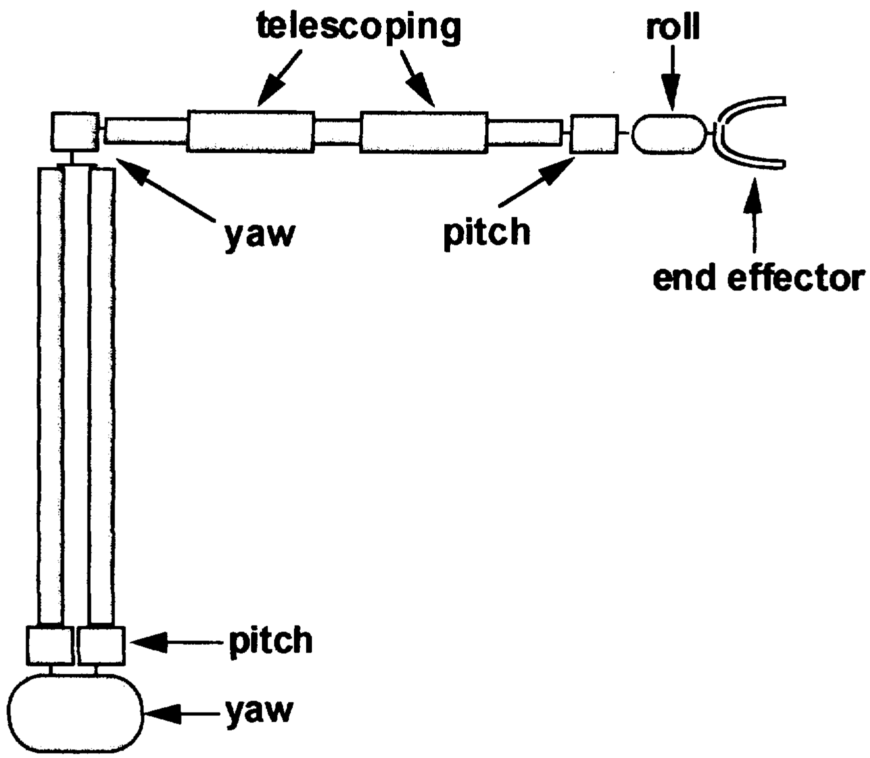


Figure 2.2.3. Manipulator

The atmosphere-collection system uses a bleed line from the fuel-production compressor to collect Martian atmosphere and deliver it at 1.3 MPa to the atmospheric sample container. The atmospheric sample container remains in the Sample-Return Capsule at all times—the bleeder line runs to the container but disconnects when the container is full. Again, using a component of the fuel-production plant to collect samples reduces the overall mass and cost of the lander.

Reliability is an extremely important consideration in the design of the collection systems, and particularly so in the case of the redundancy sample. Since the sole purpose of the redundancy sample is to serve as a backup for the dust sample, it cannot not rely on any other sample-collection devices or components and must be as simple in design as possible. For these reasons, a deployable dust collector, shown in Figure 2.2.4, has been chosen as the method for collecting the redundancy sample. A simple mechanism deploys a semi-rigid wind sock from the Sample-Return Capsule shortly after landing. Air and dust enter the wind sock through the large opening; the dust is trapped inside the wind sock and falls into the redundancy sample container in the Sample-Return Capsule. After sufficient dust has been collected, this collector is jettisoned from the spacecraft.

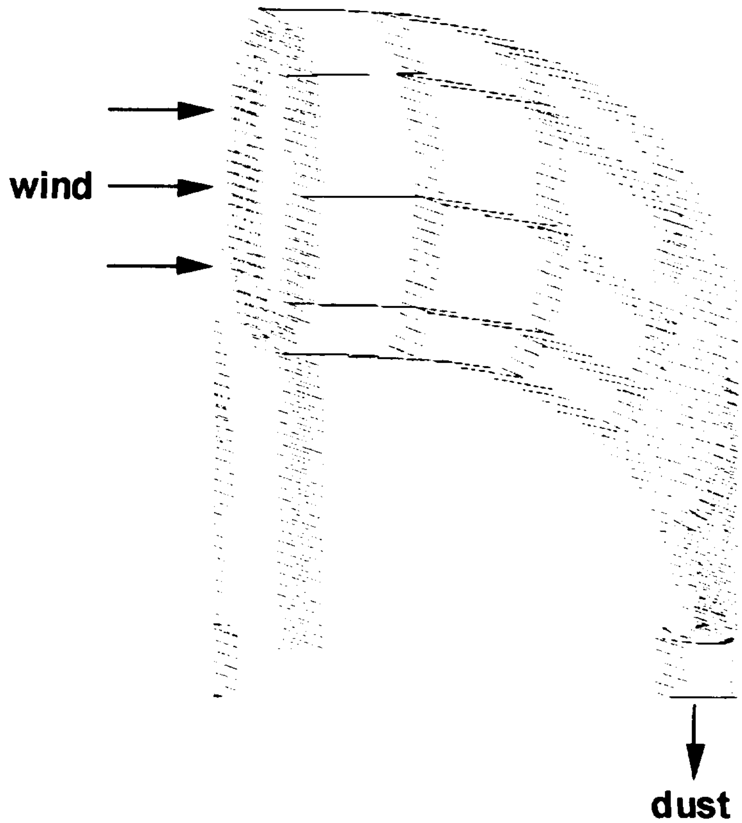


Figure 2.2.4. Wind Sock

2.2.5 Sample Containers

The samples must be carefully maintained from the time they are collected until their arrival at Earth. Volatiles in the samples must not degrade; therefore, the samples must not be exposed to contaminants from Earth, subjected to extreme temperatures, or exposed to unusual amounts of radiation. These constraints place stringent requirements on the containers that carry the samples to Earth, the environment in which the containers are kept, and their placement on the lander and Mars-Ascent Vehicle. To this end, a highly redundant protection scheme has been devised. Each of the four samples is stored in its own sample container, which is vacuum sealed as soon as the sample is collected. The four sample containers are then placed in a Sample-Container Assembly (SCA), which carries the samples in the Sample Return Capsule for the trip back to Earth. The SCA is insulated, instrumented with temperature and pressure sensors, and has two vacuum seals, all of which provide the samples with a high level of protection from the harsh environment of space.

2.3 Fuel Production

2.3.1 Fuel Choice

Exploratory Technologies considered four fuel-production systems: production of methane (CH_4) from the Martian atmosphere using imported hydrogen (H_2) from Earth; production of H_2 from the water (H_2O) in the Mars polar ice caps; production of CH_4 from H_2O extracted from the Mars permafrost layer; and production of carbon monoxide (CO) directly from carbon dioxide (CO_2) in the Martian atmosphere. CH_4 - O_2 is a fairly effective propellant, with an I_{sp} of 370 s. However, only the production of CH_4 from the Martian atmosphere with imported H_2 meets the criteria of proven design, low complexity, high reliability, and the ability to produce fuel at the chosen landing site.

2.3.2 Production-Plant Components

The CH_4 fuel-production facility includes three major components: the Sabatier reactor, an electrolyzer, and a Reverse Water-Gas Shift (RWGS) unit. This CH_4 fuel-production system also uses many sub-components such as filters, compressors, refrigeration units, and pumps, as shown in Figure 2.3.1. The following description explains the operation of the fuel-production system.

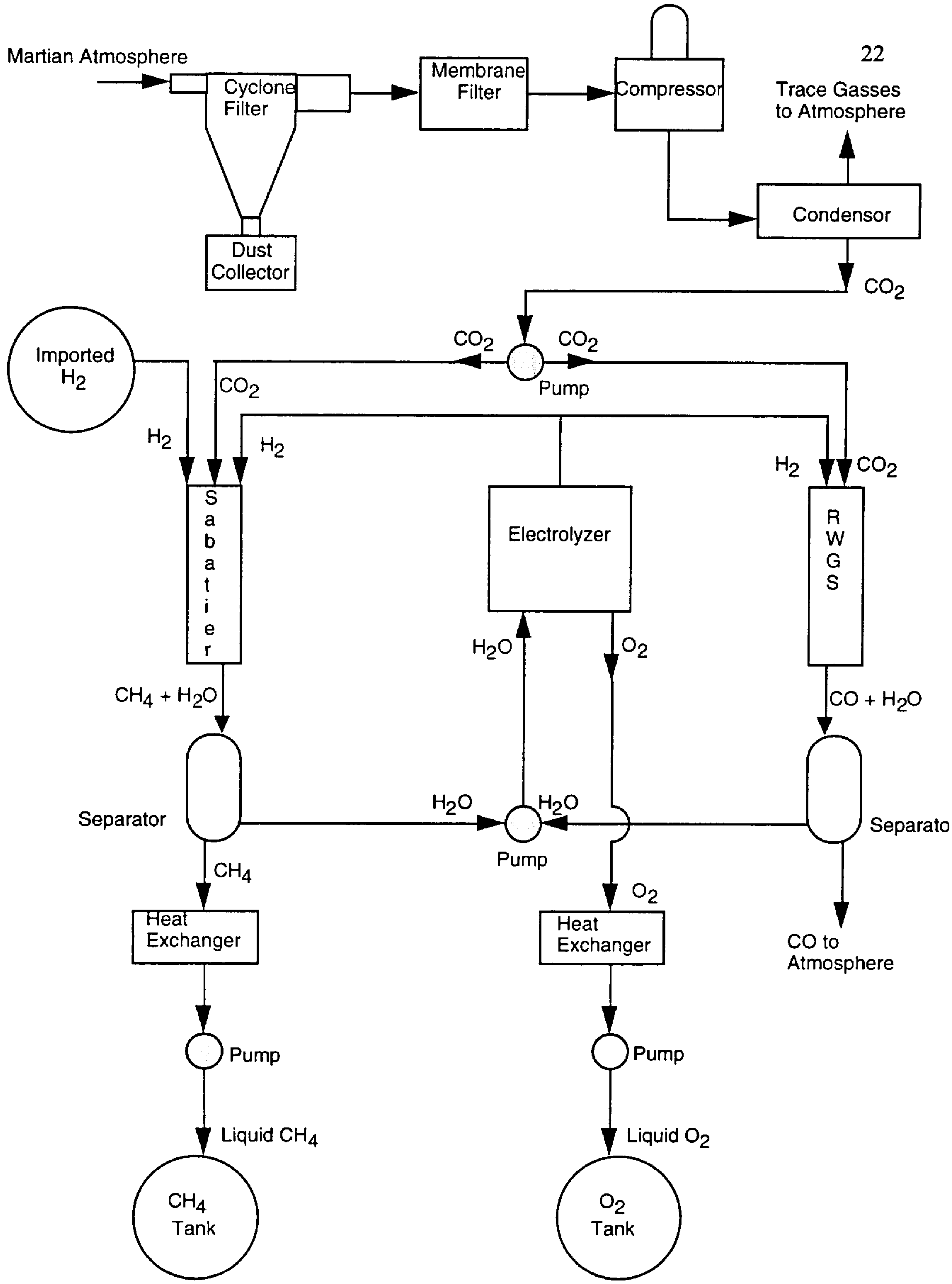


Figure 2.3.1. Fuel-Production System

The Martian atmosphere is first filtered to remove dust particles before continuing through the rest of the system. Then, the Martian atmosphere is compressed to liquefy the CO_2. Trace gases are bled off. The CO_2 is then pumped into the Sabatier reactor and the RWGS unit. In the Sabatier reactor, H_2 imported from Earth is combined with the Martian CO_2 to form CH_4 and H_2O. These substances are separated in a gravity separator. The CH_4 is pumped through a heat exchanger and refrigeration unit before being stored in the fuel tank. The H_2O is pumped into the electrolyzer. The electrolyzer splits the H_2O into H_2 and O_2. The O_2 also passes through a heat exchanger and refrigeration unit before being stored in the fuel tanks. The H_2 from the electrolyzer is pumped back into the Sabatier reactor and the RWGS unit. The H_2 that is fed into the RWGS unit combines with the Martian CO_2 to form CO and H_2O. These substances are also separated in a gravity separator. The CO is released into the atmosphere. The water is pumped to the electrolyzer.

The filter system consists of a cyclone filter in series with a membrane filter. The cyclone filter creates an internal vortex caused by high velocities of the entering atmosphere, as shown in Figure 2.3.2. Pressure gradients cause particles as small as 5 microns to be removed [5]. The cyclone filter removes approximately 95% of 5 micron particles and virtually 100% of particles larger than 10 microns in diameter [1]. As discussed in Section 2.2, the removed dust is collected for return to Earth. Since better filtration is desired, a membrane filter is used in conjunction with the cyclone filter.

