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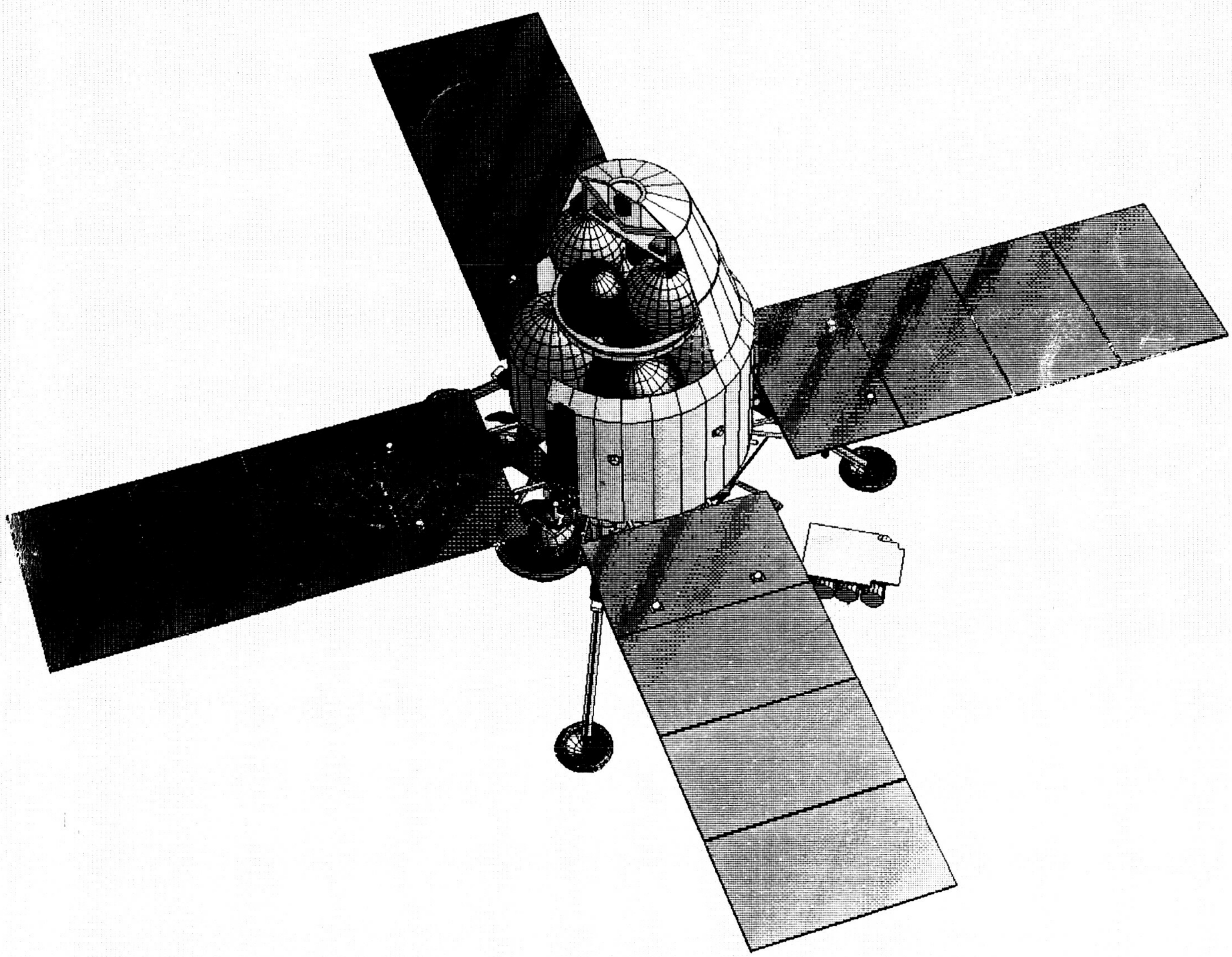
Contract NAS 9-19359

Final Report **March 31, 1995**

MARS SAMPLE RETURN MISSION UTILIZING IN-SITU PROPELLANT PRODUCTION



LOCKHEED MARTIN

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Final Report

March 31, 1995

MARS SAMPLE RETURN MISSION UTILIZING IN-SITU PROPELLANT PRODUCTION

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FOREWORD

This report, submitted to the National Aeronautics and Space Administration, Johnson Space Center, was prepared under JSC contract NAS9-19359, in response to the Mars Sample Return Mission Utilizing In-Situ Propellant Production Study. It summarizes the work that was performed, the results that were achieved and the major conclusions.

ABSTRACT

This report presents the results of a study examining the potential of in-situ propellant production (ISPP) on Mars to aid in achieving a low cost Mars Sample Return (MSR) mission. Two versions of such a mission were examined; a baseline version employing a dual string spacecraft and a light weight version employing single string architecture with selective redundancy. Both systems employed light weight avionics currently being developed by Lockheed Martin, Jet Propulsion Lab and elsewhere in the aerospace community, both used a new concept for a simple, light weight parachuteless sample return capsule, both used a slightly modified version of the Mars Surveyor lander currently under development at Lockheed Martin for flight in 1998, and both used a combination of the Sabatier-electrolysis and reverse water gas shift ISPP systems to produce methane/oxygen propellant on Mars by combining a small quantity of imported hydrogen with the Martian CO_2 atmosphere. It was found that the baseline mission could be launched on a Delta 7925 and return a 0.5 kg sample with 85% mission launch margin, over and beyond subsystem allocated contingency masses . The lightweight version could be launched on a Mid-Lite vehicle and return a 0.25 kg sample with 13% launch margin, over and above subsystem contingency mass allocations. A preliminary cost estimate was generated and it was found that the baseline mission could be flown for a total cost of $302 million, while the lightweight version could be flown for a cost of $244 million if launched on US launch vehicles. If a Russian Molniya is used to launch the mission instead, total cost of the baseline mission is estimated at $259 million while the lightweight version can be flown for $225 million.

Low Cost Mars Sample Return with In-Situ Propellant Production

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Final Report March 31, 1995

Introduction

The following is a report on a study to evaluate low cost options for a Mars Sample Return Mission utilizing in-situ propellant production. The study was carried out at Lockheed Martin in Denver, Colorado during February and March 1995. The study was funded by the Planetary Missions and Materials Branch of NASA's Johnson Space Center. David Kaplan served as the JSC Program Manager, Steve Price as the Lockheed Martin Program Manager, and Robert Zubrin served as the Lockheed Martin Technical Lead.

Assumptions and Requirements

The primary requirement for the current study was to define a minimum cost Mars Sample Return mission. In order to determine the lowest possible cost option, it was stated at the outset of the study that all requirements for system redundancy, spacecraft parts type, testing procedures, sample container environmental control, and avoidance of back contamination that have been assumed in many other MSR mission studies performed in the past would be dropped. That said, a certain amount of common sense compromise was introduced into the mission design philosophy to take measures that would significantly reduce risk and increase science return if the cost and mass impact of taking such measures would be small. Similarly, while there was no design requirement to deal with back contamination, where possible design features were adopted to minimize such a possibility. The specific objectives stated in the original statement of work were as follows:

1. The primary choice for the lander will be the Mars Surveyor lander.

2. The primary launch vehicle considered shall be the Med Lite. If that proves inadequate, other launch vehicles that may be considered include the Delta 7925 and Russian launch vehicles.

3. The mission design shall not be required to conform to standard requirements for the avoidance of back contamination.

4. The sample size shall be 0.5 kg.

5. Spacecraft design shall be single string, with selective redundancy. Parts used shall be mil-spec or better. Testing programs shall be minimized.

6. It will be assumed that an orbiter exists in Mars orbit to relay communications.

7. There shall be no requirement for a landed science payload other than that needed to obtain a grab sample.

8. Mission options shall be evaluated on the basis of cost and risk.

Other assumptions used in the course of the study were that the mission would be solar powered, that the average solar incidence on Mars during the surface stay of 500 sols is 600 W/m2, and that the delta-V to go from the Mars surface onto Trans-Earth injection is 6400 m/s.

Mission Profile

The mission profile adopted in this study is shown is fig. 1, A single launch vehicle, either a Med-Lite, a Molniya, or a Delta 7925 is used to send a single spacecraft directly from Earth to the Martian surface, refueling there during a year and a half surface stay, and then returning directly from the Martian surface to Earth. The refueling on Mars is accomplished by combining a small quantity of hydrogen transported from Earth with CO_2 from the Martian atmosphere to produce methane/oxygen bipropellant. In the process recommended, a combined Sabatier/Reverse Water Gas Shift reactor run in a loop with a water electrolysis unit, each kilogram of hydrogen transported from Earth will produce 18 kilograms of methane/oxygen bipropellant on Mars. The mass leverage provided by this in-situ propellant production system allows the mission to be accomplished without any Mars or Earth orbit rendezvous maneuvers. The elimination of the orbiter and Mars orbit rendezvous maneuver offers the potential to significantly reduce both the cost and risk of the sample return mission, since only one spacecraft need be developed, and only one spacecraft must operate successfully in order for the sample to be retrieved.

Type I conjunction class trajectories are used on both the outbound and return legs of the mission. Flight times vary depending upon the year chosen for the mission, but a typical profile would be 200 days outbound, 600 days on the surface, and 200 days on the return leg, for a total mission duration of 1000 days (2.7 years).

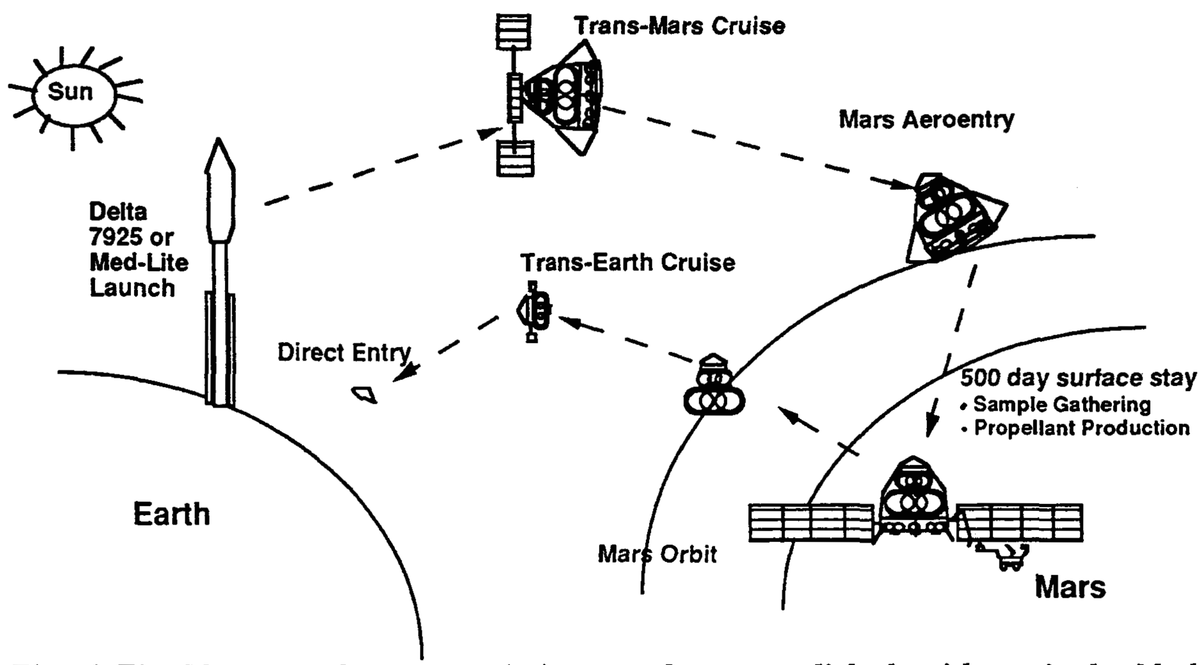

Fig. 1 The Mars sample return mission can be accomplished with a single Med-Lite or Delta launch if the return propellant is made on the Martian surface. No orbiter is required.

Methane/Oxygen Engine Performance

In Figure 2. we show the specific impulse of methane/oxygen engines as a function of the oxygen:methane mixture ratio. The data shown were generated by a standard shifting-equilibrium gas-dynamics code under the assumption of a 500 psi chamber pressure, with the values given representing 97% of the ideal performance under the chosen conditions.

Not shown are additional data for a 1000 psi chamber pressure. Under such conditions, the specific impulse achieved was found to be about 0.5 to 1.5 seconds greater than for the 500 psi case. Such marginal engine performance improvements were considered to be not worthwhile, given the structural mass impacts that would result from operating a pressure fed propulsion system at higher pressures.

Also not shown are additional data for nozzle expansions of 400:1. The use of such large nozzles were found to increase specific impulse by 7 to 10 seconds over the 200:1 cases shown, with the largest benefits occurring for the higher mixture ratios. For packaging reasons, the 400:1 expansion nozzles were avoided in the current design. However, as the performance increase resulting from the use of such nozzles (390 s Isp) is significant, the option of altering the design to include them at a later date may be regarded as an element of reserve design margin. As an additional element of conservatism, an Isp of 380 s at a mixture ratio of 3.5:1 was baselined for the mission, slightly less than the 382 s Isp indicated as feasible by the code.

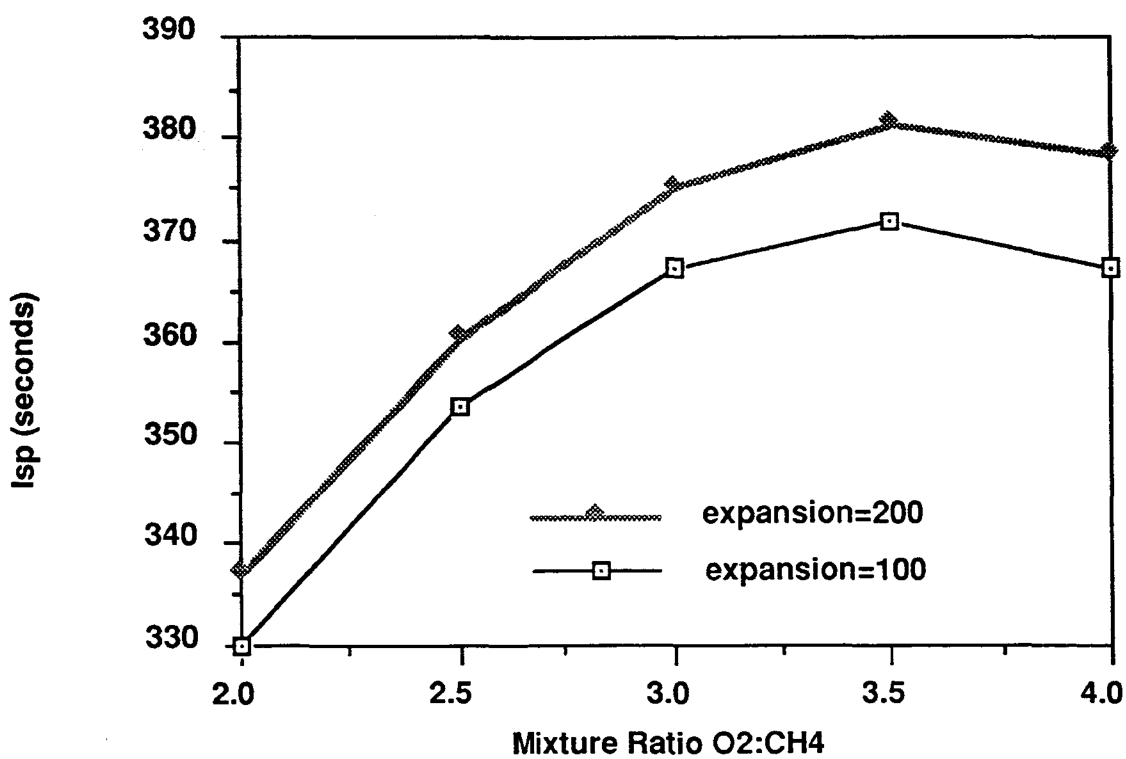

Fig. 2 The specific impulse of methane/oxygen engines as a function of mixture ratio.

The methane/oxygen engines required by the mission are in the 100 lb thrust class, with burn life time requirements on the order of 7 minutes. Recent work in programs funded by the SDIO and BMDO have produced bipropellant engines in this thrust class with short burn lifetimes with thrust/weight ratios as high as 400:1, and derated engines with long burn lifetimes with thrust/weight ratios on the order of 200:1. A conservative thrust/weight ratio of 80:1 was therefore assumed for the engines in the present study, which with 30% contingency results in a net T/W of 61, or a mass of 0.74 kg for a 100 lb thrust engine. The engines would be simple pressure fed systems, operating at a chamber pressure of 500 psi, with radiation and film cooling. The throat would have a diameter of about 1 cm, and the nozzle exit would have a diameter of 14 cm.

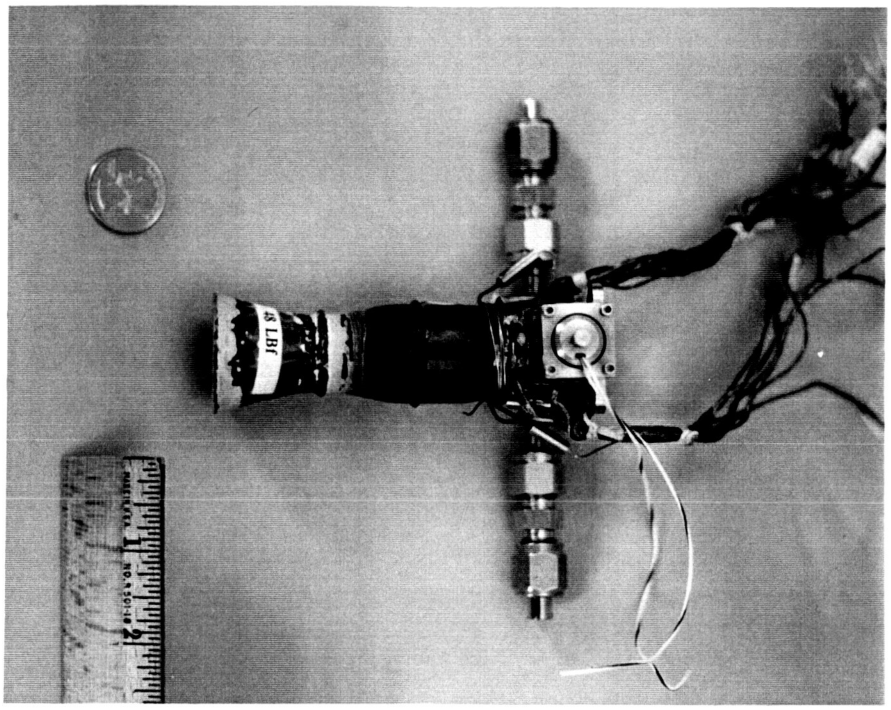

Fig. 3 This 48 lb thrust storable biprop engine built by SDIO has a demonstrated T/W ratio of 200 when derated to operate in a long-duration burn mode.

Sample Return Capsule Concept

In the course of the present study a new concept was developed for a Mars Sample Return Capsule (SRC). Previous SRC concepts developed by Martin Marietta (now Lockheed Martin) employed a parachute, which therefore required an altimeter (or redundant altimeters), mortar and mortar firing systems, all adding significant mass and complexity to the SRC. Moreover, since parachute deployment failure is a real possibly, the system would have to be designed to survive a chuteless impact anyway. Because the system was not designed for a low terminal velocity sans-chute, this then required a massive sample return canister to absorb the shock. The net result was that it was found that SRC masses about 17 kg would be required to return a 0.5 kg sample from Mars.

