FINAL REPORT

on

ADVANCED SYSTEM CONCEPT FOR TOTAL ISRU-BASED PROPULSION AND POWER SYSTEMS FOR UNMANNED AND MANNED EXPLORATION

NIAC-PHASE I CONTRACT

Research Grant 07600-020
OTC-G084-FR-2000-1

Prepared for:

Universities Space Research Association (USRA)

1 January 2000

by

Dr. Eric E. Rice, President and CEO, NIAC Fellow

Orbital Technologies Corporation
Space Center, 1212 Fourier Drive

Madison, Wisconsin 53717
608-827-5000

www.orbitec.com

EMAIL: ricee@orbitec.com

FOREWORD

This document represents the Final Report on the Phase I feasibility analysis of an "Advanced System Concept for Total ISRU-Based Propulsion and Power Systems for Unmanned and Manned Exploration" a NIAC-Phase I study contract (Research Grant #07600-020) prepared by Orbital Technologies Corporation (ORBITEC™), Madison, Wisconsin, for NASA and the NASA Institute for Advanced Concepts (NIAC), managed by the Universities Space Research Association (USRA). The work was performed from June 1999 through November 1999.

ORBITEC wishes to acknowledge the excellent communications and support from Dr. Robert Cassanova, NIAC Director and his project team. Dr. Eric E. Rice, PI and author, wishes also to acknowledge the excellent contributions of Mr. Robert Gustafson, Dr. Marty Chiaverini, Mr. Chris St.Clair, Mr. Daniel Gramer, Mr. William Knuth, Mr. Matthew Molecki, Mr. Brant White, Ms. MaryAnn Knoke, Ms. Kelley Rice, Ms. Jenny Kellogg, and Ms. Molly Mitten. It should be noted that the test firing data for CO/GOX and CH_4/GOX that are presented and were conducted under NASA/GRC contracts NAS3-27382, NAS3-99153, and in house ORBITEC IR&D. A patent has been filed by ORBITEC in the U.S. to protect the advanced solid cryogenic hybrid engine technology.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

1.0 BACKGROUND

In this feasibility study, ORBITEC has conceptualized systems and an evolving architecture for producing and utilizing Mars-based ISRU propellant combinations from the atmosphere to support ground and flight propulsion and power systems that would be part of Mars exploration and colonization. Ground transport systems included: automated unmanned roving vehicles, personal vehicles, two-person unpressurized rovers, manned pressurized transport rovers, and larger cargo transports. Flight vehicles include: Mars sample return vehicles, unmanned and manned surface-to-surface "ballistic hoppers", surface-to-orbit vehicles, interplanetary transport vehicles, powered balloons, winged aerocraft, single person rocket backpacks, and single person rocket platforms. Auxiliary power systems include: Brayton turbines and fuel cells for small Mars outposts. The vision of these systems goes from the initial human exploration missions out to 50 years beyond the early exploration initiatives.

In this Phase I study, we accomplished a preliminary systems scoping study which provides the approach and direction to fully assess the benefits of an ISRU approach (e.g., carbon/oxygen, carbon monoxide/oxygen, methane/oxygen or hydrogen/oxygen) compared to one of using all Earth-supplied propellants. There is no question that for the cost-effective human exploration of Mars, we will need to use insitu resources that are available on Mars. The real question is what propellants do we use in what applications to achieve the best economic benefit for humanity.

Probably the most cost-effective and easiest use of Martian resources is the atmosphere (95% CO_2). The CO_2 can be easily processed and converted to carbon monoxide or carbon and oxygen. Water vapor is also present in the Mars atmosphere in small proportion; soil-based water (especially in the polar regions) may likely be in much greater abundance. With the availability of C, CO, O_2, and H_2O through processing the atmosphere, excellent propellants can be made (SC/LOX, SCO/LOX, LCO/LOX, LCH_4/LOX, SC_2H_2/LOX, LH_2/SOX, LH_2/LOX, H_2O_2/CH_3OH, and etc). For this study period, we have focused upon a 50-year period beyond the initial manned Mars exploration activity. We have assumed that three different levels of activity and missions that require the use of propellants and fuels are possible, as driven by various reasons (continued presence/research/exploration, terraforming program, colonization, etc.). Therefore, we are defining what we call "low", "medium" and "high" traffic models. To define the use of propellants/fuels, we define the vehicles that would use them. ORBITEC's overall approach in this Phase I effort was to develop a feasible study methodology/approach such that a credible detailed study could be conducted in Phase II that would provide reasonable answers that would provide knowledgeable guidance to NASA technology development of systems that use the ISRU propellants as well as the definition of the ISRU processing systems themselves.

2.0 ADVANCED CONCEPT DESCRIPTION

To enable cost-effective, *in situ* production and uses of Mars atmospheric-derived oxidizers (O_2, N_2O, N_2O_4, H_2O_2) and fuels (CO, C, C_2H_2, C_2H_4, CH_4, C_3H_8 CH_3OH, H_2, etc.) and to guide technology development and unique hardware development, detailed advanced concept development and system analysis efforts are required. The use of these propellants in applications involve rocket propulsion, ground-based rovers that use turbine engines, and electric power generation systems. It is believed that by using the baseline C/O system with the addition of either Earth supplied hydrogen or Mars atmospheric derived hydrogen, in the proper fuel form (CO solid, C solid, C_2H_2 solid, CH_4 solid, etc.) that significant economic dividends can be achieved for the HEDS enterprise.

The production of oxygen and carbon monoxide through solid state electrolysis appears to be well in hand by K. R. Sridhar of the University of Arizona. Hardware is now being prepared to fly to Mars for an ISRU demonstration. ORBITEC's innovative work on this revolutionary concept, and the recent successful hot firing demonstrations of advanced cryogenic solid hybrid rocket engines, including: solid CO/GOX, solid H_2/GOX, solid O_2/GH_2, solid CH_4/GOX, and solid C_2H_2/GOX, provide the motivation to analyze, test and assess the potential of using the Mars atmosphere for ISRU propellant.

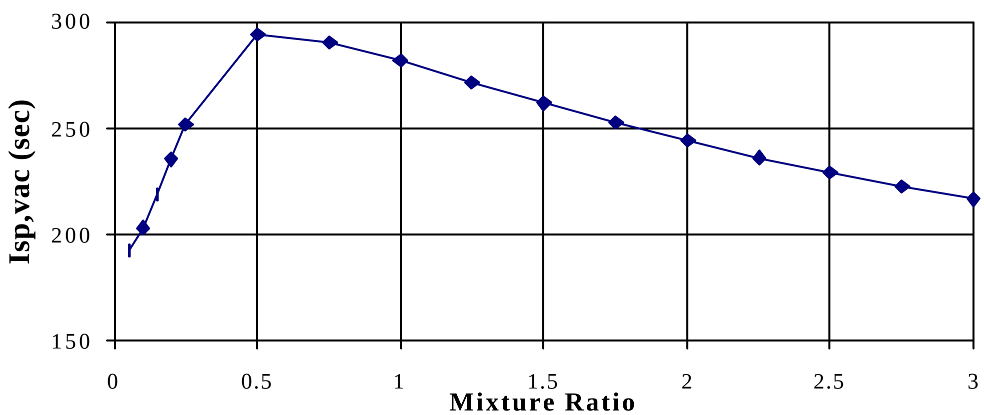

Figure 1. Solid CO/LOX Theoretical I_{sp} vs O/F Ratio for e=100, Pc- 500 psia

The beauty of our SCO/LOX propellant concept is that CO gas can be directly and quickly frozen to a solid hybrid fuel grain below the triple point temperature (68 K) by using sub-cooled LOX (with the low pressure of the Mars atmosphere (4.5 to 11.4 mm Hg, this is very easy-@ 11.4 mm LOX will be at 63 K, and @6 mm LOX will be at 60 K) as the freezing fluid and oxidizer in a cryogenic hybrid engine. The heavy tank associated with LCO goes away and a much lighter propulsion system can now be developed that will be the most simple and low cost approach. Ideal performance using the NASA/GRC CEA performance code is shown in Figure 1 for an expansion ratio of 100 and a chamber pressure of 500 psia.

Additionally, and in collaboration with Dr. Tom Sullivan (NASA/JSC), we have good indication that solid carbon, with perhaps some small amount of hydrogen, could result in an ignitable, good burning carbon hybrid fuel grain for ISRU applications on Mars. Ideal performance for carbon/LOX at e=100, and P_c=500 at differing hydrogen loadings in the carbon is indicated in Figure 2. This performance is very promising.

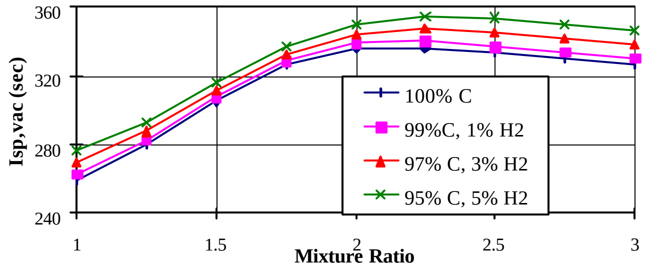

Figure 2. Solid C/LOX Theoretical I_{sp} vs O/F Ratio for 100, Pc- 500 psia and Varying H Concentrations

According to ideal performance analysis, C_2H_2 burning with O_2 at an O/F ratio of 1.4, a chamber pressure of 300 psia, with a nozzle expansion ratio of 100:1, the maximum theoretical I_{sp} exceeds 405 seconds. Figure 3 shows maximum theoretical I_{sp} levels for SC_2H_2/O_2 as a function of O/F ratio for 100:1 expansion to vacuum and for optimum expansion to atmosphere at sea level.

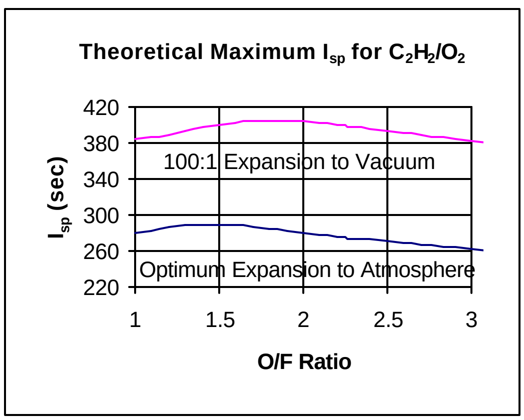

Figure 3. Theoretical Maximum I_{sp} for SC_2H_2/GOX

The results of this proposed effort are applicable to future NASA Solar System unmanned and manned exploration missions to Mars. This activity is a part of NASA's overall strategic plan. Mars-produced fuels and oxidizers will enhance and/or enable a variety of Mars exploration missions by providing a very cost-effective supply of propellants. The technology requires demonstration before propellants are selected. The establishment of practical feasibility could result absolutely huge savings to our exploration programs.

As an examples of the rocket-based vehicles that would be studied here, Figure 4 provides a summary of the vehicle systems that would be evaluated and analyzed in this proposed effort. The first concept would be for Mars sample return missions, the second for automated surface "hopper"/orbital vehicle, and the third, a large manned "hopper" /orbital vehicle.

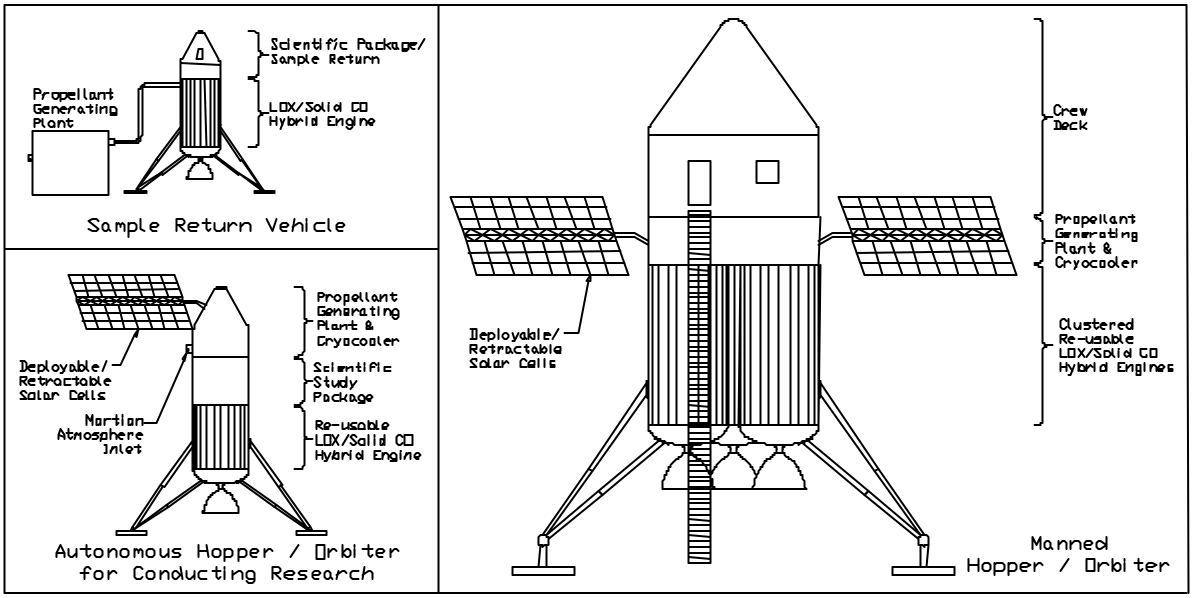

Figure 4. Solid CO/O_2-Based Hybrid Rocket Flight Vehicle Concepts

For the small vehicle, an expendable propellant production system would be outside of the vehicle shell and would include: a solar energy power generation system, an atmospheric intake system, with filters; a regenerative solid oxide fuel cell/separator system; an intermediate CO gas reservoir; an oxygen liquifer subsystem; control system, and etc.

For the medium unmanned and large manned vehicles, all of the processing systems would be contained within or on (retractable solar arrays) the flight vehicle. Part of our analysis in this proposed effort will be to develop reasonable mass breakdown projections such that the benefit assessment can be conducted. We believe that the results may show a tremendous cost/benefit for the C/O architecture.

During the Phase I effort, we developed an overall advanced concepts study approach that will be exercised in Phase II to determine the winning ISRU propellant combinations for the most economical program. We have defined the assumptions, and study guidelines and have developed the foundation for a successful Phase II effort. Our overall study approach is outlined in Figure 18, Page 22.

3.0 STUDY RESULTS AND FINDINGS

The sections that follow provide the results of the Phase I research effort, including: (1) introduction, (2) literature survey, (3) ORBITEC's SCO/GOX hybrid rocket engine firing, (4) results of ORBITEC's SCH_4/GOX hybrid rocket engine firings data, (5) system requirements definition for the architecture study, (6) Mars mission/traffic model and cost model development, (7) preliminary system concept development and analysis, (8) preliminary cost-benefit analysis, and (9) preliminary technology assessment.

3.1 Introduction

The purpose of this effort was to identify, assess and enable the cost-effective application of *In Situ* production and uses of Mars atmospheric-derived propellants and to help guide the advanced concept development, system analysis, technology development and unique hardware development efforts of the future. Mars-produced propellants will enhance and/or enable a variety of Mars exploration/exploitation missions by providing a very cost-effective supply of propellants. The most cost-effective Martian resource is the atmosphere which is comprised of 95% CO_2. Atmospheric CO_2 can be easily processed and converted to CO or C & O_2. A small amount of H_2O can be converted to H_2 and O_2, and N_2, Ar are also available from the atmosphere. With these elements, there are many propellant combinations that are possible including: SC/LOX, SCO/LOX, LCO/LOX, LCH_4/LOX, SC_2H_2/LOX, LH_2/SOX, LH_2/LOX, H_2O_2/CH_3OH, and etc).

For this study period, we have considered a 50-year period beyond the initial manned Mars exploration activity. We have assumed that three different levels of activity and missions that require the use of propellants are possible, as driven by various reasons (continued presence, research, exploration, terraforming program, colonization, etc.). Therefore, we are defining what we call "low", "medium" and "high" traffic models to define the use of propellants, we define the vehicles that would use them.

In this Phase I study, ground transport systems have included: automated unmanned roving vehicles, personal vehicles, two-person unpressurized rovers, manned pressurized transport rovers, and larger cargo transports. Flight vehicles have included: Mars sample return vehicles, unmanned and manned surface-to-surface "ballistic hoppers", surface-to-orbit vehicles, interplanetary transport vehicles, powered balloons, winged aerocraft, single person rocket backpacks, and single person rocket platforms. Auxiliary power systems include: Brayton cycle turbines and fuel cells for small Mars outposts. Implementation of this ISRU-based architecture will also greatly support logistics & base operations by providing a reliable and simple way to store solar and nuclear generated energy.

3.2 Literature Survey

The review of the literature provided background information upon which to build the data base for concepts and future design work and analysis in Phase II. Items now included in our data base include: ORBITEC work on advanced cryogenic hybrid engines, oxygen liquifaction systems, cryogenic refrigerators, the latest in light-weight cryogenic insulation, the latest studies by NASA JSC and JPL on Mars sample return propulsion and manned Mars propulsion systems, mission models for Mars base development, mission science needs, other ISRU works related to Mars, etc. Attention was also given to the on-going work on CO_2 separation technology being developed by Sridhar at the University of Arizona for CO, O_2 and CH_4 production.

Many journal articles, conference papers, detailed reports and web articles were identified and obtained and placed in our NIAC project library. These items were entered in a computer database that contains the title, author name, publication date, publication source, and a brief abstract. All of these sources were reviewed and they will serve as a valuable reference source to support future study activity in Phase II. Appendix A is a Bibliography of related works and Appendix B provides the list of references reviewed for this study.

3.3 Results of ORBITEC Experimental Work

This section briefly describes the most recent rocket engine findings that directly relate to the wok of this program, namely, test firings of CO/GOX and OH_4/GOX.

3.3.1 SCO/GOX Hybrid Rocket Firings

This section summarizes the evolutionary carbon monoxide/GOX/hybrid rocket engine hot firing tests performed to date by ORBITEC. These data are relevant and key to the conduct of the advanced concepts study.

A total of six solid carbon monoxide/GOX hybrid hot firing tests have been attempted, all in the ORBITEC Mark II engine system. Figure 5 shows a section view of the Mark II cryogenic hybrid engine assembly that has been used to test many cryogenic solid and gaseous propellant combinations. The carbon monoxide grain is deposited from the gas phase on the inner wall of the central tube in the freezer core. Coolant fluid, in this case liquid helium, fills the annular coolant chamber around this central tube and chills the tube wall to enable the cryogenic deposition. The volume inside the central tube is referred to as the grain chamber. When the firing is performed after the grain formation process, oxygen injection and the ignitor torch enter from the injector plate located at the head end of the grain chamber. The surrounding vacuum chamber provides thermal insulation to enable efficient cryogenic operation. For reference, the diameter of the grain chamber is 5.08 cm (2.00 inches).

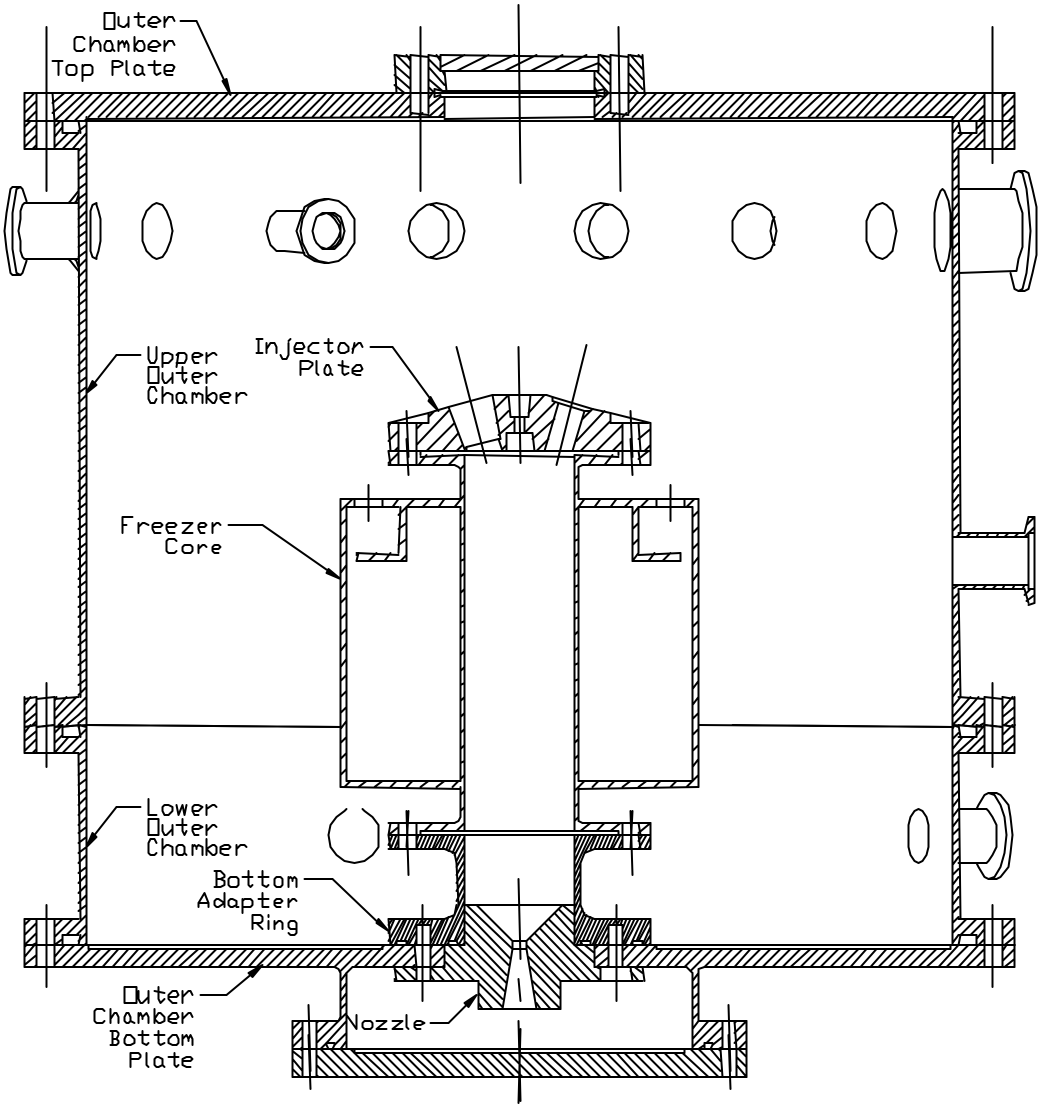

Figure 5. Mark II Engine Assembly

Table 1 summarizes the results of these tests in tabular form, and Figure 6 presents a composite pressure trace/time summarizing all successful hot firing tests.