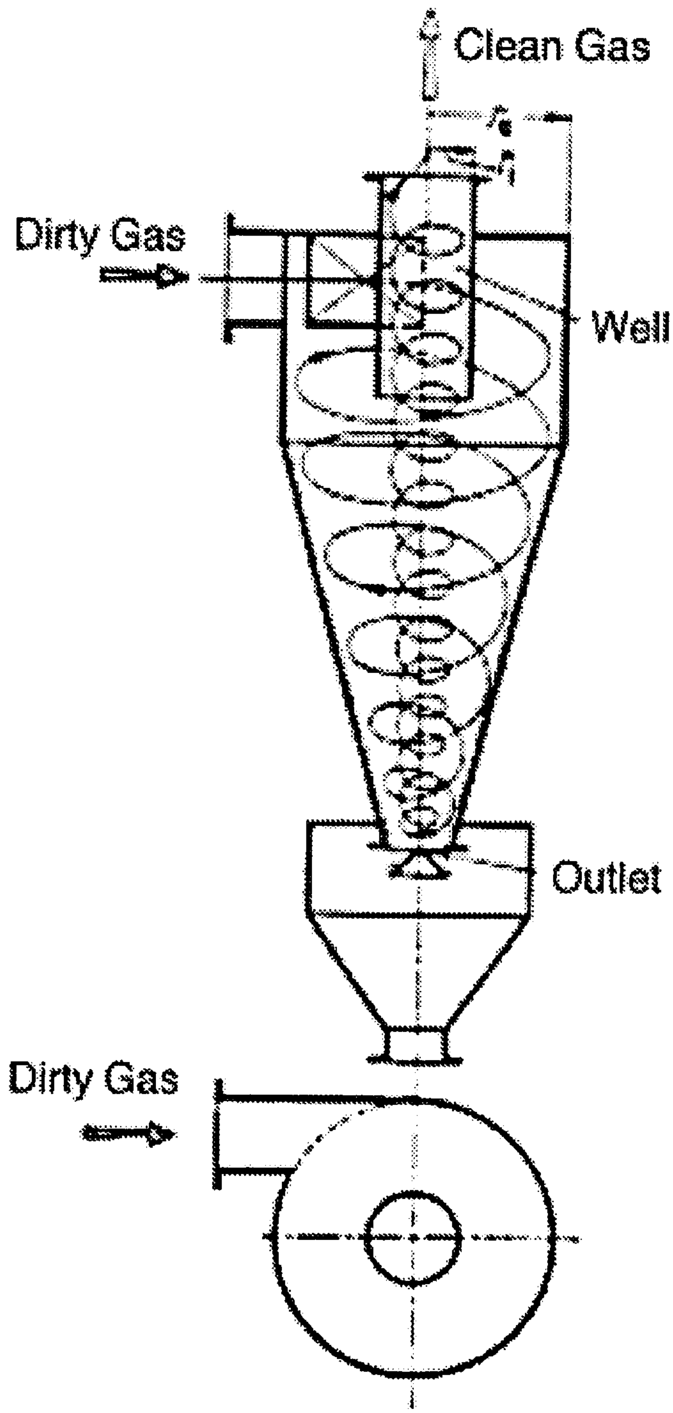


Figure 2.3.2. Cyclone Filter

To reduce problems associated with possible filter clogging, a continuous fiber-layer filter, shown in Figure 2.3.3, is used. This filter automatically forwards a roll of polyacrylonitrile fibers either at specific time intervals or when the pressure difference through the filter reaches a specified level. Use of high-efficiency fibers allows removal of at least 99% of dust particles as small as 1 micron [5]. Particles smaller than 1 micron should not be detrimental to the fuel-production process [2].

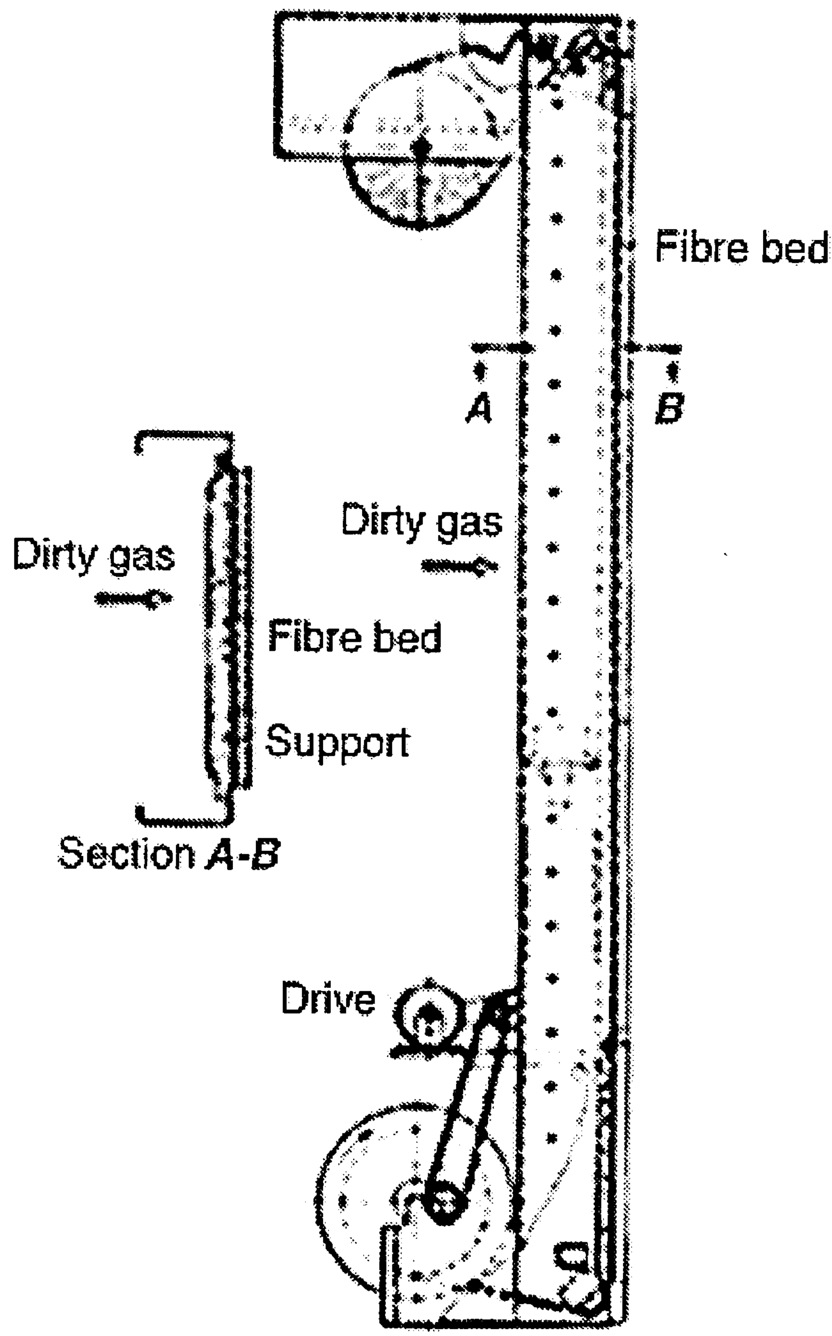


Figure 2.3.3. Fiber Filter

The filtered Martian atmosphere is compressed to 1.3 MPa [1]. This compression liquefies the CO_2 so that nitrogen, argon, and other trace gases can be bled off. High-pressure chemical metering pumps, pump the CO_2, H_2O, and O_2 through the system. The pumps are designed with an operational lifetime of five years [8,9].

The Sabatier reactor produces CH_4 by combining carbon CO_2 and H_2 to create the following exothermic reaction:

$$CO_2 + 4H_2 \Rightarrow CH_4 + 2H_2O\,.$$

The Sabatier reactor, shown in Figure 2.3.4, requires no electrical power. However, for this chemical reaction to occur, the Sabatier reactor must be maintained at or above 150°C. The reaction is exothermic and produces a temperature around 1350°C, but some small heaters are clamped around the reactor to heat the Sabatier reactor initially. The heaters may occasionally draw power, but that is unlikely [7].

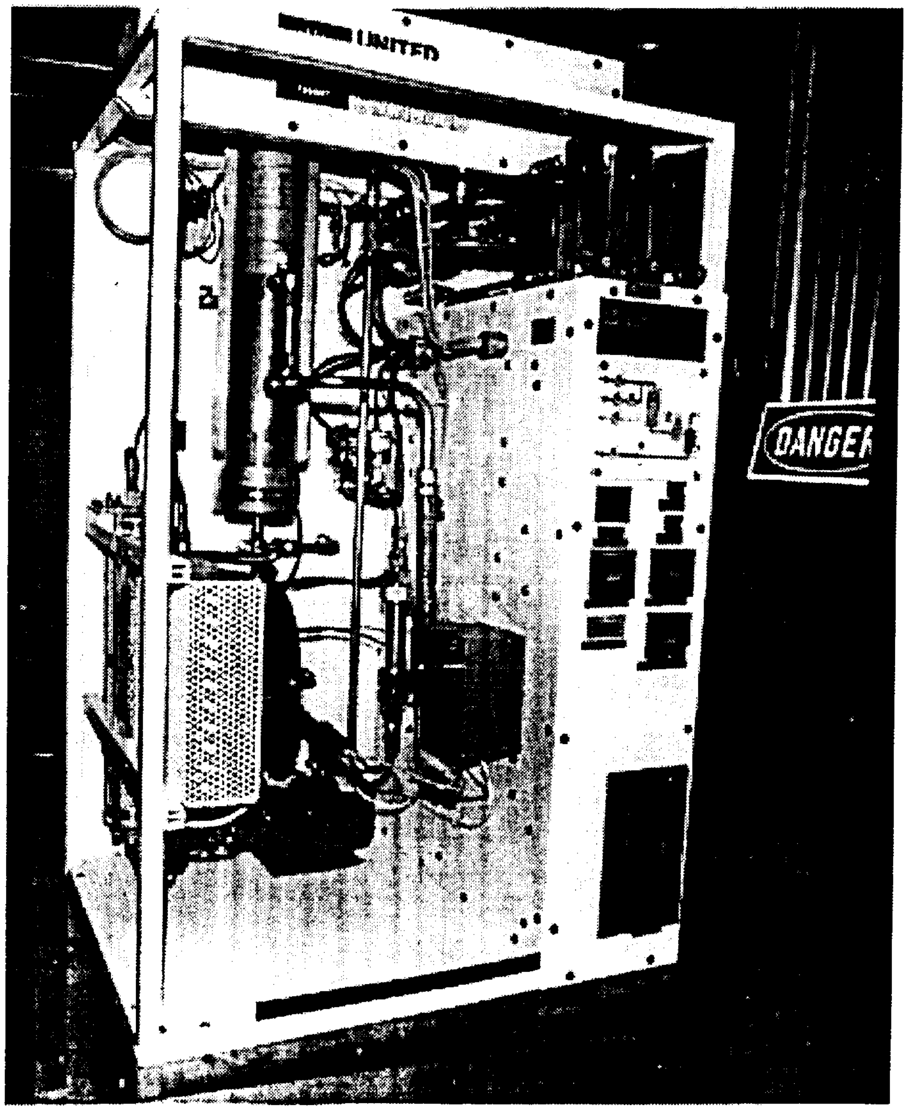


Figure 2.3.4. Sabatier Reactor in NASA's Fuel-Production Facility

The electrolyzer dissociates H_2O into H_2 and O_2 by the process

$$2H_2O \Rightarrow 2H_2 + O_2 .$$

Current unipolar electrolyzer installations have electrical current efficiencies of 99.9%. For maximum conductivity, a potassium hydroxide solution is required as an electrolyte. For sealed cells, this original electrolyte charge can be used for up to 10 years. O_2 recovered during this process has a purity of 99.7% [6].

Since the Sabatier/electrolyzer system produces only half the O_2 required for stoichiometric combustion, the system includes a RWGS unit to convert H_2 and compressed CO_2 to H_2O [2]. The unit uses thermal energy to produce H_2O and CO through the reaction

$$CO_2 + H_2 \Rightarrow CO + H_2O .$$

CO is a waste product and is vented to the Martian atmosphere [1].

Heat exchanger/refrigeration units liquefy the O_2 and the CH_4 for storage in the fuel tanks. Use of highly insulated tanks and the constant influx of the cryogenic liquids makes boil-off negligible for the CH_4 and O_2 tanks [6]. However, there is some concern about boil-off of the imported H_2. Boil-off can best be controlled with a multi-layer vacuum dewar tank configuration as shown in Figure 2.3.5 [4]. Large land-based tanks currently in use by NASA reduce boil-off down to .02% per day [6]. However, this loss in efficiency is offset by the colder temperatures associated with deep space and Mars as opposed to Earth. Boil-off during transfer to Mars should be negligible because the temperature of deep space is 3 K. Furthermore, a .004 cm layer of gold foil on the tank produces negligible radiated heat flux to the tank [6].