The concept generated in the present study (Fig. 4) represents a radical departure from this prior work. In this new, simplified concept, no parachute is employed. Instead, the capsule is simply a block of plastic foam and balsa wood, passively aerodynamically stable, and shielded on the bottom with about 4 cm of the lightweight thermal protection material SLA561. As shown in Fig. 4, the capsule used to return a 0.5 kg sample would have a diameter of 60 cm and a total mass of about 6 kg, giving a terminal velocity upon impact with the Earth's surface of 17 m/s, or about 38 miles per hour. This is the same velocity that an object falling freely on Earth will reach if dropped from an altitude of 14.7 meters. A small steel can similar to a 1 pint paint can survive such a drop without damage, even if not shielded by any foam or balsa wood. Such a can, which has a mass of

98 grams, can contain 600 grams of rock samples. As an experiment to establish a rough estimate of sample canister weight, a can of this type was filled with rocks and pumped down to vacuum and exposed to an overpressure of 800 mbar. As shown in Fig. 5, it survived this external pressure without damage or deformation. For the purpose of establishing a sample canister mass to be used in this study, the mass of the steel container used in this experiment was doubled to allow for a canister sealing mechanism.

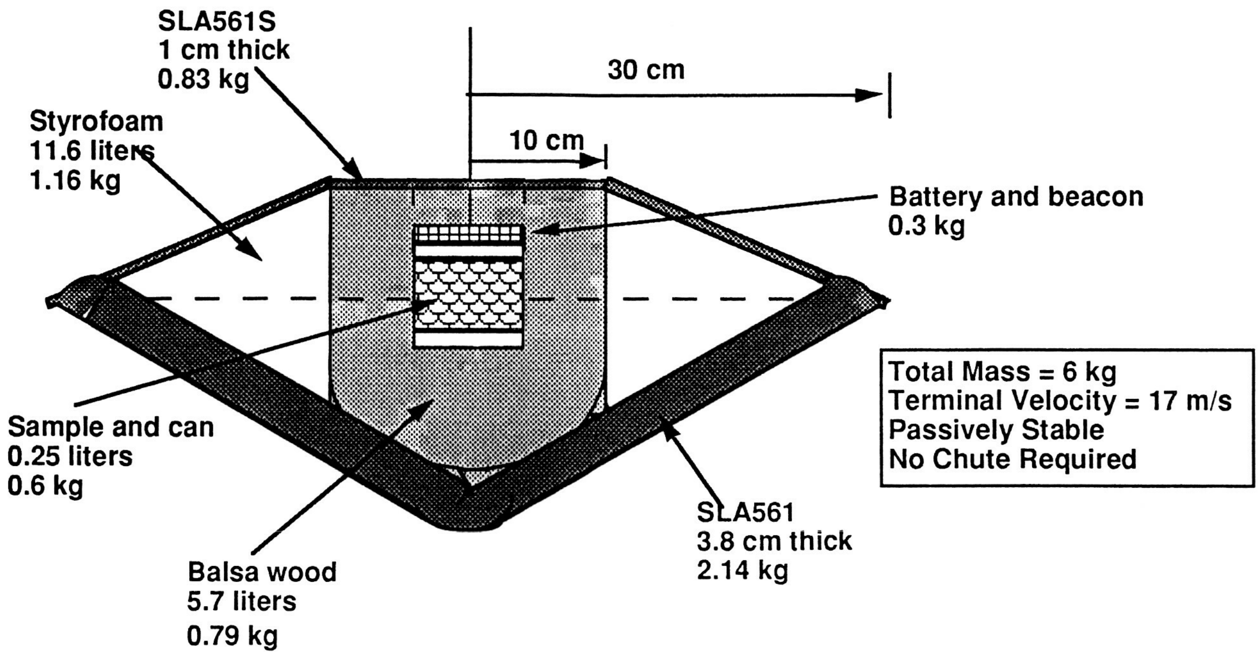

Fig. 4 Design of a Light-weight parachute-less Sample Return Capsule

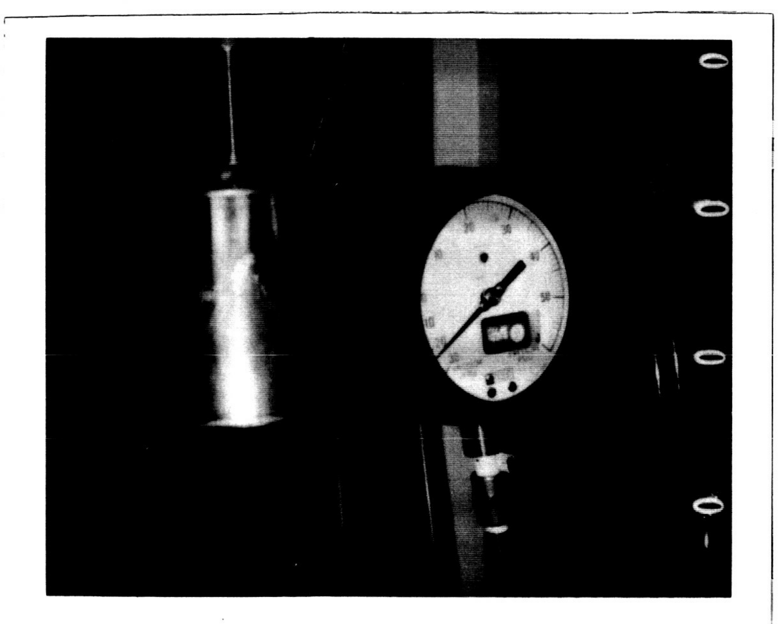

Fig. 5 A 1 pint steel can with a mass of 98 grams can contain 600 grams of rock samples and survive Earth's overpressure without damage or deformation.

The only avionics planned for use in the capsule is a simple Argos radio beacon (modified for the expected environments), similar to that currently used by the US fish and wildlife service to track the movements of peregrine falcons and other birds across the globe (Fig. 6) . Such beacons have a total mass of 25 grams, including batteries, which provide enough energy for 6 weeks of continual transmission. The beacons are received by one of two Argos satellites currently in low polar orbit around the Earth, and their signal relayed to the Argos ground centers in France and the United States. By noting the time of overpass and the Doppler shift of the received signal, the Argos satellites can locate the beacons anywhere on Earth with accuracy's of about 1 km. Because of their negligible mass and cost, the SRC is provided with 2 such beacons.

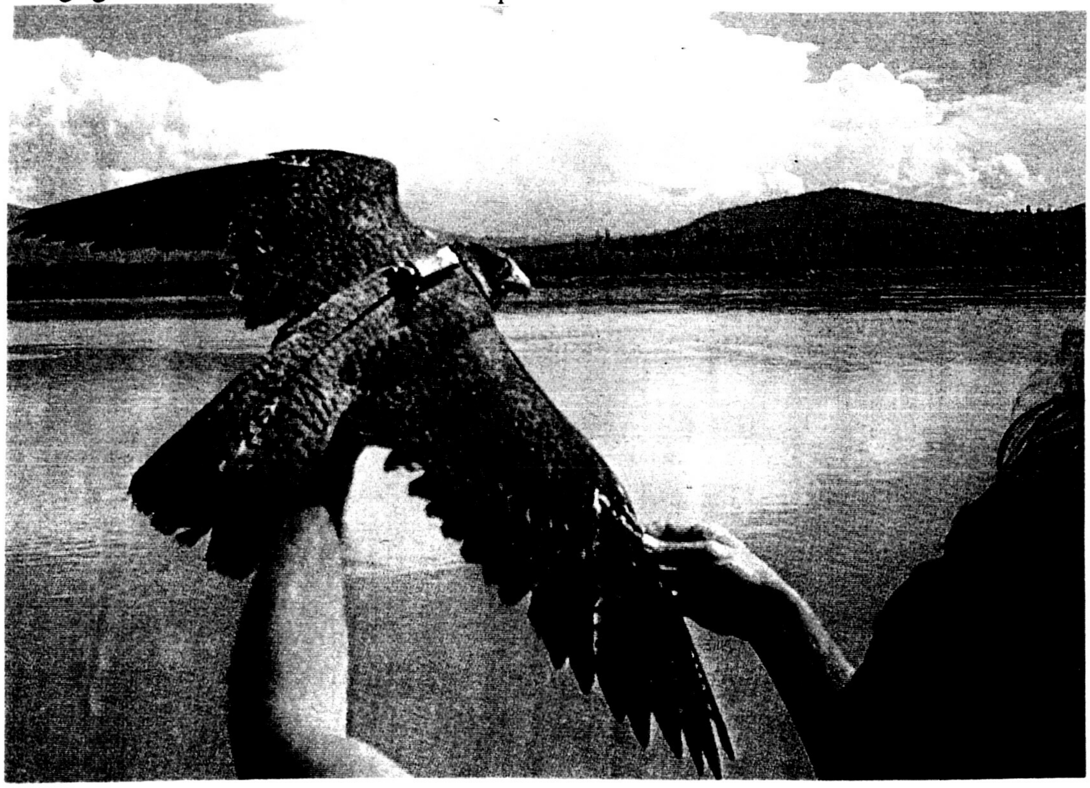

Fig. 6. Argos beacons with a total mass of 25 grams, including batteries, are currently used to track birds globally for weeks at a time. Such systems, with proper environmental modifications, can be used to locate the SRC within a kilometer anywhere on Earth.

In addition to the beacons, radar reflectors are strategically placed below the capsule thermal protection to form corner reflectors, providing the capsule with a strong radar return. This will allow the capsule to be easily tracked by ground based radar. Analysis indicates that the landing footprint of the capsules upon entry will be on the order of 100 km. If targeting even remotely approaching this degree of accuracy is achieved, ground based tracking of the falling capsule will allow it to be located even if the Argos beacons should fail completely.

Because of its simplicity, the chuteless SRC concept is quite scaleable, decreasing in mass in nearly linear fashion with sample size. This allows for significant mass savings to be achieved if smaller samples than the current 0.5 kg baseline are acceptable. Because the ballistic coefficient of scaled down systems are lower, they will hit the ground even slower than our baseline design. The relevant scaling are shown in Fig. 7 and Fig. 8. It may be noted that the total impact energy of our current baseline is 800 Joules, which about the same as the amount of energy required to kick a football 100 yards. The range safety problem associated with this kind of impact energy may be considered modest. The baseline plan is to target the SRC for White Sands, New Mexico, where radar tracking and a large uninhabited soft dry landing area is available. Water landing, however, is an alternative, as the balsa wood and plastic foam SRC will float without difficulty. A landing in a body of water the size of Lake Michigan followed by pickup using a seaplane is one low cost option worthy of consideration.

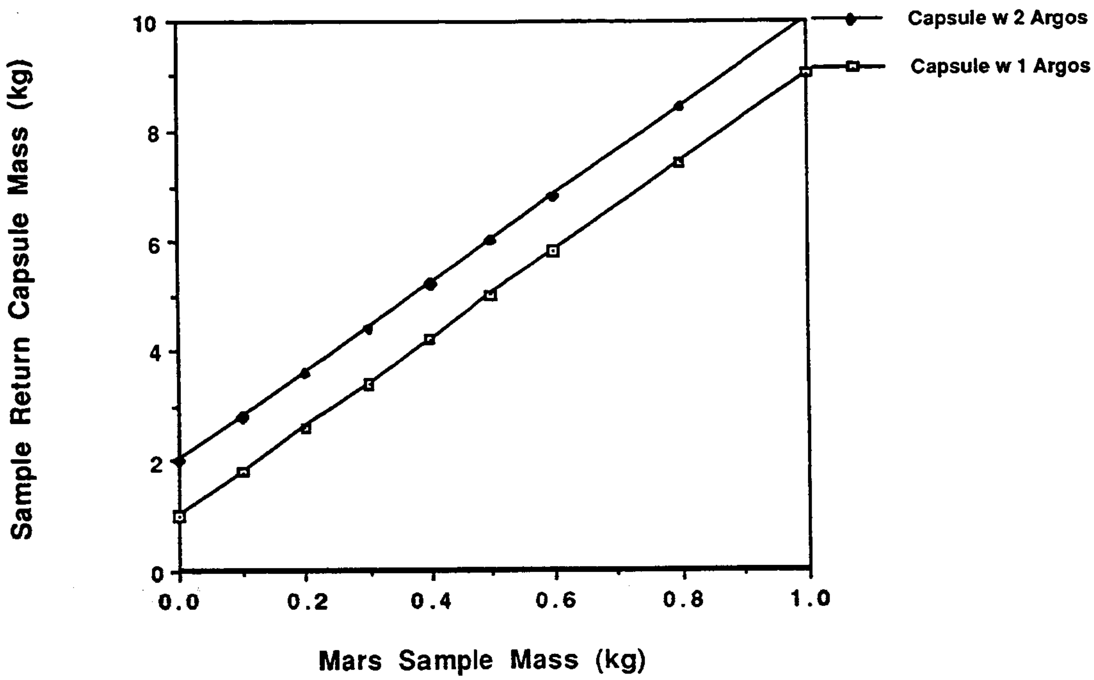

Fig. 7. Mass Scaling of Chuteless SRCs as a function of Sample mass.

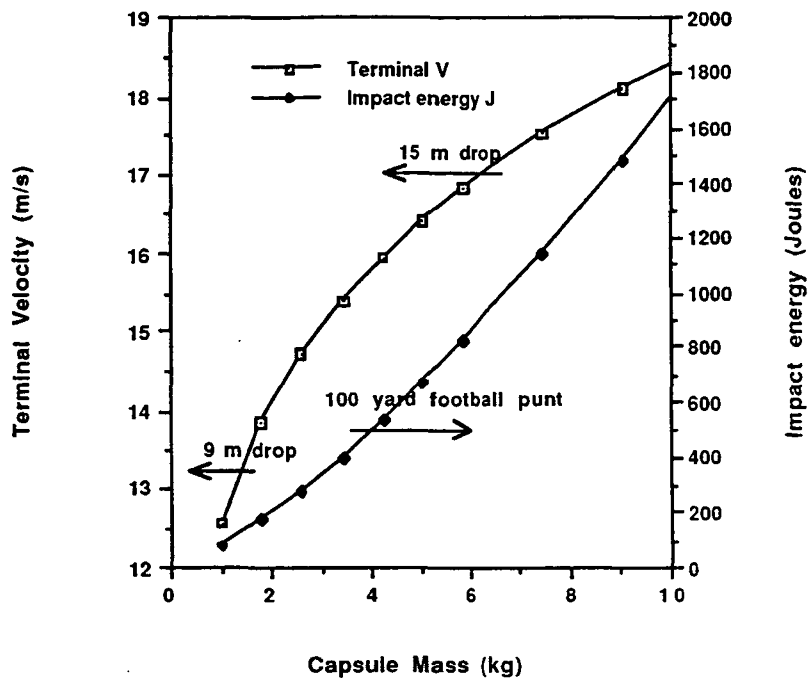

Fig. 8 Impact Velocity and Energy of Chuteless SRCs as a function of sample mass. The total impact energy of the baseline system is of the same order as that required to kick a football 100 yards.

The baseline mission strategy is to place the sample can in the lander's robotic arm at the start of the mission. A rover would be deployed to gather samples, which would be placed in the can. Once the can is filled the can would be sealed. The can would then be raised by the arm and placed into the receptacle cavity in the SRC. The SRC plug would then be pressed down upon the can, pushing it into position and closing the SRC capsule. The cavity is lined with a metal sheath, and the top of the plug, just under its 1 cm of thermal protection SLA561S is lined with solder or some similar meltable material. Calculations show that the upper surface of the capsule will reach 1000 C during Mars ascent, allowing the solder positioned just below the TPS to be heated above its melting point, thereby soldering the plug into place and putting another layer of hermetic sealing around the sample. The upper surface of the SRC is plated with a 1 molecule thick later of magnesium, which will ignite in CO_2 during ascent, thereby sterilizing the upper surface of the SRC, which is the only part of Earth return vehicle system exposed to the Martian environment.

A full-scale mockup of the chuteless SRC was built. Including a 2 kg lead weight to simulate the calculated weight of the SLA561 thermal protection system, and a 0.5 liter steel sample canister, the total mass of the mockup was less than 4 kg.

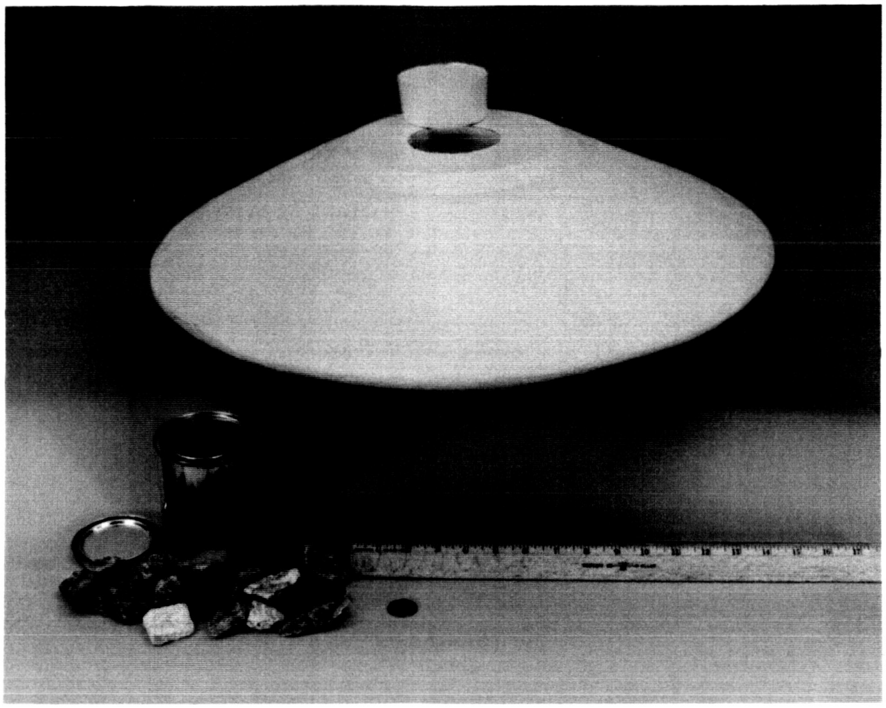

Fig. 9 The mass of a full scale mockup of the SRC is less than 4 kg. Terminal velocity of the mockup is 13 m/s.

Hydrogen Storage Concept

One of the main challenges associated with the Mars Sample Return mission using ISPP is to transport the hydrogen to Mars required to serve as feedstock for the methane/oxygen propellant production process. While the required hydrogen mass is quite small, its cryogenic properties pose storage problems while its low density makes it difficult to carry a large excess over the actual requirement so as to defeat the possibility of excessive boiloff by brute force. These challenges are much less difficult to deal with in the manned Mars Direct[1] mission that was the original inspiration for the Sabatier/Electrolysis (SE) driven MSR mission, because in the much larger vehicles that are characteristic of piloted missions, the hydrogen tank surface to volume ratio is much lower, greatly reducing the fraction of transported hydrogen likely to boiloff.