Table 1. Solid Carbon Monoxide/GOX Hybrid Firing Tests in ORBITEC's Mark II Engine

	CO-H001	CO-H002	CO-H003	CO-H004	CO-H005	CO-H006
Date:	29-Jan-98	2-Aug-99	3-Aug-99	4-Aug-99	12-Nov-99	23-Nov-99
Grain Mass (g):	100	100	100	100	100	100
CO gas delivery:	Head-end	Nozzle	Nozzle	Nozzle	Nozzle	Nozzle
CO gas cooling:	None	LN_2 bath	LN_2 bath	LN_2 bath	LN_2 bath	LN_2 bath
CO freez. press. (torr):	app. 1	9	9	9	90	90
Oxygen flow (g/s):	6.0	6.0	4.0	10.0	10.0	6.0
Oxygen phase :	GOX	GOX	GOX	GOX	GOX	GOX
Injector diameter:	Conical	0.104"	0.104"	0.104"	0.104"	0.104"
Avg. reg.rate (cm/s):	0.0576	0.0482	0.043	---	0.0608	0.0438
Freezer coolant:	LHe	LHe	LHe	LHe	LHe	LHe
Duration (sec):	9.7	11.6	13	---	9.2	12.8
Avg. O/F:	0.57	0.7	0.51	---	0.92	0.76
Avg. Pres. (psia):	71	67	52	---	95	55.4
Ignitor H_2 Flow (g/s):	0.06	0.06	0.06	0.06	0.06	0.06
Ignitor O_2 Flow(g/s):	0.24	0.24	0.24	0.24	0.24	0.24
Ignitor duration (sec):	whole test	2	2	---	2	2
C* (sec):	114	120	115	---	119	104
C* efficiency:	83	88	85	---	89	77
Max temp on TC 3 (K):	310	260	270	---	300	540
Max temp on TC 3 (K):	710	720	430	---	810	180

3.3.2 Engine-Test Operation

Many aspects of the test firings were kept constant throughout the entire test series. All grains were formed using liquid helium as a coolant fluid, and all were 100 grams in mass. The estimated grain thickness for this grain mass is 0.56 cm. All tests employed a hydrogen/oxygen/spark ignition system with an O/F ratio of 4 and a total mass flow rate of 0.3 grams/second. Also, all tests used the same basic oxygen flow profile, with an accelerating ramp for the first 0.5 seconds followed by a plateau for the remainder of the test.

The sequence timing remained virtually identical for the entire test series. A representative timeline is presented below. Note that the time defined as 't=0' corresponds to the start of the main oxygen flow ramp.

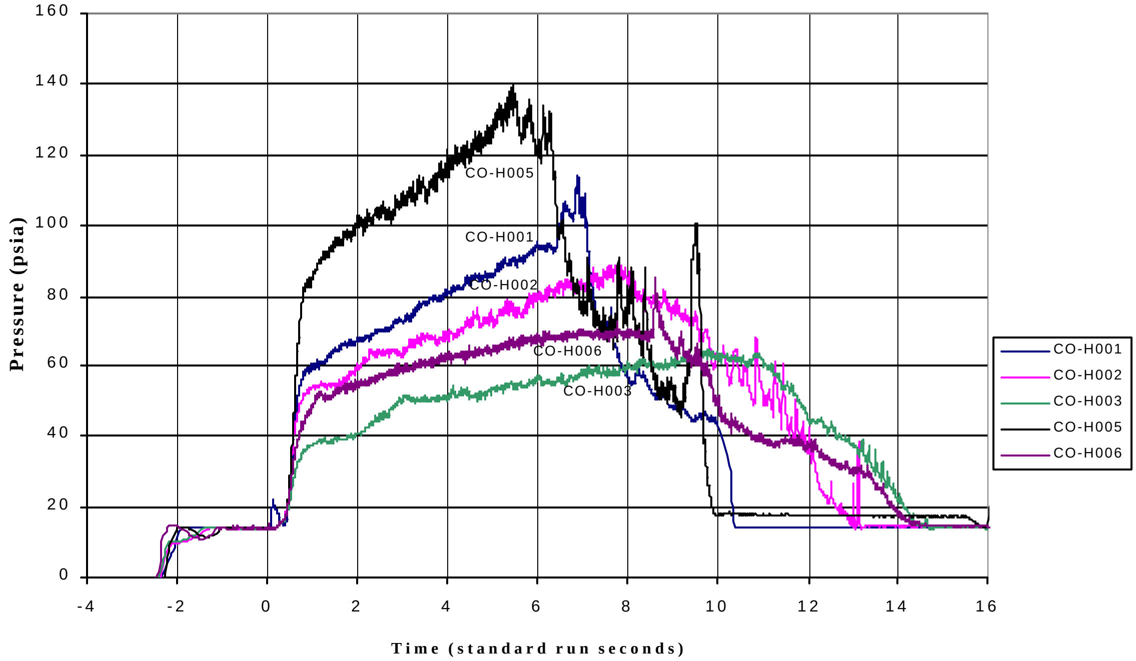

Figure 6. Pressure Traces for All Successful Carbon Monoxide Hot Firing Tests

Time (sec)	Event
-2.5	The operator initiates the firing sequence through the computer control system.
-2.5	The gaseous helium purge valve opens, dumping a calibrated volume of pressurized helium gas into the grain chamber to bring the pressure to one atmosphere.
-2.2	With the chamber now at atmospheric pressure, the computer opens the swinging door, exposing the nozzle to the atmosphere below.
-0.8	When the computer has sensed that the swinging door is completely opened and clear of the nozzle area, the ignition sequence is started. The igniter hydrogen valve is opened and the spark plug is turned on.
-0.5	The igniter oxygen valve is turned on. Several tenths of a second later, ignition occurs in the igniter and the igniter flame enters the grain chamber.
0.0	With the igniter flame on, the main oxygen flow valve is opened and the ramp-up is started. As oxygen begins entering the chamber, it encounters the igniter flame and a fuel (CO)-rich environment, and main ignition occurs.
0.5	The 'command' flow ramp of oxygen reaches the designated plateau level.

0.6 The actual ramp catches up to the command profile. Oxygen flow reaches its maximum level and remains constant for the remainder of the test.

Later, after burning for approximately 9-13 seconds, the carbon monoxide grain is depleted. Oxygen continues to flow according to the pre-programmed profile for several more seconds. Then oxygen flow is ramped down, the main oxygen flow valve is closed, and helium gas purges the combustion chamber of any remaining propellants.

3.3.3 Individual Test Procedures and Results

3.3.3.1 CO-H001 Hot Firing of SCO/GOX

During the grain freezing process, carbon monoxide gas was introduced from the head end of the engine through the igniter. A vapor pressure of approximately 1 Torr (of CO) was maintained in the grain chamber during the freeze. The actual test firing proceeded smoothly, with a good ignition followed by a test that lasted approximately 10 seconds. Figure 7 presents a pressure trace from the test firing.

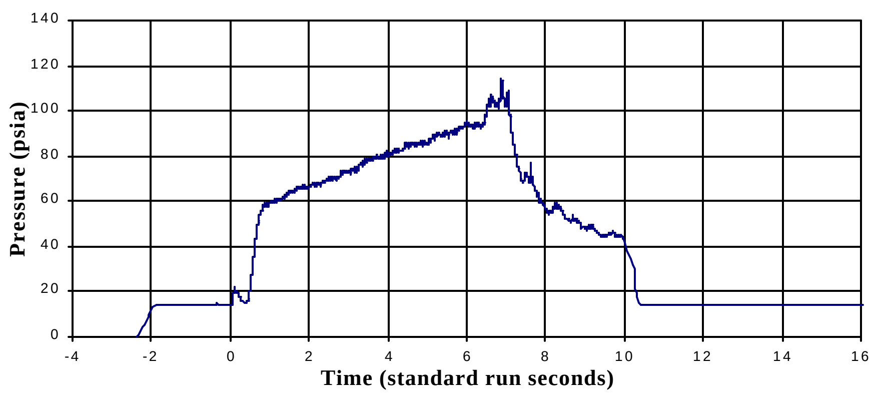

Figure 7. Pressure Trace – CO-H001

As Figure 6 illustrates, the chamber pressure reaches 60 psi shortly after the oxygen flow reaches its full level. The pressure then steadily increases for the next 6 seconds to approximately 90 psi. The end of the test is marked by several pressure spikes and an overall drop-off in pressure. The igniter torch was left on for the duration of the test. The main oxygen flow for CO-H001 was injected into the grain chamber through a single-hole injector, 0.104” in diameter. This same injector was used for CO-H001 through CO-H005.

3.3.3.2 CO-H002 Hot Firing of SCO/GOX

Several modifications were made to the freezing process for this test. One possible explanation for the rough pressure trace seen at the top of the pressure curve in CO-H001 was that the grain was forming asymmetrically, resulting in premature grain burn-through in one location during the burn and a subsequent end of burn grain break-up. Grain asymmetry could arise from a number of causes. One prime suspect was asymmetrical thermal input from the incoming gas stream. Room-temperature CO gas entering the grain chamber through the angled igniter port at high velocity could have warmed one area of the grain, resulting in a thin area prone to early grain depletion of the wall of the engine.

The following changes were made to minimize the chance of grain asymmetry caused by non-uniform convective heating. First, the location of the gas input was changed from the igniter port in the head end, which pointed into the chamber at an angle, to the nozzle at the aft end, which is directed straight up the axis of the chamber. This gave the incoming flow radial symmetry. Second, the freezing pressure during the grain formation process was increased from approximately 1 Torr to 9 Torr. This was done to reduce the velocity of the incoming gas. Last, a liquid nitrogen pre-chill heat exchanger was installed to cool the incoming gas to approximately 80 K before it reached the grain chamber. This reduced the total amount of convective warming caused by the incoming gas. It also provided the benefit of accelerating the freezing process by approximately 30%, reducing liquid helium usage.

For the firing, the oxygen flow rate was held at the same level as for CO-H001 (6.0 g/s). Figure 8 presents a summary pressure trace showing both CO-H001 and CO-H002.

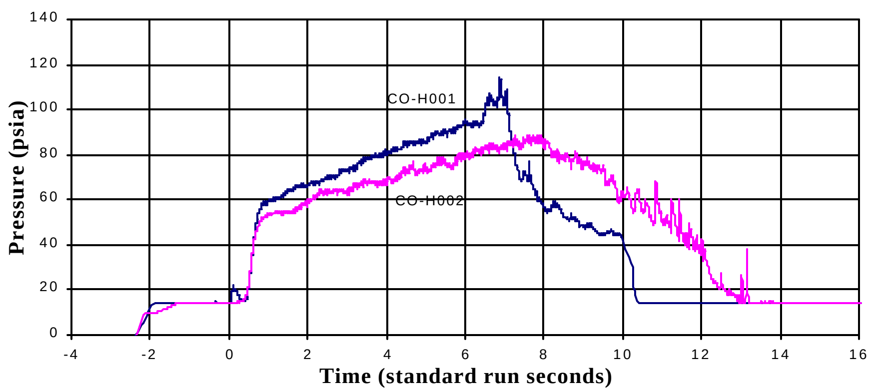

Figure 8. Pressure Trace – CO-H001 and CO-H002

As the plot illustrates, the second test had a lower pressure than the first, and a lower rise rate. The tail-off is also different; the second test goes down in pressure more smoothly, although there are still some prominent pressure spikes near the end. Referring to Figure 6, it is worth noting that CO-H002 had a higher C* efficiency than CO-H001 (88%, vs. 83%), also suggesting a cleaner burn with less ejection of solid grain material.

The igniter torch was shut off at time=1.2 seconds for CO-H002; it was left on for the duration of CO-H001. It was also shut off at time=1.2 seconds for all subsequent tests.

3.3.3.3 CO-H003 Hot Firing of SCO/GOX

The grain formation procedure for this test was the same as for CO-H002. The oxygen flow rate was reduced from 6.0 g/s to 4.0 g/s. Figure 9 shows a summary pressure trace showing both CO-H002 and CO-H003.

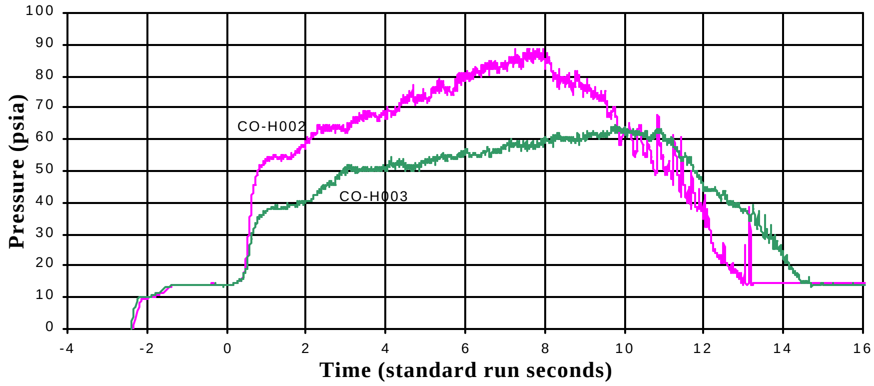

Figure 9. Pressure Trace – CO-H002 and CO-H003

As Figure 9 illustrates, CO-H003 had a lower pressure and a lower pressure rise rate than CO-H002. In addition, CO-H003 lasted longer, both until it hit its pressure peak and until the firing was completed. All of these results are consistent with a lower oxygen flow rate. CO-H003 exhibited the least pressure spiking and the flattest overall pressure trace of all the tests performed to date.

3.3.3.4 CO-H004 Hot Firing Attempt of CO/LOX

CO-H004 was set up to increase the oxygen flow rate to 10 g/s. However, the engine failed to light and the test was aborted. A close examination of the video after the test indicated that the igniter torch failed to produce a flame, causing the test failure. The cause of the igniter failure has not been determined. Possibilities include a failure in the spark generation system, a failure to deliver hydrogen or oxygen gas, or some previously unnoticed problem with the igniter design. Subsequent testing with the igniter was consistently successful.

3.3.3.5 CO-H005 Hot Firing of CO/GOX

For the grain formation process in CO-H005, a Baratron gauge with a higher range (0-100 Torr) was installed to replace the old Baratron (0-10 Torr). This enabled the freeze process to be conducted at an even higher pressure, 90 Torr, to bring inlet gas velocities lower. The triple point pressure of carbon monoxide, 116 Torr, is the upper limit for the working pressure during a gas-phase deposition process without any liquid formation. The oxygen flow rate was 10 g/s, the same flow rate attempted in CO-H004. The firing lit successfully and went to completion. Figure 10 presents pressure traces for CO-H002, CO-H003, and CO-H005. The pressure trace for CO-H005 exhibits a higher pressure and a higher pressure rise rate, both consistent with a higher oxygen mass flow. It shows relatively smooth burning for the first five seconds, followed by an extremely rough end to the burn. The test was completed faster than any of the preceding tests.

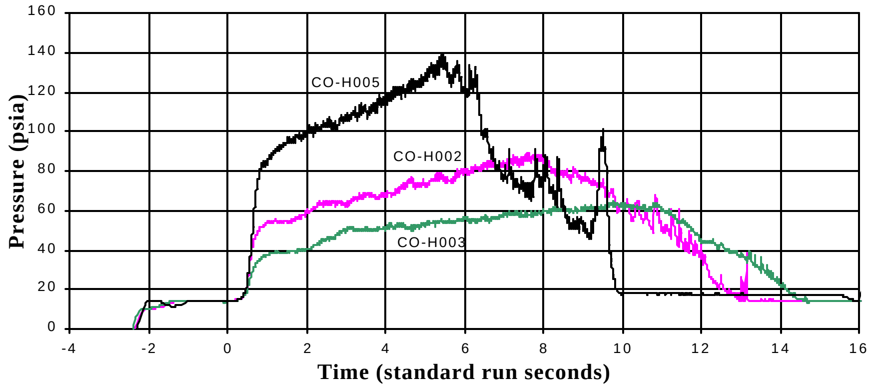

Figure 10. Pressure Trace – CO-H002, CO-H003, and CO-H005

3.3.3.6 CO-H006 Hot Firing of CO/GOX

All of the preceding tests exhibited the same overall pressure profile: a quick increase at the start, a gradual further increase for approximately 60%-70% of the test duration, and a concluding pressure tail-off which may or may not be marked with pressure spikes. Also, temperature data consistently indicated very hot temperatures near the aft end of the engine and very cold temperatures near the head end of the engine. These data supported the hypothesis that the most intense burning was occurring near the aft end of the engine, suggesting that the injected oxygen flow was jetting past the head end of the grain and burning the grain away from the bottom up. This mode of burning would be undesirable and could possibly contribute to grain break-up; ideally, the grain would regress evenly at the top and the bottom.

A new injector was designed with the intention of slowing down the incoming oxygen gas and distributing it to the head end of the grain. Figure 11 shows a section view of the engine assembly with the new conical injector in place at the head end of the grain chamber. The main oxygen flow for CO-H006 was set to 6.0 grams/second, the same flow rate used in CO-H002. Figure 12 shows the pressure traces for CO-H002 and CO-H006. The pressure trace for CO-H006 is generally lower than the trace for CO-H002. This is also reflected in the relatively poor C* efficiency for CO-H006: 77%, vs. 88% for CO-H002. In general, CO-H006 had fewer pressure spikes than CO-H002, although the overall profile remained quite non-uniform, a disappointing result. Temperature data indicated that temperatures in the head end of the grain chamber were much warmer than normal, and temperatures in the aft end of the engine were much cooler than normal.

Figure 13 shows a solid CO grain that has been formed in the Mark II engine just prior to it. Figure 14 shows the bright exhaust plume of CO_2 from one of the test firings. Figure 15 provides a summary of cryogenic hybrid regression rate results for SOX/OH_2, SH_2/GOX, SCH_4/GOX, SCH_4-Al/LOX and SC_2H_2/GOX.