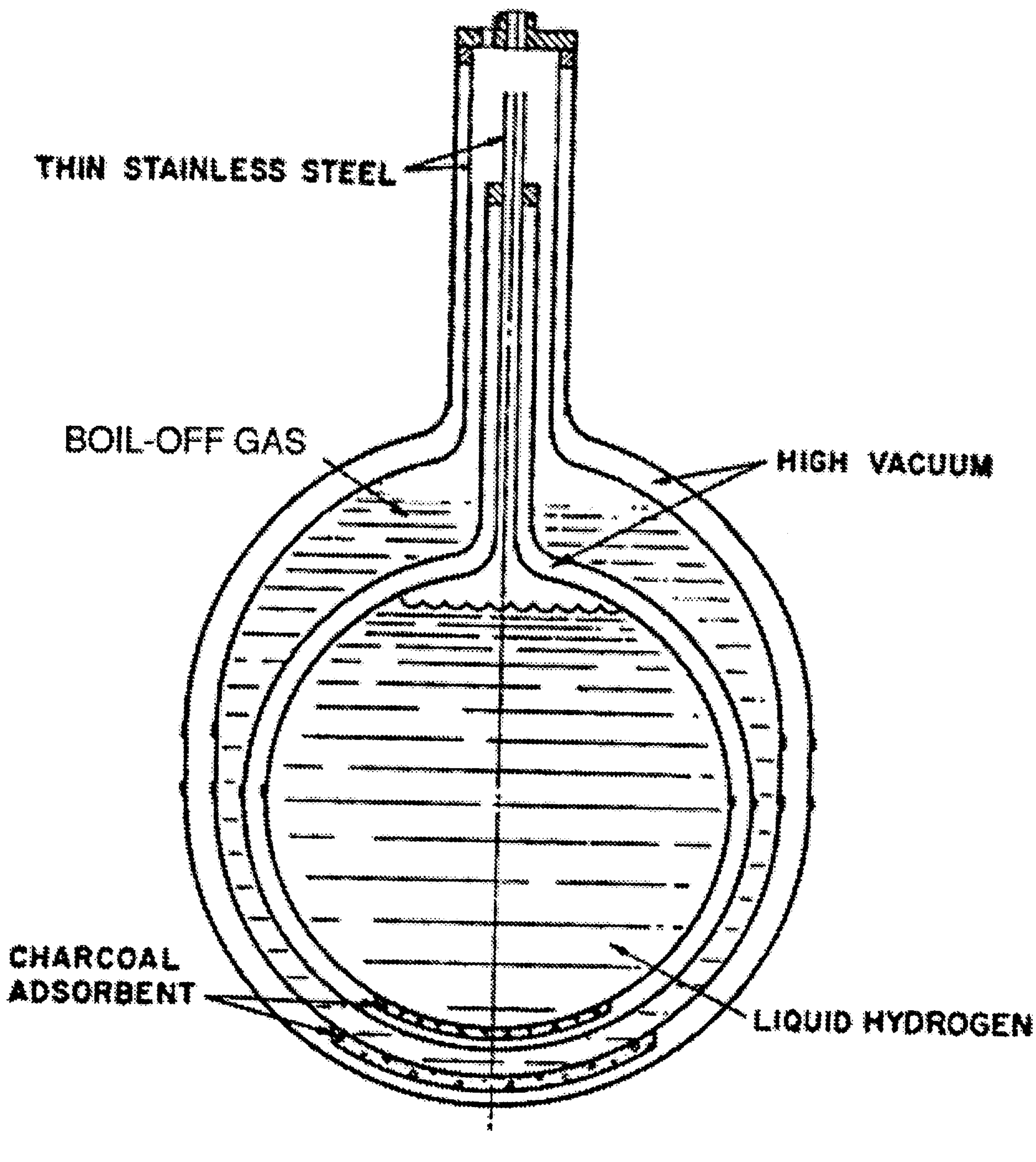


Figure 2.3.5. Dewar Tank

Using a dewar tank should limit the boil-off rate at Mars to .1% per day. Since pressure is required for transport of H_2 into the Sabatier reactor, the pressure normally associated with H_2 boil-off is actually beneficial for this process. Boil-off to the atmosphere is reduced as the high-pressure gas is fed into the fuel production system. The total boil-off for the trip should be no more than 15% and is probably significantly lower than this projected value.

Table 2.3.1. Estimate of Average Power Requirements for Fuel-Production Components [1,2,4,6]

Component	Requirement
Electrolyzer	1065 W
Heaters	150 W
Compressor	175 W
Pumps	100 W
Heat Exchangers	515 W
TOTAL	2005 W

2.3.3 Production Rates

The return trip from Mars (including 5% extra fuel) requires that the fuel-production unit produce 2801 kg of fuel on Mars. To meet this requirement, 1.03 kg of CH_4 and 4.16 kg of 0_2 is produced each day during the stay on Mars. The production rate is directly related to the amount of power supplied to the electrolyzer and the size of the Sabatier reactor. To iterate the component sizing for the spacecraft, numerous spreadsheets were developed using known component sizes, weights, and production rates.

Primarily, this sizing was performed by producing a curve-fit of the characteristics for known, tested components. Table 2.3.2 lists the mass of the primary components required for the required fuel production.

Table 2.3.2. Component mass required to produce 1.08 kg/CH_4 and 4.32 kg/O_2 per day [1,2,3,8,9]

Component	Mass (kg)
Imported Hydrogen	168
Filters	55
Sabatier Reactor	41

Reverse Water-Gas Shift Unit	41
Compressor	22
Pumps	18
Water Electrolyzer	15
Heat Exchangers	5
Other	8
TOTAL	373

Both the electrolyzer and the Sabatier reactor are proven components, and many working models exist [3]. However, no working model of the Sabatier reactor of the exact size required for the MARVIN mission exists, nor has a complete facility of this precise size been assembled and tested. Thus, the integrated system must still be designed with appropriately sized components.

2.4 Propulsion and Power

The MARVIN spacecraft uses a modified Pratt & Whitney RL-10 engine for Mars ascent and for the return trajectory to Earth. This engine is the same one that provides ΔV after the initial, Earth-departure booster and upper stage have been exhausted. Using one engine for all subsequent, propulsive mission phases not only reduces the mass of the MARVIN spacecraft over that of other designs [1]; it also enables smaller (and therefore cheaper) boosters and upper stages to be considered. A trade study suggests that, although an Earth-departure engine that uses a more efficient fuel offers a reduction in fuel mass, the structure required to attach it outside the aerobrake/heat shield is massive and complex. This structure makes such a spacecraft more expensive to launch from Earth than the chosen design, in which the merely engine extends out from inside the base of the aerobrake/heat shield. Furthermore, the size of such a structure would exclude relatively inexpensive launch vehicles, with small fairings, from consideration.

2.4.1 Earth-Departure Booster and Upper Stage

The choice of an appropriate booster/upper stage configuration depends primarily on the launch mass of the spacecraft. Current sizing calculations, shown in Appendix A, suggest that this mass is approximately 4670 kg, if the booster/upper stage can insert the spacecraft into the Earth-Mars trajectory. For this case, the RL-10 engine fires only to make small corrections and to insert the spacecraft in the elliptical Mars orbit discussed in section 2.1.

The primary engine is fueled at launch to provide additional thrust after the booster separates. The Proton, with an upper stage, can provide sufficient ΔV for this scenario. However, despite Lockheed's recent efforts to undertake a joint enterprise with Khrunichev Enterprises to offer commercial Proton launches[3], the Proton's reliability is less than 90%; its manifest is booked for 10 years [4]; and its long-term availability remains in question.

The Shuttle can certainly launch the payload, but its cost is several times that of most Titan configurations. Furthermore, the presence of the nuclear power source and several pressurized fuel tanks puts crew members at risk. Titan-class vehicles are then clearly the better choice, but most upper stages (e.g. the TOS and Transtage) do not provide sufficient performance. The Titan IV/Centaur is more appropriate. The less expensive Atlas IIAS (with two Centaur burns) is clearly too small. This cost-and-availability analysis, as well as the exact sizing estimates presented in Section 2.7, lead to the conclusion that the Titan IV/Centaur is the best choice.

2.4.2 Primary Engine

The MARVIN spacecraft uses a modified Pratt & Whitney RL-10 engine for Mars ascent and for the return trajectory to Earth. The same engine provides ΔV after the initial, Earth-departure booster and upper stage have disengaged. Multiple-engine systems can offer some redundancy, but these systems are far more massive [1] than the single RL-10, whose mass is a mere 167.8 kg [5]. Furthermore, the RL-10 has been tested with CH_4, the ISRU fuel for this mission. Thus, as a low-mass, proven technology, the RL-10 is the preferable solution.

The RL-10 is designed for applications requiring up to 20 restarts, with large variations in coast periods between firings. Also, it has a gimbaled nozzle, for controllability, and its thrust can vary to suit the application [5]. This range of thrust, 30,000 N - 98,000 N, is higher than necessary for this mission, but, as Section 2.7 shows, choosing a low thrust level (such as 30,000 N) means that the Mars-Ascent Vehicle experiences only modest acceleration (0.55 - 4.7 g) as it expends fuel, which is likely to be less than the structure withstands upon landing.

Although the restart capabilities of the reliable RL-10 are well documented, exposure to Martian dust may affect the engine's performance after its long stay on the surface. Therefore, mission planners must take steps to ensure the engine's continued health:

- Instrument critical engine components for remote fault detection
- Develop a program of *in situ* test firings
- Implement a means of clearing the fuel injectors, such as forcing a constant, slow leak or infrequent bursts of unignited fuel through the injectors .

Another means of keeping dust out of the engine, lowering the nozzle so that it touches the surface when the engine is inactive, has the disadvantage that a dune would form on the windward side of the nozzle, which may affect the spacecraft's ability to launch. Also, the local surface topography is unknown: the nozzle may be damaged if it strikes a rock as it descends to protect itself from dust. Therefore, the nozzle should not be lowered.

2.4.3 Fuel Consumption

At Earth departure, the tanks can be fueled so that the RL-10 can provide thrust after the booster and upper-stage disengage. The propellant must be methane, the ISRU fuel, because the engine's injection system can be optimized for only one fuel [6]. Fueling the tanks also enables them to be checked for leaks until just before launch. This feature helps to ensure mission success because the fuel tanks are a critical system. Section 2.7 describes these fuel tanks, which are sized specifically for the amount of fuel in the shown sizing iteration.

Fuel consumption (propellant and oxidizer) follows from the rocket equation,

$$\Delta V = g_0 I_{sp} \ln\left(\frac{m_i}{m_f}\right), \tag{1}$$

where g_o is the gravitational acceleration of Earth; I_{sp} is the specific impulse (365 - 395 s for CH_4); m_f is the final, or dry, mass; and m_i is the initial mass (m_f + fuel mass). In terms of fuel mass (m_{prop}), equation (1) becomes

$$m_{prop} = m_f\left(e^{\frac{\Delta V}{I_{sp} g_o}} - 1\right). \tag{2}$$

For this design, the sizing spreadsheet in Section 2.7 shows that an Earth-return ΔV (from the surface) of 6.1 km/s and a dry mass of 610.2 kg requires 2668 kg of fuel. Naturally, these numbers are too precise, but they are meant to serve as a likely example. The spacecraft is designed to manufacture and store an additional 5% of this fuel mass as a margin for error, making the total fuel mass 2801 kg. The fuel-production facility requires 168.1 kg of this mass in H_2 as seed fuel, which makes the dry mass of the Mars-Lander Vehicle 2289 kg. This mass and an Earth-departure ΔV (after escape) of 2.5 km/s require 2384 kg of fuel, which includes the additional 5%, and result in a total launch mass of 4673 kg.

The Titan IV/Centaur can provide enough ΔV to launch 6100 kg into the chosen Earth-Mars trajectory. However, the Centaur is structurally limited to 5220 kg. Despite this limitation, the MARVIN spacecraft is clearly light enough to be launched on a Titan IV/Centaur. Future design modifications can increase the Earth-departure mass by as much as 547 kg without exceeding structural limitations. Section 2.7 suggests some ways in which future work on this project can address this additional capacity.

Because the mass of Mars-departure fuel has a profound impact on the Earth-launch mass (as shown in Section 2.7), the final design must include a precise model of the launch from the Martian surface. To this end, Exploratory Technologies has developed a spreadsheet to model the launch of the Mars-Ascent Vehicle in a fairly realistic way, within the limits of numerical integration techniques. The spreadsheet includes gravity losses and nozzle-expansion variation with atmospheric pressure. This model also includes temperature and density profiles of the Martian atmosphere, as measured on the Viking missions. These profiles provide reasonably accurate estimates of drag, taking into account a Mach-dependent drag coefficient, in which Mach varies with the variable specific heat ratio and temperature of the Martian atmosphere. Future work in this area may focus on identifying an optimal thrust level from this data. A thrust for minimum fuel use can be chosen such that it balances atmospheric drag (associated with high thrust and high velocity) with gravity and pressure losses (associated with low thrust).

2.4.4 Power Sources

Table 2.4.1 shows the power budget for the on-Mars phase of the mission [7], which requires the most power of any phase.

Table 2.4.1. Power Budget (Peak)

System	Peak Power
Fuel Production	2005 W
CNC	100 W
Sample Collection	75 W
TOTAL	2175 W

Proven solar-cell systems can provide sufficient power, but for such high power requirements, their mass and the mass of battery systems may range from 726 kg to 961 kg [8]. This mass is unacceptably high. Furthermore, structural considerations eliminate such arrays from consideration: they may range in area from 87 m^2 to 127 m^2 [8]. RTGs are expensive, massive, and impractical, partly because they must radiate considerable waste heat [9,10].