In the prior study[2] of an MSR-ISPP mission done for JSC by Martin Marietta, the approach adopted was simply to use the lower stage Mars ascent Vehicle (MAV) tanks to carry a liquid hydrogen supply 30% greater than the import requirement at Mars. As the calculated boiloff based upon an assessment of heat leak was about 15%, this left a 15% margin against errors in the heat leak calculation. This is rather low, however in that study only a Sabatier and electrolysis system were employed in the ISPP unit, providing a net propellant leverage (CH_4/O_2 product to H_2 import ratio) of 10.3 to 1. If a third reactor, such as the reverse gas shift were employed, a leverage of 18:1 could be obtained. This would have increased the hydrogen import to requirement ratio from 1.3:1 to 2.3:1, resulting in a net margin against error in the heat leak calculation of about 100%.

In the present study, an alternative is proposed that can increase this margin further, up to about 600%, and also simplify launch operations as well. The concept is to use MAV lower stage tanks of the high strength graphite-overwrapped type, instead of conventional aluminum tanks. Liquid hydrogen is loaded into these tanks on the launch pad. However, instead of allowing hydrogen to boiloff as temperature rises and pressure increases, the strategy is to simply contain the hydrogen at increased pressure. There is thus no need to maintain an umbilical on the pad. The tank is designed to contain all the hydrogen at pressures up to 2000 psi (3000 psi burst pressure), which it will reach when the hydrogen gas contained within it reaches a temperature of 67 K. The small Stirling cycle refrigerators used in the ISPP plant are then used to prevent the temperature of the hydrogen in these tanks from rising further. These refrigerators can operate as low as 40 K, and have a refrigeration capacity of about 3 W at 65 K. As the calculated heat leak into the tank is on the order of 0.5 W, this provides a margin against error in heat leak assessment of about 600%.

The graphite overwrapped tanks proposed are based upon those made by Structural Composite Industries. Such tanks have been used in cryogenic applications and have flown on many missions. In the present application, each of the four lower stage MAV tanks is 81 cm long and 42 cm in diameter, and contains a volume of 85 liters. According to SCI analysis, each such tank will require a mass of 6.7 kg to contain hydrogen at 2000 psi. High strength tanks are frequently rated in terms of "PV/W," or pressure times volume divided by weight, with English units customarily used and the resulting parameter given in terms of inches. SCI has built and flown many tanks with PV/W as high as 1.3 million inches. The current design requires a PV/W of 704,000 inches and is thus conservative with respect to the state of the art. According to SCI, such a tank can be designed to order and developed for several hundred thousand dollars, and be ready for shipping within 8 months.

It may be noted that 67 K hydrogen at this pressure has a density of 46 kg/m3. This is less than the density of liquid hydrogen (70 kg/m3), and considerably smaller tank sizes could be obtained by going to higher pressures. However, since 67 K hydrogen at pressures above 2000 psi deviates strongly from the ideal gas law, pressure increases do not cause a proportional decrease in volume, and the net result of going to higher pressure is that a more massive tank is required to hold the same amount of gas. (Fig. 10) For this reason the large tanks holding hydrogen at 2000 psi was baselined.

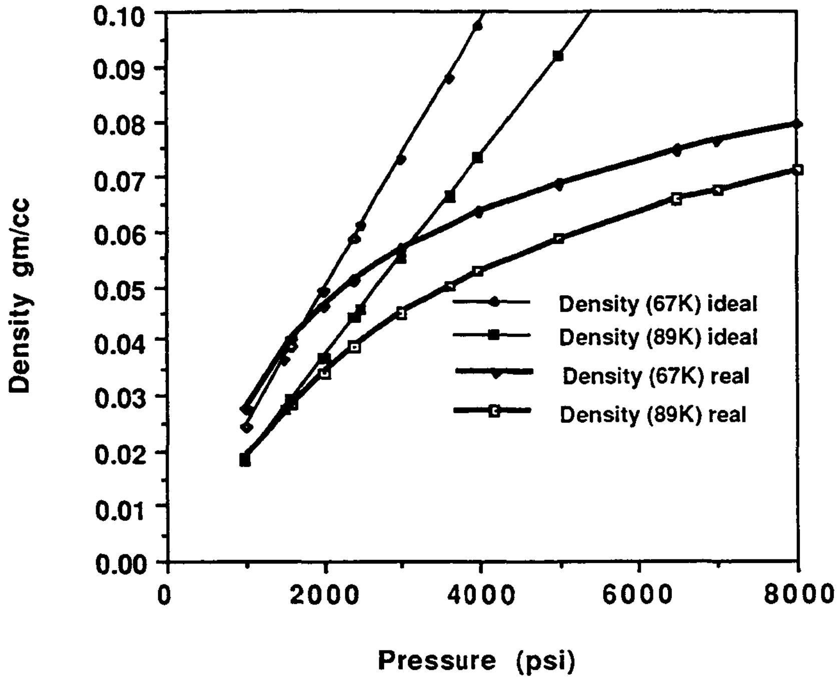

Fig.10 Behavior of high pressure hydrogen gas at low temperatures.

At the nominal calculated heat leak, the tanks will not reach 67 K until after the vehicle has arrived at Mars. If the heat leak is much greater than calculated, 67 K will be reached in transit and the Stirling cycle refrigerators will be employed to maintain the temperature at that level for the rest of the trip.

Heat leak was calculated for both the upper and lower stages of the MAV, as shown in Figs 11, and 12. Assumed in these calculations was a delta-T of 120 K, which implies an environmental temperature of 187 K in transit (the dark, highly emissive surface of the aeroshell is viewing deep space during Trans-Mars cruise) and 220 K on the Martian surface (where the issue becomes one of storing CH_4 and O_2 at 100 K for long duration). The assumed insulation system is double aluminized mylar double silk net MLI spaced with 20 layers per cm. This insulation has a thermal conductivity of 0.000045 W/m-K and a density of 45.2 kg/m3. On Mars, a light weight vacuum jacket capable of resisting Mars' 8 mbar overpressure is used to preserve the insulating properties of each tank. Once again, it should be noted that the total heat leak calculated for the tank system is on the order of 0.5 W, an order of magnitude less than the 5W capacity of the refrigeration system operating at 100 K. The refrigeration requirement to maintain the CH_4/O_2 bipropellant after it has been produced is thus much less than that required to produce the liquid CH_4/O_2 in the first place. Hydrogen storage on the surface is not a problem, as it can be reacted away into CH_4 and water soon after landing.

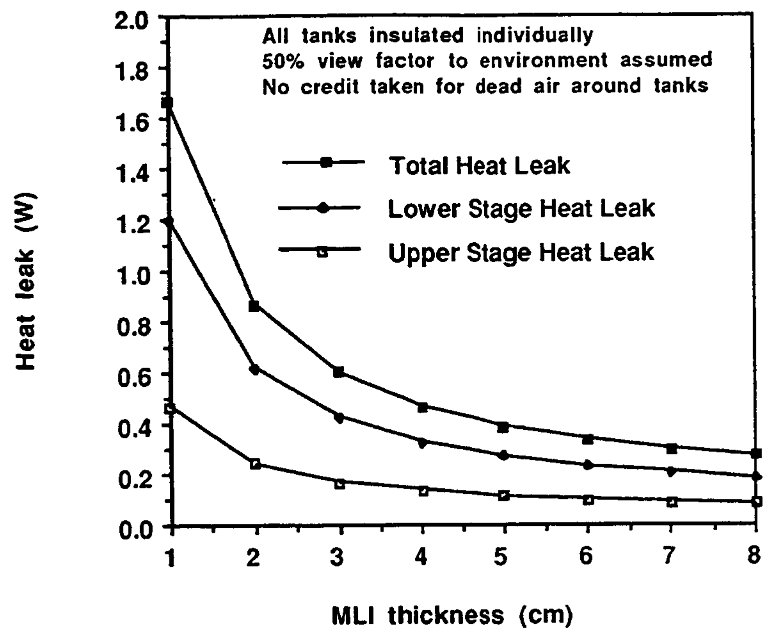

Fig. 11. Heat leak into MAV stages as a function of MLI thickness

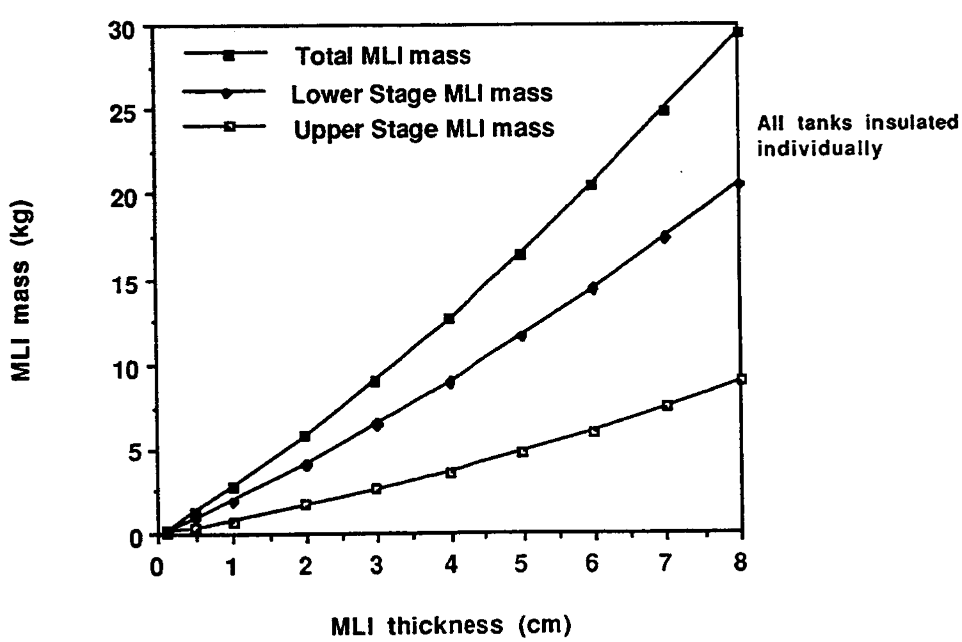

Fig. 12 Mass of MLI insulation on MAV stages as a function of thickness

On the basis of the analysis shown in Figs 11 and 12, a 3 cm MLI blanket is found to be sufficient for insulating the lower stage, while a 2 cm blanket was deemed sufficient for the upper stage.

MSR-ISPP Mission Avionics

To achieve a launch mass that is well within the capability of a Delta and potentially a Med-Lite, it was necessary to assume the use of advanced light-weight avionics. The following section describes the light-weight avionics that have been selected for use in the MSR-ISPP concept and provides the basis upon which the avionics mass numbers used in the equipment list are derived.

For the purposes of this study (and following common spacecraft subsystem classifications) MSR-ISPP avionics have been grouped into four subsystems; Power Generation, Distribution and Control, Telecommunications, Attitude Determination and Control (AD&C), and Command and Data Handling (C&DH). A key deviation from past spacecraft subsystem design practices is the integration of the power subsystem functions and the C&DH functions into one subsystem that is called the Integrated Computer and Power Subsystem (ICAPS). Realizing that many low cost missions will be enabled by very low mass avionics, Martin Marietta has for several years, been studying, developing and prototyping elements of an ICAPS system. With the recent mergers that Martin Marietta has undergone, we have also been able to exploit the miniaturization work that has been underway at several other laboratories as well. The net result is that using advanced technologies such as multi-chip modules (MCM), high density interconnects (HDI), solid state memory (SSM) and ultra-miniature connectors, Martin Marietta has been able to develop an ICAPS design that offers significant mass savings over currently available components. Elements of the ICAPS system have been and are currently being prototyped for proof of concept testing and demonstration. Figure 13 provides a drawing with dimensions of the ICAPS concept and Table 1 provides a mass breakdown of the individual modules that make up the ICAPS. To maintain a

Table 1. ICAPS Mass Breakdown

Module	X In	Y In	Z In	Volume In3	Volume cm3	Density gm/cm3	Mass gm
Computer Module							
HDI Card	4.00	3.00	0.07	0.84	13.77	3.00	41.30
Spacer	3.00	3.00	0.13	1.17	19.17	2.70	51.77
Input/Output Module							
HDI Card	4.00	3.00	0.07	0.84	13.77	3.00	41.30
Spacer	3.00	3.00	0.13	1.17	19.17	1.35	25.88
Distribution Module							
HDI Card	4.00	3.00	0.07	0.84	13.77	3.00	41.30
Spacer		3.00	0.13	1.17	19.17	2.70	51.77
Power Converter Module							
HDI Card	4.00	3.00	0.10	1.20	19.66	3.00	58.99
Spacer	3.00	3.00	0.13	1.17	19.17	2.70	51.77
Pyro Initiator Module							
HDI Card	4.00	3.00	0.07	0.84	13.77	3.00	41.30
Spacer	3.00	3.00	0.13	1.17	19.17	1.35	25.88
Power Control Module							
HDI Card	4.00	3.00	0.07	0.84	13.77	3.00	41.30
Spacer	3.00	3.00	0.13	1.17	19.17	2.70	51.77
Structure Hardware							100.00

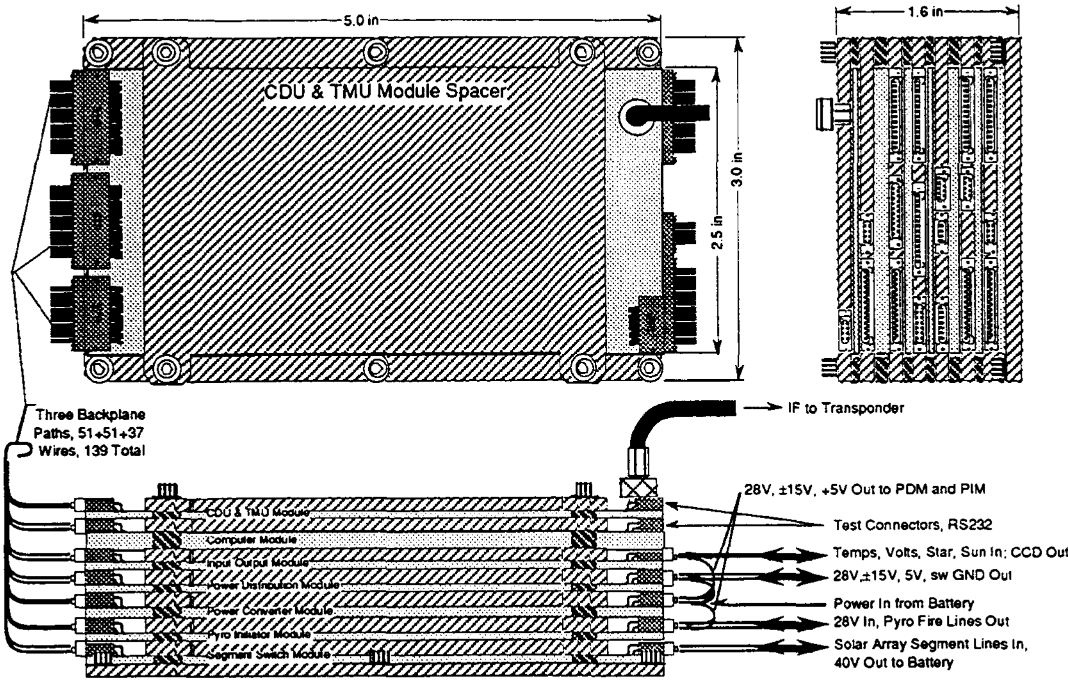

Figure 13. Integrated Computer and Power Subsystem for Mars Sample Return Mission

component mass allocation that is familiar to spacecraft designers, the ICAPS modules have been listed by function in the Low Cost MSR master equipment list rather than as one integrated unit.

Telecommunications is another avionics subsystem in which the planetary spacecraft concept design is able to benefit from technology development work currently underway for other government programs. Martin Marietta Laboratory (MML) in Syracuse is currently involved in light-weight, low power telecom. related development activities encompassing high density interconnect packaging, high efficiency microwave/millimeter wave power amplifiers, transceiver modules, modulators, X-band phased array and Ka-band landing radar.

Extensive testing of the HDI technology has been carried out at MML in order to qualify components for space. Individual light-weight components and modules that make up an X-band transponder have also been developed and tested. What remains is to integrate the individual components and modules into a fully functional deep space transponder. Using the technology and hardware currently available at MML and elsewhere, an X-band transponder weighing less than .5 kg should be available to support a Mars Sample Return mission in the 2001-2003 time frame given an adequate level of technology integration funding. In addition, mass savings as well as power savings are being achieved for the solid state power amplifier (SSPA) element of the telecom. subsystem, through the use of pseudo-morphic high electron mobility transistors (PHEMT). PHEMT performance, which holds the promise of a significant increase in amplifier efficiency, is currently being verified through testing at MML.

For this study we did not incorporate the Ka-band landing radar because the need to save mass on the lander is not as critical as on the return stages. The landing radar to be used on the Mars Surveyor Program in '98 was selected for this concept in order to exploit the cost savings and reduced technical risk associated with flight proven hardware.

The Attitude Determination and Control (AD&C) subsystem for this mission is made up of a combination of existing flight proven components that have been selected for flight on the MSP missions in '98 and advanced components. The digital sun sensors selected for incorporation in this mission and MSP are existing, flight proven components. The star sensor for this mission as well as MSP is a star camera from OCA Applied Optics that was modified and upgraded for flight and successfully demonstrated on the Clementine mission. The IMU is the only AD&C element unique to this mission, and it is based on interferometric fiber optic gyro (IFOG) technology that is currently under development by Litton.

ISPP Requirements

The primary in-situ propellant production process used in the mission is known as the Sabatier-Electrolysis or "SE" cycle. The SE cycle works as follows: Hydrogen is transported from Earth to Mars where it is combined with CO_2 acquired from the Martian atmosphere in a Sabatier reactor to produce methane and water in a 1:2 molar ratio. The methane produced by this process is drawn off and liquefied, while the water is condensed and sent to an electrolysis unit to be split into hydrogen and oxygen. The oxygen produced by the electrolysis unit is liquefied, and the hydrogen is fed back into the Sabatier reactor. The required chemical reactions are:

$4H_2 + CO_2 \text{--->} CH_4 + 2H_2O$ (1) (Sabatier reaction)

$2H_2O \text{--->} 2H_2 + O_2$ (2) (Electrolysis)

The Sabatier reaction is strongly exothermic and requires no power beyond a startup transient. Its equilibrium constant is very high (order of 1,000,000), allowing yields greater than 99% to be readily achieved, and occurs rapidly at temperatures above 570 K. The water electrolysis reaction is endothermic and energy intensive. The technology required to run both of these reactions has been completely mature for the past century. It may be noted that under nominal conditions, 50% of the hydrogen needed by the Sabatier reactor is recycled from the products of water electrolysis, while the other 50% is provided by an external source. The net propellant leverage provided by such a pure SE cycle is 12 kilograms of methane/oxygen bipropellant produced for every kilogram of hydrogen imported from Earth. A successful demonstration of a complete end-to-end brassboard SE system, including sorption pumps to acquire CO_2 from a Mars-like atmosphere, chemical reactors to transform the input hydrogen and CO_2 into CH_4/O_2, and a cryogenic refrigerator to liquefy the resulting propellant, on a scale adequate to support the MSR-ISPP mission has been conducted at Martin Marietta[4].