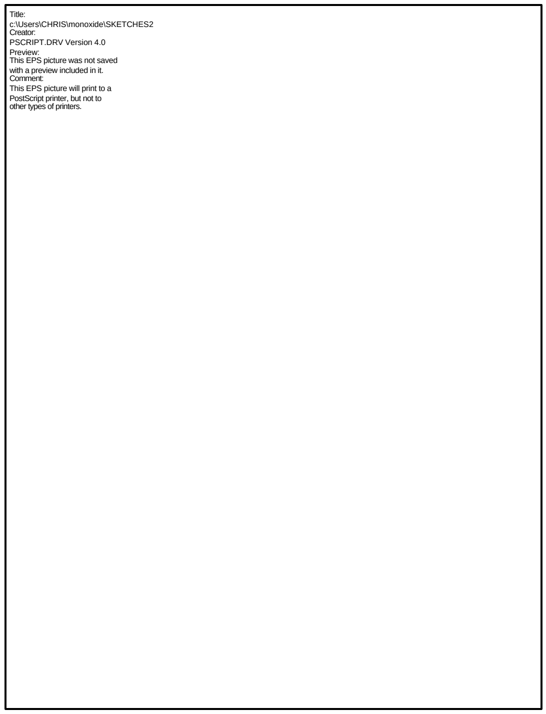

Figure 11. Engine Assembly with Conical Injector

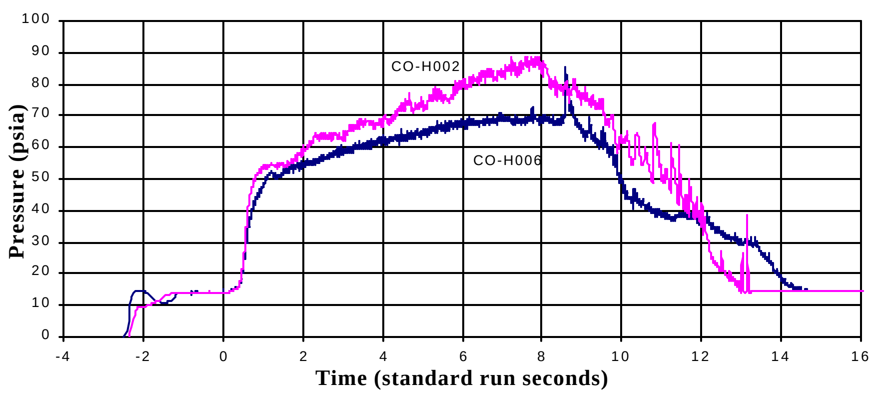

Figure 12. Pressure Trace – CO-H002 and CO-H006

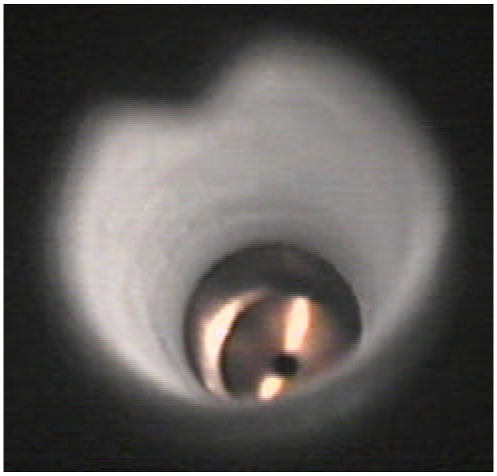

Figure 13. Solid CO Grain Formed in the Mark II Engine Prior to Firing

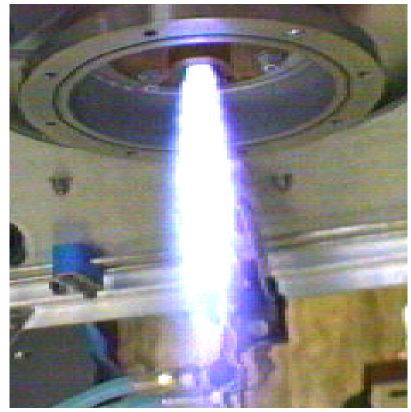

Figure 14. CO Grain Firing in the Mark II Engine

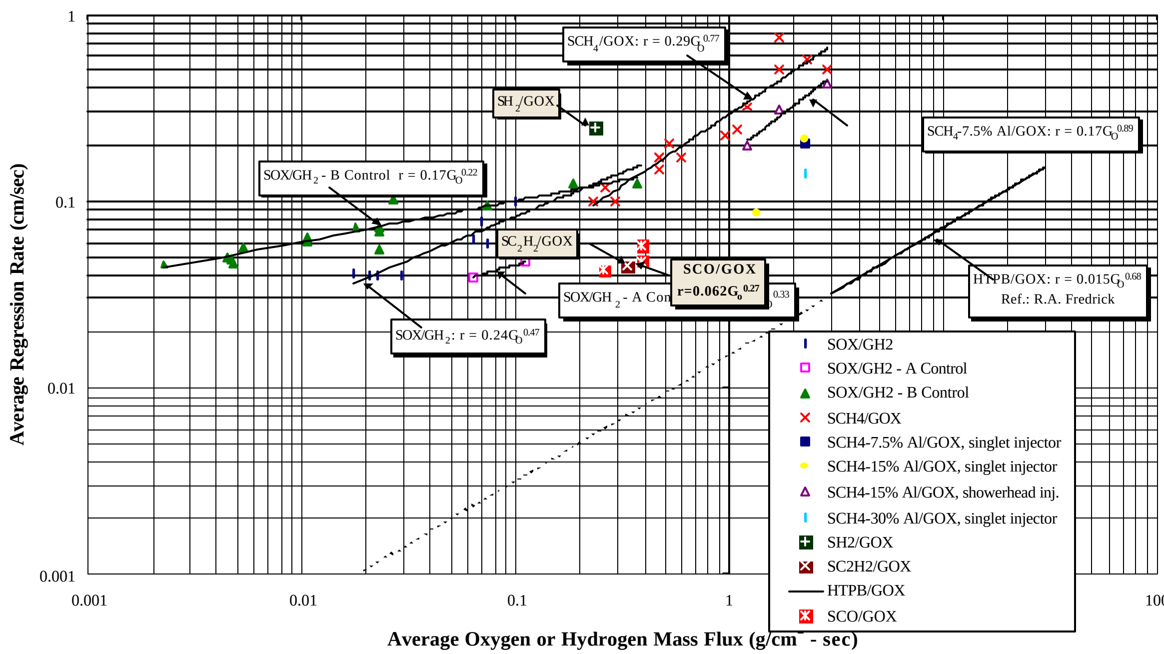

Figure 15. Summary of Cryogenic Hybrid Regression Rate Results

3.3.3.7 Summary of SCO/GOX Testing

This section summarizes the results from the ORBITEC hot firings of the SCO/GOX propellant. We discovered that SCO can be easily formed in a solid grain and the grain appears structurally sound. There were no indications of grain slipping during burns. SCO burns very well with GOX; it has been one of the smoothest burning cryogenic solids that we have tested. The pressure change with time was primarily due to the increase in area as the CO grain regressed; some contribution to the increase in grain temperature is also believed a contributor. The optimum O/F ratio was easily achieved the first time tried. The tests show great promise for the SCO/LOX propellant combination for use as a Mars sample return and a wide variety of Mars exploration applications.

3.3.4 SCH_4/GOX Hybrid Rocket Firings

ORBITEC has also completed work on a project for NASA/GRC to design, build, and test a solid methane/GOX hybrid rocket engine. A total of 24 successful test firings were performed in the final version of the Mark II engine; also an excellent Mars ISRU propellant candidate. The largest SCH_4 grain fired had a mass of 120 g. The highest steady chamber pressure attained was 240 psia, and the highest oxygen mass flow rate injected into the engine was 35 g/sec. The tests were planned using statistical experimental design (SED) to study the effects of varying grain mass, oxygen flow, grain temperature, injector type, and aluminum loading. Figure 16 shows a series of pressure curves that illustrate the effect of varying grain size. Figure 17 illustrates the effect of varying oxygen flow.

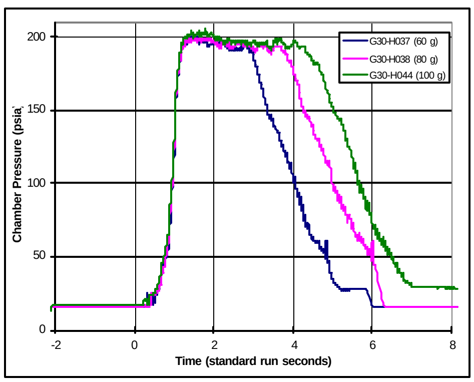

Figure 16. Pressure Curves for SCH_4/GOX Firings, Showing Effect of Varying Grain Size

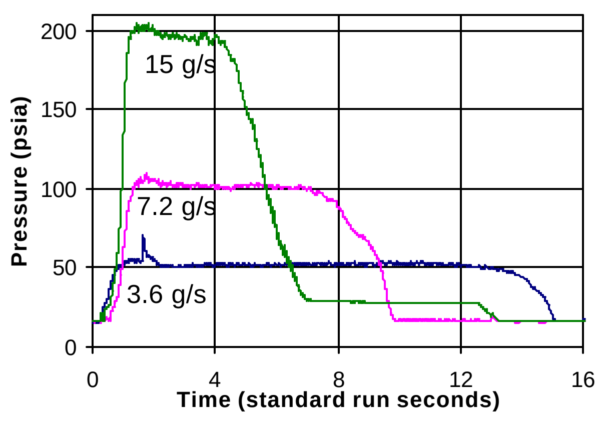

Figure 17. Pressure Curves for SCH_4/GOX Firings, Showing Effect of Varying Oxygen Flow Rate

3.4 Overall Approach for the Architecture Study

The overall approach to this architecture study is graphically displayed in Figure 18. The first step is to define the fuel and oxidizer scenarios that will be evaluated. This includes selecting fuel and oxidizer combinations, determining the planetary source of the fuel and oxidizers, and developing processing scenarios. At the beginning of the Phase I effort, the study focused on the innovative and revolutionary use of solid C and CO as fuels with LOX in both hybrid rockets and power system applications as compared to LH_2/LOX supplied from Earth. However, the focus of the study was broadened to also include ISRU propellants: SCH_4/LOX, LCH_4/LOX, SC_2H_2/LOX, LH_2/SOX, LH_2/LOX, and other secondary derivative propellants that may have significant storability advantages (e.g., H_2O_2, CH_3OH). The planetary source of each of these fuel and oxidizer combinations must be determined before the processing scenarios are developed. Some of the planetary sources of the fuel and oxidizers considered in Phase I included the atmosphere and, the regolith of the Moon (Lunar H_2/O) compared with Earth-supplied propellants. The third part of this step is the development of the processing scenarios. Conceptual designs of the processing hardware are required to determine estimates of system mass and volume, development costs, recurring costs, system life, recurring requirements, and energy requirements.

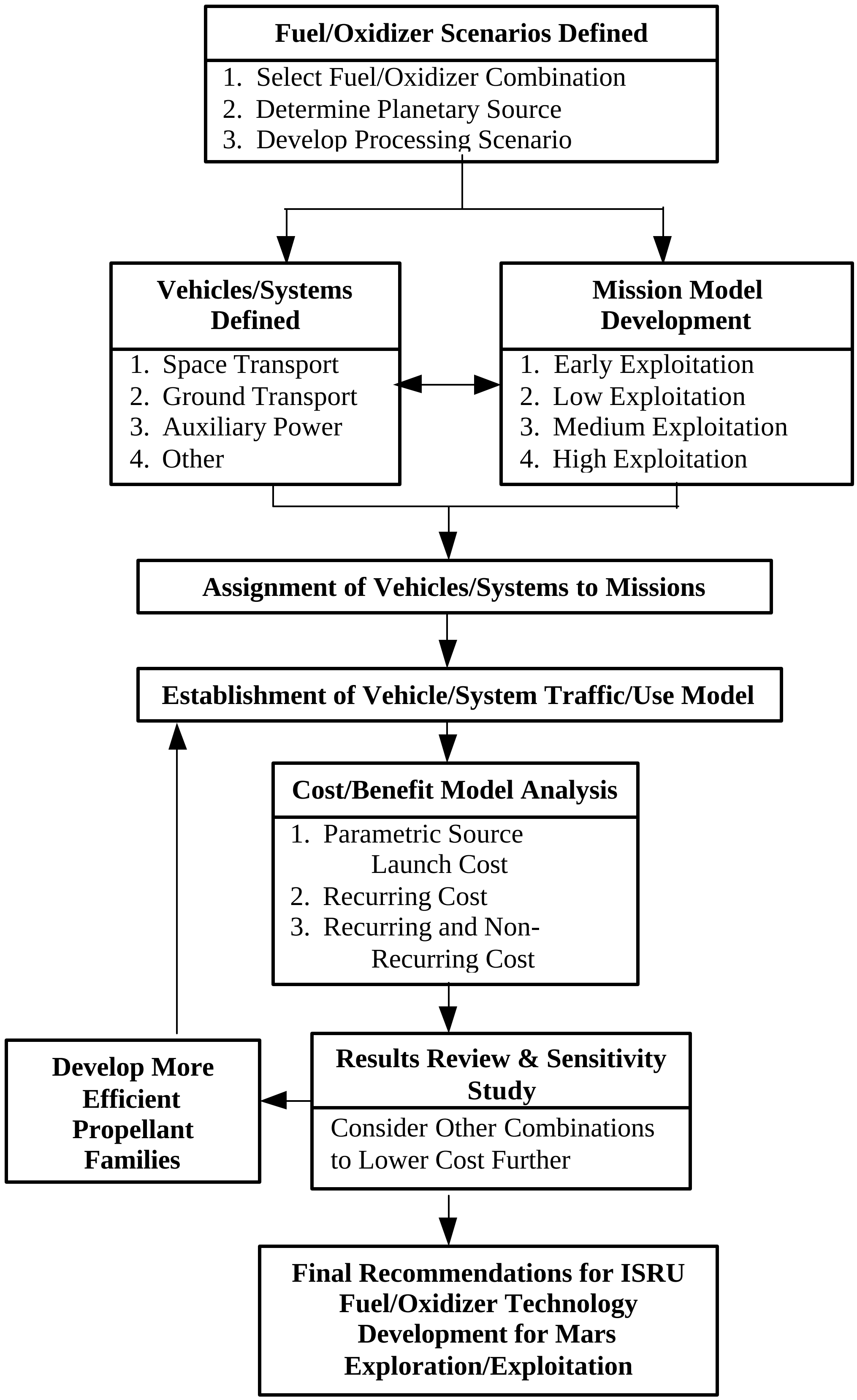

Figure 18. Overall Study Approach

The next two steps of the study occur in parallel. One is the definition of the vehicle/system families that will utilize the propellant combinations identified in first step. The general categories identified during the Phase I effort are flight vehicles, ground vehicles, and auxiliary power systems. Each propellant combination will have its own family of vehicles that cover these categories. The other step is development of the mission and traffic/use models. The mission model outlines the potential activities that require vehicle/systems. Many potential missions were identified during the Phase I effort including: scientific exploration and research, commercial exploration, terraforming, infrastructure construction, agriculture/farming, manufacturing/industrial activities, resource mining, weather/environmental, communications navigation services, surveying/mapping, personal transportation, package/mail delivery/package delivery/product delivery/food delivery/goods/services/cargo, government activity/law enforcement/emergency rescue and response, launch/space transport satellite/Earth cargo launch/space transport, auxiliary power/emergency power, life support, waste/trash management, health care/maintenance, and real/virtual travel market. The traffic/use model outlines how often these activities take place. Four different levels of human presence on Mars were defined during the Phase I effort. These levels are: (1) early exploration, (2) low presence (100 permanent inhabitants after 50 years), (3) medium presence (1,000 permanent inhabitants after 50 years) and (4) a high presence (10,000 permanent inhabitant after 50 years).

The next step is to assign vehicles/systems to the missions based upon the traffic/use model. The mission activity will help to determine the vehicle/system type that is used and the traffic/use model will help determine the number of vehicles/systems required. Four examples were developed during the Phase I effort under this step. These included: a Mars Ascent Vehicle (MAV) replacement for a Mars sample return mission, ballistic surface hopper that could travel 500 and 1,000 km distances, a rover/transporter that could travel 300 km round trip once per day, and an output auxiliary electrical generator that uses chemical power for an outpost.

The next step is the cost/benefit model and analysis.

The predicted cost-benefit of ISRU propellants and their associated production and uses is greatly affected by the assumption of what is the Earth Launch Mass (ELM) cost or Earth to Mars surface transport cost. The latter is much more difficult to estimate out into the future. The former can be straight forwardly used parametrically, once we know how much mass we must launch to Mars to support the exploration/colonization activities.

We plan to use values of $10,000, $1,000 and $400 per kg as the range of Earth Launch Mass (ELM) costs. As part of the cost-benefit analysis, we will need to understand under each mission scenario and how much mass is required from Earth. This depends on the missions that are defined, their frequency and their propellant option. We must include not only ELM propellant for Mars delivery, but all of the masses associated with storage, processing, upgrading/refurbishment, resupply, etc. of both Earth-supplied propellant and Mars-supplied propellants. We must also consider the different recurring and non-recurring costs of the flight and ground systems that are designed for each propellant use. We will estimate these costs using aerospace CER's or other software models that are available. Once the costs for all Mars-based scenarios are established, and we also know the ELM for the specific option, then we can estimate the options cost parameter. So for each given propellant family we want to analyze we will have a cost parameter for four different levels of pressure over the appropriate time period.

Once we have analyzed the costs for each option and assessed their sensitivity to the study assumptions, we would generate groups or families of propellant options that we would analyze. We expect that there may be significant benefit in selecting both high and low performance propellants for the vehicles being considered. If one looks at the modes of transportation on Earth, one can see that many different combinations exist and usually each for a good reason. We will attempt to determine the best propellant families to satisfy the lowest expected cost exploration/colonization scenarios on Mars. This family or ISRU propellant architecture will then be recommended at the end of Phase II.

3.5 System Requirements/Ground Rules Definition

Many of the system requirements and ground rules for the study have already been discussed, but a summary of the current list is provided below. This list is subject to modification from the Phase II Workshop to be held in April/May 2000.

- Purpose of the study is to assess cost-effective, in-situ production and use of Mars-derived oxidizers and fuels to guide advanced concept development, system analysis efforts, and technology and unique hardware developments
- The study timeframe includes the early manned exploration period and extends 50 years from the "end" of the initial human Mars exploration activity
- Missions to be used are those defined by the project team (as previously mentioned above)
- Earth Launch Mass (ELM) costs will be parametrically assessed at $10,000/kg, $1,000/kg, and $400/kg
- Human activity defined for the 50-year period of assessment to be 10,000 humans for high, 1000 humans for medium and 100 humans for low
- Mission vehicle assignment and mission frequency will be determined by consensus of the workshop participants and the project team and based upon the other requirements and guidelines
- All cost estimates will be in year 2000 dollars
- Flight vehicles are to include: Mars sample return vehicles, unmanned and manned surface-to-surface "ballistic hoppers", surface-to-orbit vehicles, interplanetary transport vehicles, powered balloons, winged aerocraft, single person rocket backpacks, and single person rocket platforms
- Ground vehicles are to include: automated unmanned roving vehicles, personal vehicles, two-person unpressurized rovers, manned pressurized transport rovers, and larger cargo transports
- Auxiliary power systems are to include: Brayton turbines and fuel cells for small Mars outposts
- Only propellants to be considered are those derivable from Earth (Earth delivered), the Mars atmosphere, or water resources from the Moon
- Potential propellant candidates to be considered include: CH_4/O_2, C/O_2, C_2H_2/O_2, CO/O_2, H_2O_2/CH_3OH, CH_3OH/LOX and H_2/O_2.

3.6 Propellant Family Scenarios

Propellant "family" scenarios include at least one or more propellant combinations. The propellant form can be in solid or liquid state depending upon its use as a liquid or solid propellant. We expect that up to 4 different propellant combinations could make up one family.

3.6.1 Propellant Processing Scenarios

The propellant processing scenarios that have been identified, reviewed and selected for future consideration in Phase II are shown below:

1. All Earth-supplied H_2 and O_2
2. Earth-supplied H_2, O_2 from the Mars atmosphere
3. Moon-supplied H_2 and O_2 from Lunar H_2O
4. All Mars-supplied H_2 and O_2 from H_2O in the atmosphere
5. CO and O_2 made from the Mars atmosphere
6. C_2H_2 made from Earth-supplied H_2 and Mars C and O_2 from Mars atmosphere
7. C and O_2 made from the Mars atmosphere
8. CH_4 made from Earth-supplied H_2, C and O_2 from Mars atmosphere
9. CH_4 made from Mars-supplied H_2 (atmospheric water), C and O_2 from Mars atmosphere
10. CH_3OH made from Earth H_2, Mars C and H_2O_2 from Earth H_2 and Mars O_2.

3.6.2 Propellant Performance

Analysis was performed to estimate the theoretical performance for various propellant combinations which might be used on Mars. For each propellant combination, specific impulse were generated for different expansion ratios (20:1 and 100:1) and for different atmospheric conditions (10 Torr, vacuum). NASA's CEA code was used to conduct the analysis. Table 2 presents a summary of the results. These data were used to calculate the size of different flight vehicles for different missions.

Table 2. Summary of Results – Maximum Theoretical I_{sp} for Different Propellant Combinations, Expansion Conditions

	Propellant Combination						
	CO/LOX	**LCH_4/LOX**	**LH_2/LOX**	**Carbon/LOX**	**SC_2H_2/LOX**	**CH_3OH/H_2O_2**	**Mg/CO_2**
500 psia, 100:1 to vacuum	290	402	459	335	401	359	232
100 psia, 20:1 to vacuum	260	368	429	301	367	329	209
500 psia, 100:1 to 10 Torr	276	386	445	320	387	345	220
100 psia, 20:1 to 10 Torr	240	345	408	282	346	308	193

The following seven figures (Figure 19 to Figure 25) present more detail on the performance data shown above, detailing the variation of specific impulse with O/F ratio for each case.