The Rockwell study [11] suggests a Dynamic Isotope Power System (DIPS), which uses a closed Brayton cycle to extract work from the radioactive decay of plutonium (^{238}Pu). Such a system can provide 2500 W with only 352 kg of mass. The power output exceeds the requirements of the fuel-production facility, but DIPS systems are sized with a step function that makes 2500 W the nearest design point. In fact, this design has not been optimized for spacecraft use. Some components might be made even lighter. Despite the scarcity of nuclear fuels, ^{238}Pu fuel is readily available, and the supply is expected to exceed the demand for the foreseeable future. Finally, the DIPS system being proposed is not only a proven technology, it is highly reliable. Its reliability is greater than 99% over 10 years, and its design life is approximately 30 years. This reliability and longevity, coupled with the extra power it provides, suggests that future work on this project should address the possibility of ensuring that the DIPS from this mission can be used by future Mars missions.

The spacecraft requires considerably less power during the Earth-Mars and Mars-Earth trajectories. Therefore, a small photovoltaic system provides electrical power to the craft during these phases. This system allows the DIPS to be discarded on Mars before liftoff. Communications and control needs are not expected to exceed 100 W. Therefore, a 1.5 m^2 array of gallium arsenide solar cells and a lightweight nickel-hydrogen battery serve these needs. One reason that the battery can be fairly small is that the depth of discharge can be high: the spacecraft is rarely occulted from the sun, so the battery requires only infrequent and minimal recharging. However, these systems are delicate, and one cannot expect the deployed array to remain intact during aerobraking and landing. Retracting the array before landing may prove equally difficult. However, the system is light enough that an additional solar array can be included for use on the Mars-Earth trajectory. For these reasons, the complete power system includes two virtually identical solar arrays and one battery. One array powers the spacecraft before Mars descent, and the other after liftoff from Mars. The DIPS charges the battery before the DIPS is jettisoned at liftoff. The total mass of the system is less than 22 kg.

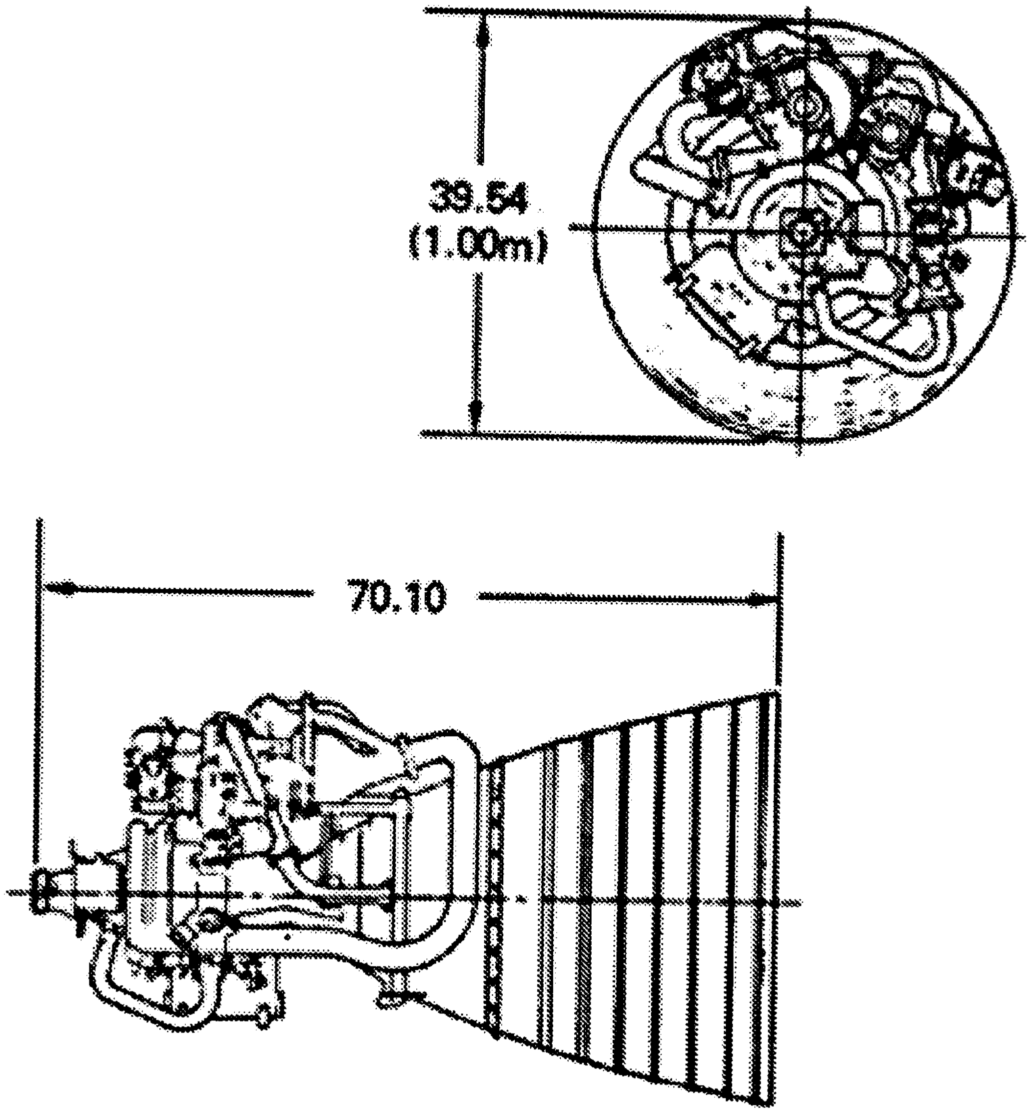


Figure 2.4.1. Pratt & Whitney RL-10 Engine

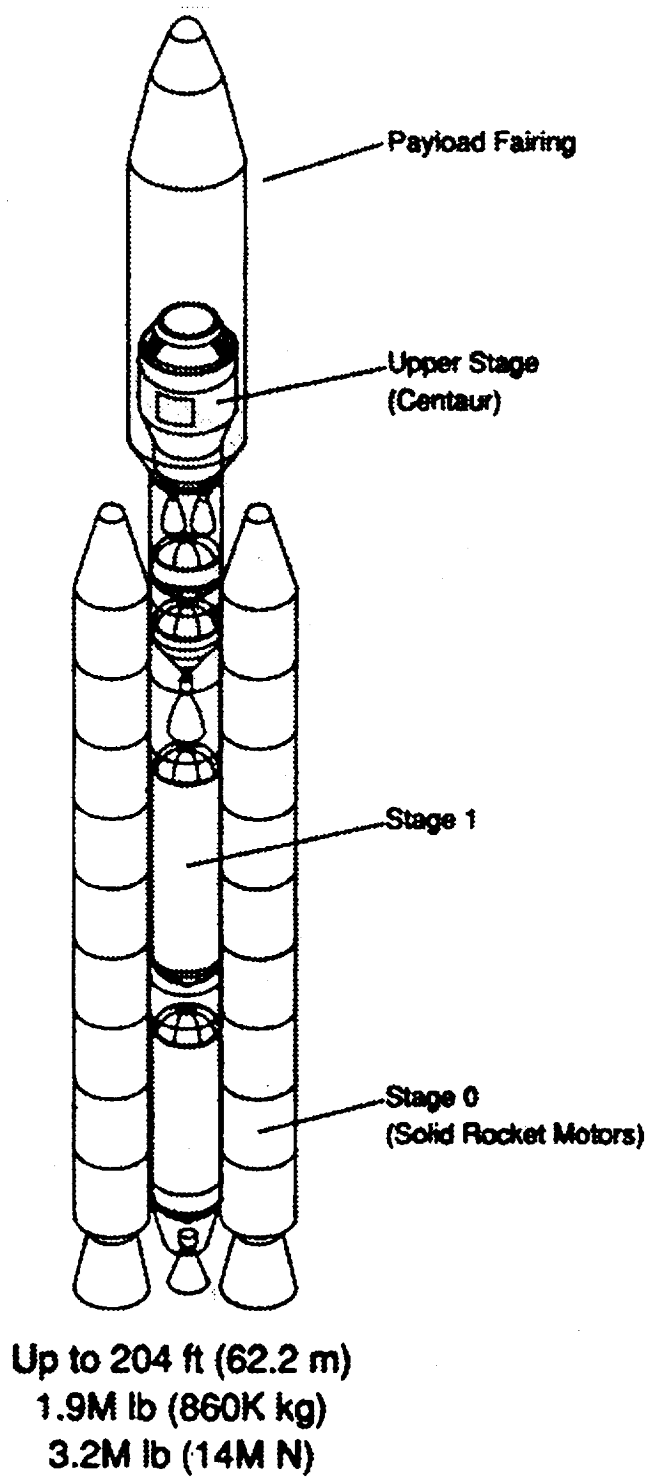


Figure 2.4.2 Titan IV/Centaur

2.5 Control and Communications

This section outlines requirements for communications, data storage, and guidance, navigation and control (GNC). These subjects are grouped together because they are interdependent, electrical systems. Communications and control serves to support and monitor the primary mission phases, such as sample collection and fuel production.

2.5.1 Control

Control for MARVIN is automated, yet reprogrammable from Earth. Onboard sensors determine the position of the spacecraft and its orbit. Actuators are used to alter the orientation of the spacecraft.

2.5.1.1 Automation

Because the round-trip light time from Mars to Earth ranges from 10 to 40 minutes, real-time remote control of the MARVIN mission is impossible. Real-time, onboard navigation is necessary to complete landing, takeoff and attitude-change maneuvers. Autonomy also reduces personnel requirements on Earth [1]. However, this automation cannot be the sole means of control. In case of emergency, Earth-based control must be possible.

Furthermore, handshaking from Earth-based computers provides frequent checks on spacecraft maintenance. In order to react to unforeseen contingencies, any automated process in the MARVIN mission must be reprogrammable. This balance of automation and control from Earth is essential to mission success.

2.5.1.2 Stabilization

Like many previous interplanetary missions, the MARVIN spacecraft is primarily three-axis-stabilized. Unlike spin stabilization, three-axis-stabilization simplifies the structure and the instrumentation of the spacecraft. Instruments like antennas and cameras can be mounted onto the spacecraft frame, rather than be despun on a special platform. Three-axis-stabilization also simplifies the design of the MAV solar array. [2]

However, during orbit transfer thrusting procedures, the spacecraft requires spin stabilization. Spinning the MARVIN vehicle creates a gyroscopic rigidity that reduces torque due to thruster misalignment. [2] After thrusting is completed, the attitude-control system returns the MARVIN spacecraft to three-axis-stabilization.

2.5.1.3 Sensors

The following equipment provides real-time guidance during orbit insertion, landing, takeoff and slewing procedures [3]:

- *Rate-integrated gyroscopes (RIG)*. These gyroscopes can be calibrated with star mappers in order to precisely track the angular orientation of the spacecraft(s) in real-time. They have a calibrated drift rate less than .05 degrees per hour. [4]
- *Accelerometer*. A closed-loop accelerometer oriented in the direction of the thruster integrates thrust to provide ΔV estimates.

2.5.1.4 Actuators

Control actuators, controlled by the onboard computer, correct the attitude of the spacecraft(s). These actuators also perform midcourse correction maneuvers. The MARVIN design uses monopropellant hydrazine control jets. A network of sixteen hydrazine jets provide fine attitude control. Eight jets have a thrust of one newton each, and the other eight jets have thrust of ten newtons each.

The MARVIN mission requires two independent control jet systems. One system is housed on the MLV, and the other on the MAV. Because the MAV is nested within the MLV, they cannot share control-jet systems. Therefore, two separate systems must be used.

2.5.1.5 Health-Monitoring System

Most spacecraft emit a health message along with their other communications. The MARVIN mission also incorporates an independent health-monitoring system. This system communicates the status of MARVIN hardware to both an on-board control computer and an Earth-based computer.

2.5.2 Navigation

Both Earth-based mission control and onboard devices estimate position and determine the orbit. On the spacecraft, sensors provide data for automatic position determination. Simultaneously, Earth-based computers use NASA's Deep Space Network (DSN) to determine the position and orbit of the spacecraft.

As did the previous Viking and Mars Observer Missions, the MARVIN mission uses the DSN for both orbit determination and communication. Periodically, the 70 meter antenna subnetwork uses interferometry to precisely determine the orbit of the MARVIN craft. This subnetwork is strategically located at 120 degree intervals of longitude around the Earth, in

Goldstone, California; Madrid, Spain; and Canberra, Australia.[1] This subnetwork is remotely controlled from Jet Propulsion Laboratory (JPL) in Pasadena, California.