The SE cycle working alone produces O_2 and CH_4 in a mixture ratio (by weight) of 2:1. The optimal mixture ratio for burning this propellant combination, however, is 3.5:1. This can be achieved simply by discarding some of the product methane, however such a strategy lowers the net propellant leverage to 10.3:1. An alternative strategy is to provide an additional reactor that can produce additional oxygen from the Martian atmosphere, thereby raising the net leverage to 18:1. One possible alternative for doing so is to employ zirconia cells to directly dissociate CO_2 into CO and O_2. Such technology has been demonstrated by the University of Arizona[5]. However the energy cost per unit oxygen produced by Zr cells is about an order of magnitude higher than that required by the SE system. As power is at a premium on a solar powered Mars Sample Return-ISPP mission, this approach was therefore rejected. An alternative approach is to employ the reverse water gas shift (RWGS) reaction, familiar to chemical engineers since the gaslight era.

The RWGS reaction is:

$CO_2 + H_2 \text{--->} H_2O + CO$ (3)

The water produced by this reaction can be electrolyzed in accordance with equation (2), with the resulting hydrogen recycled back into the RWGS reactor, the oxygen liquefied as propellant product, and the CO discarded as waste. Aside from leakage, no hydrogen would be consumed by such a cycle. The RWGS reaction (3) is mildly endothermic. Like the Sabatier reaction, it can be conducted in a simple steel vessel containing a catalyst bed, allowing for robust and compact system construction. When properly catalyzed it is a fast reaction, but unfortunately has a low equilibrium constant (order of 0.1) when conducted in the 570 K range. Therefore, in order to operate the RWGS system as an oxygen machine in cycle with an electrolyzer, measures must be taken to drive the reaction to the right, the most obvious being the use of desiccant beds or cold traps to remove water from the system. Additional efficiencies can be gained by using a selective membrane to separate any waste hydrogen from the RWGS reactor effluent stream and then recycle it back into either the RWGS or Sabatier reactors. Since the Sabatier reactor can produce enough waste heat to drive the RWGS reactor, the primary power requirement is for water electrolysis. This gives such a system about the same product/power ratio as an SE system, i.e. much better than that offered by Zr cells.

The mass and complexity increase caused by adding such a system can be avoided, however, by running both the Sabatier and RWGS reactions in the same reactor. Such a system was first suggested in reference 1. Analysis done by the Allied Signal company, presented as an Appendix in reference 4 shows that if the Sabatier reactor is run at 870 K, reactions (1) and (3) will occur in tandem, with the net reaction resulting given by:

$$3CO_2 + 6H_2 \dashrightarrow CH_4 + 4H_2O + 2CO \qquad (4)$$

Reaction (4) is mildly exothermic, and provided efficient heat exchangers are used to recycle heat from the effluent gases back to the input gases, it can be run without input power beyond the startup transient. It also has a fairly good equilibrium constant (order of 100 at 870 K), allowing it to be used as a simple single pass reactor. The CO waste product can be separated downstream, as it will not liquefy at the same temperatures as CH_4. When operated in tandem with an electrolyzer performing reaction (2), reaction (4) will produce O_2:CH_4 in a mixture ratio of 4:1, with a net propellant leverage of 20:1 of CH_4/O_2 product to hydrogen import.

The simplicity of this single combined Sabatier/RWGS reactor system speaks for itself. However, until such a system is demonstrated, we have decided to err on the side of conservatism by assuming ISPP system mass based upon separate Sabatier and RWGS reactors. The Sabatier reactors are still used to heat the RWGS reactors however.

The baseline mission requires a total of 277 kg of propellant to be produced, consisting of 62 kg CH_4 and 215 kg O_2. If performed in the baselined 500 day surface stay, this results in a propellant production requirement of 0.554 kg/day, or about 75% of the rate demonstrated by the Martin Marietta brassboard unit reported in reference 4. Based upon the results of that demonstration, the following is the estimated mass and power requirements for the ISPP system.

Table 2. Mass and Power of ISPP System

Subsystem	Mass	Power
CO_2 Acquisition (2 Carbon Sorption Pumps)	3 kg	0
Sabatier Reactors (2 units)	4 kg	0
RWGS Reactors (2 units)	4 kg	0
Electrolysis Units (2 units)	6 kg	214 W
Refrigerators (3 units)	5 kg	106 W
Electronics and Controls	2 kg	10 W
Lines and Valves	2 kg	0
Margin	4 kg	20 W
Total	30 kg	350 W

Power requirements for both the carbon sorption pumps and RWGS reactors are zero because both are heated by waste heat from the Sabatier reactor. This can readily be done either with a heat pipe arrangement or more elegantly by placing the Sabatier reactors within the RWGS reactors, and the RWGS reactors within the sorption pumps, in a concentric arrangement. In operation, the sorption pump is allowed to cool at night to Mars ambient temperatures of 180 K, where it's activated carbon bed will absorb 40% of its weight in CO_2. In general, the ISPP system is operated only in the daytime, when it is desired to heat the sorption bed so that it will outgas. Inserting the Sabatier reactor inside the bed thus allows the outgassing to be done at no cost in power beyond the initial startup transient. The system in Table 2 provides for redundant sorption pumps, Sabatier and RWGS reactors, and electrolyzers, and doubly redundant refrigerators. The power requirement for the electrolyzer and refrigeration system given in Table 2 are daytime (daylight) averages. As explained below, they will vary in the course of the mission.

Assuming an average solar incidence of 600 W/m2 on Mars during the course of the mission, and an 18% collector efficiency, an 8 square meter fixed solar array can generate an average daytime power of 432 W. Such an array was therefore judged sufficient to power the ISPP system with enough left over to power communications and other necessary vehicle hardware.

ISPP Production Timeline

The ISPP production system goes through three phases in the course of the 500 surface stay of the missions.

In Phase A of the production cycle, the only parts of the SE system that operate will be the sorption pumps, the combined Sabatier/RWGS reactor, and the methane liquefaction system. The electrolysis and the oxygen liquefaction system are not operated. The purpose of operating in this mode is to as rapidly as possible convert 25% of the landed hydrogen supply in all the lower stage tanks into methane and water. This allows the temperature in the lower stage tanks to rise to 90 K without increase in tank pressure. Because the heat sink in the hydrogen used is much greater than that required to liquefy the product methane, no refrigeration power is required during this phase, and since the electrolyzer is not operating, the total power required to operate the ISPP system is very small, and not dependent upon the rate of propellant production. During Phase A, therefore, operation around the clock is feasible, although probably not necessary, as the sorption pumps can be cycled several times per day to greatly increase the rate of CO_2 acquisition over the baseline requirement of 0.5 kg/day. Each time the sorption pump is cycled 0.8 kg of CO_2 is acquired. Assuming 5 cycles per day, 4 kg of CO_2 will be acquired per day. It will thus take 11 days to acquire the 44 kg of CO_2 required to react away the 4 kg of hydrogen that constitutes 25% of the 16 kg imported hydrogen supply.

At the end of 11 days, each of the lower stage tanks will contain 3 kg of hydrogen. Phase A then continues draining 2.5 kg of hydrogen out of one of the lower stage methane tanks. This will take another 7 days. Thus at the end of 18 days the MAV will have on board 8.7 kg of liquid methane, which will be stored in the upper stage methane tanks, and 9.5 kg of hydrogen gas stored in the lower stage MAV tanks. There will also be 39 kilograms of water, which will be stored in auxiliary tanks on the lander.

Phase B then begins, with the entire system in operation, producing 0.49 kg of oxygen and 0.14 kg of methane per day. Net hydrogen consumption is 0.035 kg per day. At the beginning of Phase B, all the hydrogen used is taken from the methane tank which had been reduced to 0.5 kg of hydrogen during the final part of Phase A. It will thus take 14 days to empty this tank. At the end of that time, 2 more kilograms of methane will have been produced, which will still not completely fill the upper stage methane tanks. Also, 7 kilograms of oxygen will have been produced, which will be stored in the upper stage oxygen tanks.

In the next part of Phase B, one of the lower stage oxygen tanks is completely emptied of its 3 kg of hydrogen. This will produce another 12 kilograms of methane, which will complete the filling of the upper stage tanks and begin to fill the vacated lower stage methane tank. 42 kilograms of oxygen will also be produced, making a total of 49 kilograms, which will nearly complete the filling of the upper stage oxygen tanks. This part of Phase B will take 84 days.

In the final part of Phase B, the remaining 6 kilograms of hydrogen contained in the MAV lower stage tanks will be converted into 24 kilograms of methane and 84 kilograms of oxygen, all of which can be stored in the two lower stage tanks that have been emptied of hydrogen previously. This part of Phase B will take 168 days.

Phase B power requirements are much higher than Phase A because the electrolyzer is operating and oxygen is being liquefied. However, as there is still cold hydrogen to use as a heat sink which contains nearly enough refrigeration capacity to liquefy both the product methane and oxygen, refrigeration power requirements are still significantly reduced. At the end of Phase B all hydrogen has been emptied from the lower stage tanks. The only remaining source of hydrogen is the 39 kilograms of water that had been stored during the course of Phase A.

Phase C now begins. During this phase the operating temperature of the Sabatier/RWGS reactor is allowed to fall to 600 K, so that only the Sabatier reaction occurs. The 39 kilograms of water is converted into 17.3 kilograms of methane and 69.3 kilograms of oxygen. This takes 173 days. Because there is no cold hydrogen to provide a heat sink, refrigeration power requirements are maximized in Phase C. To compensate, the rate of water electrolysis is somewhat reduced.

The overall timeline of ISPP operation for the MSR mission is summarized in Table 3.

Table 3 ISPP Operations During the Mars Surface Stay

		Material Produced/Used (kg)					
Phase	#days	H_2	H_2O	CH_4	O_2	Electrolyzer Pwr	Refrigerator Pwr
A1	11	-4	24	5.3	0	0 W	0 W
A2	7	-2.5	15	3.4	0	0	0
B1	16	-0.5	0	2	8	240	80
B2	96	-3	0	12	48	240	80
B3	192	-6	0	24	96	240	80
C1	173	0	-39	17.3	69	192	160
Total	495	-16	0	64	221	214 (Ave)	106(Ave)

Power numbers given in Table 3 are daytime averages based upon a 12 hour daylight period. It is assumed that no ISPP operations other than sorbing are occurring at night.

CAD Drawings

Figs 14 and 15 are CAD drawings showing the primary mission vehicles deployed on the Martian surface and stowed in its aeroshell during the outbound trip to Mars.

Fig. 14 shows the vehicle deployed on the Martian surface. At its base is a Mars Surveyor Lander, modified from the version that Lockheed Martin Astronautics is currently building for NASA's Mars 98 mission by the addition of an extra pair of solar panels. The use of this lander adds to system engineering heritage and minimizes cost and risk. An extra pair of tanks, identical to the hydrazine propellant tanks are attached to the lander to be used for temporary storage of the water intermediate product during the ISPP process. A microrover, similar to the JPL Pathfinder rover is visible. It is used to collect samples which are then transferred to the sample canister held in the lander's robotic arm. At the conclusion of sample collection, that arm is used to transfer the sample canister to the receptacle cavity in the sample return capsule positioned at the top of the Mars Ascent Vehicle.

Four large graphite overwrapped tanks together with four pressure-fed 150 lb thrust CH_4/O_2 engines comprise the core of the MAV lower stage. Each tank is surrounded by a layer of 3 cm of MLI, encased in a light weight vacuum jacket. A lightweight fairing surrounds the four tanks, keeping dust out of the system and creating a layer of dead air around the tanks which adds to the insulating effect of the MLI. The MAV's liftoff mass is 373 kg, including contingency, giving it a lift off thrust/weight ratio of 2. The lower stage is used to lift the vehicle into space, giving it a suborbital delta-V of about 3.1 km/s.

Above the first stage is an adapter, and then the upper stage, which is composed of four off-the shelf light weight aluminum tanks, each surrounded by 2 cm of MLI and a vacuum jacket, and one 150 lb CH_4/O_2 engine. The upper stage fires from a suborbital condition, delivering a delta-V of 3.3 km/s that drives the vehicle to Mars orbit and then on to Earth. The stage, together with the Earth Return Vehicle that sits above it, are surrounded by a light weight fairing that keeps out dust and reinforces the systems thermal insulation from the Martian environment. Thrust vector control for the MAV is accomplished using hydrazine thrusters contained in the Earth Return Vehicle, (ERV, also termed the Trans-Earth Cruise Stage) which also contains all avionics needed for both Mars ascent and the trans-Earth cruise portion of the mission. A small set of unfolding solar arrays can be seen stowed on either side of the Earth Return Vehicle. Atop the ERV is the Sample Return Capsule, a solid disk of plastic foam and balsa wood surrounded by a thin graphite composite shell and SLA561 thermal protection material. It can be seen that the upper surface of the SRC is the only portion of the system delivered onto trans-Earth trajectory that is at any time exposed to the Martian environment. As the upper surface is sterilized during ascent, when aeroheating raises it to 1000 C and ignites a thin magnesium layer plated upon it, the possibility of back contamination to Earth is minimized.

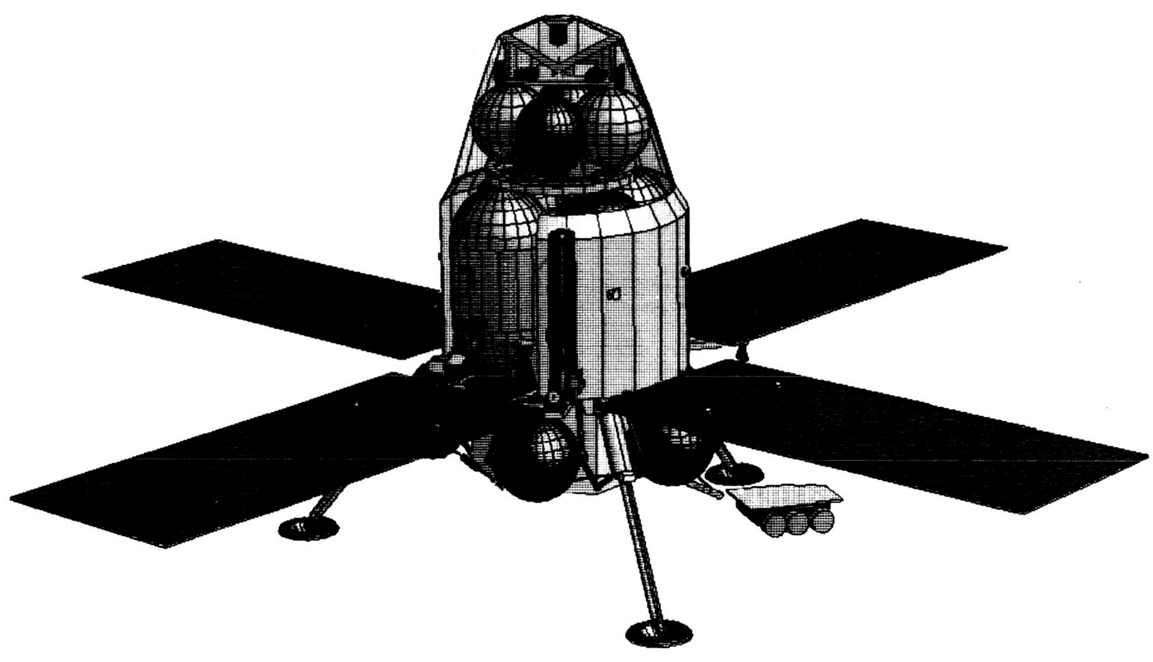

Fig. 14. Mars Ascent Vehicle Deployed on Mars Atop Modified Mars 98 Lander

Fig. 15 shows the same vehicle stowed in its entry aeroshell. At first glance it may appear top heavy, but this is not the case as the majority of the mass is in the lander and aeroshell, placing the system's center of gravity well below it's center of pressure. The aeroshell is identical to that which will be used on the Mars 98 mission. It is 2.5 meters in diameter, allowing the vehicle to fit within the launch fairing of a Delta, Molniya, or Med-Lite launch vehicle. It will be noted that there is no parachute. Instead, the aeroshell is used to decelerate the system to about 190 m/s terminal velocity, after which the lander's rocket engines are used to provide a delta-V of 400 m/s to achieve a soft landing. The elimination of the parachute greatly simplifies system packaging and has the potential to reduce mission mass, cost, and risk as well. The primary concern parachute-elimination creates is the need to provide for clean separation between the MAV/lander combination and the aeroshell. In the current design this is achieved by providing a set of rails which are used to guide the back-shell off the lander after terminal velocity is reached. Clean separation between the lander and the aeroshell beneath it is achieved by using a Pyro device to blow holes in the aeroshell beneath the lander engines. The engines are then used to decelerate the system to negligible air speed, and thus dynamic pressure, allowing the aeroshell to be dropped from the lander without difficulty.

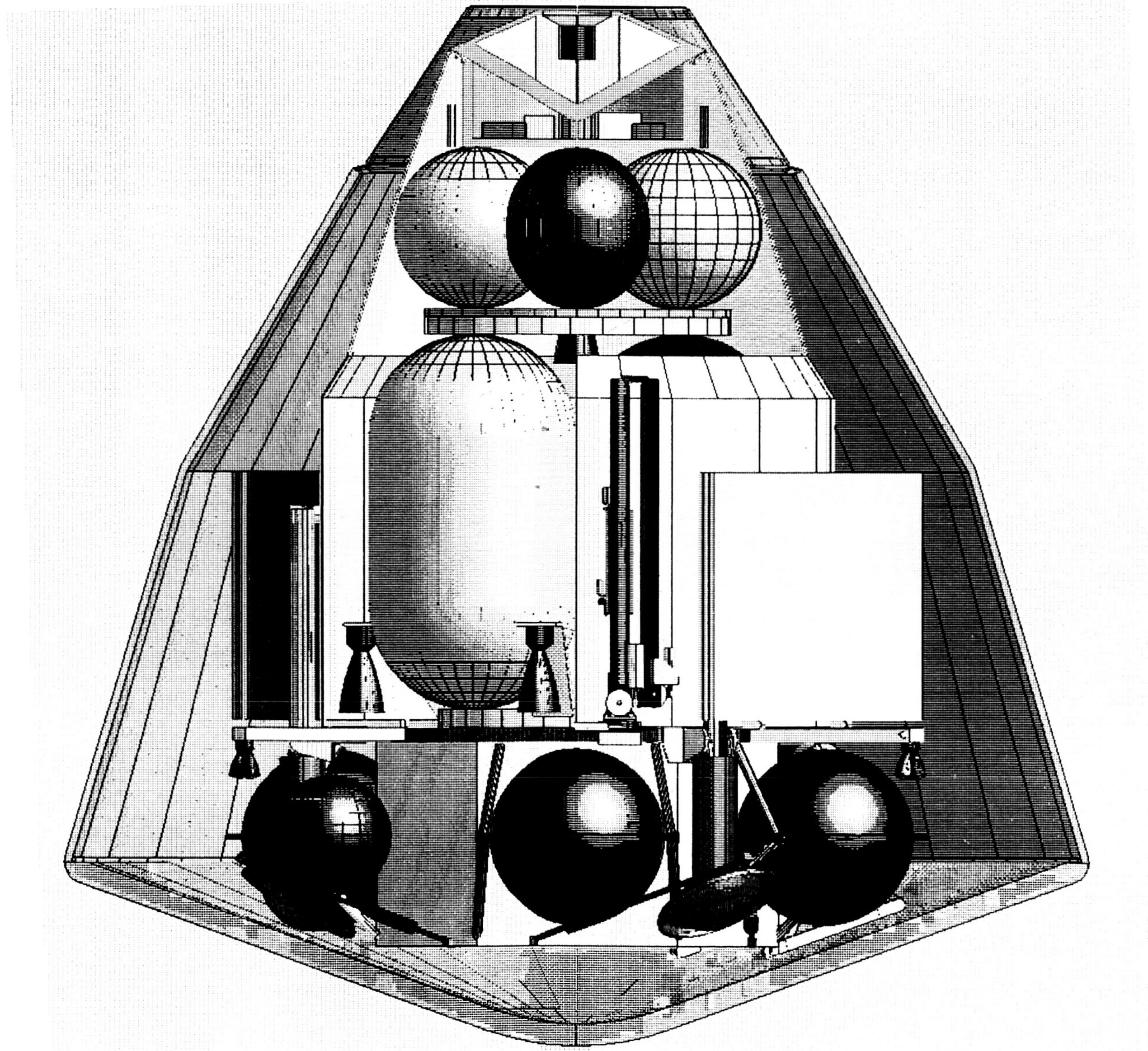

Fig. 15. MAV and Lander stowed within a Mars 98 Derived Aeroshell.