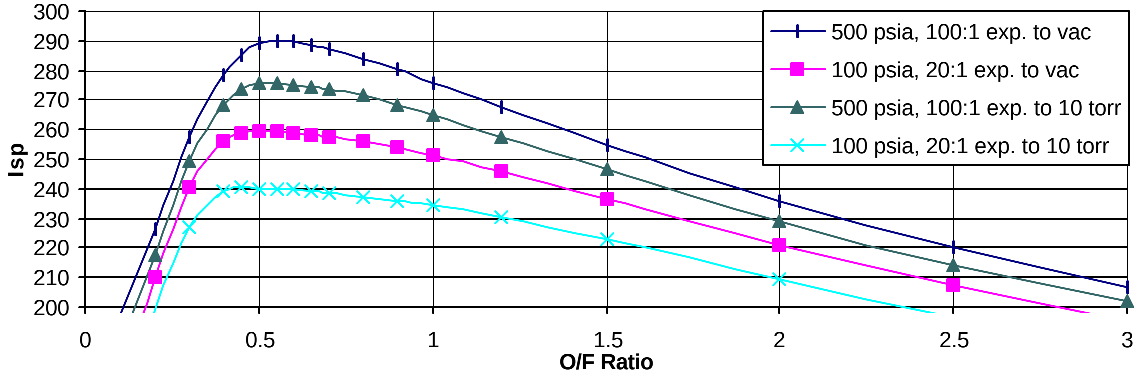

Figure 19. CO/LOX – Theoretical I_{sp}

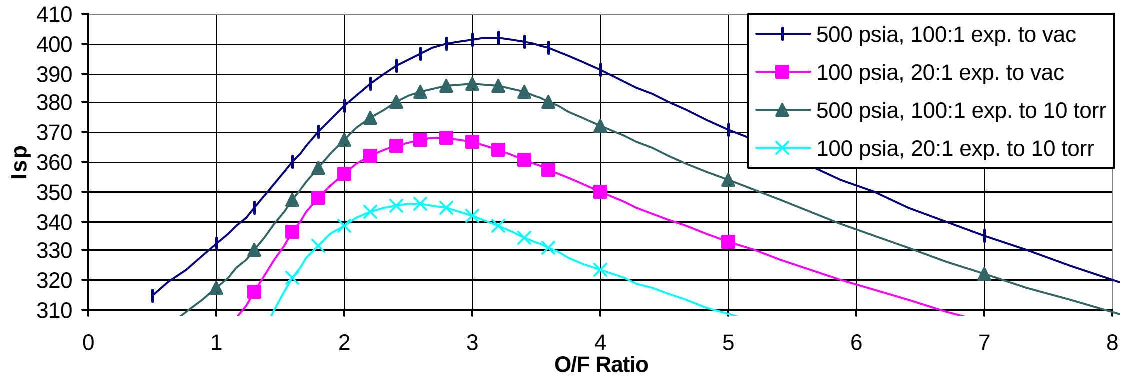

Figure 20. LCH_4/LOX – Theoretical I_{sp}

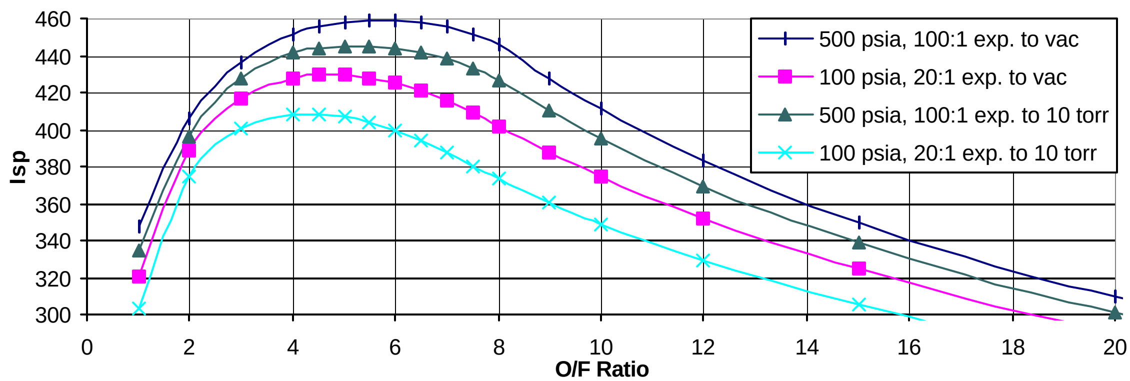

Figure 21. LH_2/LOX – Theoretical I_{sp}

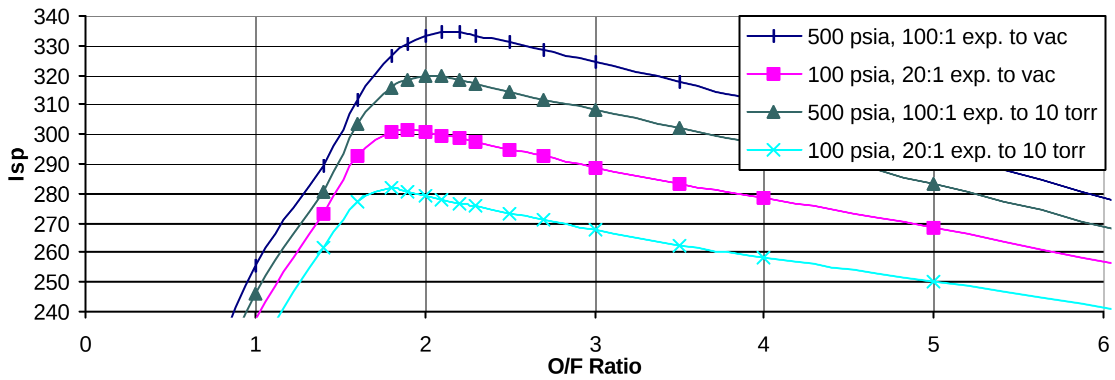

Figure 22. Carbon/LOX – Theoretical I_{sp}

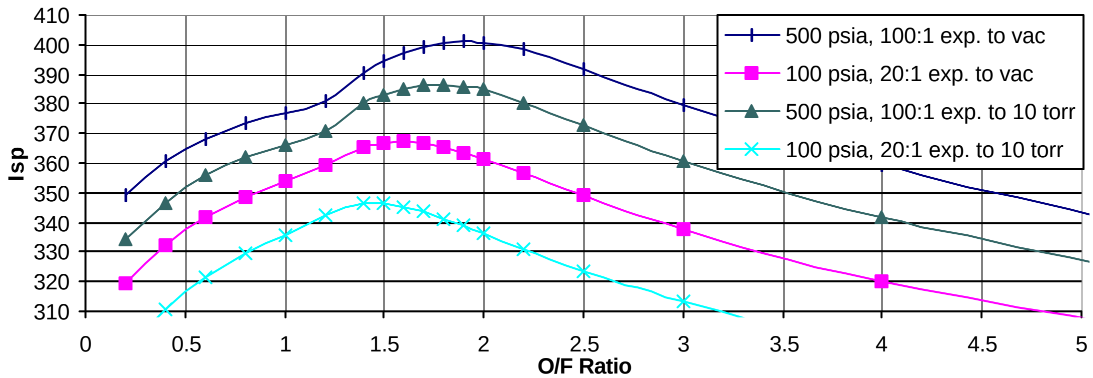

Figure 23. SC_2H_2/LOX – Theoretical I_{sp}

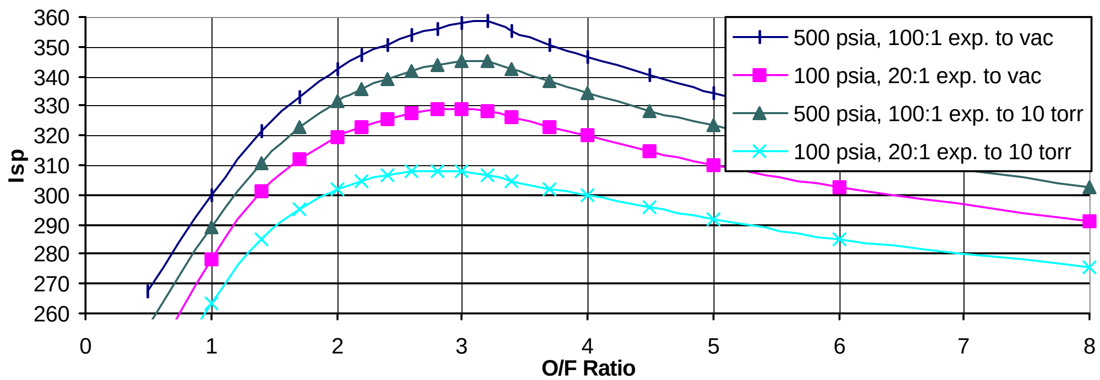

Figure 24. CH_3OH/H_2O_2 – Theoretical I_{sp}

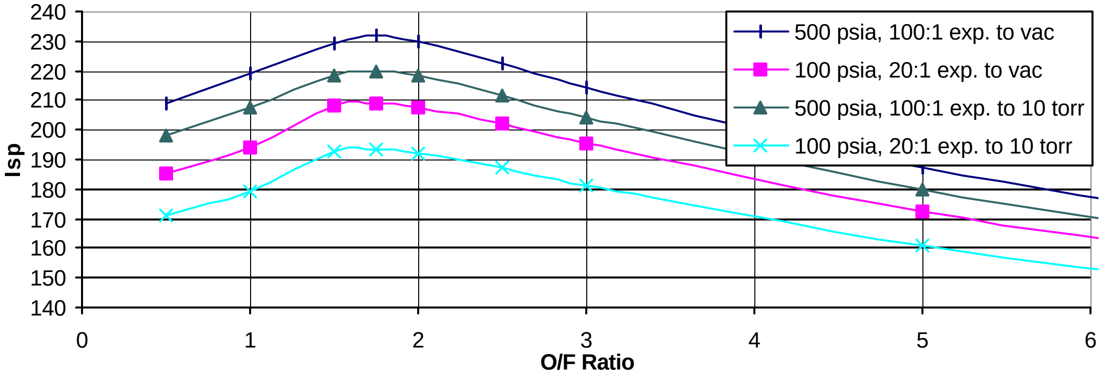

Figure 25. Mg/CO_2 – Theoretical I_{sp}

3.6.3 CO/O_2 Production Plant

The masses and energy requirements for two different sizes of CO/O_2 production plants were developed during the Phase I effort. The small production plant will convert 10 kg of CO_2 into 10 kg of CO and O_2 (combined mass) per day. The large plant will convert 1,000 kg of CO_2 into 1,000 kg of CO and O_2 (combined mass) per day. The analysis for the small production plant assumes that the CO_2 compressor and CO/CO_2 separator will operate for one cycle per day. The analysis for the large plant assume that the CO_2 compressor and CO/CO_2 separator will operate for 8 cycles per day. This will decrease the launch mass requirements when compared to a simple linear scaling. The O_2 generator in the small production plant would operate four 7 hours each day, while the large production plant would operate 24 hours per day. This again reduces the launch mass required. The results of this analysis are summarized in Table 3. The mass listed is the total launch mass required. The energy listed represents the total energy required to operate the plant each day.

Table 3. CO/O_2 Production Plant Mass and Energy Estimates

	Small	**PLANT 10 KG/DAY**	**Large**	**Plant 1,000 kg/day**
Component	**Mass (kg)**	**Energy (kW-hr)**	**Mass (kg)**	**Energy (kW-hr)**
CO_2 Compressor	120	24	1,500	2,400
Oxygen Generator	15	15	437	1,500
CO/O_2 Separator	120	24	1,500	2,400
Support Equipment	75	-	950	-
TOTAL	**330**	**63**	**4,387**	**6,300**

3.7 Mission and Traffic/Use Model

With the assumption that man has explored Mars for a 10 to 20 year period and has decided to stay and grow the population at the colony for a wide variety of reasons, we took on the task to try to identify realistic missions that require a flight or ground vehicles or chemical power/storage systems. We held brain storming sessions with the project staff and discussed possible missions with outside aerospace experts. The list of some 19 mission categories is shown below in Figure 26.

- **Scientific Exploration & Research**
- **Commercial Exploration**
- **Terraforming**
- **Infrastructure Construction**
- **Agriculture/Farming**
- **Manufacturing/Industrial Activities**
- **Resource Mining**
- **Weather/Environmental**
- **Communications Navigation Services**
- **Surveying/Mapping**
- **Personal Transportation**
- **Package/Mail Delivery/Product Delivery/Food Delivery/Goods/Services/Cargo**
- **Government Activity/Law Enforcement/Emergency Rescue/Response**
- **Launch/Space Transport Satellite/Earth Cargo Launch/Space Transport**
- **Auxiliary Power/Emergency Power**
- **Life Support**
- **Waste/Trash Management**
- **Health Care/Maintenance**
- **Virtual Travel Market**

Figure 26. Far-Term Mars Mission Categories

Work sheets were developed for each mission category; Figure 27 provides an example for "Scientific Exploration and Research."

Data needs included: number of crew, robotic or manned, mission duration, distance from base, travel time, payload and vehicle type required. Once these data are developed, then we can configure a ground or flight vehicle that can satisfy the mission need. The approach would be to develop only a few sets of ground and flight vehicles that can satisfy all the missions. Appendix B provides the "draft" worksheets for the other mission categories. Completion of these sheets will be accomplished in Phase II.

Once we have assigned a defined vehicle to the mission, the next task is to identify how often the mission needs to be accomplished. Figure 28 shows a cell printout from Excel spreadsheet that was created in Phase I to eventually provide the means to input the traffic model of all vehicles for the low, medium and high scenarios for the given mission categories and propellant family.

Mission Category: Scientific Exploration and Research

Mission/Submission Scope?	# of Crew/ Robotic	Mission Duration	Distance from Base (km)	Travel Time	Payload Mass (kg)	Vehicle Type Required
Past/Current Life on Mars – search for evidence of past life, geology of the planet, ice at poles or permafrost (tools, sample boxes, life support, rover, sample rocks/dust, measure seismic activity)	2/Robotic 2/Robotic 2/Robotic	1-5 days 1 day 3-7 days	4000 km 500 km 10,000 km	Minutes Hours Minutes	300 300 300	Ballistic Flight Ground Ballistic Flight
Meteorology – study/characterize atmosphere, dust storms, other weather Phenomena (temperate, pressure, wind velocity, solar radiation, humidity) — Delivery Vehicle Recovery Vehicle Sounding Rocket Astronomy – any orbiting systems supplied from Earth - any ground-based systems located at base, so no requirement for transport	Robotic Robotic Robotic	1 day 1 day < day	10,000 km 10,000 km ? altitude	Minutes Minutes Minutes	10 10 2	Ballistic Flight Ballistic Flight Ballistic Flight
Solar Monitoring – located at base, so no need for transport Other Science – study meteorites, characterize poles	2/Robotic 2/Robotic 2/Robotic	1-5 days 1 day 3-7 days	4000 km 500 km 10,000 km	Minutes Hours Minutes	200 50 200	Ballistic Flight Ground Ballistic Flight
Mars Moon Exploration (landing equipment, tools similar to the search for life/geology mission)	3/Robotic	1 week	Moon Orbits	Hours	100	Flight vehicle
Mission to Asteroid Belt	3/Robotic	Months	Asteroid Belt	Hours	100	Flight vehicle

Figure 27. Example of a Mars Mission Worksheet for Scientific Exploration and Research

[Note: each period represents 5 years]

Propellant: Solid CO/LOX

Mission Area	Time Period (low model)											Time Period (medium model)											Time Period (high model)											Total Period Summary	
	1	2	3	4	5	6	7	8	9	10	Totals	1	2	3	4	5	6	7	8	9	10	Totals	1	2	3	4	5	6	7	8	9	10	Totals	Low	Medium
1 Scientific Exploration & Res																																			
FV1											0											0											0	0	0
FV2											0											0											0	0	0
FV3											0											0											0	0	0
FV4											0											0											0	0	0
FV5											0											0											0	0	0
Totals:	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
GS1											0											0											0	0	0
GS2											0											0											0	0	0
GS3											0											0											0	0	0
GS4											0											0											0	0	0
GS5											0											0											0	0	0
Totals:	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Figure 28. Cell Printout from Preliminary Traffic Model Worksheet

3.8 Vehicle/System Families Scenarios

After we have developed the mission, we will assign the appropriate type of vehicle to the mission. The types of vehicles are listed below in Figure 29 under the three vehicle categories.

Flight Vehicles	Ground Vehicles	Power Systems
MAVs for Sample Return	Automated Rovers	Turbine
Ballistic Hoppers	Personal Closed Rovers	Fuel Cell
Surface to Orbit	2-Person Open Rovers	
Interplanetary	Multi-Person Closed Rover	
Powered Balloons	Large Cargo Transports	
Winged Aerocraft		
Single Rocket Backpacks		
Single Rocket Platforms		

Figure 29. Vehicle Categories to Be Considered to Satisfy Mission Requirements

For each of these vehicles, we will define a vehicle concept/design with an estimated dry payload Mars and propellant storage/use capability that will allow the mission to be satisfied. For example a hopper vehicle for LH_2/LOX would be expected to be smaller and carry less propellant than an equivalent SCO/LOX-based hopper. In addition to these vehicles, the infrastructure/support systems also have to be conceptualized.

Examples of vehicles defined for mission applications are provided in the next section of the report.

3.9 Assignment of Vehicles/Systems to Missions and Traffic/Use Model

In the all-up study that we are planning for Phase II, the assignment of the defined vehicle types and sizes (propellant dependant) and infrastructure/ISRU systems would be made and a traffic model developed such that costs can be determined. Obviously, in Phase I we did not have the resources to do a complete analysis. Therefore, we chose to pick 4 missions and define vehicles/systems that could provide us some cost-benefit answers based on an estimated ELM cost. Four systems were defined; they were:

- MAV Replacement for Mars Sample Return Mission
- Ballistic Surface Hopper, Assuming H/O, CO/O,CH_4/O,C/O and Single Stage, 1000 Kg Payload, Fly to 500, 1000 Km Distances
- Rover/Transporter to 300 Km Distance Once Per Day, Using Fuel Cell or Brayton Cycle
- Outpost Chemical Power Using Fuel Cell or Brayton Cycle and H/O, CO/O, CH_4/O, CH_3OH/O

In the sections that follow, these four applications are analyzed and characterized.

3.9.1 TSTO Mars Ascent Vehicle (MAV)

The effects of using different terrestrial and ISRU propellants were explored by determining the mass characteristics of a two-stage-to orbit (TSTO) Mars Ascent Vehicle (MAV) for a variety of propellant and engine combinations. The analysis included an estimation of the propellant, vehicle, and ELM required for each system. The mission characteristics and vehicle assumptions common to all propellants are summarized in Table 4. The overall MAV mission is to place a 3.6-kg payload into orbit for retrieval by an Earth-bound vehicle. The initial launch velocity refers to the velocity supplied by the rotation of Mars at the launch site. As a first approximation for the Phase I effort, the subsystem masses were assumed to be the same for each MAV, comprised of the power systems; guidance, navigation, and control; thermal control; auxiliary cold gas propulsion; and structures. A rendering of one possible MAV configuration is shown in Figure 30.

Table 4. Mission Characteristics and Assumptions

Orbit:	600 km
Orbit Type:	Circular
Payload Mass:	3.6 kg
Stage 1 Subsystem Mass:	16.9 kg
Stage 2 Subsystem Mass:	1.7 kg
Initial Launch Velocity:	241 m/s
First Stage Delta-V:	2382 m/s
Second Stage Delta-V:	1514 m/s
Total Delta-V:	4137 m/s

Figure 30. TSTO Mars Ascent Vehicle

Performance characteristics for each system are summarized in Table 5. The propulsion system mass fractions, defined by Equation 1, were assumed to be the same for bi-propellant and hybrid systems while those used for the solid system were slightly lower. The ideal I_{SP} was calculated for each propellant combination using the NASA/GRC Chemical Equilibrium Analysis (CEA) program and the delivered I_{SP} was determined by applying the appropriate efficiency for each type of propulsion system. No attempt was made to optimize the mixture ratio from a systems point of view; the engines were assumed to run at the mixture ratio yielding the highest I_{SP}. An explanation of the CEA performance calculations and results is given in Section 3.6.2.

$$(1) \qquad MF_{ps} = \frac{M_{ps,dry}}{M_p}$$

where:

$MF_{ps} = propulsion\ system\ mass\ fraction$

$M_{ps,dry} = dry\ mass\ of\ primary\ propulsion\ system$

$M_p = total\ propellant\ mass$

Table 5. Performance Characteristics for Each Propellant Combination

Propellants	Propulsion System	Mixture Ratio	Stage 1 PSMF*	Stage 2 PSMF*	I_{SP} Efficiency (%)	Stage 1 Delivered I_{SP} (sec)	Stage 2 Delivered I_{SP} (sec)
SCO/LOX	Hybrid	0.57	0.27	0.66	92	253.7	267.0
SC/LOX	Hybrid	2.1	0.27	0.66	92	294.3	307.7
SC-H_2/LOX**	Hybrid	2.2	0.27	0.66	92	311.1	324.6
SC_2H_2/LOX	Hybrid	1.8	0.27	0.66	92	355.6	368.6
HTPB/LOX	Hybrid	2.5	0.27	0.66	92	331.2	341.3
LCH_4/LOX	Bi-Propellant	3.2	0.27	0.66	95	367.0	381.6
CTPB binder	Solid	-	0.24	0.63	95	268.0	279.4

*PSMF = Propulsion System Mass Fraction
**Solid carbon with 5% H_2 additive by mass

The Engineering Equation Solver (EES) software was employed to simultaneously solve for the MAV system unknowns for each propellant combination. The required program input and resulting output are listed below.

MAV Program Input

- Delivered I_{SP} for stage 1
- Delivered I_{SP} for stage 2
- Payload mass
- Subsystem mass for stage 1
- Subsystem mass for stage 2
- Propulsion system mass fraction for stage 1
- Propulsion system mass fraction for stage 2
- Delivered Delta-V required by stage 1

- Delivered Delta-V required by stage 2

MAV Program Output

- Mass of propellant required for stage 1
- Mass of propellant required for stage 2
- Mass of stage 1 propulsion system
- Mass of stage 2 propulsion system

The results of the analysis are summarized in Table 6**,** including the propellant and vehicle dry masses; GLOW (Gross Lift Off Weight of the Mars Launch Mass); and ELM (Earth Launch Mass - the total mass of the MAV systems and propellant transported from Earth). Terrestrial propellants are those which are transported to Mars from Earth, and ISRU propellants are assumed to be manufactured by existing Martian infrastructure, and hence, are not reflected in the ELM. While carbon is available on Mars, it was assumed that the hydrogenated solid carbon grain was manufactured on Earth. The resulting grain is a highly storable, compact, and inert package, and eliminates the need to transport LH_2, or manufacture it on Mars.

In this case it was assumed that the entire MAV system (minus any ISRU propellants) is brought from Earth. However, these results do not take into account the various subsystems that would be required for transporting the different propellants. In this regard, the more storable solid carbon, HTPB, and solid grains would require less thermal control and packaging than the liquid hydrogen. Note that in the case of a pure ISRU propellant all of the chemicals are derived from the Martian environment, as in the case of SCO/LOX, rendering the vehicle dry mass identical to the ELM. The ELM is reduced between 42-68% through the use of ISRU, depending upon the propellant combination. Per the aforementioned ground rules, the lowest ELM for the cases considered here would be realized by the use of solid acetylene, reducing the ELM to 32.1 kg. As expected, the largest ELM is associated with the only non-ISRU system considered (solid) where the ELM is well over twice as high as for the heaviest of the ISRU systems. More detailed mass breakdowns for most systems are given in Tables 7a through 7f (mass is in kg).