2.5.2.1 Sensors

The following sensors are used for position determination [3]:

- *Star mappers.* Two fixed star mappers can provide vectors to bright stars, such as the star Canopus, tracked by the Voyager mission [1]. Star mappers were chosen over star trackers because star mappers map several stars at once. The error in a star mapper's angular-position reading ranges from ±1 arc minute to ±1 arc second [5].
- *Charge-couple device (CCD) camera.* This device produces digital images of the MARVIN vehicles and their environments. On the surface of Mars it monitors the Martian landscape and sample-collection activity. It also serves as a low-performance, redundant star mapper in orbit [1].
- *High-quality sun sensor.* This device provides a vector to the sun. This information can be used to protect the charge-couple device and star mappers from direct exposure to the sun.

2.5.2.2 Collision-Avoidance System

In order to land safely on the rocky surface of Mars, the MAV needs a collision-avoidance system. Like the Viking lander's, this system can employ radar to determine precise altitude and to avoid obstacles such as boulders. This system can also determine the slope of the terrain as the MAV descends. Design of this system, while important, is beyond the scope and resources of Exploratory Technologies. This important design feature must be investigated before MARVIN is implemented.

2.5.3 Communications

Communications systems transmit scientific and mission-control data back to Earth. The MARVIN mission uses a communications system similar to most of the interplanetary missions before it. Earth-based personnel use these data to monitor the progress of the MARVIN mission. Several types of data are communicated to Earth:

- Fuel-production data
- Sample-collection data
- Power drawn
- Video images
- Hardware-health status message

- Navigation and attitude-sensor measurements
- Onboard position and attitude determination

2.5.3.1 Orbiter Usage

An analysis of communications needs suggests that the MARVIN spacecraft does not need to deploy an orbiter. The success of the Viking mission demonstrates some of the benefits of using an orbiter; however, the Viking orbiter increased the data exchange from the Viking Lander only 50 percent [6]. Such an increase does not merit the increased complexity and cost of an orbiter in this mission.

2.5.3.2 Data Rates and Frequency Selection

The MARVIN mission uses standard DSN frequencies. Most interplanetary missions—from Voyager to Viking to Mars Observer—have used these frequencies and data rates. Table 2.5.1 summarizes the uplink and downlink frequencies and data rates [1,6,7]:

Table 2.5.1. Uplink and Downlink Frequencies and Data Rates

	Frequency	**Data rate (maximum)**
Downlink	S-band	85 kbits/s
Uplink	X-band and S-band	500 bits/s

2.5.3.3 Antenna Sizing and Selection

The MARVIN mission employs one high-gain, 1.5 meter diameter antenna, as well as one low-gain, .1 m diameter antenna. This selection is based on the earlier, successful Viking and Mariner missions. [1,6] However, the structural design of the high-gain antenna involves some complex considerations. The antenna must withstand two launches: one from Earth and one from Mars. A rugged antenna design is therefore necessary. The high-gain antenna is an umbrella structure, deployed when the spacecraft is in three-axis stabilization mode and on the surface of Mars. It is retracted during launch, landing and orbit-transfer phases.

On Earth, the MARVIN mission communicates with the 34 meter DSN subnetwork. This subnetwork is located at the same sites as the 70 meter subnetwork mentioned in Section 2.5.2. Like the 70 meter subnetwork, the 34 meter subnetwork is remotely controlled by JPL. Because its antennae are distributed at intervals of roughly 120 degrees of latitude about the Earth, one antenna is always in view of Mars.

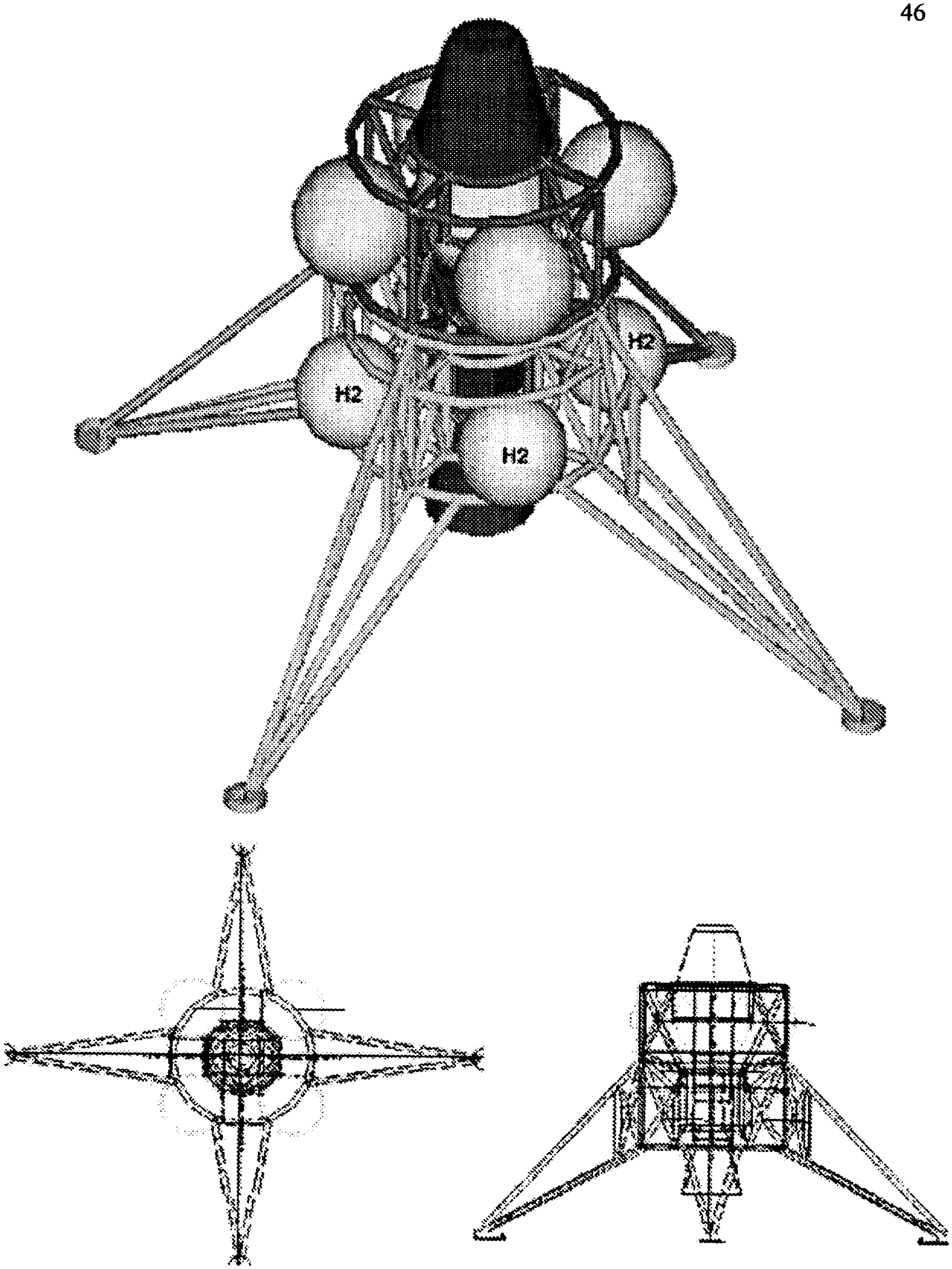
2.6 Structures

The MARVIN spacecraft consists of three major components: the Mars-Lander Vehicle (MLV), the Mars-Ascent Vehicle (MAV), and the Sample-Return Capsule (SRC). The design of the MARVIN spacecraft incorporates the landing gear, the heat shields, the fuel-production assembly, the sample-collection/storage devices, and the propulsion system into the overall design of the spacecraft. A scaled model of the MARVIN spacecraft, showing these systems, has been constructed.

The spacecraft module was designed to accommodate all of the components. It takes into account their masses and sizes, according to the information in Section 2.7. Furthermore, the design is simple, lightweight, reliable, and compact. The following sections present a more detailed discussion of the design and development of the three structural components. The overall dimensions of the final MARVIN spacecraft design are driven by the payload-mass estimates and component sizing.

2.6.1 Mars-Lander Vehicle (MLV)

The MLV is the module that travels from Earth to Mars. Its structure includes the central frame and the landing gear. Payload elements are attached to the structural frame of the MLV, as shown in Figure 2.6.1. This frame supports the MAV, the avionics, the fuel-production plant, the robotic manipulator, and the seed H_2 tank. Furthermore, the design of the MLV takes into consideration that certain components should be near one another to minimize the mass of piping, wiring, etc. The height of the MLV, with the landing gear retracted, is 5.0 m, and its diameter is 3.5 m. This design easily fits within the selected Titan IV booster fairing, which is capable of housing packages 6.0 m high and 4.5 m in diameter. The additional, radial 1.0 m in the fairing has been reserved for an aerobrake/heat shield. With its landing gear fully extended, the MLV is 6.0 m high and 9.0 m wide. This configuration leaves a 1.0 m clearance between the engine nozzle and the surface of Mars.



TOP VIEW **SIDE VIEW**

Figure 2.6.1 Mars-Lander Vehicle

2.6.1.1 MLV Central Frame

The central frame includes three 2.4 m diameter structural rings connected by vertical beams and other cross-members. The MLV consists of two parts. The lower structural frame is 1.0 m high and supports all the seed H_2 tanks, the power facility, and parts of the fuel production facility. The upper structural frame is 0.5 m high and houses part of the fuel production facility, the sample collection device (robotic arm), and the radiators. In addition to supporting the various scientific components, the MLV also acts as a launch pad for the MAV. The MLV central truss frame and the components it houses remain on Mars once the MAV launches off the surface. Furthermore, components supported by the central frame are shielded to reduce heat loading as the spacecraft passes through the Martian atmosphere and to protect them from the Martian environment.

2.6.1.2 MLV Landing Gear

The landing gear is a very important system for the MARVIN spacecraft. Its most important role is to ensure that the lander either touches down in a nearly vertical position or is able to orient the MAV so that it can take off from a vertical configuration. To accomplish these tasks, landing-gear struts incorporate extensible members. Figure 2.6.2 shows the upper struts of the landing gear under consideration. The upper strut is a telescoping member. Initially, the MLV lands with its landing gear in a fixed position. Once the MLV has landed, hydraulic actuators autonomously adjust the extendible struts to ensure the MAV is in a vertical position. These landing-gear legs are not fully designed and should be further investigated.

2.6.2 Mars-Ascent Vehicle

The center of mass of the entire structure must be near the base for Mars landing and launch considerations. Thus, the MAV is nested within the MLV and is attached to the latter's lower support frame. The MAV has an overall height of 5.0 m and diameter of 3.5 m. The MAV is not merely the Mars-launch vehicle; it also includes the SRC. The design of the MAV is presented in Figure 2.6.3. The MAV consists mainly of a support frame, avionics, solar cells, the rocket engine, four O_2 tanks, one CH_4 tank, and the SRC. When the MAV is launched, pyrotechnic bolts separate it from the MLV frame.

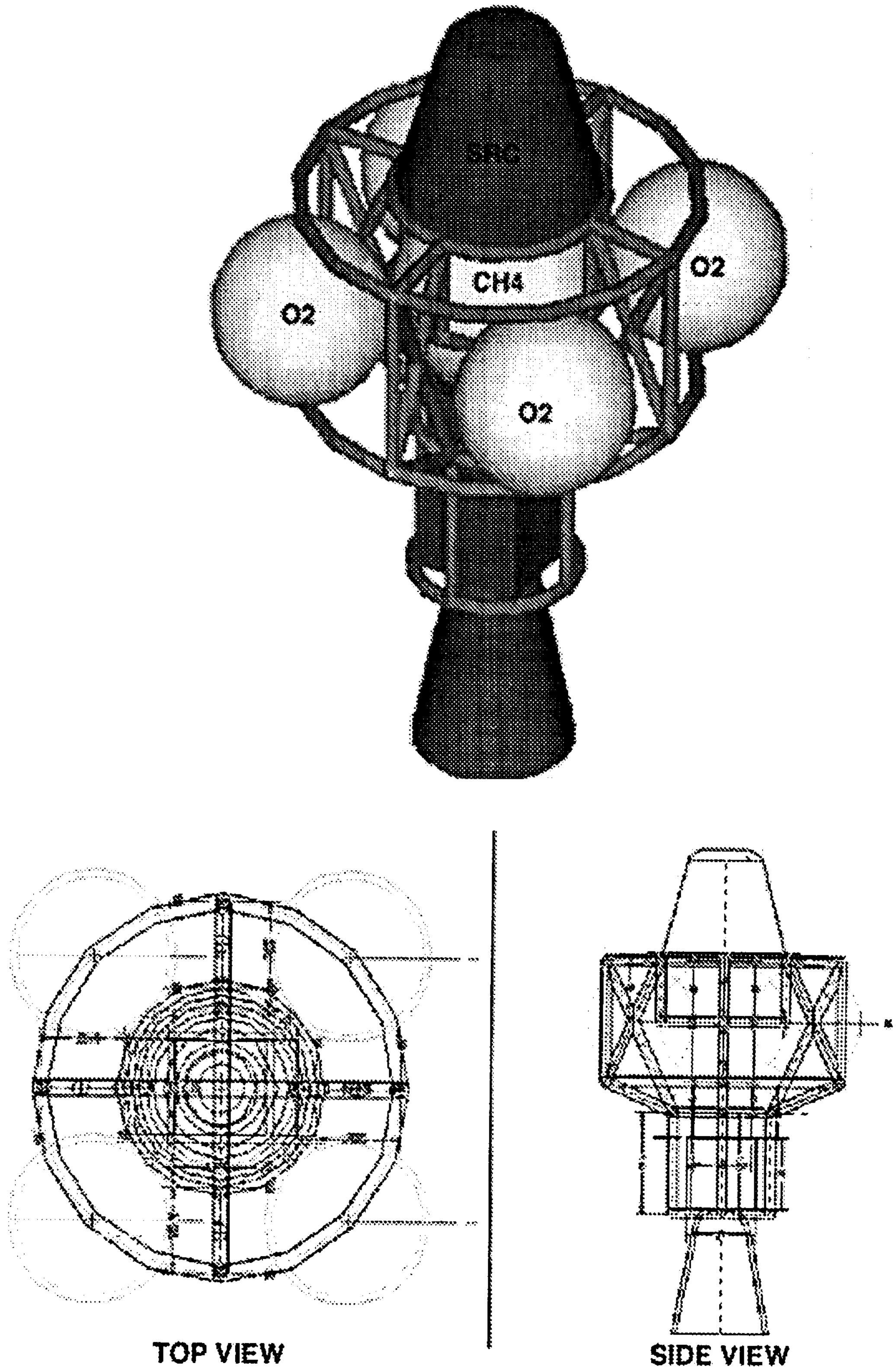


Figure 2.6.3 Mars Ascent Vehicle with Sample-Return Capsule Attachment.