Not shown in the drawing is the Trans-Mars cruise back pack, consisting of the usual set of solar panels, antenna, star sensors, propulsion system and other avionics used to guide the capsule during its outbound trip to Mars. The trans-Mars cruise back pack is attached to the back shell of the aeroshell during outbound cruise, and feeds its data to a C&DH system contained within the lander. The back pack is expended shortly before Mars aero-entry.

Mass Breakdown of Baseline Mission

Table 4 provides a summary mass breakdown for the baseline MSR-ISPP mission. A detailed mass breakdown is provided in the appendix, pages A1-14. Rather than using a single percentage for contingency, as is sometimes done in preliminary studies, contingency masses were assigned to each subsystem on the basis of the technological maturity of that system. In the baseline system, the entire spacecraft is dual string, with one set of redundant avionics "brains" assigned to the outbound system (Trans-Mars Cruise Stage and lander) and another redundant dual avionic "brain" set assigned to the return system (MAV and ERV).

The total system mass, including contingency, is 540 kg, making it a bit too heavy to fly on a Med-Lite but providing about 85% launch margin if flown on a Delta-7925 or a Molniya launch vehicle (about 1000kg Trans-Mars Injection (TMI) capability each).

Table 4 Mass Breakdown of Baseline MSR-ISPP Mission

Item	Mass Estimate (kg)	Mass Estimate with Contingency (kg)
Sample Reentry Capsule	5.698	6.657
Trans Earth Cruise stage	21.083	24.812
MAV Stage II	10.898	12.663
MAV Stage I	48.248	55.497
Hydrogen	15.550	18.131
Lander	130.283	150.155
ISRU System	26.000	29.900
Science Payload	15.000	15.000
Mars Entry Capsule	90.480	104.152
Trans Mars Cruise Stage	48.397	55.483
Total	471.126	540.735

Mass Breakdown of Bare Bones Mission

Table 5 provides a summary mass breakdown for the "bare bones" light weight version of the MSR-ISPP mission. A detailed mass breakdown is provided in the appendix, pages B1-14. Once again contingency masses were assigned to each subsystem on the basis of the technological maturity of that system, but in this case aggressive contingency estimates were used, based upon an assumption that much of the needed light weight sub-system development would have been achieved by other programs before the commencement of the MSR program. The entire spacecraft is single string, with one set of non-redundant avionics "brains" governing both the outbound and return legs of the mission. Sample mass has also been reduced from the baseline version's 0.5 kg to 0.25 kg in order to save weight.

The total system mass, including contingency, is 444 kg, allowing it to be launched on a Med-Lite (500 kg TMI capability) with about 13% launch margin, or 122% launch margin if either a Delta 7925 or Molniya launch vehicle are used instead.

Table 5 Mass Breakdown of Bare Bones MSR-ISPP Mission

Item	Mass Estimate (kg)	Mass Estimate with Contingency (kg)
Sample Reentry Capsule	2.585	2.945
Trans Earth Cruise Stage	18.826	21.263
MAV Stage II	10.218	11.474
MAV Stage I	45.448	50.670
Hydrogen	13.544	15.224
Lander	112.459	123.292
ISRU System	26.000	28.600
Science Payload	10.000	10.000
Mars Entry Capsule	89.480	98.528
Trans Mars Cruise Stage	37.787	41.530
Total	403.284	444.204

Cost Estimating Models and Techniques

Lockheed Martin uses parametric estimating techniques to provide senior management with independent cost estimates for reviews of all major programs, both in progress and proposals. The models and techniques are in continuous use and upgrading to incorporate the best available estimating techniques, the most accurate reflection of recent Lockheed Martin experience and new ways of doing business that are being implemented on an on-going basis. We utilized the best combinations of parametric estimating techniques, bottoms-up and historical relationships to estimate the hardware/software development, delivery and launch support costs. Estimates were iterated and reconciled with the engineering team and with other similar program estimates. Where possible, more than one estimate for various end items and tasks were generated and compared; any significant differences between the estimates were reconciled. This reconciliation processes occurred at the lowest level of estimating detail. In particular, this estimate was compared and reconciled with the applicable portions of the detailed bottoms up estimates completed for the SSTI, MGS and MARS98 programs.

Models and Techniques

The hardware estimates were generated using a combination of Lockheed Martin's (formerly GE) PRICE H commercial hardware model, in-house cost estimating relationship equations and historical factors. The software was estimated using Martin Marietta's developed SASET (Software Architecture Sizing Estimating Tool). DM-Data's VLSI Cost model is used for integrated circuit estimating including ASIC design and production costs.

We have continuously refined the cost data bases we use with our own and other industry experience. We develop a detailed Master Equipment List to support estimating at the component level. The cost analyst uses the best available data from the models, procurement inputs and bottoms up and then iterates the estimates with the project technical experts to produce a very refined estimate that takes into account the specific technology and management approach for the project. This methodology produces more accurate and justifiably estimates than the traditional approach.

The PRICE H model is a commercially available computer tool capable of estimating budgets and schedules on a wide range of electronic, mechanical and electromechanical components. Properly calibrated with cost and programmatic data from previously completed contracts the model will yield accurate estimating results. The PRICE Model is the tool used to provide parametric hardware estimates on virtually all Lockheed Martin Denver programs (used on a daily basis). The PRICE Model methodology applies as an industry standard and the results and complexity values are representative of most aerospace companies. The PRICE models have been calibrated to Lockheed Martin and aerospace industry data. The model is extremely flexible. The PRICE model estimates all phases of hardware development and production and software development and coding.

The SASET model was used to estimate the software budgets and schedules. SASET was developed by Martin Marietta Denver Astronautics for the Navy and Air Force Cost Centers using actual software development data collected over a number of years. The equations and resulting software parametric model has been validated by comparing labor actuals to SASET estimates and projected schedules to actual software development time spans. The SASET generates software cost by program phase, by functional element and by discipline and produces schedules by software lifecycle phase. SASET cost estimates are based on software lines of code, complexity, schedule, operational environment, type of software, etc. SASET also provides the ability to generate an estimate of the number of lines of code that would be required to develop the system based upon the required software functions. This capability was used as a check against the engineering derived number of lines of code.

Lockheed Martin has recognized that the most important step in developing a credible parametric cost estimate on a new effort is collecting and evaluating the relevant technical data. The data should be collected at the lowest level of conceptual design detail. The estimate for Mars Sample Return Mission was developed at the component level from the Master Equipment List as derived from the conceptual design.

Assumptions

The cost estimates are based upon the mission and spacecraft design as described in this report. They are also based on the contemporary management approaches currently being practiced at Lockheed Martin. The bottom line estimate encompasses the complete life cycle costs of the mission. Minimum sparing has been included for critical and long lead component items only. The ground support equipment includes all the necessary tooling fixtures, handling fixtures, lifting slings, test equipment/sets, test fixtures, shipping container and other required equipment and accounts for items that are available.

New Ways of Doing Business

The parametric estimating data also includes Lockheed Martin's emphasis on cost reduction initiatives. One of these approaches is characterized by improved requirements derivation. The process of requirements definition and flow of these requirements to the respective disciplines at lower requirements/design definition levels will be thoroughly addressed. This avoids excessive resources (time, manpower and cost) being spent on redesign, rebuild and retest of hardware late in the lifecycle of the program. This initiative requires more systems/generalist type engineers (and more experience) in the early requirements phase of a program, resulting in a 20% increase in this type of labor costs during this phase. The benefit of this investment is approximately a 20% decrease in design time, a decrease in manufacturing time of 10% and a decrease in test time of 10% (through the life of the program). An added benefit is a nearly 50% decrease in time spent on redesign, rebuild and retest. The up front investment with a relatively small group pays large dividends in reducing costs in the highly man loaded later portions of the program.

Producibility and test issues are considered up front during the design process. This reduces the complexity, and the resulting cost and schedule time, for build/test of items. This initiative requires more production/test type support in the design phase and generally results in a 10-15% increase in labor costs for these activities. The resulting benefits are improved communication between systems/design and production/test personnel effectively reducing overall design time by 20%, manufacturing time by 10% and test time by 10% (through the life of the program). Redesign, rebuild and retest activities are also significantly decreased as the baseline design established by this process can be produced and tested essentially without modification. Since production and test issues have been addressed, tooling and test fixture complexity is reduced resulting in additional savings.

Another important initiative is testing at the appropriate level. Tests are scoped and performed at levels consistent with the "true objectives" of the test. Lower level tests focus on functionality of a circuit or box and workmanship, while higher level test focus on inter-operability and performance of subsystems at the system level, and workmanship. The investment required is increased involvement of test personnel in the design phases of a program (approximately 20% more than traditional) and increased testing at lower levels. The results are a decrease in test failures and the cost and schedule time associated with correction and retest. The net effect is an overall program total decrease in costs of over 5%.

Management is one of the most important issues to address in controlling costs. The management approach utilizes integrated product development teams (IPDT) and concurrent engineering. IPDT effectiveness is enhanced by working in the Space Technology Center (STC) which is a fully integrated and self-contained work environment that has the hardware and software to provide development, integration and documentation support to the teams. The STC is a controlled access area with all work disciplines interlinked, a voice net and electronically driven visualization center. As requirements and design are refined and documented the integrated system provides the necessary data communicated among the various disciplines and development teams. This information flow eliminates the communication complexity inherent to the design and development of complex mission hardware and software.

Cost Estimate Groundrules and Assumptions

The cost estimating data that is provided in this report is based upon the assumptions enumerated below. Additional technical assumptions are discussed in the body of this document.

Content Assumptions:

- The estimate includes the analysis, development and test of the Mars lander, the Mars and Earth transfer vehicles, the Mars entry system, the sample reentry system, ground support equipment, software, test management, payload integration and other support services.

- Protoflight development approach - engineering model development will be executed where necessary to mitigate risks and reduce costs.

- The estimate includes all management and non-engineering labor costs.

- The Systems Engineering effort represents system level analysis and trades. The System Design activity is system level preliminary design tasks.

- An evaluation of mission requirements/design is included to support spacecraft requirements/ design tasks.

- Labor costs to support the integration of the spacecraft to the launch vehicle, the launch vehicle itself, flight support and operations costs are included in cost estimate.

- JPL/JSC labor/material costs - and other appropriate monitoring and support costs have been included. Post-mission data analysis tasks are in the cost numbers.

- Heritage has been accounted for with particular emphasis on using Mars Global Surveyor and Mars 98 program design information and equipment.

- The C&DH subsystem is based upon ICAPS - it is assumed that this processor concept will be mature prior to Mars Sample Return Mission.

- The attitude control system and communication systems will use significant elements from the Mars 98 program.

- There is significant software heritage from the Mars 98 program. This software will be extensively evaluated and modified as required.

- Some support equipment will be obtained from that used on Mars Global Surveyor and Mars 98; other unique items will be designed and built by Lockheed Martin. The ground support equipment includes tooling fixtures, handling fixtures, lifting slings, test equipment/sets, test fixtures, shipping container and other required equipment.

Cost Assumptions:

- Results represent a preliminary engineering ROM cost estimate.

- Costs are provided in constant 1995 dollars through G&A

- IPD and TQM management/development approaches assumed

- A 10% manufacturing overage/spares factor was used where appropriate to insure uninterrupted development and test.

- No cost for spares included.

- The design/development costs for the methane engines and the methane production plant has been included in the estimate. Other mission components are assumed flight qualified before they are needed.

- No cost for STL (Spacecraft Test Lab) included.

- No contingency included in the Lockheed Martin hardware/software subsystem numbers. Contingency is included at the bottom line at 25%.

- No costs included for special tests and test facilities.

Mission Cost

The following pages provide a detailed breakdown of the cost estimate for the MSR-ISPP mission based upon the methodology and assumptions described above. The bottom line is that the baseline dual-string mission is estimated to have a total cost of $301 million if flown on a Delta 7925 or $258 million if flown on a Molniya. The lightweight single string mission is estimated to have a total cost of $248 million if flown on a Med-Lite or $229 million if flown on a Molniya.

MARS SAMPLE RETURN MISSION
PARAMETRIC COST SUMMARY
($'S x1000)

NAME	BASIC CONCEPT	MINIMUM COST CONCEPT
SAMPLE RETURN VEHICLE	$29,804	$27,300
POWER SYSTEM	$1,148	$1,162
COMM SYSTEM	$5,628	$3,316
GN & C SYSTEM	$3,168	$3,168
C & DH SYSTEM	$3,557	$3,304
STRUCTURES/MECH	$801	$801
PROPULSION	$2,558	$2,558
THERMAL SYSTEM	$477	$477
SRV SYSTEM DESIGN	$969	$878
ASSY & TEST	$3,428	$3,269
SRC INTEGRATION	$342	$342
SRV SOFTWARE	$1,845	$2,490
MAGE	$342	$342
EAGE	$1,359	$1,359
SYST/LOGIS/PROG MGMT	$4,183	$3,835
LANDER SYSTEM	$51,889	$36,692
POWER SUBSYSTEM	$3,701	$3,701
RF COMM SUBSYSTEM	$5,934	$767
GN&C SUBSYSTEM	$4,764	$3,100
C&DH SUBSYSTEM	$4,086	$0
THERMAL SUBSYSTEM	$1,129	$1,129
MECH & STRUCT SUBSYSTEM	$2,280	$2,280
PROPULSION SUBSYSTEM	$3,092	$3,092
SAMPLE ARM	$2,353	$2,353
SURFACE IMAGER	$556	$556
ISRU	$9,373	$9,373
SOFTWARE	$1,398	$0
PAYLOAD INTEGRATION	$783	$953
LANDER SCIENCE I/F MOD	$138	$138
LANDER SYSTEM DESIGN	$969	$627
VEH ASSY & TEST	$3,492	$2,687
MAGE	$305	$305
EAGE & TEST EQUIP	$924	$624
SYSTEMS & MGMT	$6,613	$5,008

MARS SAMPLE RETURN MISSION
PARAMETRIC COST SUMMARY
($'S x1000)

NAME	BASIC CONCEPT	MINIMUM COST CONCEPT
MAV SYSTEM	$21,501	$21,501
MAV STAGE II		
THERMAL SUBSYSTEM	$446	$446
STRUCTURE SUBSYSTEM	$414	$414
PROPULSION SUBSYSTEM	$2,678	$2,678
MAV STAGE I		
THERMAL SUBSYSTEM	$565	$565
STRUCTURE SUBSYSTEM	$948	$948
PROPULSION SUBSYSTEM	$10,043	$10,043
MAV DESIGN & TEST	$2,599	$2,599
MAGE/EAGE TEST EQUIP	$330	$330
MGMT & SYSTEMS	$3,478	$3,478
MARS ENTRY CAPSULE	$2,229	$2,229
STRUCTURE SUBSYSTEM	$1,207	$1,207
THERMAL SUBSYSTEM	$666	$666
SYSTEMS/MGMT/SUPT	$355	$355
REENTRY CAPSULE	$1,275	$930
POWER SYSTEM	$134	$0
COMM SYSTEM	$136	$0
STRUCTURES/MECH	$431	$395
THERMAL SUBSYSTEM	$185	$185
INTEG & TEST	$158	$158
MAGE	$14	$14
SYST/LOGIS/PROG MGMT	$217	$178

MARS SAMPLE RETURN MISSION
PARAMETRIC COST SUMMARY
($'S x1000)

NAME	BASIC CONCEPT	MINIMUM COST CONCEPT
TRANS MARS CRUISE STAGE	$12,173	$12,173
POWER SYSTEM	$1,061	$1,061
COMM SYSTEM	$410	$410
ACS SUBSYSTEM	$2,291	$2,291
STRUCTURE SUBSYSTEM	$760	$760
PROPULSION SUBSYSTEM	$2,794	$2,794
THERMAL SUBSYSTEM	$411	$411
SYSTEM DESIGN	$741	$741
ASSY & TEST	$2,055	$2,055
SYST/LOGIS/PROG MGMT	$1,651	$1,651
LAUNCH OPS/MISSION OPS	$1,072	$1,072
MISSION OPS	$1,072	$1,072
SPECIALTY TESTS		
STERILIZATION		
TOTAL MSRM HARDWARE	$119,942	$101,897

MARS SAMPLE RETURN MISSION
PARAMETRIC COST SUMMARY
($'S x1000)

NAME	MSRM BASIC CONCEPT	MINIMUM COST CONCEPT
EARTH RETURN VEH. - ERV		$27,300
LANDER SYSTEM	$51,889	$36,692
MAV SYSTEM	$21,501	$21,501
MARS ENTRY CAPSULE	$2,229	$2,229
RETURN CAPSULE	$1,275	$930
TRANS MARS CRUISE STAGE	$12,173	$12,173
MISSION OPS	$1,072	$1,072
LAUNCH OPS/STACK	$2,484	$2,285
TOTAL MSRM HARDWARE	$122,426	$104,182
PROFIT & CAS @10%	$12,243	$10,418
GRAND TOTAL (GT)	$134,669	$114,600
JPL/JSC MONITOR (@10%)	$13,467 - BASE GT	$11,460
JPL/JSC BURDEN (@8.5%)	$11,447 - BASE GT	$9,741
S/C TOTAL COSTS	$159,583	$135,801
MISSION DESIGN (@2.5%)	$3,367 - BASE GT	$2,865
MOS DEVELOPMENT (@3.5%)	$4,713 - BASE GT	$4,011
PROJECT MGMT (@5%)	$6,733 - BASE GT	$5,730
PROJECT RESERVES (@25%)	$43,599 - SUMx25%	$37,102
PREPROJECT EXPENSES (@2%)	$4,360	$3,710
TOTAL DEVELOPMENT COSTS	$222,355	$189,219
LAUNCH COSTS	$58,000	$34,000
MO&DA COSTS & DSN COSTS	$17,000	$17,000
MO&DA RESERVES (@25%)	$4,250	$4,250
TOTAL POST LAUNCH COSTS	$79,250	$55,250
TOTAL COSTS (US LAUNCH)	$301,605	$244,469
MOLNIYA ALTERNATIVE OPTION		
MOLNIYA LAUNCH COSTS	$15,000	$15,000
TOTAL COSTS (MOLNIYA LAUNCH)	$258,605	$225,469

Summary of Cost Reduction Strategy

The cost estimates given above indicate that the MSR-ISPP mission design presented in this study are among the lowest cost options for a Mars Sample Return mission that have ever been presented. it is useful to review the means by which these savings have been achieved.