Table 6. Summary of MAV Analysis

Propellant Combination	Propulsion System	Terrestrial Propellants	Propellant Mass (kg)	Dry Mass* (kg)	GLOW (kg)	ELM (kg)
SCO/LOX	Hybrid	-	107.2	50.9	161.7	50.9
SC/LOX	Hybrid	-	65.9	38.7	108.2	38.7
SC-H_2/LOX**	Hybrid	C, H_2	56.3	36.0	95.9	53.3
SC_2H_2/LOX	Hybrid	H_2	40.4	31.0	75.0	32.1
HTPB/LOX	Hybrid	HTPB	48.2	33.5	85.3	47.3
LCH_4/LOX	Bi-Propellant	H_2	37.5	30.2	71.3	32.4
CTPB binder	Solid	Solid	81.3	40.7	125.6	122.0

*Dry mass does not include 3.6 kg payload

**SC with 5% H_2 additive by mass

Table 7a. Vehicle Mass Breakdown
Fuel: SCO
Oxidizer: LOX
Engine Type: Hybrid

Stage 1 Mass	
Main Propulsion:	26.6
Subsystems:	16.9
Propellant:	98.6
Stage 1 Total:	142.1
Stage 2 Mass	
Main Propulsion:	5.7
Subsystems:	1.7
Propellant:	8.6
Stage 2 Total:	16.0
Payload Package:	3.6
Total Propellant Mass:	107.2
Total Dry Mass:	50.9
Total Launch Mass:	161.7

Table 7b. Vehicle Mass Breakdown
Fuel: SC
Oxidizer: LOX
Engine Type: Hybrid

Stage 1 Mass	
Main Propulsion:	16.1
Subsystems:	16.9
Propellant:	59.8
Stage 1 Total:	92.8
Stage 2 Mass	
Main Propulsion:	4.0
Subsystems:	1.7
Propellant:	6.1
Stage 2 Total:	11.8
Payload Package:	3.6
Total Propellant Mass:	65.9
Total Dry Mass:	38.7
Total Launch Mass:	108.2

Table 7c. Vehicle Mass Breakdown
Fuel: SC with 5% H_2 additive (by mass)
Oxidizer: LOX
Engine Type: Hybrid

Stage 1 Mass	
Main Propulsion:	13.8
Subsystems:	16.9
Propellant:	50.9
Stage 1 Total:	81.6
Stage 2 Mass	
Main Propulsion:	3.6
Subsystems:	1.7
Propellant:	5.4
Stage 2 Total:	10.7
Payload Package:	3.6
Total Propellant Mass:	56.3
Total Dry Mass:	36.0
Total Launch Mass:	95.9

Table 7d. Vehicle Mass Breakdown
Fuel: LCH_4
Oxidizer: LOX
Engine Type: Bi-Propellant

Stage 1 Mass	
Main Propulsion:	9.0
Subsystems:	16.9
Propellant:	33.6
Stage 1 Total:	59.5
Stage 2 Mass	
Main Propulsion:	2.6
Subsystems:	1.7
Propellant:	3.9
Stage 2 Total:	8.2
Payload Package:	3.6
Total Propellant Mass:	37.5
Total Dry Mass:	30.2
Total Launch Mass:	71.3

Table 7e. Vehicle Mass Breakdown
Propellant: CTPB
Engine Type: Solid

Stage 1 Mass	
Main Propulsion:	17.5
Subsystems:	16.9
Propellant:	74.0
Stage 1 Total:	108.4
Stage 2 Mass	
Main Propulsion:	4.6
Subsystems:	1.7
Propellant:	7.3
Stage 2 Total:	13.6
Payload Package:	3.6
Total Propellant Mass:	81.3
Total Dry Mass:	40.7
Total Launch Mass:	125.6

Table 7f. Vehicle Mass Breakdown
Fuel: SC_2H_2
Oxidizer: LOX
Engine Type: Hybrid

Stage 1 Mass	
Main Propulsion:	9.8
Subsystems:	16.9
Propellant:	36.2
Stage 1 Total:	62.9
Stage 2 Mass	
Main Propulsion:	2.8
Subsystems:	1.7
Propellant:	4.2
Stage 2 Total:	8.7
Payload Package:	3.6
Total Propellant Mass:	40.4
Total Dry Mass:	31.0
Total Launch Mass:	75.0

3.9.2 Martian Hopper Analysis

The effects of using different terrestrial and ISRU propellants were also explored by determining the mass characteristics of a Mars ballistic hopper for four propellant combinations. The analysis included an estimation of the amount of propellant, vehicle mass, and earth launch mass required for each system. The hopper is designed to transport personnel, supplies, and equipment from one base to another, using a powered landing, refueling at each base.

Analysis was conducted to determine the optimum launch angle and required propulsion system delta-V for a given trip distance. The geometry of a ballistic trajectory is defined in Figure 31. From the initial value problem (Thomson, pp. 93), the angular position can be shown to be:

$$\tan\left(\frac{\phi}{2}\right) = \frac{-\left(Rv_o^2/K\right)\sin\beta_o\cos\beta_o}{\left(Rv_o^2/K\right)\cos^2\beta_o - 1}$$

The optimum β_o and minimum ΔV can be found by solving the above equation for v_o, taking the derivative of the resulting equation with respect to β_o, and setting the numerator equal to zero:

$$\beta_o = \tfrac{1}{2}\arctan\left[\cot\left(\frac{\phi}{2}\right)\right]$$

$$v_{min} = \sqrt{\left[\frac{K}{R}\frac{2\cos(2\beta_o)}{1+\cos(2\beta_o)}\right]}$$

The eccentricity of the elliptical path, the height of apogee, and the time of flight can also be determined by following Thomson's analysis in Section 4.17.

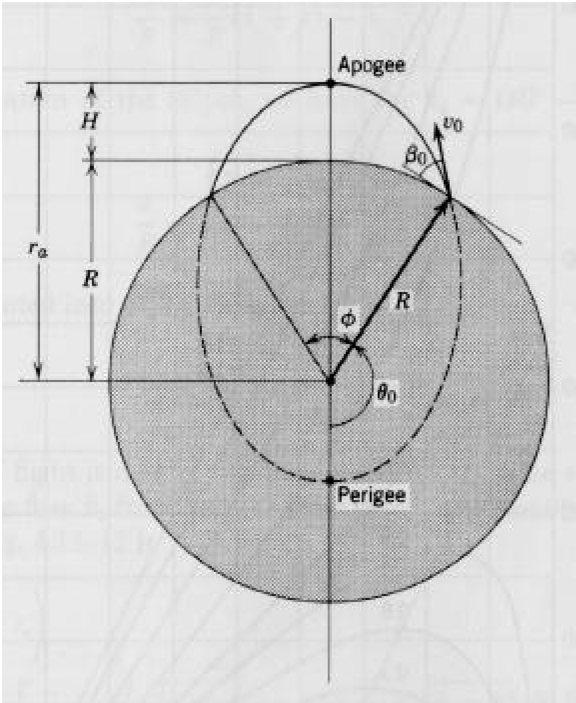

Figure 31. Geometry of Ballistic Trajectory

With the velocity increment known, the rocket equation can be used to determine the vehicle mass fraction (neglecting gravity and drag losses):

$$\frac{M_i}{M_f} = \exp\left(\frac{\Delta v}{I_{sp}\, g}\right)$$

The following charts (Figures 32-40) were produced using a spreadsheet analysis of this basic approach and allows the payload mass, structural mass fraction, and propellant specific impulse to vary parametrically.