2.6.3 Sample-Return Capsule

The SRC, which separates from the MAV at Earth and is retrieved after atmospheric entry, consists mainly of a re-entry shield, a sample-storage facility, and perhaps parachute and propulsion systems. The SRC is approximately 1 m high and 1 m in diameter. The sample container is placed in such a way that the heat shields protect it from solar radiation. This placement eliminates the need for a refrigeration system. Figure 3 suggests a design for the SRC. Once the SRC has entered the Earth's atmosphere, air snatch or splash down ensues for sample retrieval.

2.6.4 Propellant Tanks

Since the Mars-Earth portion of the spacecraft journey requires the most fuel, the propellant tanks are sized according to the total volume of fuel that is needed for this transfer. Spherical propellant tanks are chosen because it reduces the surface area to volume ratio and the mass to volume ratio. This optimization helps to reduce boil-off and helps to reduce mass. Table 2.6.1 shows the quantity and the sizes of the tanks used in the MARVIN spacecraft.

Table 2.6.1 Fuel Tank Outer Radius

H_2 Tanks (4)	0.491 m
O_2 Tanks (4)	0.490 m
CH_4 Tank (1)	0.541 m

2.6.5 Materials

Aluminum-Lithium 2090-T83 was chosen as the material for all truss members because this material exhibits desirable properties: high stiffness-to-density ratio, high ductility, excellent workability, non-magnetism, and high corrosion resistance. Aluminum-Lithium is better than conventional Aluminum 2024 because these materials can offer a 30% reduction in weight and can have a tensile strength above 100 ksi.

Titanium is chosen for all fittings between structural components. This material exhibits a substantially greater yield strength than aluminum alloys, which makes it an advantageous material at joints, where stress concentrations are likely to appear. In addition, titanium is selected for the inner wall of the O_2 and CH_4 propellant tanks. However, titanium reacts with H_2 and becomes brittle. Thus, steel is chosen for the H_2 propellant tanks.

Heat shields can be made from a combination of HTP-6 tiles and AVCO-5026 ablator. These tiles are new-generation Shuttle tiles, and the ablative material is basically the same material that Apollo used.

2.7 Sizing

2.7.1 Methods

Estimating the mass of the three configurations of the MARVIN spacecraft is, of course, an iterative process. For this reason, exploratory technologies has designed an iteratively solved spreadsheet to calculate the mass of the MLV and that of the MAV. The mass of the SRC is virtually independent of the sizing of other equipment. However, the cost and, in fact, the feasibility of this mission are very sensitive to some parameters. Analysis of the mission has shown that the ΔV required for Mars departure and Earth return is such a parameter. For example, a 20% increase in this ΔV more than doubles the required fuel-production rate; and, because the fuel-production plant is sized for a certain production rate, the Earth-launch mass is ultimately greater.

Cells in the spreadsheet include component mass as functions of a few quantities, such as fuel-production rate, DV and H_2 mass. Fuel tanks are also sized with some care: since the fuel tanks are nearly empty when the spacecraft lands on Mars, they need not be as strong as the H_2 tank, which is full at landing and thus more likely to burst. Structural mass (along with cabling, shielding, etc.) is estimated from an empirical formula based on other designs, coupled with information from the structural analysis discussed in Section 2.6.

This spreadsheet has enabled Exploratory Technologies to choose trajectories that use the fuel tanks most efficiently: ideally, the tanks should be sized for Mars Ascent to minimize that mass, and they are. However, since the engine also operates between Earth and Mars, the trajectory and booster design have been chosen such that the capacity of these tanks is not wasted. Furthermore, analysis of the spreadsheet can identify those systems in which weight reduction offers the greatest overall benefits.

2.7.2 Design Perturbations

One way to perform the analysis suggested above is to perturb some element of the design from its nominal state and examine the overall effects of these perturbations. Such an analysis was undertaken for wet mass because this parameter is an effective measure of the cost of the mission. The analysis identifies two important types of component mass:

- Mass jettisoned before takeoff
- Mass returned to Earth

Each one contributes differently to the overall mass of MLV and the MAV. The mass returned to Earth affects the overall mass the most. Figure 2.7.1 shows that a 1% increase in this parameter (from the nominal value, shown as 0.000 in the figure) increases the mass of the MAV by 6% and that of the MLV (which includes the mass of the MAV) by 9%. Clearly, the SRC must be sized with some care.

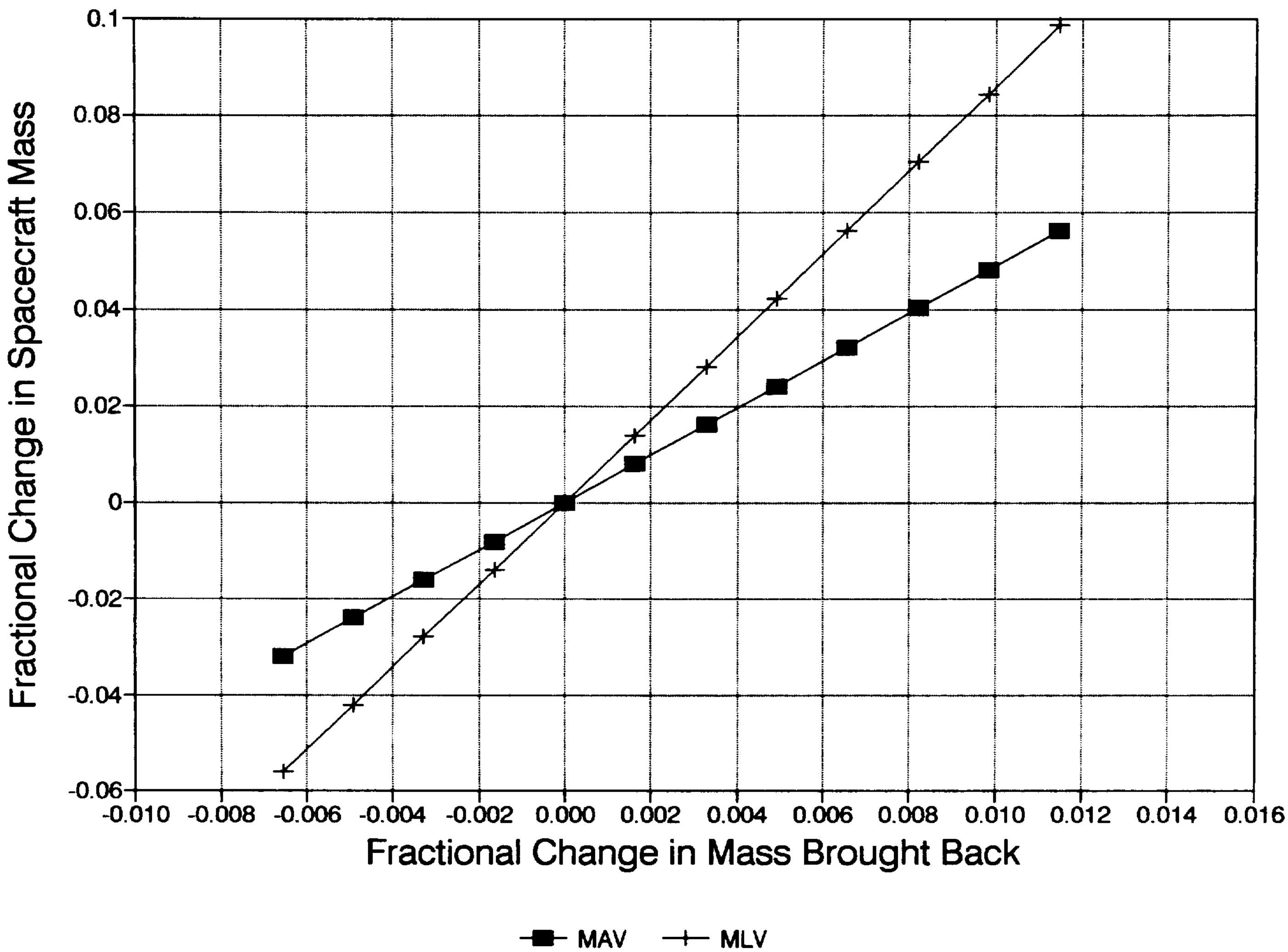


Figure 2.7.1 Fractional Change in Overall Spacecraft Mass as a Result of a Fractional Change in Mass Returned to Earth.

The mass brought to Mars but not returned, such as the 352 kg DIPS, also affects the MLV mass in a substantial way. Interestingly, it also affects the MAV mass, partly because it affects structural mass (a function of fuel mass). Figure 2.7.2 shows that a 1% increase in this parameter increases the mass of the MAV by 2% and that of the MLV by 4%. Of course, mass left on Mars is not wasted; it may play a role in future missions, providing infrastructure.

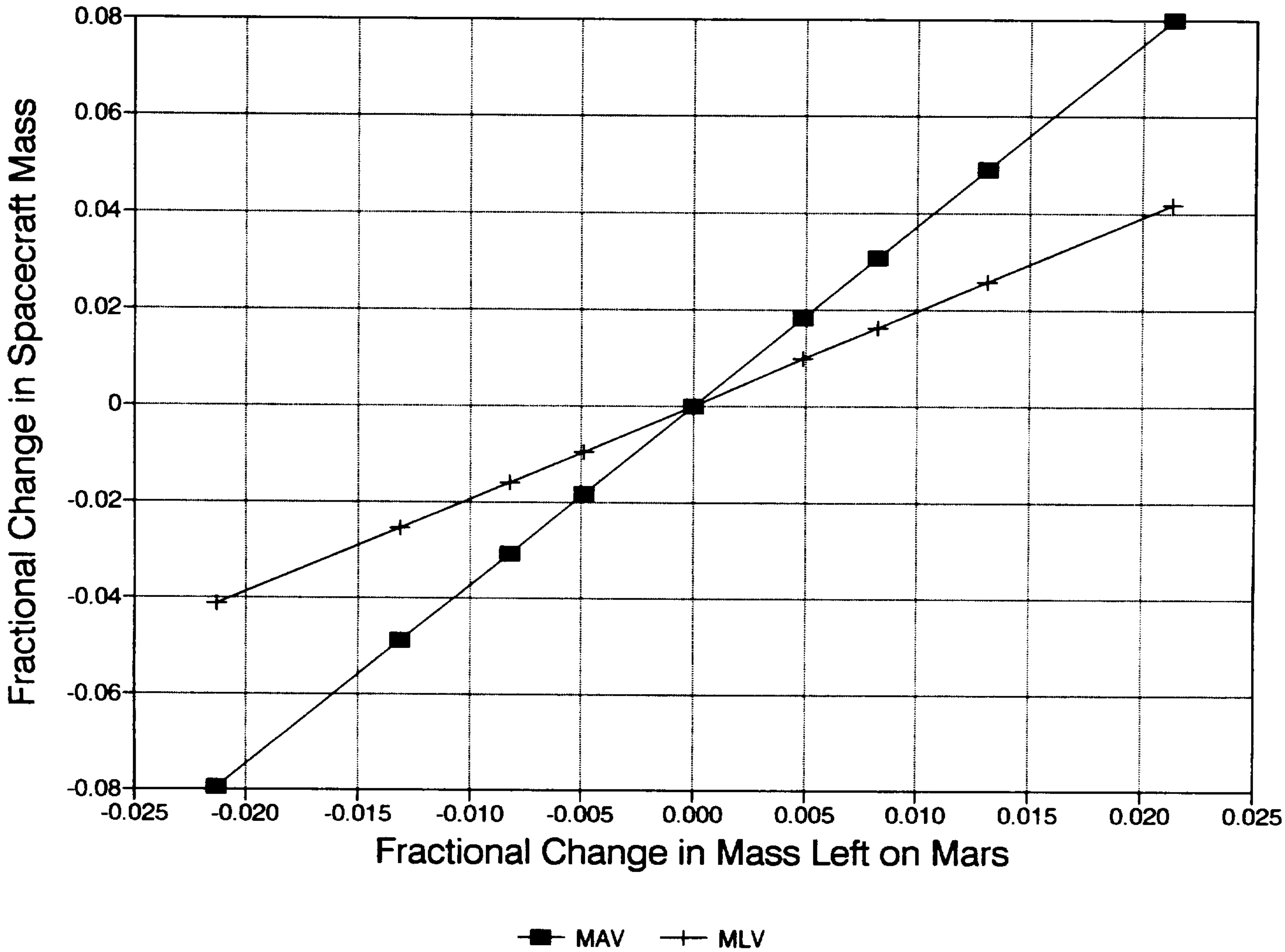


Figure 2.7.2 Fractional Change in Overall Spacecraft Mass as a Result of a Fractional Change in Mass Jettisoned at Mars.