1. The use of ISPP to provide Mars ascent and Earth return propulsion.

The use of ISPP gives this mission great mass leverage, as a combination of 16 kilograms of hydrogen feedstock with a 30 kilogram ISPP plant and 10 extra kg of solar panels are used to produce 277 kilograms of propellant on the surface of Mars. This net gain of 221 kg of landed payload is by itself larger than the entire payload that can be delivered to the Martian surface by a dedicated Delta launch. Using this leverage, mission launch mass is greatly reduced, the need for a Mars orbiter spacecraft and the development of the capability for an autonomous Mars orbit rendezvous and dock maneuver are eliminated.

2. The use of a Mars Surveyor Lander and aeroshell

The reduction in mission mass enabled by ISPP allows the required mass to conduct a direct-return Mars Sample Return mission to be delivered to the Martian surface by a Mars Surveyor Lander and aeroshell system. This lander and aeroshell are being developed by Lockheed Martin to support JPL's Mars 98 mission, and so the development cost for a new lander and aeroshell to support the MSR mission are greatly reduced.

3. The use of a new type of Sample Return Capsule

The new SRC design presented in this study is lightweight and cheap, and can function without any electronics, instruments, or pyro devices.

4. The use of new lightweight avionics

Lightweight avionics, including ICAPS and light weight telecommunications systems are currently being developed for use in the Mars Surveyor, Discovery, and New Millennium programs. By taking advantage of these developments, much non-recurring cost is eliminated. Also, together with the new lightweight SRC design, the use of such light weight avionics reduces the propellant production requirements and thus the power requirements of the ISPP system to the point where solar energy can be used for power. This eliminates the need to carry an RTG on the mission.

5. The use of new means of doing business

In accord with NASA's "faster-cheaper-better" strategy, the burdens, overheads, mission operations costs and reserves employed in this mission are lower than those generally used in the past.

6. The dropping of extraneous requirements

While the proposed mission design does as good a job as any in assuring sample preservation and preventing back contamination, the dropping of any formal requirements in these areas eliminates the need for costly engineering work to deal with an endless number of speculative "what-if" possibilities.

7. The use of international capabilities

Significant cost savings can accrue to the mission by using a Russian Molniya for launch.

By combining all of these advantages, the MSR-ISPP design offers the possibility of a Mars Sample Return mission whose costs approach the Discovery class.

Conclusions

We have presented two designs for a Mars Sample Return mission utilizing in-situ propellant production. The dual string system design returns a 0.5 kg sample can be flown for an estimated cost of $302 million using a Delta 7925, or $259 million, including launch and mission operations, if a Russian Molniya launch vehicle is used. The single string mission returns a 0.25 kg sample and can be flown for an estimated cost of $244 million on a Med-Lite or $225 million on a Molniya, including launch and mission operations. While no formal risk analysis has been done, it seems clear that the large reduction in risk associated with the dual string design more than justifies its 15% increase in cost over the single string option. We therefore recommend the dual string system as the preferred option.

In the course of this study we encountered no technical show-stoppers for an MSR-ISPP mission. In particular, a solution was found to the most troublesome problem associated with the Sabatier/electrolysis MSR-ISPP option. The new approach, shipping the hydrogen to Mars as low temperature supercritical gas, keeping it cool with the ISPP system refrigerators, allows for margins of about 600% in error in heat leak calculation during transport, effectively putting the problem to rest. All other required systems, including the ISPP system and the lightweight avionics were found to be credible for use in a relatively near-term mission. A major political concern was also eliminated when it was found possible to support the ISPP process with photovoltaics on the surface of Mars, instead of the RTG previously deemed necessary. Using the mass leverage of the ISPP system, the need for a Mars orbiter and Mars orbit rendezvous generally necessary in non-ISPP missions was eliminated, and it was found possible to land the complete Mars Ascent and Earth Return Vehicle on Mars using a Mars Surveyor lander. These features resulted in very large cost savings to the mission.

On the basis of the present study it appears that in-situ propellant production can enable a Mars Sample Return mission whose costs approach the Discovery class. This is an extremely exciting prospect. It is strongly recommended that this mission be studied further as a possibility for launch in the 2001 to 2005 timeframe.

Acknowledgments

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APPENDIX A

Low Cost MISR Baseline Concept Master Equipment List

Sample Reentry Capsule (SRC)

Component	Units	Mass /unit (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Prog. Common.	Descrip/Comments
Power and Cabling										
Battery	2	0.025	0.050	2	7.0	0.054		Tadiran		Li Thionyl Chloride Primary Cell Battery
Cabling	1	0.010	0.010	4	15.0	0.012		MMC/Nanonics		
Subtotal			0.060		8.3	0.065				
Telecom										
Argos/Beacon Transmitter	2	0.010	0.020	4	15.0	0.023		various		
Antenna	2	0.010	0.020	4	15.0	0.023		various		
Subtotal			0.040		15.0	0.046				
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Heaters	4	0.006	0.024	2	7.0	0.026	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	2	0.006	0.012	1	5.0	0.013	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	4	0.013	0.052	1	5.0	0.055	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
MLI	1	0.050	0.050	5	20.0	0.060		Sheldahl	MSP	sized to fit battery and beacon
Subtotal			0.138		10.8	0.153				
Structures/Mechanisms										
Mechanisms										
Bio bag										
Structure										
Heat Shield	1	0.500	0.500	5	20.0	0.600	60 D	MMC	PF/MSP	
Head Shield TPS	1	2.140	2.140	5	20.0	2.568		MMC	PF/MSP	3.8 cm thck, .25 gm/cm, SLA561
Back Shell	1	0.300	0.300	5	20.0	0.360		MMC	PF/MSP	
Back Shell TPS	1	0.830	0.830	5	20.0	0.996		MMC	PF/MSP	1.0 cm thick, SLA 561S
Crushable Material	1	0.790	0.790	3	10.0	0.869				
Sample Canister Retention	1	0.200	0.200	5	20.0	0.240		MMC		

Component	Units	Mass /unit (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Prog. Common.	Descrip/Comments
Subtotal			4.760		18.3	5.633				
Propulsion										
Subtotal										
Science / Payload										
Sample Canister	1	0.200	0.200	6	30.0	0.260		MMC		
Sample	1	0.500	0.500	0	0.0	0.500		Mars		
Subtotal			0.700		8.6	0.760				
Summary										
Power and Cabling			0.060			0.065				
Telecom			0.040			0.046				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			0.138			0.153				
Structures/Mechanisms			4.760			5.633				
Propulsion			0.000			0.000				
Science/Payload			0.700			0.760				
Flight Element Total			5.698		16.8	6.657				

Trans Earth Cruise Stage (TECS)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Batteries	1	0.900	0.900	5	20.0	1.080		Eagle-Picher		Li-Ion Secondary Cell, 80 whrs, 90 whrs/kg
Solar Array Cells	1	1.280	1.280	5	20.0	1.536	.88 m2	Spectro-Lab		80 watts at Mars, 18% eff., 95% EPD eff.
Power Electronics										
Power Dist. Module	2	0.101	0.202	5	20.0	0.242	7.6 x 12.7	MMC		Part of Integr. Comp. and Power System (ICAPS)
Power Converter Module	2	0.109	0.218	5	20.0	0.262	7.6 x 12.7	MMC		Part of ICAPS
Pyro Initiation Module	2	0.072	0.144	5	20.0	0.173	7.6 x 12.7	MMC		Part of ICAPS
Segment Switch Module	2	0.101	0.202	5	20.0	0.242	7.6 x 12.7	MMC		Part of ICAPS
Cabling	1	1.600	1.600	6	30.0	2.080	—	MMC/Nanonics		approx. 8% of flight element dry total masss
Subtotal			4.546		23.5	5.615				
Telecom										
Transponder	2	0.500	1.000	6	30.0	1.300	7.6x12.7x3.8	—		X-Band
CDU/TMU Module	2	0.072	0.144	5	20.0	0.173	7.6x12.7	MMC		Part of ICAPS
Diplexer	2	0.200	0.400	3	10.0	0.440		Loral		
Patch Antenna	2	0.060	0.120	2	7.0	0.128		MMC		874C0401300
Medium Gain Antenna	1	0.300	0.300	3	10.0	0.330		Boeing		
RF Switch	2	0.200	0.400	2	7.0	0.428				
SSPA	1	0.400	0.400	6	30.0	0.520	7.6x12.7x3.8	—		X-Band
Waveguides, filters, couplers	1	0.900	0.900	6	30.0	1.170				
Subtotal			3.664		22.5	4.489				
AD&C										
Star Sensor	2	0.600	1.200	4	15.0	1.380	5.3x5.3x11	OCA Applied Optics	Clementine	
Digital Sun Sensor, Head	2	0.250	0.500	1	5.0	0.525	8.1x8.1x2.0	Adcole	Diverse	Model 29970
Digital Sun Sensor, Electro.	1	0.640	0.640	1	5.0	0.672	11.4x12.4x5.4	Adcole	Diverse	Model 29720
IMU, Incl. Accelerometers	1	0.650	0.650	2	7.0	0.696		Litton		FOG
Subtotal			2.990		9.4	3.273				
C&DH										
Processor	2	0.130	0.260	5	20.0	0.312	7.6x12.7	MMC		Part of ICAPS
I/O	2	0.072	0.144	5	20.0	0.173	7.6x12.7	MMC		Part of ICAPS
Subtotal			0.404		20.0	0.485				
Thermal Control										
Heaters	12	0.006	0.072	2	7.0	0.077	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	12	0.006	0.072	1	5.0	0.076	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	12	0.013	0.156	1	5.0	0.164	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
MLI	1	0.500	0.500	5	20.0	0.600		Sheldahl	MSP	
Subtotal			0.800		14.6	0.916				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	4	0.120	0.480	3	10.0	0.528	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
Solar Array Deploy	2	0.200	0.400	1	5.0	0.420	2.5x6x6	StarSys	MSP	SH-9010

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Ordinance Initiators	8	0.010	0.080	1	5.0	0.084	4Dx6	Hi-Shear	MSP	
SRC Sep Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	MSP	9421-2
Structure										
Solar Array Substrate	2	0.500	1.000	4	15.0	1.150		MMC	MSP,MGS	
SRV Structure	1	3.000	3.000	5	20.0	3.600		MMC		
Misc Ftgs and Brkts	1	0.500	0.500	5	20.0	0.600		MMC		
Subtotal			5.960		15.9	6.907				
Propulsion										
N2H4 Tank w/ PMD	1	0.400	0.400	3	10.0	0.440	Cyl. 15 x 25	MMC		Vane Type PMD
N2H4 Lines, Misc	1	0.350	0.350	4	15.0	0.403		MMC		
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Filters	1	0.050	0.050	2	7.0	0.054		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	1	0.040	0.040	3	10.0	0.044		Valvetech		
GHe Tank	1	0.212	0.212	3	10.0	0.233		SCI		
Pyro Valve	1	0.059	0.059	2	7.0	0.063		Pyronetics		Redundant Bridgewire
Regulator	1	0.023	0.023	4	15.0	0.026		Futurecraft		
Check Valve / Burst Disk	1	0.045	0.045	3	10.0	0.050		Valvetech		
N2H4 Thruster	8	0.160	1.280	5	20.0	1.536	1 x 9	Rocket Research		5N, MR-133 mod
Dry Subtotal			2.719		15.0	3.126				
Science / Payload										
Sample Return Capsule			5.698		16.8	6.657				
Subtotal			5.698		16.8	6.657				
Summary										
Power and Cabling			4.546			5.615				
Telecom			3.664			4.489				
AD&C			2.990			3.273				
C&DH			0.404			0.485				
Thermal Control			0.800			0.916				
Structures/Mechanisms			5.960			6.907				
Propulsion			2.719			3.126				
Flt Element Subtotal (Dry)			21.083		17.7	24.812				
Science/Payload			5.698			6.657				
Element Total (Dry)			26.781		17.5	31.469				
GHe Pressurant	1	0.034	0.034	2	7.0	0.036	0.000			
N2H4 Propellant			2.356			2.769				TCM 120 m/sec + 60 m/sec ascent, Isp 222, 2% trapped
Wet Total			29.171		17.5	34.273				

Mars Ascent Vehicle, Stg II (MAV II)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	0.350	0.350	4	15.0	0.403		MMC/Nanonics		3% of dry mass
Subtotal			0.350		15.0	0.403				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Heaters	6	0.006	0.036	2	7.0	0.039	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	6	0.006	0.036	1	5.0	0.038	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	6	0.013	0.078	1	5.0	0.082	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Plume Shield	1	0.300	0.300	5	20.0	0.360				
MLI	1	2.000	2.000	5	20.0	2.400		Sheldahl	MSP	
Subtotal			2.450		19.1	2.918				
Structures/Mechanisms										
Mechanisms										
Structure										
MAV Stg II Structure	1	1.500	1.500	6	30.0	1.950		MMC		
Misc Ftgs and Brkts	1	0.500	0.500	6	30.0	0.650		MMC		
Subtotal			2.000		30.0	2.600				
Propulsion										
CH4 Tanks	2	1.000	2.000	3	10.0	2.200		Arde, Inc.		Modified Model E3899
O2 Tanks	2	1.200	2.400	1	5.0	2.520		Arde, Inc.		Model E3899
Lines, Misc	1	0.300	0.300	1	5.0	0.315		MMC		
Fill and Drain Valves	4	0.020	0.080	1	5.0	0.084		Pyronetics		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Propellant Filter	2	0.050	0.100	2	7.0	0.107		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	10.0	0.088		Valvetech		
CH4/O2 Engine	1	0.750	0.750	6	30.0	0.975		—		150 lbf thrust
Pressure Regulator	2	0.023	0.046	4	15.0	0.053		Futurecraft		
Temperature Sensors	2	0.006	0.012	2	7.0	0.013		Yellowfin		
Coolant Loop	1	0.150	0.150	6	30.0	0.195		MMC		
Dry Subtotal			6.098		10.6	6.742				
Science / Payload										
Sample Return Vehicle			29.171		17.5	34.273				
Subtotal			29.171		17.5	34.273				
Summary										
Power and Cabling			0.350			0.403				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			2.450			2.918				
Structures/Mechanisms			2.000			2.600				
Propulsion			6.098			6.742				
Flt Elem Subtotal (dry)			10.898		16.2	12.663				
Science/Payload			29.171			34.273				wet payload total
Flt Elem Total (dry)			40.069		17.1	46.937				
CH4 Propellant			12.842			15.043				ΔV = 3284 m/sec, ISP = 380, 3.5/1
O2 Propellant			45.005			52.718				ΔV = 3284 m/sec, ISP = 380, 3.5/1
Flt Elem Total (wet)			97.916		17.1	114.697				

Mars Ascent Vehicle, Stg I (MAV I)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	1.400	1.400	5	20.0	1.680		MMC/Nanonics		3% of dry mass
Subtotal			1.400		20.0	1.680				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Heaters	10	0.006	0.060	2	7.0	0.064	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	10	0.006	0.060	1	5.0	0.063	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	10	0.013	0.130	1	5.0	0.137	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Plume Shield	1	1.000	1.000	5	20.0	1.200				
MLI	1	5.000	5.000	5	20.0	6.000		Sheldahl		
Subtotal			6.250		19.4	7.464				
Structures/Mechanisms										
Mechanisms										
Stg II Sep Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	diverse	
Cable Cutters	1	0.200	0.200	2	7.0	0.214	1.9Dx7.4	Hi-Shear	diverse	Model SL1063
Tube Cutters	8	0.200	1.600	2	7.0	1.712				
Structure										
MAV Stg I Structure	1	4.000	4.000	6	30.0	5.200		MMC		
Misc Ftgs and Brkts	1	1.000	1.000	6	30.0	1.300		MMC		
Ascent Fairing	1	2.000	2.000	6	30.0	2.600		MMC		
Subtotal			9.300		24.2	11.551				
Propulsion										
CH4 Tanks	2	6.700	13.400	3	10.0	14.740		SCI		Graphite Overwrap, 2000 psi
O2 Tanks	2	6.700	13.400	3	10.0	14.740		SCI		Graphite Overwrap, 2000 psi
Lines, Misc	1	0.500	0.500	5	20.0	0.600		MMC		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Propellant Filter	2	0.100	0.200	2	7.0	0.214		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	10.0	0.088		Valvetech		
CH4/O2 Engine	4	0.750	3.000	5	20.0	3.600		—		150 lbf thrust
Pressure Regulator	2	0.023	0.046	2	7.0	0.049		Futurecraft		
Temperature Sensors	2	0.006	0.012	2	7.0	0.013		Yellowfin		
Coolant Loop	1	0.400	0.400	5	20.0	0.480		MMC		
Dry Subtotal			31.298		11.2	34.802				
Science / Payload										
MAV Stg II										
Earthbound			97.916			114.697				
Marsbound			40.069			46.937				
Earthbound Subtotal			97.916		17.1	114.697				
Marsbound Subtotal			40.069		17.1	46.937				
Summary										
Power and Cabling			1.400			1.680				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			6.250			7.464				
Structures/Mechanisms			9.300			11.551				
Propulsion			31.298			34.802				
Flt Elem Subtotal (dry)			48.248			55.497				
Science/Payload										
Earthbound			97.916			114.697				
Marsbound			40.069			46.937				
Flt Elem Total (Earthbound)			146.164		16.4	170.194				
Flt Elem Total (Marsbound)			88.317		16.0	102.433				
CH4 Propellant			43.647		16.4	50.823				ΔV = 3132 m/sec, ISP 380
O2 Propellant			152.962		16.4	178.110				ΔV = 3132 m/sec, ISP 380
Flt Elem Total Wet (Earth)			342.773		16.4	399.128				
Hydrogen			15.550			18.131				
Flt Elem Total Wet (Mars)			103.867		16.1	120.565				