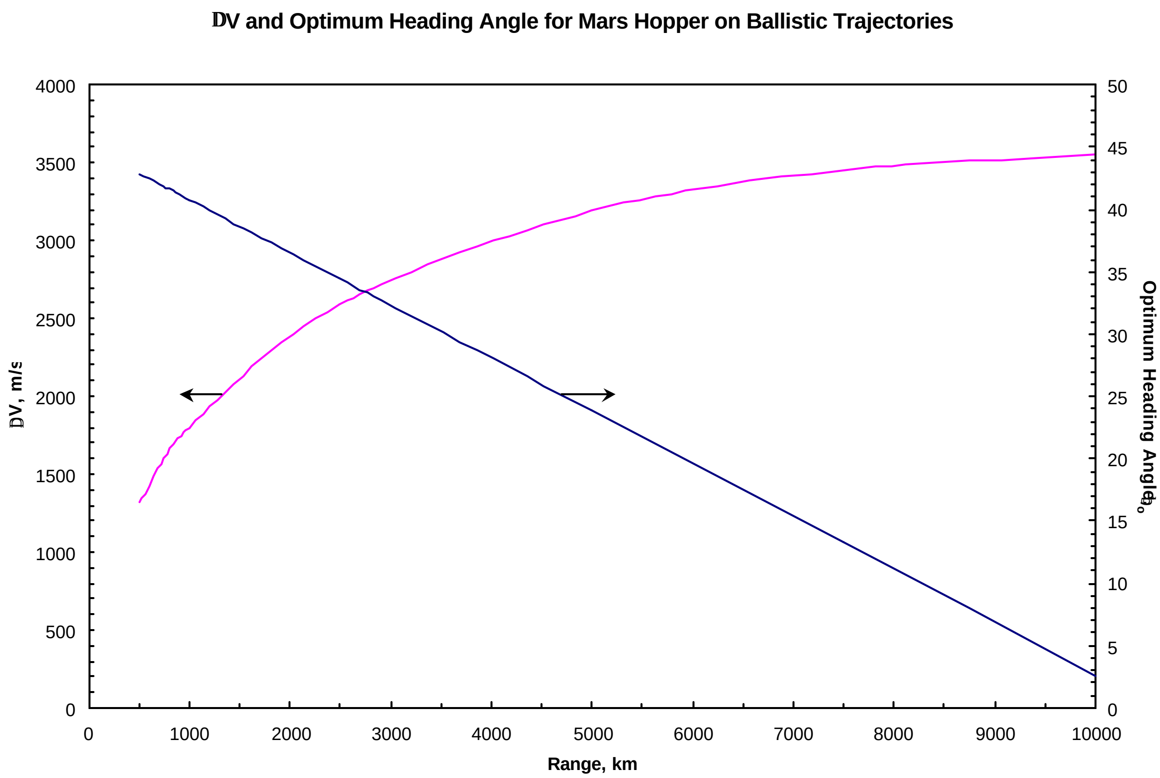

Figure 32. Delta V and Optimum Heading Angle for Mars Hopper

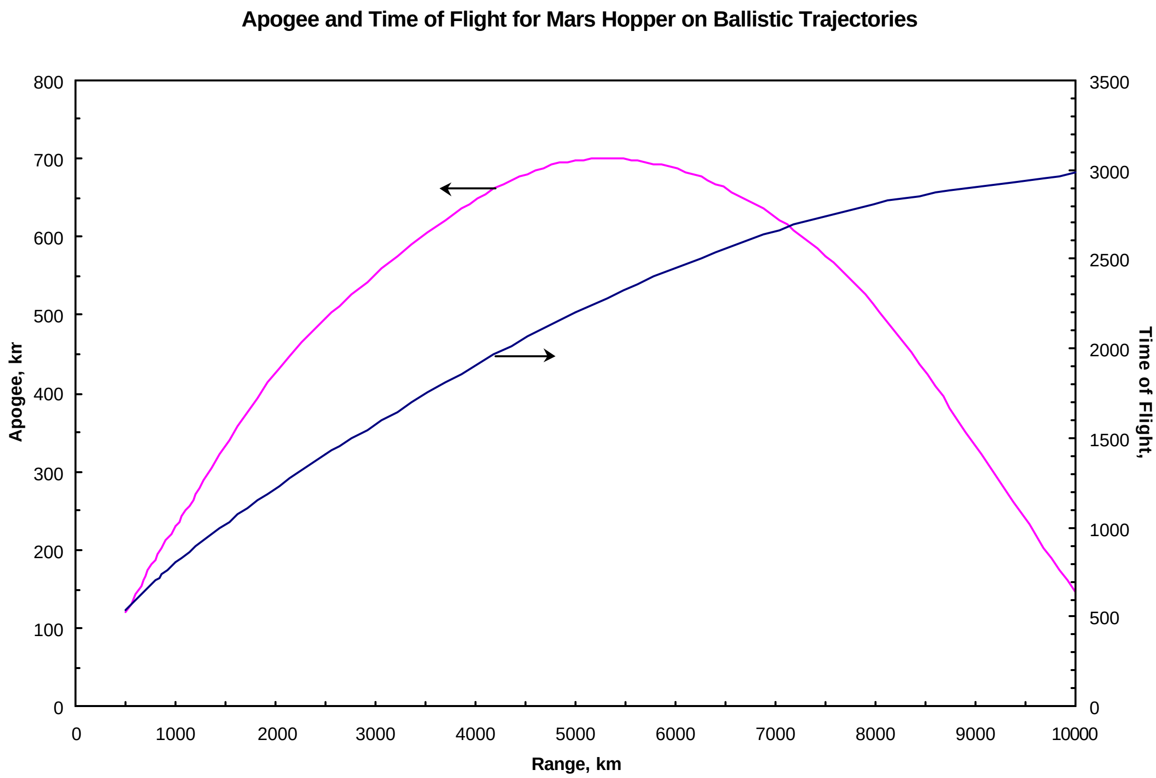

Figure 33. Apogee and Time of Flight for Mars Hopper

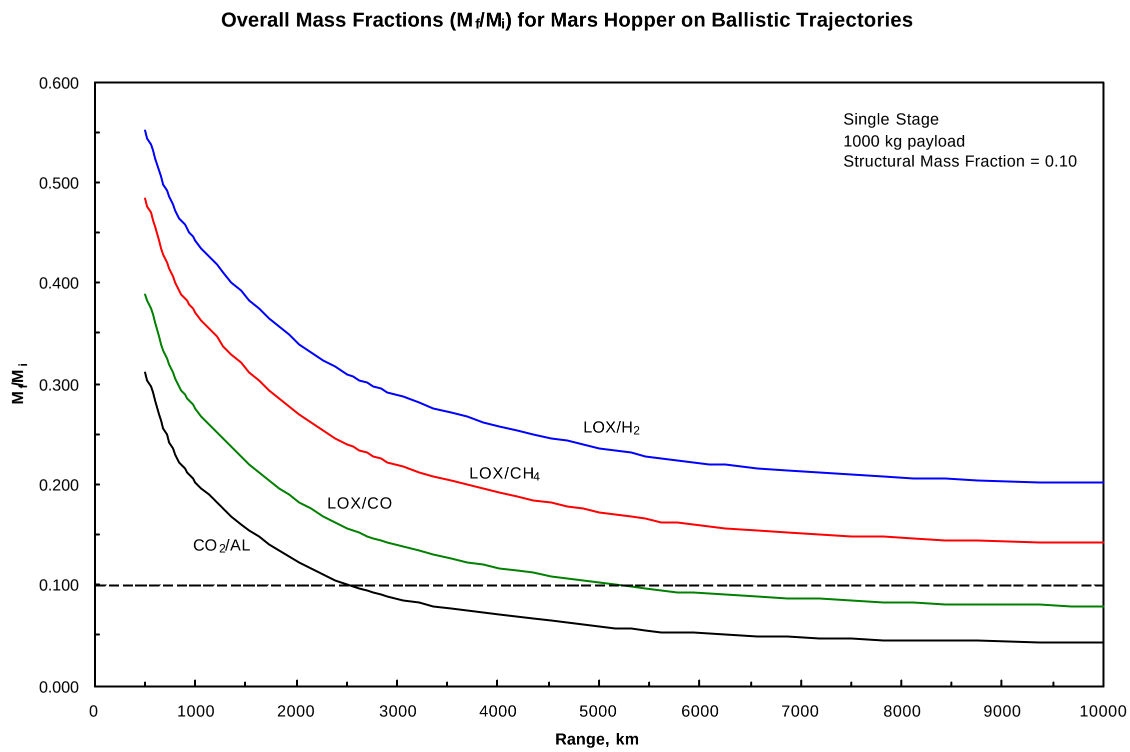

Figure 34. Overall Mass Fractions for Mars Hopper

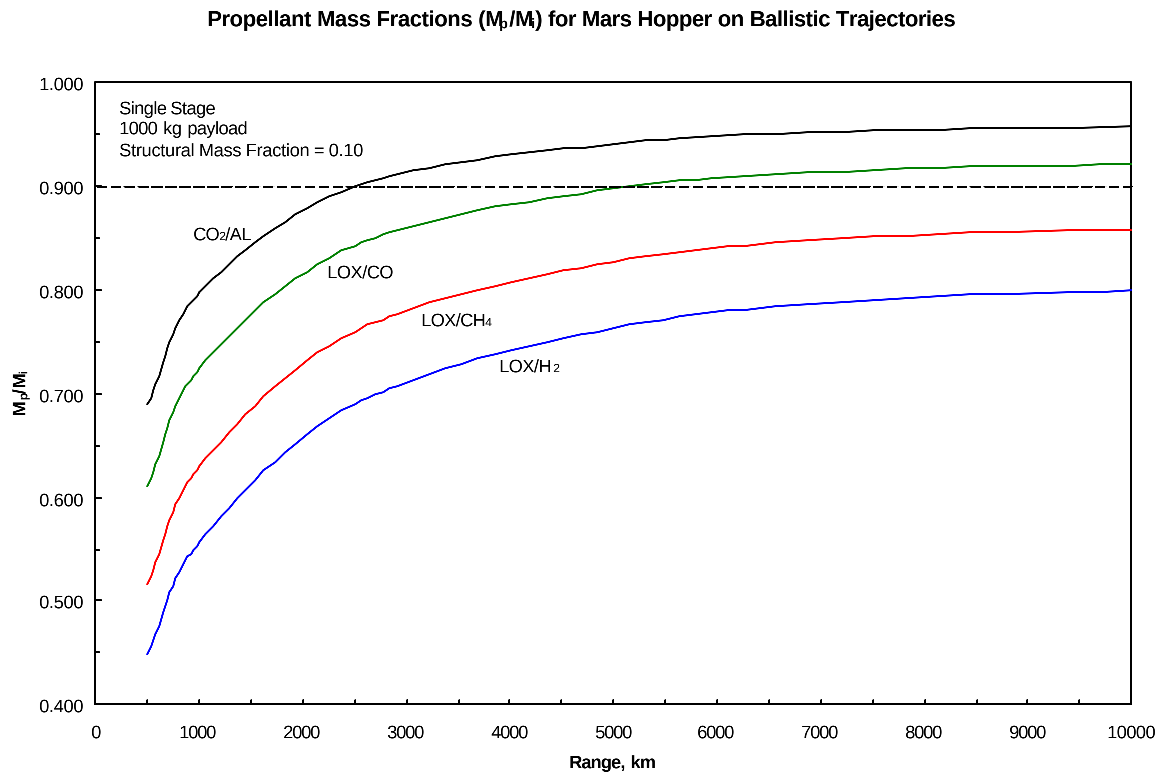

Figure 35. Propellant Mass Fractions for Mars Hopper

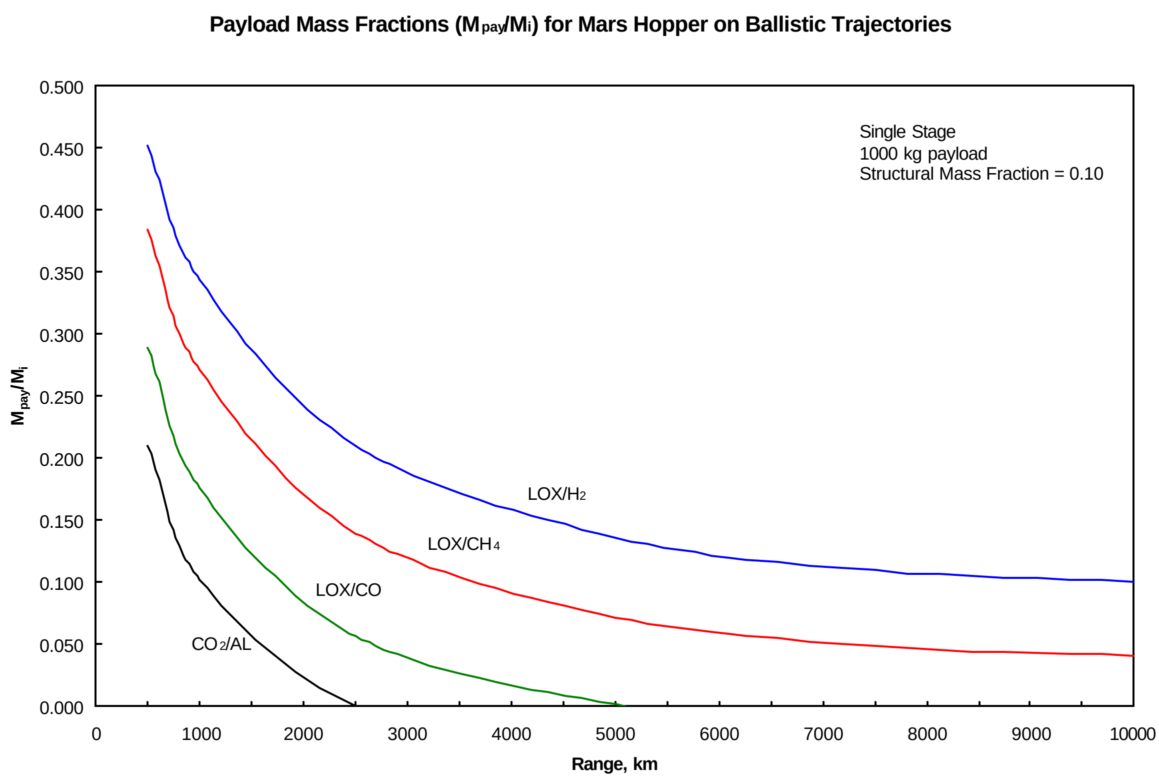

Figure 36. Payload Mass Fractions for Mars Hopper

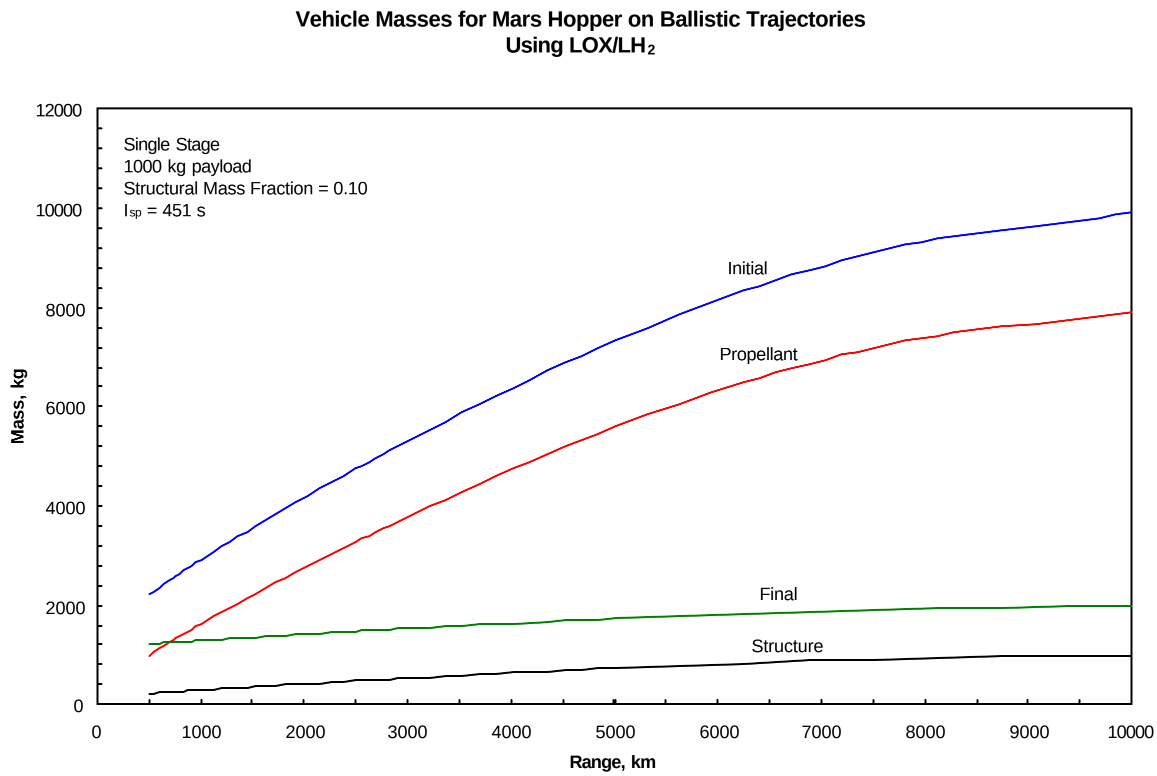

Figure 37. Vehicle Masses for Mars Hopper using LOX/LH_2

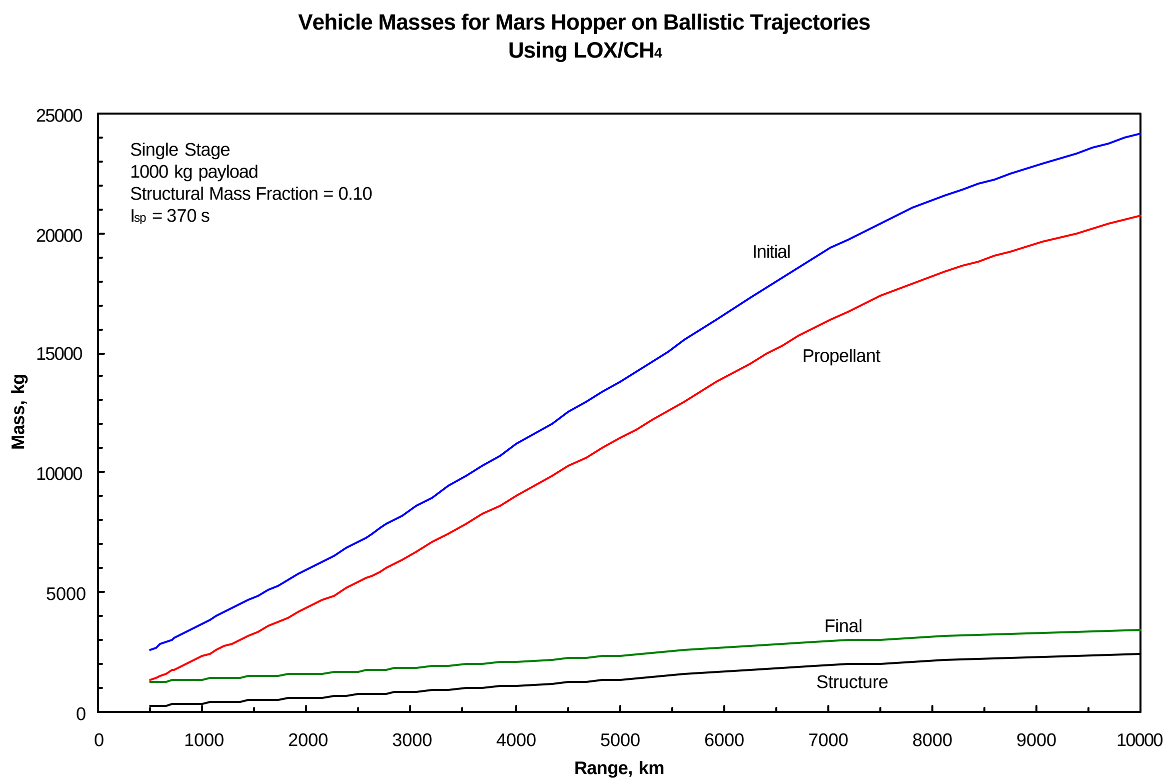

Figure 38. Vehicle Masses for Mars Hopper using LOX/CH_4

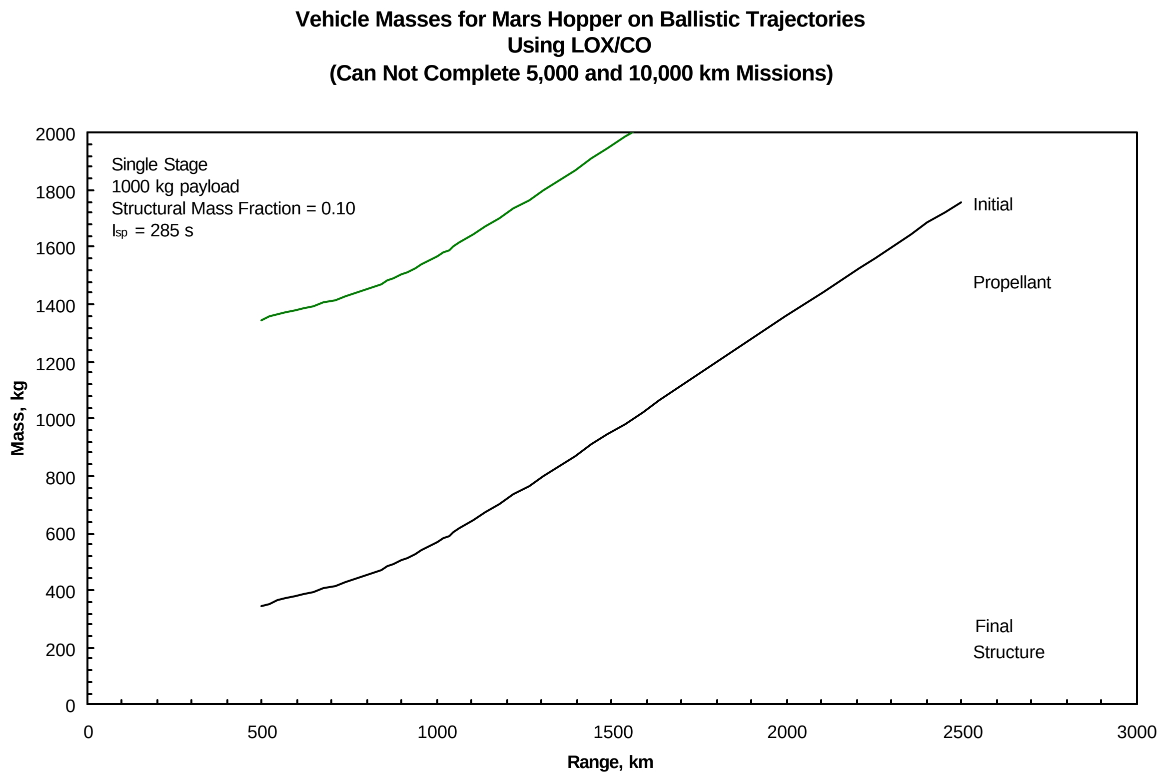

Figure 39. Vehicle Masses for Mars Hopper using LOX/CO

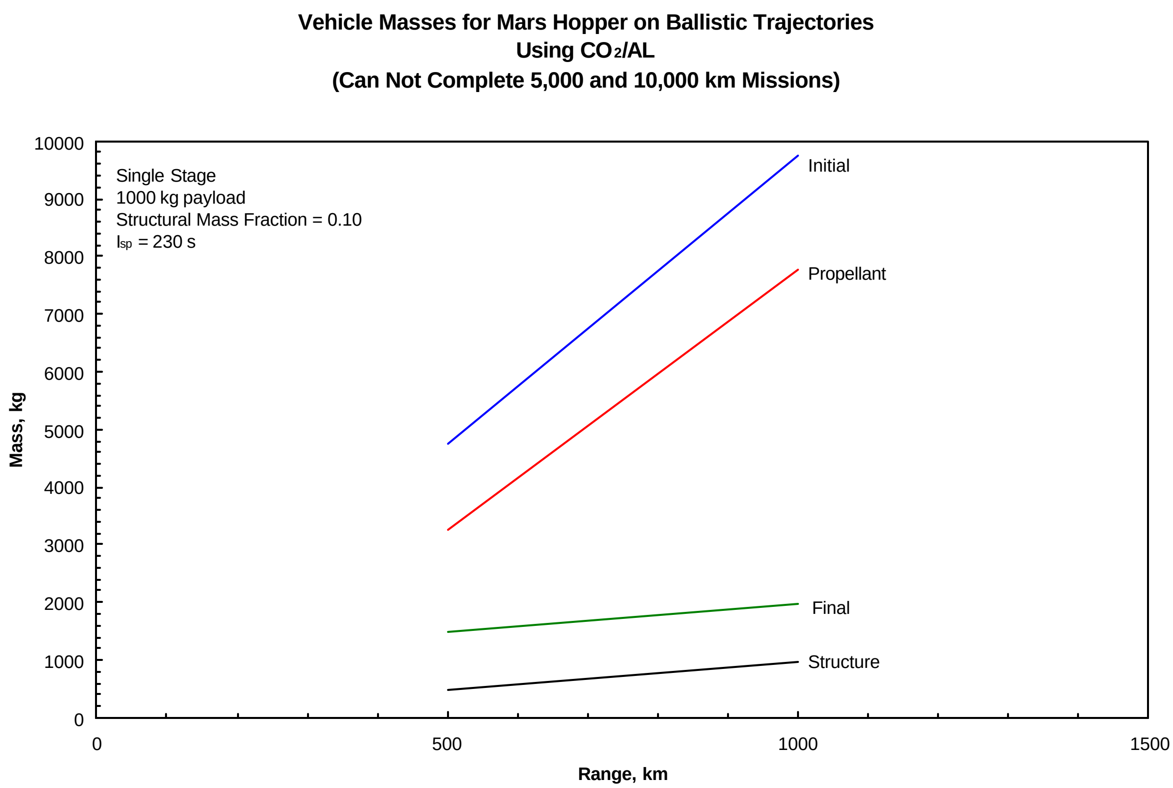

Figure 40. Vehicle Masses for Mars Hopper using CO_2/Al

Three specific hopper distances were selected for comparison: 500, 1000, and 2500 km. The total vehicle delta-V required for a one-way trip with a powered landing is shown in Table 8 for each case. The total non-propulsion hopper mass, comprised of a two-man crew, life support, transport payload, and miscellaneous vehicle systems and supplies, was assumed to be the same for all distances and propellants. A mass summary is shown in Table 9 and a rendering of one possible hopper configuration is shown in Figure 41. This mass breakdown is assumed for all hopper results presented in this section.

Table 8. Delta-V Required

Trip Distance (km)	Total Vehicle Delta-V Required (km/sec)
500	2636
1000	3600
2500	5168

Table 9. Hopper Mass Breakdown

Description	Mass (kg)
Two Man Crew:	158.8
Crew Cabin and Life Support:	453.6
Transport Payload:	1000
Misc. Vehicle Systems and Supplies:	260.3
Total Non Propulsion Hopper Mass:	1873

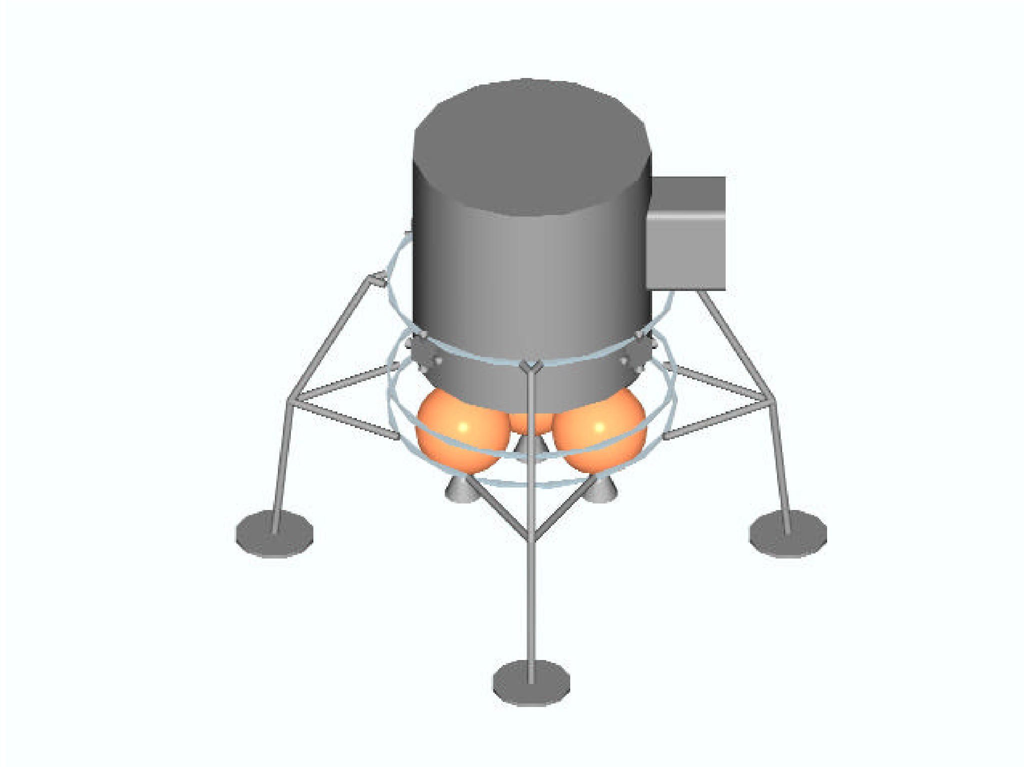

Figure 41. Mars Ballistic Hopper

The amount of propellant required, propulsion system mass (propellant tanks, engine, fluid delivery system, and nozzle) and total vehicle mass were parametrically calculated as a function of the propellant combination, structural mass fraction (defined by Equation 2), and trip distance, using the rocket equation. Specific impulse, efficiency, and mixture ratio data for the propellants were the same as shown in Table 2 for the second stage of the MAV, for LH_2/LOX: I_{SP} = 436 sec, delivered I_{SP} efficiency = 95%, and O/F ratio = 6.0.

(2) $$MF_s = \frac{M_{ps,dry}}{M_p + M_{ps,dry}}$$

where:

MF_s = *structural mass fraction*

$M_{ps,dry}$ = *dry mass of primary propulsion system*

M_p = *total propellant mass*

The results for a structural mass fraction of 0.10 are shown in Table 10 including the propellant and vehicle dry masses; GLOW (Ground Lift Off Weigh - the Mars Launch Mass); and ELM (Earth Launch Mass). Terrestrial propellants are those which are transported to Mars from Earth, and ISRU propellants are assumed to be manufactured by existing Martian infrastructure and hence are not reflected in the ELM. In this case it is assumed that the fully equipped hopper vehicles are also Martian infrastructure, and do not factor into the ELM calculation.

The GLOW increases exponentially with the distance traveled due to the need to accelerate larger amounts of propellants, including the propellant required for the powered landing. It also appears that the lower performing propellants, which are more easily derived from the Martian environment, are more attractive for the shorter trips. For example, the mass of SCO/LOX required for a 500 km trip is 2.3 times that of the LH_2/LOX system whereas this ratio jumps to 6.3 for the 2500-km trip. There is a similar affect on the mass of the propulsion system that is reflected in the dry mass listed in the table.

Table 10. Summary of Martian Hopper Analysis for a Structural Mass Fraction = 0.10

Propellant	PROPULSION System	Terrestrial Propellants	DISTANCE (km)	Propellant Mass (kg)	DRY Mass (kg)	GLOW (kg)	ELM (kg)
SCO/LOX	Hybrid	-	500	4040	2320	6360	0
SC/LOX	Hybrid	-	500	3090	2220	5310	0
LCH_4/LOX	Bi-Propellant	H_2	500	2160	2110	4270	129
LH_2/LOX	Bi-Propellant	H_2, O_2	500	1760	2070	3830	1760
SCO/LOX	Hybrid	-	1000	8250	2790	11,040	0
SC/LOX	Hybrid	-	1000	5770	2520	8290	0
LCH_4/LOX	Bi-Propellant	H_2	1000	3690	2280	5970	220
LH_2/LOX	Bi-Propellant	H_2, O_2	1000	2900	2190	5090	2900
SCO/LOX	Hybrid	-	2500	37,470	6040	43,510	0
SC/LOX	Hybrid	-	2500	17,170	3780	20,950	0
LCH_4/LOX	Bi-Propellant	H_2	2500	8320	2800	11,120	500
LH_2/LOX	Bi-Propellant	H_2, O_2	2500	5950	2530	8480	5950

The effects of varying the structural mass fraction were also explored. Figures 42 through 44 are a plot of the mass of propellant required vs. structural mass fraction for the three trip distances. Note that as the structural mass fraction increases, the lower performing SCO/LOX pays a higher penalty than the other two systems. Conversely, the SCO/LOX system will benefit much more by future materials and system technology that reduce the overall weight of the propulsion system. This is most clearly illustrated by Figure 44.

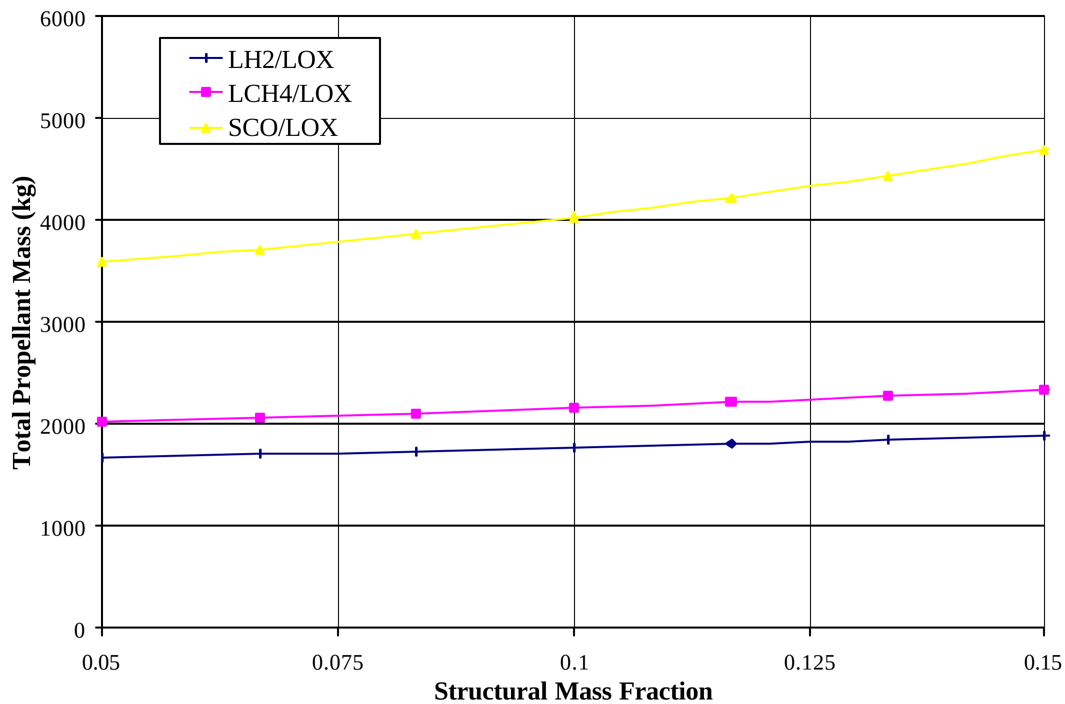

Figure 42. Total Propellant Mass Required for a 500 km Ballistic Hop vs. Structural Mass Fraction