ISRU makes the MARVIN project more cost-efficient than it would otherwise be. However, the I_{sp} of the ISRU fuel is comparatively low, and the fuel mass is substantial. If the Isp were even a little higher, the exponential nature of fuel use would drop significantly. Figure 2.7.3 illustrates this point. It shows that, for example, a 5% increase in I_{sp} reduces the mass of the MLV by 8% and that of the MAV by 6%.

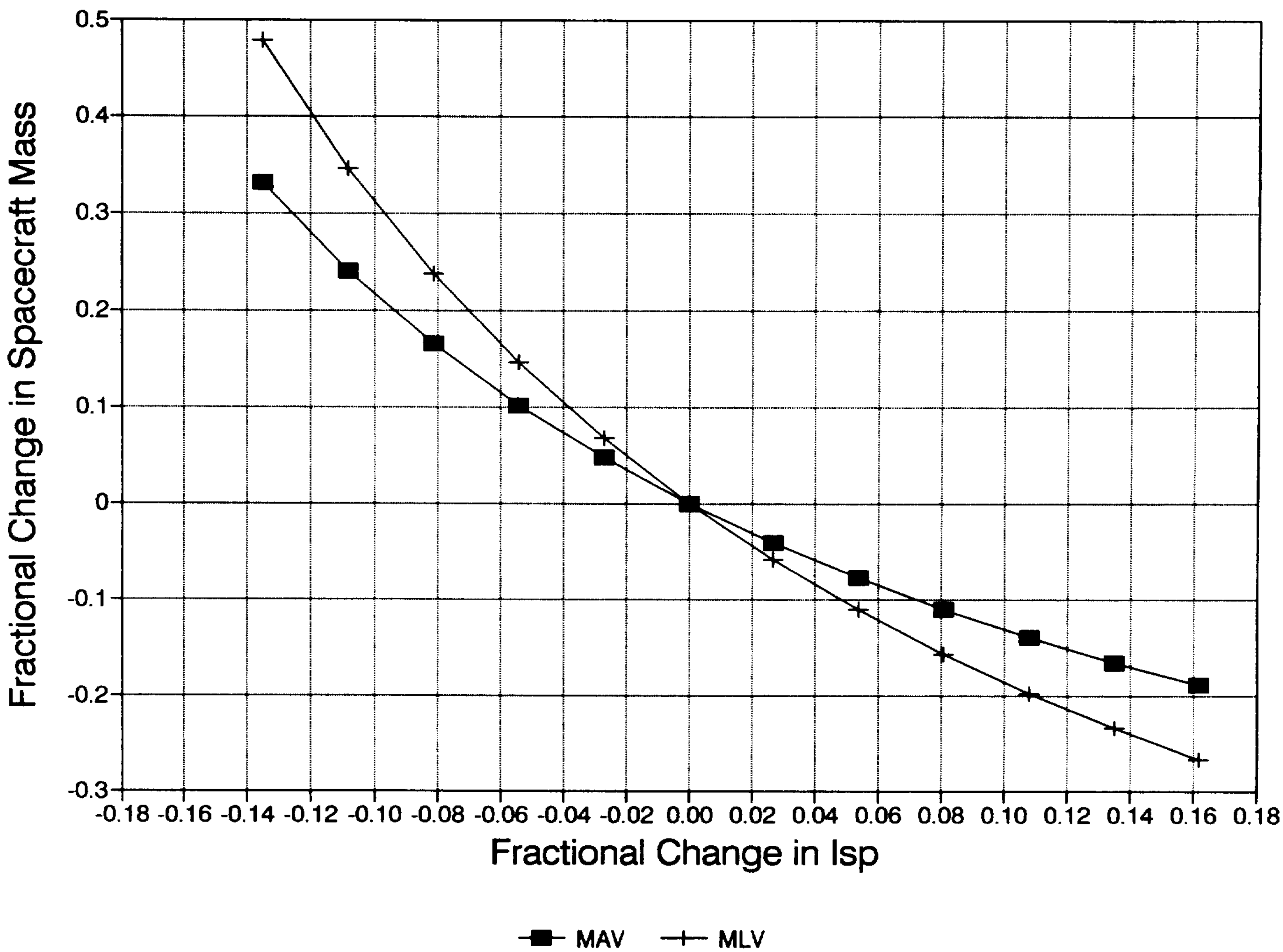


Figure 2.7.3 Fractional Change in Overall Spacecraft Mass as a Result of a Fractional Change in I_{sp}.

Again, the fuel mass, which grows exponentially with dry mass, contributes the most to the overall wet mass of the spacecraft. In turn, fuel mass is an explicit function of the ΔV required for orbital maneuvers. As an example, Figures 2.7.4 and 2.7.5 shows the effect on spacecraft mass of the ΔV for insertion into the Mars-Earth orbit and the Earth-Mars trajectory, respectively. In the Figure 2.7.4, the function is almost linear, and, as one would expect, this ΔV has virtually no affect on the mass of the MLV for small perturbations of the design. However, if the perturbations were larger, the graph would show that MLV mass increases because it must carry a larger fuel-production system (and perhaps a larger power supply) in addition to the structural mass and the associated fuel and tank masses to accommodate it. Figure 2.7.5 shows that the ΔV for Earth return has an even greater effect on the mass of the configurations.

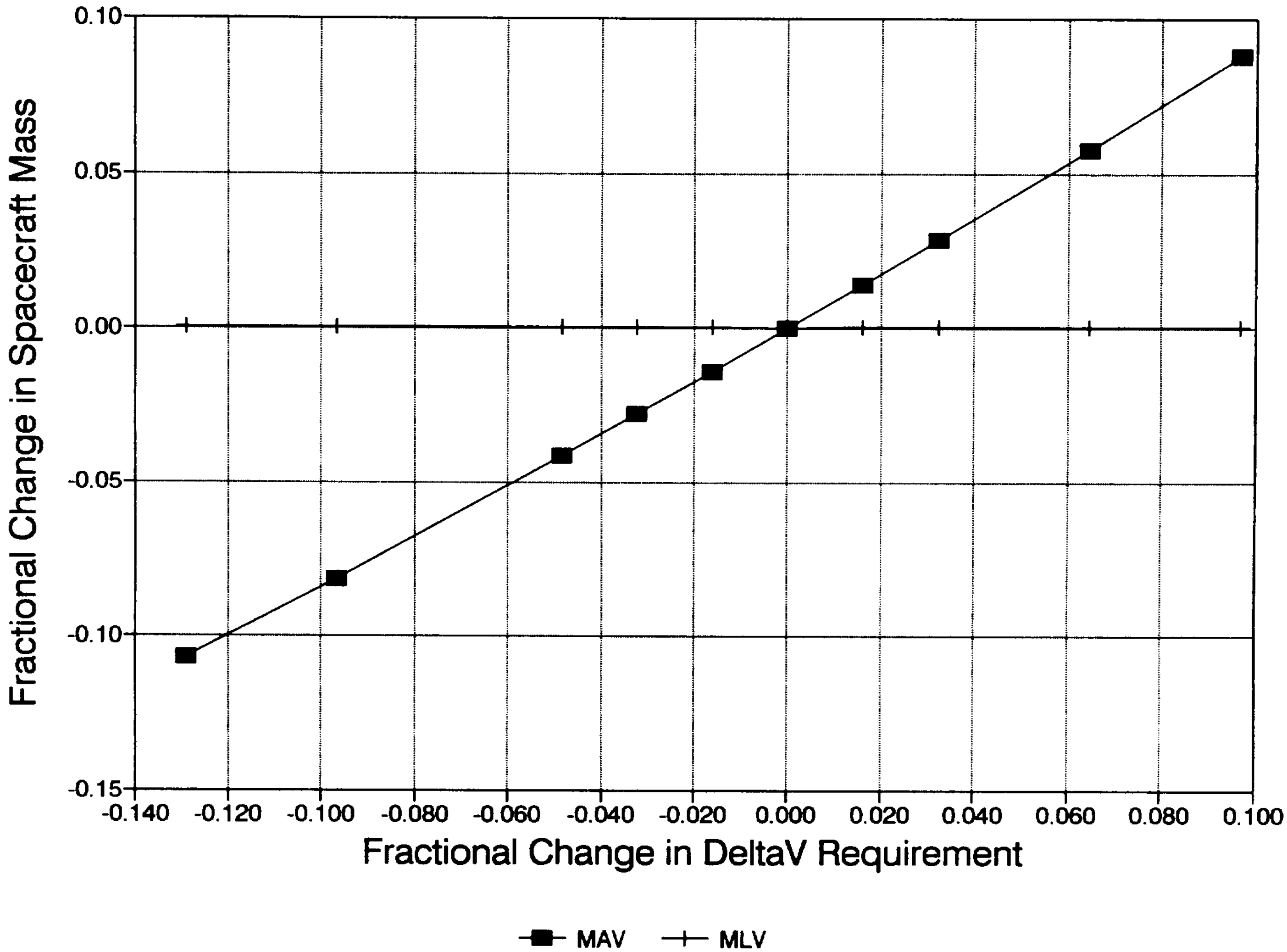


Figure 2.7.4. Fractional Change in Overall Spacecraft Mass as a Result of a Fractional Change in Mars-Earth ΔV.

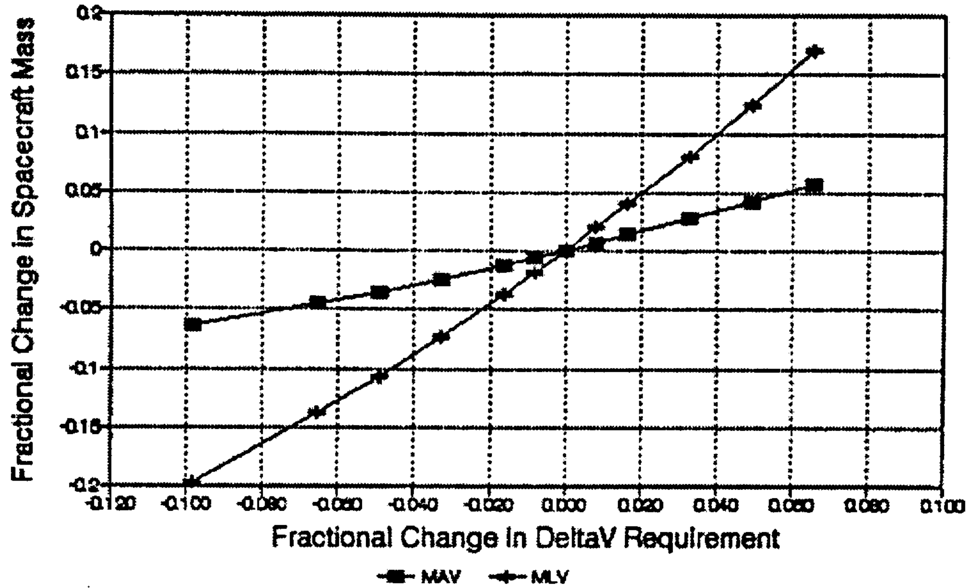


Figure 2.7.5. Fractional Change in Overall Spacecraft Mass as a Result of a Fractional Change in Earth-Mars ΔV.

Although this design-perturbation analysis concerns small variations from the nominal values shown in the appendix, other ISRU missions with sample return can use these results. They ought to be valid for small perturbations of any similar design. Furthermore, with is information, more effective trade studies can be performed during the initial stages of planning a mission like MARVIN, in which the fuel, trajectory and sample-return subsystems are chosen early in the design process.

2.7.3 Expansion Possibilities

The design that this report describes includes some margin for safety. For example, in addition to the redundancy options presented in Section 1.0, the spacecraft carries 5% extra fuel; its fuel tanks are sized with appropriate bursting safety factors; and 20% extra hydrogen is included for boil-off. However, even with these and other provisions, the mass of the spacecraft is less than the maximum capacity of the Titan IV/Centaur. Exploratory Technologies presents some suggestions for enhancing the mission in an effort to take advantage of the 547 kg margin. Section 2.7.2 provides data that shows what percent increase in the different types of components can make up the difference. This section provides some specific suggestions.