Lander

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Battery	1	2.400	2.400	4	15.0	2.760		Eagle-Picher		Li-Ion Secondary, 215 whrs
Solar Array Cells	4	2.850	11.400	4	15.0	13.110	8 m^2	Spectro-Lab		750 watts, 18% eff., 95% EPD
Power Electronics										
Power Distribution Mod.	2	0.101	0.202	5	20.0	0.242	7.6x12.7	MMC		Part of ICAPS
Power Converter Module.	2	0.109	0.218	5	20.0	0.262	7.6x12.7	MMC		Part of ICAPS
Pyro Initiation Module	2	0.072	0.144	5	20.0	0.173	7.6x12.7	MMC		Part of ICAPS
Segment Switching Module	2	0.101	0.202	5	20.0	0.242	7.6x12.7	MMC		Part of ICAPS
Cabling	1	10.000	10.000	5	20.0	12.000		MMC/Nanonics		Approx. 8% of flt elem dry mass
Subtotal			24.566		17.2	28.789				
Telecom										
Transponder	2	0.500	1.000	6	30.0	1.300	7.6x12.7x3.8	—		X-band
CDU/TMU Module	2	0.072	0.144	5	20.0	0.173	7.6x12.7	MMC		Part of ICAPS
Diplexer	2	0.200	0.400	3	10.0	0.440		Loral		
Medium Gain Antenna	1	0.300	0.300	3	10.0	0.330				
RF Switch	2	0.200	0.400	2	7.0	0.428				
SSPA	1	0.400	0.400	6	30.0	0.520	7.6x12.7x3.8			X-band
Wave Guides, filters, couplers	1	0.900	0.900	6	30.0	1.170				
UHF Transceiver	1	1.200	1.200	5	20.0	1.440				
UHF Antenna	1	0.100	0.100	5	20.0	0.120				
Subtotal			4.844		22.2	5.921				
AD&C										
Term. Descent Landing Radar	1	4.800	4.800	3	10.0	5.280		Honeywell	MSP	
IMU, Incl. Accelerometers	2	0.650	1.300	2	7.0	1.391		Litton		FOG
Subtotal			6.100		9.4	6.671				
C&DH										
Processor	2	0.130	0.260	5	20.0	0.312	7.6x12.7	MMC		Part of ICAPS
I/O	2	0.072	0.144	5	20.0	0.173	7.6x12.7	MMC		Part of ICAPS
Subtotal			0.404		20.0	0.485				
Thermal Control										
Heaters	36	0.006	0.216	2	7.0	0.231	5 x 20 typ	Tayco	MSP	sized to fit
Heat Pipe Loops	2	1.500	3.000	3	10.0	3.300		Dynatherm	MSP	
Heat Pipe, Var. Cond.	2	1.000	2.000	3	10.0	2.200		Dynatherm	MSP	
Thermistors	36	0.006	0.216	1	5.0	0.227	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	36	0.013	0.468	1	5.0	0.491	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Radiator	2	1.500	3.000	4	15.0	3.450	.2m^2	MMC	MSP	
MLI	1	2.500	2.500	5	20.0	3.000		Sheldahl	MSP	
Subtotal			11.400		13.2	12.899				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	8	0.120	0.960	3	10.0	1.056	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
MGA R&R	2	0.120	0.240	3	10.0	0.264	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
Solar Array Deploy	12	0.290	3.480	1	5.0	3.654	2.5x6x6	StarSys	MSP	SH-9010
Physical Surface Sensor	3	0.200	0.600	5	20.0	0.720		MMC	MSP	
Legs	3	4.490	13.470	5	20.0	16.164		MMC	MSP/Vkg	

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Cable Cutter	1	0.200	0.200	2	7.0	0.214	1.9Dx7.4	Hi-Shear	MSP	SL1063
Tube Cutter	8	0.200	1.600	2	7.0	1.712			MSP	
Ordinance Initiators	40	0.010	0.400	1	5.0	0.420	4Dx6	Hi-Shear	Diverse	
Structure										
Solar Array Substrate	4	1.780	7.120	4	15.0	8.188		MMC	MSP,MGS	
Instrument Deck	1	7.160	7.160	4	15.0	8.234		MMC	MSP	
Component Deck	1	3.350	3.350	4	15.0	3.853		MMC	MSP	
Equip Enclosure	1	8.460	8.460	4	15.0	9.729		MMC	MSP	
Prop. Tank Support	2	1.480	2.960	4	15.0	3.404		MMC	MSP	
SA Stow Ftgs	4	0.250	1.000	5	20.0	1.200		MMC	MSP	
Leg Mounts	3	1.910	5.730	5	20.0	6.876		MMC	MSP	
Misc Ftgs & Brkts	1	3.000	3.000	5	20.0	3.600		MMC	MSP	
Subtotal			59.730		16.0	69.288				
Propulsion										
N2H4 Tank w/ PMD	2	5.900	11.800	2	7.0	12.626		PSI	MSP	80358-1
N2H4 Lines, Misc	1	2.500	2.500	4	15.0	2.875		MMC	MSP	
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Filters	2	0.100	0.200	2	7.0	0.214		Vacco		
Pressure Transducer	4	0.090	0.360	2	7.0	0.385		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	10.0	0.088		Valvetech		
GHe Tank	1	0.212	0.212	3	10.0	0.233		Brunswich	MSP	220088-1
Pyro Valve	1	0.059	0.059	2	7.0	0.063		Pyronetics		
Regulator	1	0.023	0.023	4	15.0	0.026		Futurecraft		
Check Valve / Burst Disk	1	0.045	0.045	3	10.0	0.050		Valvetech		
Descent Eng. Mod. / Roll	2	0.840	1.680	5	20.0	2.016		Rocket Research	MSP,MO	
Descent Eng. Mod. / Pitch,Yaw	4	1.550	6.200	5	20.0	7.440		Rocket Research	MSP,MO	
Dry Subtotal			23.239		12.3	26.102				
Science / Payload										
MAV Stg I	1		103.867			120.565				
Surface Science	1	15.000	15.000	0	0.0	15.000				
ISRU System	1	26.000	26.000	4	15.0	29.900				
Subtotal			144.867		14.2	165.465				
Summary										
Power and Cabling			24.566			28.789				
Telecom			4.844			5.921				
AD&C			6.100			6.671				
C&DH			0.404			0.485				
Thermal Control			11.400			12.899				
Structures/Mechanisms			59.730			69.288				
Propulsion			23.239			26.102				
Flt Element Subtotal (dry)			130.283			150.155				
Science/Payload			144.867			165.465				Wet Payload
Flight Element Total (Dry)			275.150		14.7	315.620				
N2H4 Propellant			56.648			64.980				ΔV 400 m/s, ISP 222 sec
GHe Pressurant			0.450			0.500				
Wet Subtotal			332.249		14.7	381.100				

Mars Entry Capsule (MEC)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	0.200	0.200	4	15.0	0.230		MMC/Nanonics		
Subtotal			0.200		15.0	0.230				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Subtotal										
Structures/Mechanisms										
Mechanisms										
HS Sep Device	1	4.000	4.000	4	15.0	4.600		MMC	PF,MSP	818A1410326
BS Sep Device	1	2.500	2.500	4	15.0	2.875		MMC	PF, MSP	818A1410326
BS Sep Rails	1	2.000	2.000	5	20.0	2.400		MMC		
Structure										
Heat Shield	1	23.900	23.900	4	15.0	27.485		MMC	PF, MSP	
Head Shield TPS	1	20.300	20.300	4	15.0	23.345		MMC	PF, MSP	SLA-561
Back Shell	1	25.000	25.000	4	15.0	28.750		MMC	PF, MSP	

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Back Shell TPS	1	7.000	7.000	4	15.0	8.050		MMC	PF, MSP	SLA-561S
Lander Support Structure	6	0.430	2.580	4	15.0	2.967		MMC	PF, MSP	
Ballast	1	2.000	2.000	4	15.0	2.300				
Misc Ftgs & Brkts	1	1.000	1.000	4	15.0	1.150		MMC		
Subtotal			90.280		15.1	103.922				
Propulsion										
Dry Subtotal										
Science / Payload										
Lander			332.249			381.100				
Subtotal			332.249		14.7	381.100				
Summary										
Power and Cabling			0.200			0.230				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			0.000			0.000				
Structures/Mechanisms			90.280			103.922				
Propulsion			0.000			0.000				
Flt Elem. Subtotal (dry)			90.480			104.152				
Science/Payload			332.249			381.100				
Flight Element Total			422.729		14.8	485.252				

Trans Mars Cruise Stage (TMCS)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Solar Array Cells	1	1.600	1.600	4	15.0	1.840	1.1 m^2	Spectro-Lab		100 watts at Mars, 18% effic., 95% EPD eff.
Cabling	1	1.800	1.800	5	20.0	2.160				8% of flight element dry weight
Subtotal			3.400		17.6	4.000				
Telecom										
Patch Antenna	3	0.060	0.180	2	7.0	0.193		MMC	Diverse	874C0401300
Medium Gain Antenna	1	0.300	0.300	2	7.0	0.321		Boeing	MSP	
RF Switch	1	0.200	0.200	2	7.0	0.214				
Subtotal			0.680		7.0	0.728				
AD&C										
Star Sensor	2	0.600	1.200	4	15.0	1.380	5.3x5.3x11	OCA Applied Optics	MSP	
Digital Sun Sensor, Head	2	0.250	0.500	1	5.0	0.525	8.1x8.1x2.0	Adcole	Diverse	Model 29970
Digital Sun Sensor, Electro.	1	0.640	0.640	1	5.0	0.672	11.4x12.4x5.4	Adcole	Diverse	Model 29720
Subtotal			2.340		10.1	2.577				
C&DH										
Subtotal										
Thermal Control										
Heaters	12	0.006	0.072	2	7.0	0.077	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	12	0.006	0.072	1	5.0	0.076	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	12	0.013	0.156	1	5.0	0.164	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
MLI	1	1.000	1.000	5	20.0	1.200		Sheldahl	MSP	
Subtotal			1.300		16.6	1.516				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	4	0.120	0.480	3	10.0	0.528	2.5Dx5	Fokker/Hi-Shear	MSP	
Solar Array Deploy	2	0.250	0.500	1	5.0	0.525	2.5x6x6	StarSys	MSP	
Ordinance Initiators	8	0.010	0.080	1	5.0	0.084	4Dx6	Hi-Shear	MSP	
MEC Sep. Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	MSP	9421-2
Structure										
Solar Array Substrate	2	0.500	1.000	4	15.0	1.150		MMC	MSP	
Cruise Stage Structure	1	4.000	4.000	5	20.0	4.800		MMC		
Misc Ftgs and Brkts	1	1.000	1.000	5	20.0	1.200		MMC		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Subtotal			7.560		16.6	8.812				
Propulsion										
N2H4 Tank w/ PMD	2	2.230	4.460	3	10.0	4.906		PSI	GPS II	
N2H4 Lines, Misc	1	0.500	0.500	5	20.0	0.600		MMC		
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Propellant Filter	2	0.050	0.100	2	7.0	0.107		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	10.0	0.088		Valvetech		
GHe Tank	1	0.300	0.300	3	10.0	0.330		SCI		
Pyro Valve	1	0.059	0.059	2	7.0	0.063		Pyronetics		redundant bridgewire
Regulator	1	0.023	0.023	4	15.0	0.026		Futurecraft		
Check Valve / Burst Disk	1	0.045	0.045	3	10.0	0.050		Valvetech		
N2H4 Thruster, Pitch - Yaw	4	0.160	0.640	5	20.0	0.768	1x9	Rocket Research		5N, MR-133 mod
N2H4 Thruster, Roll	4	0.160	0.640	5	20.0	0.768	1x9	Rocket Research		5N, MR-133 mod
Dry Subtotal			7.107		12.3	7.984				
Science / Payload										
Mars Entry Capsule			422.729			485.252				
Subtotal			422.729		14.8	485.252				
Summary										
Power and Cabling			3.400			4.000				
Telecom			0.680			0.728				
AD&C			2.340			2.577				
C&DH			0.000			0.000				
Thermal Control			1.300			1.516				
Structures/Mechanisms			7.560			8.812				
Propulsion			7.107			7.984				
Flt Element Subtotal (dry)			22.387			25.617				
Science/Payload			422.729			485.252				
Flight Element Total			445.116		14.8	510.869				
GHe Pressurant	1	0.250	0.250	5	20.0	0.300				
N2H4 Propellant			25.760			29.566				ΔV 120 m/sec, ISP 222 sec
Wet Subtotal			471.126		14.8	540.735				

Maturity Index Table

Maturity Index	Contingency
0	0
1	5
2	7
3	10
4	15
5	20
6	30
7	40
8	50
9	75
10	100

Abbreviations

MGS	Mars Global Surveyor (Mars '96)
MMC	Martin Marietta (Lockheed Martin)
MSP	Mars Surveyor Program (Mars '98)
PF	MESUR Pathfinder
Vkg	Viking

APPENDIX B

Low Cost MISR Minimum Cost (bare bones) Concept Master Equipment List

Sample Reentry Capsule (SRC)

Component	Units	Mass /unit (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Prog. Common.	Descrip/Comments
Power and Cabling										
Subtotal			0.000		####	0.000				
Telecom										
Subtotal			0.000		####	0.000				
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Subtotal			0.000		####	0.000				
Structures/Mechanisms										
Mechanisms										
Radar Reflector	4	0.050	0.200	5	15.0	0.230				
Structure										
Heat Shield	1	0.250	0.250	5	15.0	0.288	60 D	MMC	PF/MSP	
Head Shield TPS	1	1.070	1.070	5	15.0	1.231		MMC	PF/MSP	3.8 cm thck, .25 gm/cm, SLA561
Back Shell	1	0.150	0.150	5	15.0	0.173		MMC	PF/MSP	
Back Shell TPS	1	0.415	0.415	5	15.0	0.477		MMC	PF/MSP	1.0 cm thick, SLA 561S
Crushable Material	1	0.400	0.400	3	8.0	0.432				
Sample Canister Retention	1	0.100	0.100	5	15.0	0.115		MMC		
Subtotal			2.585		13.9	2.945				
Propulsion										

Component	Units	Mass /unit (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Prog. Common.	Descrip/Comments
Subtotal										
Science / Payload										
Sample Canister	1	0.100	0.100	6	20.0	0.120		MMC		
Sample	1	0.250	0.250	0	0.0	0.250		Mars		
Subtotal			0.350		5.7	0.370				
Summary										
Power and Cabling			0.000			0.000				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			0.000			0.000				
Structures/Mechanisms			2.585			2.945				
Propulsion			0.000			0.000				
Science/Payload			0.350			0.370				
Flight Element Total			2.935		12.9	3.315				

Trans Earth Cruise Stage (TECS)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Batteries	1	0.900	0.900	5	15.0	1.035		Eagle-Picher		Li-Ion Secondary Cell, 80 whrs, 90 whrs/kg
Solar Array Cells	1	1.280	1.280	5	15.0	1.472	.88 m2	Spectro-Lab		80 watts at Mars, 18% eff., 95% EPD eff.
Power Electronics										
Power Dist. Module	1	0.101	0.101	5	15.0	0.116	7.6 x 12.7	MMC		Part of Integr. Comp. and Power System (ICAPS)
Power Converter Module	1	0.109	0.109	5	15.0	0.125	7.6 x 12.7	MMC		Part of ICAPS
Pyro Initiation Module	1	0.072	0.072	5	15.0	0.083	7.6 x 12.7	MMC		Part of ICAPS
Segment Switch Module	1	0.101	0.101	5	15.0	0.116	7.6 x 12.7	MMC		Part of ICAPS
Cabling	1	1.600	1.600	6	20.0	1.920	—	MMC/Nanonics		approx. 8% of flight element dry total masss
Subtotal			4.163		16.9	4.867				
Telecom										
Transponder	1	0.500	0.500	6	20.0	0.600	7.6x12.7x3.8	—		X-Band
CDU/TMU Module	1	0.072	0.072	5	15.0	0.083	7.6x12.7	MMC		Part of ICAPS
Diplexer	1	0.200	0.200	3	8.0	0.216		Loral		
Patch Antenna	2	0.060	0.120	2	7.0	0.128		MMC		874C0401300
Medium Gain Antenna	1	0.300	0.300	3	8.0	0.324		Boeing		
RF Switch	1	0.200	0.200	2	7.0	0.214				
SSPA	1	0.400	0.400	6	20.0	0.480	7.6x12.7x3.8	—		X-Band
Waveguides, filters, couplers	1	0.900	0.900	6	20.0	1.080				
Subtotal			2.692		16.1	3.125				
AD&C										
Star Sensor	1	0.600	0.600	4	10.0	0.660	5.3x5.3x11	OCA Applied Optics	Clementine	
Digital Sun Sensor, Head	2	0.250	0.500	1	5.0	0.525	8.1x8.1x2.0	Adcole	Diverse	Model 29970
Digital Sun Sensor, Electro.	1	0.640	0.640	1	5.0	0.672	11.4x12.4x5.4	Adcole	Diverse	Model 29720
IMU, Incl. Accelerometers	1	0.650	0.650	2	7.0	0.696		Litton		FOG
Subtotal			2.390		6.8	2.553				
C&DH										
Processor	1	0.130	0.130	5	15.0	0.150	7.6x12.7	MMC		Part of ICAPS
I/O	1	0.072	0.072	5	15.0	0.083	7.6x12.7	MMC		Part of ICAPS
Subtotal			0.202		15.0	0.232				
Thermal Control										
Heaters	12	0.006	0.072	2	7.0	0.077	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	12	0.006	0.072	1	5.0	0.076	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	12	0.013	0.156	1	5.0	0.164	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
MLI	1	0.500	0.500	5	15.0	0.575		Sheldahl	MSP	
Subtotal			0.800		11.4	0.891				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	4	0.120	0.480	3	8.0	0.518	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
Solar Array Deploy	2	0.200	0.400	1	5.0	0.420	2.5x6x6	StarSys	MSP	SH-9010

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Ordinance Initiators	8	0.010	0.080	1	5.0	0.084	4Dx6	Hi-Shear	MSP	
SRC Sep Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	MSP	9421-2
Structure										
Solar Array Substrate	2	0.500	1.000	4	10.0	1.100		MMC	MSP,MGS	
SRV Structure	1	3.000	3.000	5	15.0	3.450		MMC		
Misc Ftgs and Brkts	1	0.500	0.500	5	15.0	0.575		MMC		
Subtotal			5.960		12.0	6.672				
Propulsion										
N2H4 Tank w/ PMD	1	0.300	0.300	3	8.0	0.324		MMC		Vane Type PMD
N2H4 Lines, Misc	1	0.350	0.350	4	10.0	0.385		MMC		
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Filters	1	0.050	0.050	2	7.0	0.054		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	1	0.040	0.040	3	8.0	0.043		Valvetech		
GHe Tank	1	0.212	0.212	3	8.0	0.229		SCI		
Pyro Valve	1	0.059	0.059	2	7.0	0.063		Pyronetics		Redundant Bridgewire
Regulator	1	0.023	0.023	4	10.0	0.025		Futurecraft		
Check Valve / Burst Disk	1	0.045	0.045	3	8.0	0.049		Valvetech		
N2H4 Thruster	8	0.160	1.280	5	15.0	1.472	1 x 9	Rocket Research		5N, MR-133 mod
Dry Subtotal			2.619		11.6	2.922				
Science / Payload										
Sample Return Capsule			2.935		12.9	3.315				
Subtotal			2.935		12.9	3.315				
Summary										
Power and Cabling			4.163			4.867				
Telecom			2.692			3.125				
AD&C			2.390			2.553				
C&DH			0.202			0.232				
Thermal Control			0.800			0.891				
Structures/Mechanisms			5.960			6.672				
Propulsion			2.619			2.922				
Flt Element Subtotal (Dry)			18.826		12.9	21.263				
Science/Payload			2.935			3.315				
Element Total (Dry)			21.761		12.9	24.578				
GHe Pressurant	1	0.034	0.034	2	7.0	0.036	0.000			
N2H4 Propellant			1.915			2.162				TCM 120 m/sec + 60 m/sec ascent, Isp 222, 2% trapped
Wet Total			23.710		12.9	26.777				