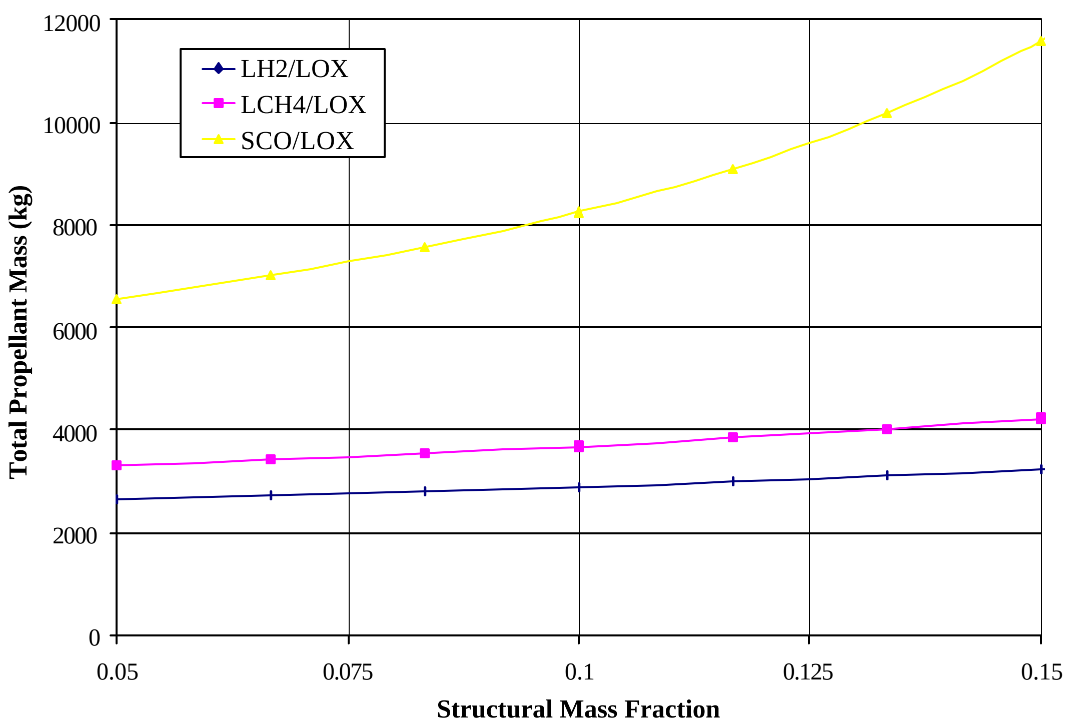

Figure 43. Total Propellant Mass Required for a 1000 km

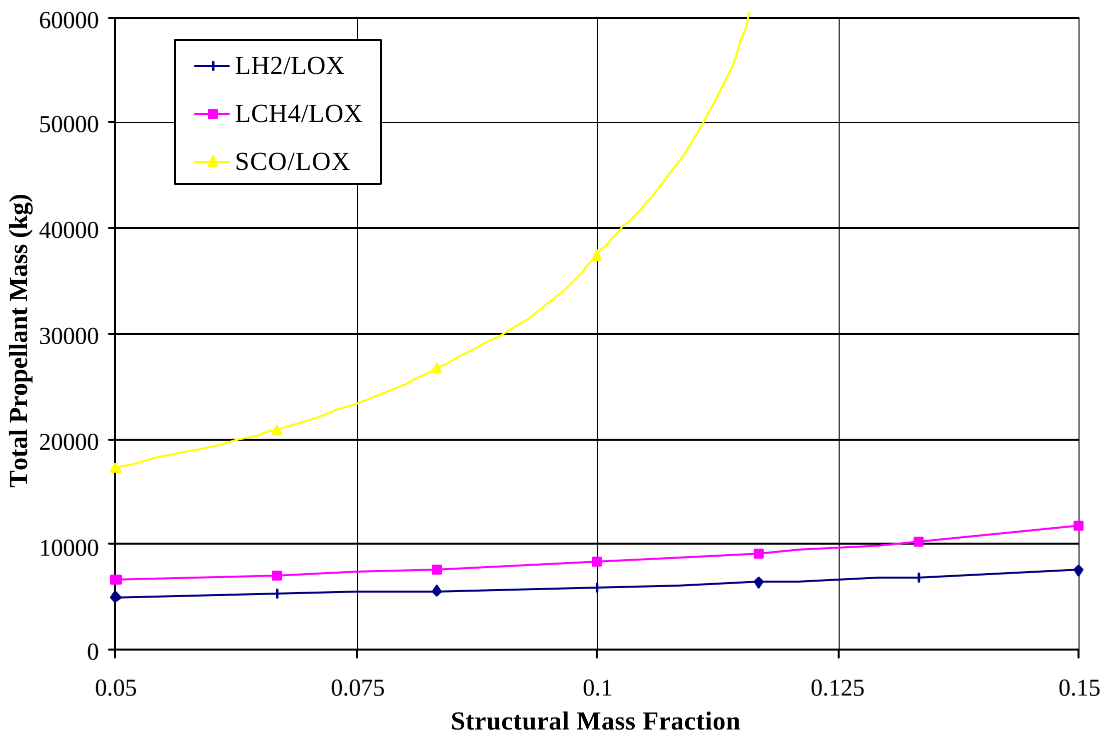

Figure 44. Total Propellant Mass Required for a 2500 km Ballistic Hop vs. Structural Mass Fraction

3.9.3 Rover

A rover was conceptualized and is shown is Figure 45.

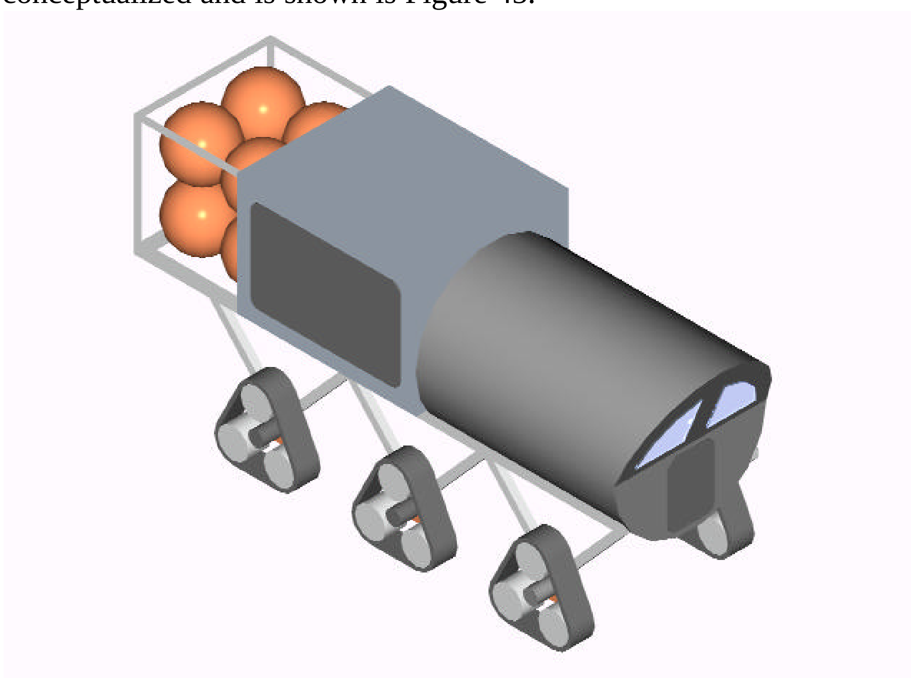

Figure 45. ISRU Powered Rover

The rover mission was defined as a 300 km one day trip. A system of gravel roads was assumed. Based on a trip distance of 300 km and a transit time of 10 hours, a velocity of 30 km/h was used. Turbine efficiency was assumed same for all fuels at 65% chemical potential to mechanical energy conversion efficiency.

For the initial iteration, the structural mass penalty for storing propellant was a 100 kg storage tank. The propellant mass penalty was ½ the mass of the propellant. For example, a 2000 kg rover that burns 200 kg of propellant and stores the byproducts has a mass of 2300 kg for the entire duration of the trip (with the tank mass penalty), while a 2000 kg venting rover has a mass of 2200 kg at the beginning of the trip and 2000 kg at the end. The mass used to calculate the propellant usage was therefore 2100 kg.

The rolling resistance per wheel was determined using the following equation:

$$R_c = \frac{\left(\frac{3Mg}{\sqrt{D}}\right)^{\frac{2n+2}{2n+1}}}{\left[3-n\right]^{\frac{2n+2}{2n+1}}\left[n+1\right]\left[K_c + bK_f\right]^{\frac{1}{2n+1}}}$$

The three constants in equation one, K_c, K_f, n were, until further data can be acquired, left at the values given in the source, 20855 Pa, 8100 Pa/m, and 1, respectivley.

The total resistance is $6R_c$, which gives the force needed to keep the vehicle at a constant velocity. Multiplying by that velocity gives the power needed to keep the vehicle at that velocity. Mutiplying by trip time in seconds gives the total energy needed throughout the course of the trip. Energy densities of the various propellants have already been calculated. This analysis assumes a 65% efficiency in the usage of that energy density. Taking the energy needed and dividing it by this scaled energy density give the amount of propellant needed to make the trip. The summary follows:

$$6R_c = F_{Total}$$
$$F_{Total} v = P$$
$$Pt = E$$
$$\frac{E}{ed} = m_{fuel}$$

where R_c is rolling resistance per wheel; F_{Total} is the force required to maintain a constant velocity; P is the power requirement to mainatin the velocity, v; t is the total vehicle powered run time; E is the total energy required; ed is the energy density of the fuel; m_{fuel} is the total mass of fuel needed to power the rover.

The mass estimates for the various rover components are listed in Table 11.

Table 11. Rover Mass Estimations

Rover Mass Summary			
	Light Duty	Mid Duty	Heavy Duty
Life Support Hardware	32	32	32
Pressurized Cabin	454	454	454
Waste Management	2.3	2.3	2.3
Medical Supplies	0.1	0.1	0.1
Windows	0.5	0.5	0.5
Crew Accommodations	11.5	11.5	11.5
Lighting	0.1	0.1	0.1
Radiation Shielding	18.1	18.1	18.1
Water, Food and Oxygen	4	4	4
External Lighting	0.5	0.5	0.5
Controls, Navigation, Avionics, Instruments, GPS	6	6	6
Wheels/Tires (Each)	5	7.5	10
Frame	150	600	1500
Mechanical Delivery System	50	100	200
Cargo Bay (un-pressurized)	80	160	500
Insulation	9.1	9.1	9.1
Electric Power Distribution	5	5	5
Turbine (kg/kJ)	1000	1500	2000
Vehicle Mass	1339	2422	4264
Payload	1000	4000	10000
Total Dry Mass	2339	6422	14264

Assessment of fuel needs for H_2/O_2, CH_4/O_2, and CO/O_2 was made and is shown in Table 12 below.

Table 12. Fuel Needs for a 300km, Ten-Hour Turbine-Powered, Rover Mission

Fuel Type	H_2/O_2	CH_4/O_2	CO/O_2
Fuel Use, Exhaust Recovered (kg)	113*	154 (13*)	249
Fuel Use, Exhaust Not Recovered (kg)	104*	142 (12*)	223

*Mass Supplied from Earth or Moon as Hydrogen

3.9.4 Auxiliary Power

Use of CO/O_2 in fuel cell energy production has already been proven feasible (AIAA Paper 98-0650, K. R. Sridhar), over the next century technology advances driven by attractiveness of this process on Mars should drive efficiencies to 60% and higher with combined cycle processes utilizing waste heat for base/rover heating. Molten Carbonate fuel cells already have efficiencies close to 85% for combined cycle hydrogen fuels (DOE), indicating that high efficiencies could be attained for an in situ CO/O_2 system. The molten carbonate fuel cell has to be run in an electrolyzer mode to remove chemisorbed CO from the catalyst, but this process is only necessary daily and occurs quickly. New catalysts can lessen or eliminate this problem as well.

Fuel cells have the capability of supplying power on demand without the rundown associated with low batteries and a modular design would provide for daily power fluctuations and for the increased loads of a growing outpost. An RTG would be used for startup heating of the fuel cell and for powering essential systems when the base is uninhabited; a battery or solar system would most likely be invoked on a rover.

Parametric equations have been developed for varying fuel cell efficiencies and fuels of H_2/O_2, CO/O_2, and CH_4/O_2. Calculations were preformed to determine the amounts of various fuels needed to power a Mars base at varying energy loads and production efficiencies. These calculations only consider the energy density of the fuels, the base's power load requirements, and the efficiency of the process in determining fuel requirements so the results can be applied to both a fuel cell and turbine energy system. Existing power loads for Mir and the International Space Station are given in Table 13 as a reference, and power requirements are given in kWh/day. Fuel energy densities are given in kWh/kg fuel. The fuel requirements are then calculated using the formula:

$$F = \frac{E}{r_e h}$$

where:

F = Fuel Mass Required per Day
E = Daily Base Energy Requirement
r_e = Fuel Energy Density
h = Efficiency

Table 13. Fuel Requirements of a proposed Mars Outpost

Current Space Requirements		
ISS:	105 kW	2520 kWh/day
Mir:	30 kW	720 kWh/day

Fuel Energy Densities			
H_2/O_2	13500 kJ/kg	=	3.75 kWh/kg
CO/O_2	6540 kJ/kg	=	1.82 kWh/kg
CH_4/O_2	10080 kJ/kg	=	2.80 kWh/kg

Turbine/Fuel Cell Combined Cycle Fuel Requirements

Facility	kg Fuel at 40% Efficiency			kg Fuel at 50% Efficiency			kg Fuel at 60% Efficiency			kg Fuel at 70% Efficiency			kg Fuel at 80% Efficiency			kg Fuel at 90% Efficiency		
kWh/day	CO/O_2	CH_4/O_2	H_2/O_2	CO/O_2	CH_4/O_2	H_2/O_2	CO/O_2	CH_4/O_2	H_2/O_2	CO/O_2	CH_4/O_2	H_2/O_2	CO/O_2	CH_4/O_2	H_2/O_2	CO/O_2	CH_4/O_2	H_2/O_2
1200	1651	1071	800	1321	857	640	1101	714	533	944	612	457	826	536	400	734	476	457
1440	1982	1286	960	1585	1029	768	1321	857	640	1132	735	549	991	643	480	881	571	549
1680	2312	1500	1120	1850	1200	896	1541	1000	747	1321	857	640	1156	750	560	1028	667	640
1920	2642	1714	1280	2114	1371	1024	1761	1143	853	1510	980	731	1321	857	640	1174	762	731
2160	2972	1929	1440	2378	1543	1152	1982	1286	960	1699	1102	823	1486	964	720	1321	857	823
2400	3303	2143	1600	2642	1714	1280	2202	1429	1067	1887	1224	914	1651	1071	800	1468	952	914
2640	3633	2357	1760	2906	1886	1408	2422	1571	1173	2076	1347	1006	1817	1179	880	1615	1048	1006
2880	3963	2571	1920	3171	2057	1536	2642	1714	1280	2265	1469	1097	1982	1286	960	1761	1143	1097
3120	4294	2786	2080	3435	2229	1664	2862	1857	1387	2453	1592	1189	2147	1393	1040	1908	1238	1189
3360	4624	3000	2240	3699	2400	1792	3083	2000	1493	2642	1714	1280	2312	1500	1120	2055	1333	1280
3600	4954	3214	2400	3963	2571	1920	3303	2143	1600	2831	1837	1371	2477	1607	1200	2202	1429	1371
3840	5284	3429	2560	4228	2743	2048	3523	2286	1707	3020	1959	1463	2642	1714	1280	2349	1524	1463
4080	5615	3643	2720	4492	2914	2176	3743	2429	1813	3208	2082	1554	2807	1821	1360	2495	1619	1554
4320	5945	3857	2880	4756	3086	2304	3963	2571	1920	3397	2204	1646	2972	1929	1440	2642	1714	1646
4560	6275	4071	3040	5020	3257	2432	4183	2714	2027	3586	2327	1737	3138	2036	1520	2789	1810	1737
4800	6606	4286	3200	5284	3429	2560	4404	2857	2133	3775	2449	1829	3303	2143	1600	2936	1905	1829

An analysis of fuel requirements for base energy loads at various efficiencies were calculated for each prospective fuel system, with the results for the CO/O_2 system given in Figure 46, CH_4/O_2 in Figure 47, and H_2/O_2 in Figure 48. For a 70% efficient process, the fuels' energy densities were contrasted in Figure 49.