First, if the design presented in this report is incomplete in some way—for example, if the structural mass has been underestimated—the 547 kg may enable the problem to be resolved without altering the basic design. Also along these lines, if aerobraking requires more fuel mass or if the aerobrake/heat shields prove to be heavier than expected, the basic design may still be valid. If the mass increases beyond 547 kg, a different booster/upper stage must be chosen, although most other aspects of the mission would be unaffected.

Several other options can improve redundancy. One of the simplest of these options is to increase the fuel-production rate, which must be accompanied by an increase in the mass of the fuel-production system and associated systems. This increase would follow the trend suggested in Figure 2.7.2. An higher fuel-production rate would provide some safety in the event that a critical system were to fail after some time on Mars. The spacecraft might already have produced enough fuel to return the samples.

Another option is to bring additional fuel to Mars, which the spacecraft could use on either the Earth-Mars trajectory or the Mars-Earth Trajectory. The fuel might even be left on Mars for future missions to use. Finally, any system (even the DIPS) might be duplicated for greater redundancy.

Up to 547 kg of scientific equipment might instead be built into the spacecraft so that it could analyze samples on Mars. If such equipment is used, the mission would not be a complete failure if no samples are returned. Scientific equipment might also be used for other purposes, such as astronomy (on-orbit). Furthermore, this equipment might include life-science or materials-science experiments in preparation for a manned Mars mission. Video cameras, in addition to those MARVIN already incorporated, could be included to provide public-relations material. All of these suggestions deserve careful scrutiny, but the flexible design of the MARVIN mission makes them worth considering.

3.0 Management Costs

This section compares estimated costs to actual costs incurred in Exploratory Technologies' research of Project MARVIN and its management of the project. This section discusses neither the cost of the technical design nor that of its components.

3.1 Personnel Costs

The staff of Exploratory Technologies assigned to Project MARVIN are listed in Table 3.1.1. Their positions on the project and their hourly work rates are shown.

Table 3.1.1. Staff, Positions, and Hourly Rates

Name	Position	Hourly Rate ($)
Ursula Callaway	Program Manager / Engineer	45
Julia Shane Oltman	Administrative Officer / Engineer	40
Ben Harris	Engineer	38
Greg Merritt	Engineer	38
Michael Parker	Engineer	38
Mason Peck	Engineer	38
Julian Turner	Engineer	38
Norton Wong	Engineer	38
Anthony Perez	Co-Op/Intern	15

Throughout the project each person submitted a time card that showed how many hours he or she worked on a particular team. Each team submitted a weekly timecard on Fridays. The hours spent on team tasks were added to group-meeting hours and presentation or technical writing hours. The Administrative Officer recorded the total hours each group member worked on Project MARVIN each week.

In order to provide a cost estimate for the proposal, the Administrative Officer developed a method for calculating the estimated personnel costs. This method was based on an average of 10 working hours per week, per person, for a normal week and an average of 15 working hours per week, per person, for weeks during which a presentation occurred. Of the fifteen weeks spent on Project MARVIN, four were presentation weeks and eleven were normal weeks.

Exploratory Technologies had expected that each team member would have worked 170 hours on Project MARVIN by the end of the project. The actual hours that each team member has accumulated are shown in Table 3.1.2, as is a comparison of the budgeted and the actual personnel costs.

Table 3.1.2. Hours, Budgeted Personnel Costs and Actual Personnel Costs

Name	# of hours	Personnel Costs—Budgeted	Personnel Costs—Actual
Callaway	191	7,650	8,595
Harris	135	6,460	5,130
Merritt	165	6,460	6,270
Oltman	153	6,800	6,120
Parker	187	6,460	7,106
Peck	231	6,460	8,796
Perez	44	2,550	660
Turner	163	6,460	6,194
Wong	207	6,460	7,866
Subtotal	1,476	55,760	56,737

Figure 3.1.2. displays the estimated and actual accumulated hours through each week of the project. As the figure shows, Exploratory Technologies did not meet the expected accumulated hours of 1530: the actual accumulated hours totaled only 1476.

Exploratory Technologies' original budget neglected to include an estimate for consulting costs. Consulting costs have now been included as additional personnel costs. All consultants were compensated for time spent providing information to the engineers working on this project. Each consultant received a base rate of $80 an hour. The consulting fees totaled $880. Table 3.1.3 lists the consultants that were contacted, the hours they spent with Project MARVIN engineers, and their total fees.

Table 3.1.3. Consultants Contacted for Project MARVIN and Their Fees

Consultant	Number of Hours	Total Fee ($)
Adam Bruckner	1	80
John Bosak	1	80
David Garza	1	80
Dr. Gooding	1	80
John Jones	1	80
David Kaplan	1	80

Elfego Piñon	1	80
Thomas Sullivan	1	80
Joe Sovie	1	80
Dr. Delbert Tesar	2	160
Subtotal	11	880

3.2 Computer Costs

The budgeted computer costs of computer time and supplies for Project MARVIN are shown in Table 3.2.1

Table 3.2.1. Budgeted Costs of Computer Time and Supplies

Computer	Cost/Week per Employee ($)	Total Usage Cost ($)	Supply Cost ($)	Total Cost ($)
Macintosh	20	2,700	50	2,750
IBM PC	5	675	10	685
UNIX (Sun) Mainframes	5	675	5	680
Subtotal	30	4,050	65	4,115

The actual computer costs were slightly lower than the budgeted computer costs, with a total of $3,980, because the usage time was less than expected.

3.3 Material Costs

Table 3.3.1 provides a list of estimated, total material costs incurred during Project MARVIN and the actual material expenses for the project.

Table 3.3.1 Estimated Total Cost of Materials and Actual Material Expenses

Item	Estimated Expense ($)	Actual Expense ($)
Photocopies @ $0.08 ea.	250	240
Design Model	35	52
Project Poster	15	5

Project Notebooks	20	20
Long Distance Telephone Calls	120	115
Travel	150	135
Miscellaneous Supplies	25	24
Subtotal	615	591

3.4 Total Project Cost

Exploratory Technologies initially estimated that the total cost of Project MARVIN, including personnel, computer, and material costs would be **$60,490**. As the information in Table 3.4.1 shows, Exploratory Technologies went slightly over budget with an actual project cost of **$62,188**.

Table 3.4.1. Actual Expenses of Project MARVIN

Type of Expense	**Budgeted Cost ($)**	**Actual Cost ($)**
Personnel Costs	55,760	57,618
Computer Costs	4,115	3,980
Material Costs	615	591
Total	60,490	62,189

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Glossary of Terms

Accelerometer: a sensor that measures acceleration.

Charge Couple Device (CCD): a digital camera that uses photons to charge cells in a super-cooled semiconductor array. The charge of each cell determines the amount of light on that cell. This array forms a digital image.

Deep Space Network (DSN): a communications network used by NASA for communications and orbit determination for interplanetary craft.

Delta V (ΔV): a change in velocity of a spacecraft during orbital maneuvers.

Downlink: communication from a spacecraft to ground control. See uplink.

Electrolyzer: a device that produces chemical changes, such as the dissociation of water into hydrogen and oxygen, by passage of an electrical current through an electrolyte.

Fortran (FORTRAN): a symbolic, algebraic, and logical language for programming a computer (*for*mula *tran*slation).

Gyroscope: a wheel or disc mounted to spin rapidly about an axis and also free to rotate about one or both of two axes perpendicular to each other and to the axis of spin so that a rotation of one of the two mutually perpendicular axes results from a torque applied to the other when the wheel is spinning and so that the entire apparatus opposes torques that would change the spin axis.

Hohmann Transfer: a minimum-energy, two-body orbital transfer technique, in which the satellite transfers from an initial circular orbit to the periapsis of an elliptical orbit, whose apoapsis distance is equal to the radius of the desired final, circular orbit. The second burn takes place at the apoapsis of the intermediate, elliptical orbit.

Hydrazine: A propellant. MARVIN control jets thermally decompose liquid, anhydrous hydrazine (N2H4) into nitrogen, ammonia, and hydrogen to produce thrust. The specific impulse of hydrazine ranges from 200 seconds to 250 seconds. Liquid hydrazine has the same density as water.

Hyperbolic Excess Velocity Squared (C3): the square of the velocity (with respect to some body) of an object at quasi-infinite distance from the body. If an object is traveling at exactly escape velocity, C3=0. If the object is traveling any faster, C3>0. C3 = twice the launch or arrival energy at a planet (per unit mass) and equals to the square of *Vinfinity*.

Interferometry: use of two or more radio telescopes to amplify a radio signal. Basically, the precision of a radio signal is proportional to antenna diameter. By using two antennae, one can simulate a "virtual" antenna the size of the distance between two original antennae. This is an important key to the incredible orbit determination capabilities of the NASA Deep Space Netwrok, which has antenna networks distributed around the globe.

Isp (Specific Impulse): the impulse per unit mass (of the spacecraft) that an engine imparts to a rocket.

Lambert Targeting: a method of trajectory analysis based on Lambert's Theorem, which states that a given trajectory is defined by only two position vectors and a time of flight.

LEO: low Earth orbit.

Light Minute: the distance light travels in a vacuum in 60 solar seconds.

Mars-Ascent Vehicle (MAV): the vehicle, which consists of parts of the MLV, that travels from the Martian surface to LEO.

Mars-Lander Vehicle (MLV): the module that travels from Earth to Mars. It consists of a structural frame to which the payload elements are attached.

Moment of Inertia: the ratio of the torque applied to a rigid body free to rotate about a given axis to the angular acceleration thus produced about that axis; a measure of the body's resistance to angular acceleration.

Reverse Water-Gas Shift (RWGS): Combines CO_2 and H_2 to produce CO and H_2O.

Round-trip light time: total time for a command to be sent to a remote spacecraft. This not only includes the time to reach the spacecraft, but also the time it takes for a verification message to be sent to ground control.

S-band: radio communication frequency of 2.3 GHz.

Sabatier Reactor: a reactor that uses a catalyst to convert hydrogen and carbon dioxide to methane and water.

Sample-Return Capsule (SRC): the structure that separates from the MAV and enters Earth's atmosphere.

Slewing: spacecraft reorientation.

Spin stabilization: stability mode in which a spacecraft rotates about an axis of minimum or maximum rotational inertia. This spinning creates a gyroscopic rigidity which reduces the effect of disturbing torques.

Star Mapper: a sensor that views a small region of the sky, providing vectors to the brightest stars. It outputs the relative positions of stars to Earth and to the onboard processor.

Sun Sensor: a sensor that provides a vector to the center of the sun's disk. This data is used to protect sun-sensitive equipment.

Three-axis-stabilized: stability condition requiring spacecraft attitude is constant with respect to the stars.

Uplink: communication to a spacecraft from ground control. See downlink.

V infinity: the excess velocity of a spacecraft in a hyperbolic orbit relative to a planet.

X-band: radio communication frequency of 8.4 GHz.

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Appendix A
Sizing Spreadsheet

Mass Budget (summary)

Mass (kg)	
GNC	200.00
Sample Collection	75.10
Fuel Production (from UW report & Ursula)	372.93
Propulsion	298.43
Power Systems	353.57
Subsystems TOTAL	1300.03

Fuel Use, Wet & Dry Masses

MAV

Total Excluding Structures		1300.03
Structural Mass from empirical formula		254.15
Mass of cabling, etc. (Wertz & Larson)		25.41
Landing Gear (based on Lunar Polar Coring Lander)		272.50
Parachute (linearly, from U. Wash. Report)		127.19
(MLV shield area, m ^ 2)	40	
Heat Shield = 1% 1522K, 30% 900-1000K, 69% 640-900K		205.12
SRC Structural Weight		40.00
(SRC shield area, m ^ 2)	6	
Heat Shield = 10% 1522 K, 90% 900 - 1000 K		64.96
(from UT Subsystems Guide)		
MLV Total (dry mass)		2289.36
Total Earth - Mars Delta-V (m/s):	2500.00	
Minimum Fuel Use Earth-Mars		2270.61
+5% for control, adjustments		113.53
Total Earth - Mars Fuel		2384.14
MLV Total (wet mass)		4673.50

MAV

Dry MAV Mass = MLV - fuel prod - all sample coll. -		610.21
landing gear - parachute - h2 - DIPS - MLVgnc - shield		
extra h2 tank + 5 kg samples - 70% structure + containers		
Total Mars - Earth Delta-V (m/s):	6100.00	
Fuel Use Mars-Earth		2668.03
+5% for control, adjustments		133.40
Total Mars - Earth Fuel		2801.43
MAV Total (wet mass)		3411.64

Phase with Most Fuel:

	2801.43
max o2	1867.62
max ch4	933.81

Launch Environment:

T= 30000

* Isp= 370

Min G	0.55
Max G	4.67
Kg fuel/day	5.19
# Days:	540.00