Mars Ascent Vehicle, Stg II (MAV II)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	0.350	0.350	4	10.0	0.385		MMC/Nanonics		3% of dry mass
Subtotal			0.350		10.0	0.385				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Heaters	6	0.006	0.036	2	7.0	0.039	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	6	0.006	0.036	1	5.0	0.038	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	6	0.013	0.078	1	5.0	0.082	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Plume Shield	1	0.300	0.300	5	15.0	0.345				
MLI	1	2.000	2.000	5	15.0	2.300		Sheldahl	MSP	
Subtotal			2.450		14.4	2.803				
Structures/Mechanisms										
Mechanisms										
Structure										
MAV Stg II Structure	1	1.500	1.500	6	20.0	1.800		MMC		
Misc Ftgs and Brkts	1	0.500	0.500	6	20.0	0.600		MMC		
Subtotal			2.000		20.0	2.400				
Propulsion										
CH4 Tanks	2	0.860	1.720	3	8.0	1.858		Arde, Inc.		Modified Model E3899
O2 Tanks	2	1.000	2.000	1	5.0	2.100		Arde, Inc.		Model E3899
Lines, Misc	1	0.300	0.300	1	5.0	0.315		MMC		
Fill and Drain Valves	4	0.020	0.080	1	5.0	0.084		Pyronetics		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Propellant Filter	2	0.050	0.100	2	7.0	0.107		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	8.0	0.086		Valvetech		
CH4/O2 Engine	1	0.750	0.750	6	20.0	0.900		—		150 lbf thrust
Pressure Regulator	2	0.023	0.046	4	10.0	0.051		Futurecraft		
Temperature Sensors	2	0.006	0.012	2	7.0	0.013		Yellowfin		
Coolant Loop	1	0.150	0.150	6	20.0	0.180		MMC		
Dry Subtotal			5.418		8.6	5.886				
Science / Payload										
Sample Return Vehicle			23.710		12.9	26.777				
Subtotal			23.710		12.9	26.777				
Summary										
Power and Cabling			0.350			0.385				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			2.450			2.803				
Structures/Mechanisms			2.000			2.400				
Propulsion			5.418			5.886				
Flt Elem Subtotal (dry)			10.218		12.3	11.474				
Science/Payload			23.710			26.777				wet payload total
Flt Elem Total (dry)			33.928		12.7	38.251				
CH4 Propellant			10.874			12.259				ΔV = 3284 m/sec, ISP = 380, 3.5/1
O2 Propellant			38.106			42.962				ΔV = 3284 m/sec, ISP = 380, 3.5/1
Flt Elem Total (wet)			82.908		12.7	93.472				

Mars Ascent Vehicle, Stg I (MAV I)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	1.400	1.400	5	15.0	1.610		MMC/Nanonics		3% of dry mass
Subtotal			1.400		15.0	1.610				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Heaters	10	0.006	0.060	2	7.0	0.064	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	10	0.006	0.060	1	5.0	0.063	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	10	0.013	0.130	1	5.0	0.137	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Plume Shield	1	1.000	1.000	5	15.0	1.150				
MLI	1	5.000	5.000	5	15.0	5.750		Sheldahl		
Subtotal			6.250		14.6	7.164				
Structures/Mechanisms										
Mechanisms										
Stg II Sep Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	diverse	
Cable Cutters	1	0.200	0.200	2	7.0	0.214	1.9Dx7.4	Hi-Shear	diverse	Model SL1063
Tube Cutters	8	0.200	1.600	2	7.0	1.712				
Structure										
MAV Stg I Structure	1	4.000	4.000	6	20.0	4.800		MMC		
Misc Ftgs and Brkts	1	1.000	1.000	6	20.0	1.200		MMC		
Ascent Fairing	1	2.000	2.000	6	20.0	2.400		MMC		
Subtotal			9.300		16.7	10.851				
Propulsion										
CH4 Tanks	2	6.000	12.000	3	8.0	12.960		SCI		Graphite Overwrap, 2000 psi
O2 Tanks	2	6.000	12.000	3	8.0	12.960		SCI		Graphite Overwrap, 2000 psi
Lines, Misc	1	0.500	0.500	5	15.0	0.575		MMC		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Propellant Filter	2	0.100	0.200	2	7.0	0.214		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	8.0	0.086		Valvetech		
CH4/O2 Engine	4	0.750	3.000	5	15.0	3.450		—		150 lbf thrust
Pressure Regulator	2	0.023	0.046	2	7.0	0.049		Futurecraft		
Temperature Sensors	2	0.006	0.012	2	7.0	0.013		Yellowfin		
Coolant Loop	1	0.400	0.400	5	15.0	0.460		MMC		
Dry Subtotal			28.498		8.9	31.046				
Science / Payload										
MAV Stg II										
Earthbound			82.908			93.472				
Marsbound			33.928			38.251				
Earthbound Subtotal			82.908		12.7	93.472				
Marsbound Subtotal			33.928		12.7	38.251				
Summary										
Power and Cabling			1.400			1.610				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			6.250			7.164				
Structures/Mechanisms			9.300			10.851				
Propulsion			28.498			31.046				
Flt Elem Subtotal (dry)			45.448			50.670				
Science/Payload										
Earthbound			82.908			93.472				
Marsbound			33.928			38.251				
Flt Elem Total (Earthbound)			128.356		12.3	144.143				
Flt Elem Total (Marsbound)			79.376		12.0	88.921				
CH4 Propellant			38.329		12.3	43.044				ΔV = 3132 m/sec, ISP 380
O2 Propellant			134.326		12.3	150.847				ΔV = 3132 m/sec, ISP 380
Flt Elem Total Wet (Earth)			301.011		12.3	338.034				
Hydrogen			13.544			15.224				
Flt Elem Total Wet (Mars)			92.920		12.1	104.145				

Lander

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Battery	1	2.400	2.400	4	10.0	2.640		Eagle-Picher		Li-ion Secondary, 215 whrs
Solar Array Cells	4	2.850	11.400	4	10.0	12.540	8 m^2	Spectro-Lab		750 watts, 18% eff., 95% EPD
Power Electronics										
Segment Switching Module	2	0.101	0.202	5	15.0	0.232	7.6x12.7	MMC		Part of ICAPS
Cabling	1	5.000	5.000	5	15.0	5.750		MMC/Nanonics		Approx. 8% of flt elem dry mass
Subtotal			19.002		11.4	21.162				
Telecom										
UHF Transceiver	1	1.200	1.200	5	15.0	1.380				
UHF Antenna	1	0.100	0.100	5	15.0	0.115				
Subtotal			1.300		15.0	1.495				
AD&C										
Term. Descent Landing Radar	1	4.800	4.800	3	8.0	5.184		Honeywell	MSP	
Subtotal			4.800		8.0	5.184				
C&DH										
Subtotal			0.000		####	0.000				
Thermal Control										
Heaters	36	0.006	0.216	2	7.0	0.231	5 x 20 typ	Tayco	MSP	sized to fit
Heat Pipe Loops	2	1.500	3.000	3	8.0	3.240		Dynatherm	MSP	
Heat Pipe, Var. Cond.	2	1.000	2.000	3	8.0	2.160		Dynatherm	MSP	
Thermistors	36	0.006	0.216	1	5.0	0.227	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	36	0.013	0.468	1	5.0	0.491	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
Radiator	2	1.500	3.000	4	10.0	3.300	.2m^2	MMC	MSP	
MLI	1	2.500	2.500	5	15.0	2.875		Sheldahl	MSP	
Subtotal			11.400		9.9	12.524				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	8	0.120	0.960	3	8.0	1.037	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
MGA R&R	2	0.120	0.240	3	8.0	0.259	2.5Dx5	Fokker/Hi-Shear	MSP	R06720-429/431
Solar Array Deploy	12	0.290	3.480	1	5.0	3.654	2.5x6x6	StarSys	MSP	SH-9010
Physical Surface Sensor	3	0.200	0.600	5	15.0	0.690		MMC	MSP	
Legs	3	4.490	13.470	4	10.0	14.817		MMC	MSP/Vkg	
Cable Cutter	1	0.200	0.200	2	7.0	0.214	1.9Dx7.4	Hi-Shear	MSP	SL1063
Tube Cutter	8	0.200	1.600	2	7.0	1.712			MSP	
Ordinance Initiators	40	0.010	0.400	1	5.0	0.420	4Dx6	Hi-Shear	Diverse	
Structure										
Solar Array Substrate	4	1.780	7.120	4	10.0	7.832		MMC	MSP,MGS	
Component Deck	1	3.350	3.350	4	10.0	3.685		MMC	MSP	

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Equip Enclosure	1	8.460	8.460	4	10.0	9.306		MMC	MSP	
Prop. Tank Support	2	1.480	2.960	4	10.0	3.256		MMC	MSP	
SA Stow Ftgs	4	0.250	1.000	4	10.0	1.100		MMC	MSP	
Leg Mounts	3	1.910	5.730	4	10.0	6.303		MMC	MSP	
Misc Ftgs & Brkts	1	3.000	3.000	4	10.0	3.300		MMC	MSP	
Subtotal			52.570		9.5	57.585				
Propulsion										
Ox/Fuel Tanks	2	5.900	11.800	2	7.0	12.626		PSI	MSP	80358-1
Lines, Misc	1	2.500	2.500	4	10.0	2.750		MMC	MSP	
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Filters	2	0.100	0.200	2	7.0	0.214		Vacco		
Pressure Transducer	4	0.090	0.360	2	7.0	0.385		Sensometrics		
Latching, Isolation Valve	4	0.040	0.160	3	8.0	0.173		Valvetech		
GHe Tank	1	0.212	0.212	3	8.0	0.229		Brunswich	MSP	220088-1
Pyro Valve	1	0.059	0.059	2	7.0	0.063		Pyronetics		
Regulator	2	0.023	0.046	4	10.0	0.051		Futurecraft		
Check Valve / Burst Disk	2	0.045	0.090	3	8.0	0.097		Valvetech		
Descent Eng. Mod. / Roll	2	0.840	1.680	4	10.0	1.848				Biprop
Descent Eng. Mod. / Pitch,Yaw	4	1.550	6.200	4	10.0	6.820				Biprop
Dry Subtotal			23.387		8.4	25.341				
Science / Payload										
MAV Stg I	1		92.920			104.145				
Surface Science	1	10.000	10.000	0	0.0	10.000				
ISRU System	1	26.000	26.000	4	10.0	28.600				
Subtotal			128.920		10.7	142.745				
Summary										
Power and Cabling			19.002			21.162				
Telecom			1.300			1.495				
AD&C			4.800			5.184				
C&DH			0.000			0.000				
Thermal Control			11.400			12.524				
Structures/Mechanisms			52.570			57.585				
Propulsion			23.387			25.341				
Flt Element Subtotal (dry)			112.459			123.292				
Science/Payload			128.920			142.745				Wet Payload
Flight Element Total (Dry)			241.379		10.2	266.037				
Propellant			33.495			36.916				ΔV 400 m/sec, ISP 320 sec, UDMH/NTO4
GHe Pressurant			0.450			0.500				
Wet Subtotal			275.324		10.2	303.453				

Mars Entry Capsule (MEC)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Cabling	1	0.200	0.200	4	10.0	0.220		MMC/Nanonics		
Subtotal			0.200		10.0	0.220				
Telecom										
Subtotal										
AD&C										
Subtotal										
C&DH										
Subtotal										
Thermal Control										
Subtotal										
Structures/Mechanisms										
Mechanisms										
HS Sep Device	1	4.000	4.000	4	10.0	4.400		MMC	PF,MSP	818A1410326
BS Sep Device	1	2.500	2.500	4	10.0	2.750		MMC	PF, MSP	818A1410326
BS Sep Rails	1	2.000	2.000	5	15.0	2.300		MMC		
Structure										
Heat Shield	1	23.900	23.900	4	10.0	26.290		MMC	PF, MSP	
Head Shield TPS	1	20.300	20.300	4	10.0	22.330		MMC	PF, MSP	SLA-561
Back Shell	1	25.000	25.000	4	10.0	27.500		MMC	PF, MSP	

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Back Shell TPS	1	7.000	7.000	4	10.0	7.700		MMC	PF, MSP	SLA-561S
Lander Support Structure	6	0.430	2.580	4	10.0	2.838		MMC	PF, MSP	
Ballast	1	1.000	1.000	4	10.0	1.100				
Misc Ftgs & Brkts	1	1.000	1.000	4	10.0	1.100		MMC		
Subtotal			89.280		10.1	98.308				
Propulsion										
Dry Subtotal										
Science / Payload										
Lander			275.324			303.453				
Subtotal			275.324		10.2	303.453				
Summary										
Power and Cabling			0.200			0.220				
Telecom			0.000			0.000				
AD&C			0.000			0.000				
C&DH			0.000			0.000				
Thermal Control			0.000			0.000				
Structures/Mechanisms			89.280			98.308				
Propulsion			0.000			0.000				
Flt Elem. Subtotal (dry)			89.480			98.528				
Science/Payload			275.324			303.453				
Flight Element Total			364.804		10.2	401.981				

Trans Mars Cruise Stage (TMCS)

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Power and Cabling										
Solar Array Cells	1	1.600	1.600	4	10.0	1.760	1.1 m^2	Spectro-Lab		100 watts at Mars, 18% effic., 95% EPD eff.
Cabling	1	1.800	1.800	5	15.0	2.070				8% of flight element dry weight
Subtotal			3.400		12.6	3.830				
Telecom										
Patch Antenna	3	0.060	0.180	2	7.0	0.193		MMC	Diverse	874C0401300
Medium Gain Antenna	1	0.300	0.300	2	7.0	0.321		Boeing	MSP	
RF Switch	1	0.200	0.200	2	7.0	0.214				
Subtotal			0.680		7.0	0.728				
AD&C										
Star Sensor	1	0.600	0.600	4	10.0	0.660	5.3x5.3x11	OCA Applied Optics	MSP	
Digital Sun Sensor, Head	2	0.250	0.500	1	5.0	0.525	8.1x8.1x2.0	Adcole	Diverse	Model 29970
Digital Sun Sensor, Electro.	1	0.640	0.640	1	5.0	0.672	11.4x12.4x5.4	Adcole	Diverse	Model 29720
Subtotal			1.740		6.7	1.857				
C&DH										
Subtotal										
Thermal Control										
Heaters	12	0.006	0.072	2	7.0	0.077	5 x 20 typ	Tayco	MSP	sized to fit
Thermistors	12	0.006	0.072	1	5.0	0.076	.2 x .2 x .001	Yellowfin	MSP	
Thermostats	12	0.013	0.156	1	5.0	0.164	1.5D x 2.0	Texas Instruments	MSP	Model SC71D27
MLI	1	1.000	1.000	5	15.0	1.150		Sheldahl	MSP	
Subtotal			1.300		12.8	1.466				
Structures/Mechanisms										
Mechanisms										
Solar Array R&R	4	0.120	0.480	3	8.0	0.518	2.5Dx5	Fokker/Hi-Shear	MSP	
Solar Array Deploy	2	0.250	0.500	1	5.0	0.525	2.5x6x6	StarSys	MSP	
Ordinance Initiators	8	0.010	0.080	1	5.0	0.084	4Dx6	Hi-Shear	MSP	
MEC Sep. Device	2	0.250	0.500	1	5.0	0.525	4Dx7	G&H	MSP	9421-2
Structure										
Solar Array Substrate	2	0.500	1.000	4	10.0	1.100		MMC	MSP	
Cruise Stage Structure	1	4.000	4.000	4	10.0	4.400		MMC		
Misc Ftgs and Brkts	1	1.000	1.000	5	15.0	1.150		MMC		

Component	Units	Unit Mass (kg)	Total Mass (kg)	Matur. Index	Cont. %	Total w/ Cont.	Size (cm)	Vendor	Program Common.	Descrip/Comments
Subtotal			7.560		9.8	8.302				
Propulsion										
Ox/Fuel Tank	2	2.230	4.460	3	8.0	4.817		PSI	GPS II	
Lines, Misc	1	0.500	0.500	5	15.0	0.575		MMC		
Fill and Drain Valves	4	0.020	0.080	2	7.0	0.086		Pyronetics		
Propellant Filter	2	0.050	0.100	2	7.0	0.107		Vacco		
Pressure Transducer	2	0.090	0.180	2	7.0	0.193		Sensometrics		
Latching, Isolation Valve	2	0.040	0.080	3	8.0	0.086		Valvetech		
GHe Tank	1	0.300	0.300	3	8.0	0.324		SCI		
Pyro Valve	2	0.059	0.118	2	7.0	0.126		Pyronetics		redundant bridgewire
Regulator	2	0.023	0.046	4	10.0	0.051		Futurecraft		
Check Valve / Burst Disk	2	0.045	0.090	3	8.0	0.097		Valvetech		
Thruster, Pitch - Yaw	4	0.160	0.640	4	10.0	0.704				Biprop
Thruster, Roll	4	0.160	0.640	4	10.0	0.704				Biprop
Dry Subtotal			7.234		8.8	7.869				
Science / Payload										
Mars Entry Capsule			364.804			401.981				
Subtotal			364.804		10.2	401.981				
Summary										
Power and Cabling			3.400			3.830				
Telecom			0.680			0.728				
AD&C			1.740			1.857				
C&DH			0.000			0.000				
Thermal Control			1.300			1.466				
Structures/Mechanisms			7.560			8.302				
Propulsion			7.234			7.869				
Flt Element Subtotal (dry)			21.914			24.053				
Science/Payload			364.804			401.981				
Flight Element Total			386.718		10.2	426.034				
GHe Pressurant	1	0.250	0.250	5	15.0	0.288				
Propellant			15.396			16.962				ΔV 120 m/sec, ISP 320 sec, UDMH/NTO4
Wet Subtotal			402.364		10.2	443.284				

Low Cost MISR, bare bones

Maturity Index	Contingency
0	0
1	5
2	7
3	8
4	10
5	15
6	20
7	40
8	50
9	75
10	100

Maturity Index Table

Abbreviations

MGS	Mars Global Surveyor (Mars '96)
MMC	Martin Marietta (Lockheed Martin)
MSP	Mars Surveyor Program (Mars '98)
PF	MESUR Pathfinder
Vkg	Viking

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13. ABSTRACT (Maximum 200 words)

This report presents the results of a study examining the potential of in-situ propellant production (ISPP) on Mars to aid in achieving a low cost Mars Sample Return (MSR) mission. Two versions of such a mission were examined; a baseline version employing a dual string spacecraft and a light weight version employing single string architecture with selective redundancy. Both systems employed light weight avionics currently being developed by Lockheed Martin, Jet Propulsion Lab and elsewhere in the aerospace community, both used a new concept for a simple, light weight parachuteless sample return capsule, both used a slightly modified version of the Mars Surveyor lander currently under development at Lockheed Martin for flight in 1998, and both used a combination of the Sabatier-electrolysis and reverse water gas shift ISPP systems to produce methane/oxygen propellant on Mars by combining a small quantity of imported hydrogen with the Martian CO2 atmosphere. It was found that the baseline mission could be launched on a Delta 7925 and return a 0.5 kg sample with 82% mission launch margin, over and beyond subsystem allocated contingency masses. The lightweight version could be launched on a Mid-Lite vehicle and return a 0.25 kg sample with 11% launch margin, over and above subsystem contingency mass allocations.

14. SUBJECT TERMS
Mars, Mars Sample Return, In-Situ Propellant Production, ISRU

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