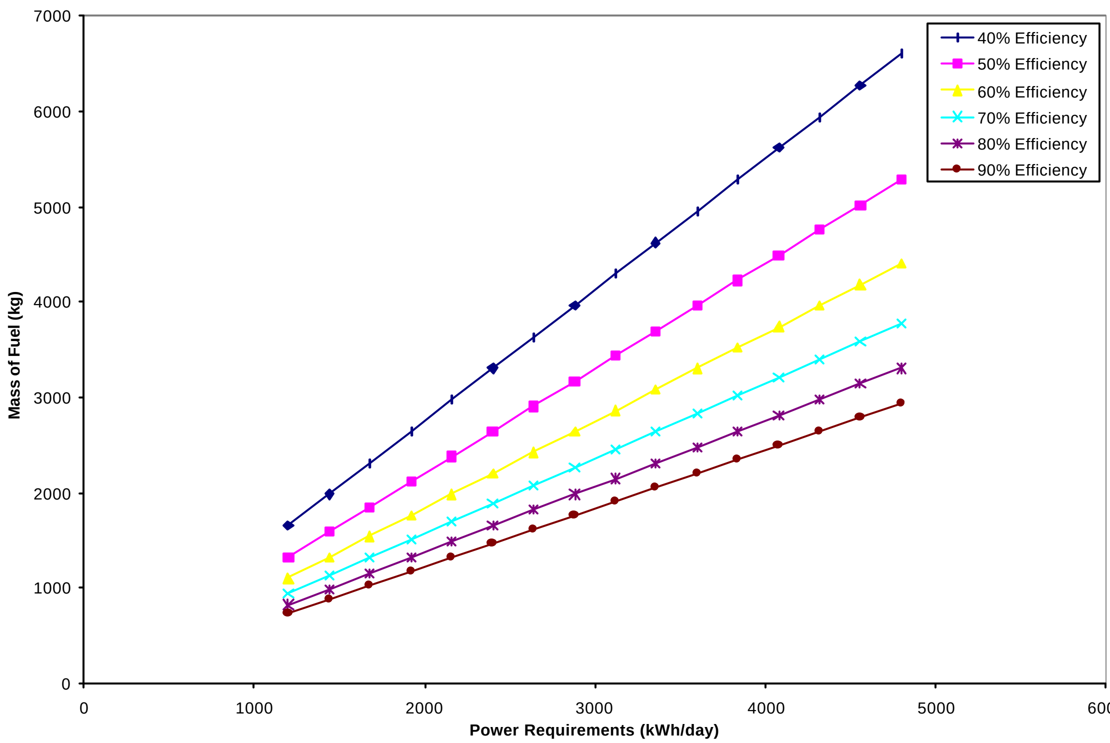

Figure 46. CO/O_2 System Fuel Requirements at Various Efficiencies

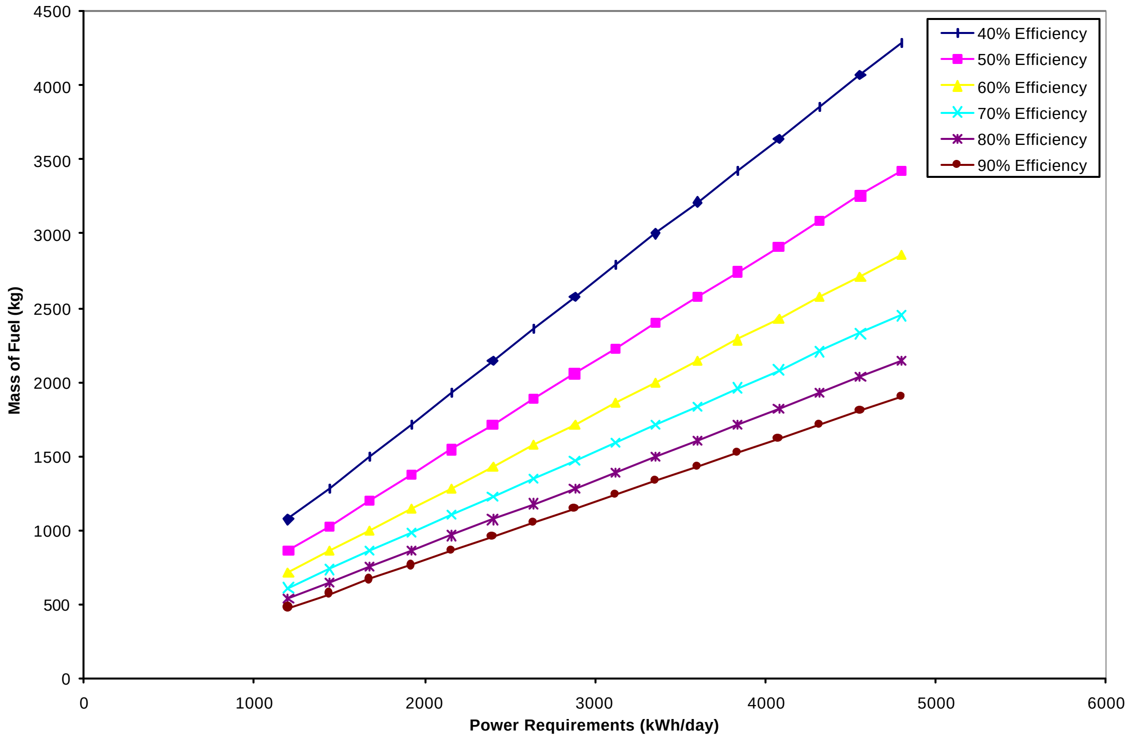

Figure 47. CH_4/O_2 System Fuel Requirements at Various Efficiencies

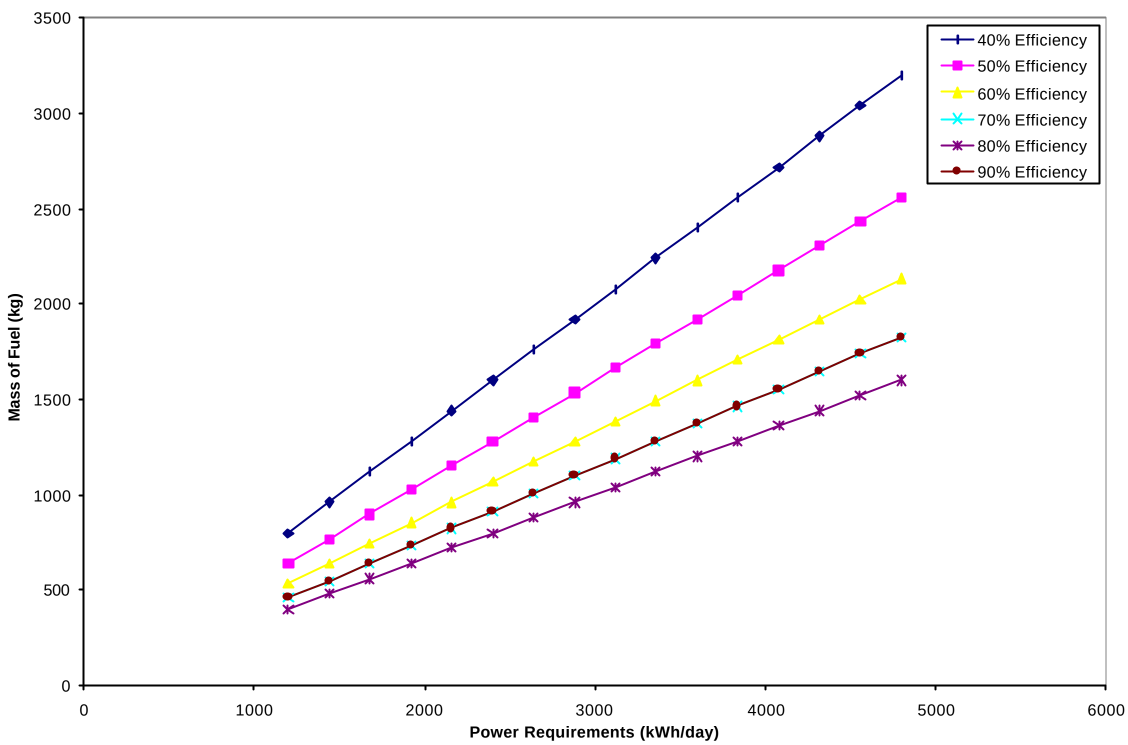

Figure 48. H_2/O_2 System Fuel Requirements at Various Efficiencies

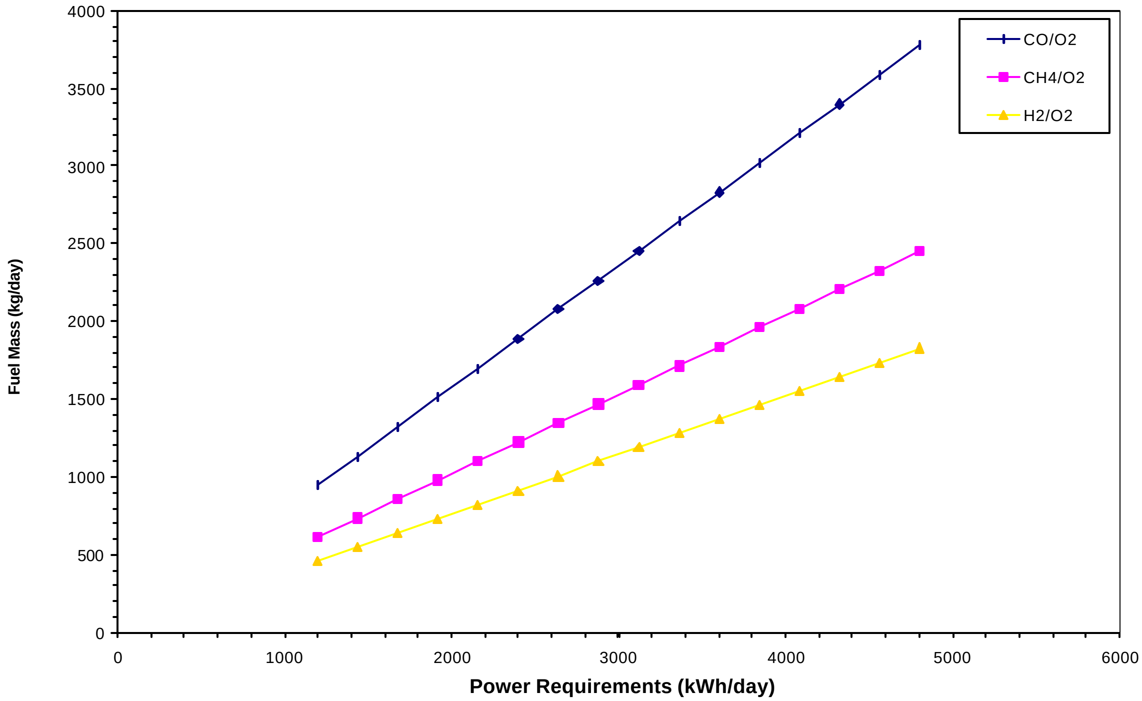

Figure 49. Fuel Requirements for System Power at 70% Overall Efficiency

3.10 Cost Models/Cost-Benefit Analysis

3.10.1 Cost Models

Several cost estimating models were investigated during the Phase I effort for use in Phase II. A brief description of some of these models is listed below.

Advanced Missions Cost Model

- This is a simple online advanced missions cost model (AMCM) that provides a useful method for quick turnaround, rough-order-of-magnitude estimating. The model can be used for estimating the development and production cost of spacecraft, space transportation systems, aircraft, missiles, ships, and land vehicles.

Cost Estimating Guidelines

- Online guide to cost estimating practices and principles

Cost Spreading Calculator

- This is a simple online cost spreading calculator that can be used to spread the estimated cost of a program up to 8 years. The calculator uses a beta curve to determine the amount of money to be spent in each year based on the fraction of the total time that has elapsed. The user enters the total cost to be spread, the beginning and ending years.

DSN Cost Estimating Cost Model

- This is a simple on-line model for estimating the cost required to do a cost estimate for Deep Space Network (DSN) projects that range from $0.1 to $100 million. The cost of the cost estimate in thousands of dollars, CE, is found to be approximately given by CE = K*CP^ 0.35 where CP is the cost of the project being estimated in millions of dollars and K is a constant depending on the accuracy of the estimate. For an order-of-magnitude estimate, K = 24; for a budget estimate, K = 60; and for a definitive estimate, K = 115. That is, for a specific project, the cost of doing a budget estimate is about 2.5 times as much as that for an order-of-magnitude estimate, and a definitive estimate costs about twice as much as a budget estimate. Use of this model should help provide the level of resources required for doing cost estimates and, as a result, provide insights towards more accurate estimates with less potential for cost overruns.

Expendable Launch Vehicles - International and US

- Online data
- Gives an estimate of launch cost vs. vehicle used, orbit achieved, and payload weight in FY94$

Inflation Collectors

- Wide variety of different inflation calculators

Learning Curve Calculator

- The calculator uses the learning curve to estimate the unit, average, and total effort required to produce a given number of units. Effort can be expressed in terms of cost, man-hours, or any other measure of effort. The calculator can be set to compute the Wright learning curve or the Crawford learning curve. The user is required to enter the effort (in terms of cost, man-hours, etc.) required to produce the first unit, the total number of units, and the learning percent.

Missions Cost Operations Cost Model

- This is a simple online mission operations cost model (MOCM) that provides a useful method for quick turnaround, rough-order-of-magnitude cost estimating. The model can be used for estimating the mission operations cost of manned, unmanned, and planetary spacecraft. The MOCM is based on NASA data for spacecraft flown between 1962 and 1990. The MOCM provides an estimate of the basic mission operations and data analysis (MODA) cost for a given spacecraft. MODA is defined as the cost of: maintaining and upgrading ground systems; mission control; tracking; telemetry; command functions; mission planning; data reduction and analysis; crew training and related activities. The MOCM does not include the cost of launch vehicles or launch services. The model estimates the average annual MODA based on the type of mission and the investment cost of the spacecraft. The investment cost is defined as the total development and production cost of the spacecraft, experiments and ground systems. Note that the investment cost does not include launch vehicle or service costs.

NASA/Air Force Cost Model

♦ The NAFCOM96 Cost Model is an innovative computer model for estimating aerospace program costs. NAFCOM96 is a user friendly estimating tool which operates in the Microsoft Windows environment. The model gives users flexibility in estimating by accommodating up to five systems and ten WBS levels, and by providing the user with the option of inputting hardware or integration cost or allowing the model to calculate the cost using NAFCOM96 estimating methodology or user defined equations.

Small Satellite Cost Model

♦ The Small Satellite Cost Model, SSCM, is a parametric cost model which runs on any Microsoft Excel-supported platform. The latest version, SSCM98, estimates the development and production costs of a small satellite bus for Earth-orbiting or near-planetary spacecraft.

Space Operations Cost Model

♦ NASA's Space Operations Cost Model (SOCM) study team is currently developing a suite of tools to estimate space mission operations costs for future NASA projects. The estimating methodology is based on a mix of parametric estimating relationships derived from collected data and constructive approaches capturing assessments of advanced technology impacts and reflecting experience from current mission planning teams. The study team includes cost, technical, and programmatic experts from each NASA Center.

Spacecraft/Vehicle Level Cost Model

♦ This is a simple on-line cost model that provides a useful method for quick turnaround, rough-order-of-magnitude cost estimating. The model can be used for estimating the development and production cost of spacecraft, launch vehicle stages, engines and scientific instruments. The SVLCM is a top-level model derived from the NASA/Air Force Cost Model (NAFCOM) database.

TRANSCOST Model

♦ This model is designed for the initial conceptual design phase of all propulsive space transportation system elements and engines. It is a "transparent model" with graphical display of the reference data that is based on a comprehensive 30-year database from US and European space vehicle and engine projects. TRANSCOST is a system-level model, based on the actual cost of completed projects with careful data evaluation for analytical processing and application of specific regression factors.

3.10.2 Cost-Benefit Analysis

Table 14 presents a cost-benefit comparison of propellants under consideration for use on Mars. An Earth launch mass (ELM) cost of $10,000/kg was assumed. The ISRU savings is the difference between the cost to launch a baseline (completely Earth-supplied) mission and the cost to launch the ISRU mission. Carbon and oxygen availability on Mars is assumed; hydrogen would be from Earth. Propellant/propulsion combinations, mass requirements and mission profiles were originally presented for the MAV (Table 6), hopper (Table 10), rover (Table 12) and outposts (Table13). The outpost is assumed occupied 50 days per year. The ISRU cost savings for the Mars hoppers and rovers are significant.

Table 14. Cost-Benefit Comparison of ISRU Propellants

Mission	ELM per Mission (kg)	ELM Cost ($M)	ISRU Savings per Mission ($M)	Missions per Year	ISRU Savings per Year ($M)
MAV Sample Return					
Baseline Solid	122	1.22	----	1	----
SC_2H_2/LOX Hybrid	32.1	0.32	0.90	1	0.90
LCH_4/LOX Bi-Prop	32.4	0.32	0.90	1	0.90
SC/LOX Hybrid	38.7	0.39	0.83	1	0.83
HTPB/LOX Hybrid	47.3	0.47	0.75	1	0.75
SCO/LOX Hybrid	50.9	0.51	0.71	1	0.71
SC-H_2/LOX Hybrid	53.3	0.53	0.69	1	0.69
One Way Hopper (1000 km)					
LH_2/LOX Bi-Prop Baseline	2.900	29.0	----	10	----
LCH_4/LOX Bi-Prop	220	2.20	26.8	10	268
SC/LOX Hybrid	0	0	29.0	10	290
SCO/LOX Hybrid	0	0	29.0	10	290
Turbine Powered Rover (300km)					
LH_2/LOX Turbine Baseline	113	1.13	----	100	----
LCH_4/LOX Turbine	7.7	0.08	1.05	100	105
SCO/LOX Turbine	0	0	1.13	100	113
Outpost Auxiliary Power					
LH_2/LOX Turbine Baseline	1400	14.00	----	50	----
LCH_4/LOX Turbine	100	1.00	13.00	50	650
LCO/LOX Turbine	0	0	14.00	50	700

Note: Processing equipment not amortized over ISRU derived propellant ELM

3.11 Recommendations for ISRU Propellant Technology Development

Significant technology development will be required before the Mars ISRU architecture can be put into place. ORBITEC is proposing to determine the feasibility of two of the promising propellant performers (SC/LOX and SC_2H_2/LOX) by conducting small-scale rocket engine test firings early in the Phase II so that they remain possible or are dropped from consideration. Propulsion technology for advanced cryogenic hybrids needs to be developed further than ORBITEC has taken it to date. The ORBITEC flight type LH_2/SOX engine tests should prove very useful here. More work is needed to develop efficient cryogenic coolers to support ISRU missions.

A significant amount of technology development in space-qualified ISRU processing systems will be necessary and should begin now. Very small-scale-systems for the Mars 2001 Lander are to be flown; however, larger systems need to be flight qualified.

As a result of the Phase II architecture study, we expect to identify and subsequently recommended technology development for critical times. The technology for processing the propellants from the atmosphere appears to be in hand for CO and O_2 production via the University of Arizona. K. R. Sridhar has provided us data that indicates that electrolytic processing scales favorably as it get larger (see Section 3.6.3).

4.0 CONCLUSIONS

The conclusions that have been made as a result of this Phase I NIAC study are as follows:

- ISRU will be a significant benefit to the Mars Exploration Program
- The SCO/LOX propellant system is likely good for short ballistic hops and wide use in ground systems; it will likely require staging or other propellant saving measures for large orbital operations
- Improving mass fraction helps lower-performance systems
- Cryogenic solid grains can be made and stored in Mars propellant facilities
- CH_4/LOX propellants are excellent for large orbital operations
- Carbon/LOX and acetylene hybrids also are excellent for more demanding missions and require further consideration
- H_2/O_2 systems would be best suited for high-performance missions, if Mars can supply it
- Large cargo transports are best accommodated by ground transport vehicles
- Ballistic rocket flight makes sense for high priority missions
- O/F choice can make a significant cost-benefit difference
- For ground-based systems, hydrogen in the exhaust can and should be recovered
- Consider savings attributed to wings, aeroshells, parachutes, etc
- Likely need nuclear power systems in many sizes
- The ISRU analysis approach we are pursuing is a complex problem
- Need to do a reasonable concept design on vehicles and process equipment to arrive at correct answer.

5.0 RECOMMENDATIONS

This section presents ORBITEC's study recommendations that have been based on the Phase I work. Our recommendations are:

1. NIAC fund ORBITEC's proposed Phase II effort
2. Gain participation of key NASA staff in the Phase II Workshop
3. Gain participation of Dr. Robert Cassanova, NIAC Director, in the development of the Mars propellant architecture study workshop
4. Conduct feasibility testing for SC/GOX and SC_2H_2/GOX in Phase II
5. Carry out the system architecture study to determine the best overall ISRU approach
6. Study, analyze, and develop the most promising ISRU processing techniques
7. NIAC develop contact with Dr. Gerald Sanders, NASA/JSC, because of his significant interest and responsibilities in NASA's ISRU program.

APPENDIX A

BIBLIOGRAPHY

AIAA, “Need for a NASA Indigenous Space Resource Utilization (ISRU) Program,” An AIAA Position Paper, Prepared by the AIAA Space Processing Technical Committee, January 6, 1997.

Ash, R., Dowler, W., and Varsi, G., “Feasibility of Rocket Propellant Production on Mars,” Acta Astronautica, Vol 5, pp. 705-724, 1978.

Carrick, P. G., “Specific Impulse Calculations of High Energy Density Solid Cryogenic Rocket Propellants 1: Atoms in Solid H_2,” Air Force Technical Report, PL-TR-93-3014, 1993. (a)

Clark, D.L., “In-Situ Propellant Production on Mars: A Sabatier/Electrolysis Demonstration Plant”, AIAA 97-2764, 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, July, 1997.

Crow, Steven C., “The MOXCE Project: New Cells for Producing Oxygen on Mars”, AIAA 97-2766, 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, July 1997.

Forch, B., and Merrow, C., “Photodissociation Processes in Carbon Monoxide at 193 nm,” *Journal of Chemistry Physics*, Vol. 95, 1991.

French, J., “The Impact of Martian Propellant Manufacturing on Early Manned Exploration,” AAS 84-178, Proceedings of the Second Case for Mars Conference, Boulder, CO 1984.

Frisbee, R., French, J., and Lawton, E., “A New Look at Oxygen Production on Mars - In-Situ Propellant Production (ISPP)” AIAA-87-0236, AIAA 25th Aerospace Sciences Meeting, January 1987.

Guernsey, C, Thunnissen, D., and Adler, M, “Evaluation of Some Candidate Propulsion Technologies for Mars Ascent”, AIAA 98-0651, 36th Aerospace Sciences Meeting & Exhibit, January 12-15, 1998, Reno NV.

Hepp, A., Landis, G., and Linne, D., “Material Processing with Hydrogen and Carbon Monoxide on Mars,” NASA TM 104405, May, 1991.

Jacobs, M., Deal, S., German, D., and Stancati, M., “An Assessment of In-Situ Propellant Production for Low Cost Mars Sample Return,” SAIC Final Report SAIC-95/1158 on Contract NASW-4491. Presented to the Solar System Exploration Division, NASA HQ, April, 1995.

Linne, D., “Experimental Evaluation of the Ignition Process of Carbon Monoxide and Oxygen in a Rocket Engine,” NASA TM 107267, AIAA-96-2943, 33rd Joint Propulsion Conference, June 1996.

Linne, D., “Performance and Heat Transfer Characteristics of a Carbon Monoxide/Oxygen Rocket Engine,” NASA TM 105897, 28th Joint Propulsion Conference (June 1992), February 1993.

Linne, D., “Carbon Monoxide and Oxygen Combustion Experiments: A Demonstration of Mars In-Situ Propellants,” NASA TM 104473, AIAA-91-2443, 27th Joint Propulsion Conference, June 1991.

Linne, D., Roncace, J., and Groth, M., "Mars InSitu Propellants: Carbon Monoxide and Oxygen Ignition Experiments," NASA TM 103202, AIAA-90-1894, 26th Joint Propulsion Conference, June 1990.

Meyer, T., and McKay, C., "The Atmosphere of Mars - Resources for the Exploration and Settlement of Mars," Proceedings of the Case for Mars Conference, Boulder, CO 1981.

Rapp, D., Karlmann, P.B., Clark, D.L., and Carr, C.M., "Adsorption Compressor for Acquisition and Compression of Atmospheric CO_2 on Mars," AIAA 97-2763, 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, July 1997.

Rice, E.E., Knuth, W.H., St. Clair, C.P, and Gramer, D.J., "Advanced Cryogenic Solid Hybrid Rocket Engine Developments for HEDM and NON-HEDM Demonstrations" 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Cleveland, OH, July 13-15, 1998, Orbital Technologies Corporation, Madison, WI, 1998.

Rice E.E., Gramer, D.J., Knuth W.H., St. Clair, C.P. "Experimental Investigation of a Metallized Cryogenic Hybrid Rocket Engine" 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Cleveland, OH, July 13-15, 1998, Orbital Technologies Corporation, Madison, WI, 1998.

Rice, E.E., "ORBITEC Advanced Cryogenic Solid Hybrid Rocket Engine and Propellant Developments" 50th OSU/AAE Anniversary, Orbital Technologies Corporation, Madison, WI, June 5, 1998.

Rice, E.E., St. Clair, C.P., Knuth, W.H., Gustafson, R.J., Gramer, D.J., and Chiaverini, M.J., "ORBITEC's First Test Firing Results for Solid CO/GOX", ORBITEC Technical Memorandum OTC-TM-R499-98-1, Orbital Technologies Corporation, Madison, WI, February 1998.

Rice, E.E., Knuth, W.H., and St.Clair, C. "Status Report of Cryogenic Solid Hybrid Rocket Engine Development for HEDM Demonstrations", Orbital Technologies Corporation, Madison, WI, paper presented at the 1997 HEDM Conference, VA, June 1997.

Rice, E.E., Knuth, W.H., "Status Report of Cryogenic Solid Hybrid Rocket Engine Development for HEDM Demonstrations", Orbital Technologies Corporation, Madison, WI, paper presented at the 1996 HEDM Conference, Boulder, CO, June 1996.

Rice, E.E., Knuth, W.H., St. Clair, C.P., Gramer, D.J., Rothbauer, W.J., and Heinen, G.W., "Storage and Delivery Device for Solid Oxygen Final Report", PL-TR-96-3022, Orbital Technologies Corporation, Madison, WI, October 1996.

Rice, E.E., Gramer, D.J., St. Clair, C.P., Knuth, W.H., Bemowski, P.A., and Hermes, P.A., "Metallized Cryogen for Advanced Hybrid Engines", OTC-G030-FR-96-1, Orbital Technologies Corporation, Madison, WI, December 1996.

Rice, E.E., Knuth, W.H., Kohles, D.R., Rothbauer, W.J., Heinen, G.W., St. Clair, C.P., Vermaak, A.G., "Cryogenic Hydrogen/Oxygen Solid Propellant Rocket Motor", OTC-G034-FR-95-1, NASA/LeRC Phase I SBIR Contract NAS3-27503, Orbital Technologies Corporation, Madison, WI, June 14, 1995.

Rice, E E., et al., "Storage and Delivery Device for Solid Hydrogen," NASA Contract F04611-93-C-0085, OTC-G019-FR-94-1, Orbital Technologies Corporation (ORBITEC), Madison, WI, April 1994.

Rice, E. E., Crabb, T. M., Hanley, G. M., and Teeter, R. R., "Aluminum/Oxygen Propellant Rocket Engines for Lunar Transport Applications", NASA/HQ Contract NASW-4476, OTC-FR-G007-90-1, Orbital Technologies Corporation (ORBITEC), Madison, WI, May 23, 1990.

Rice, E.E., T.M. Crabb, G.M. Hanley and R.R. Teeter, "Use of Tethered Platforms to Recover, Store and Utilize CO_2 from the Mars Atmosphere for On-Orbit Propellants," NASA/HQ Contract NASW-4478, OTC-FR-G008-90-1, Orbital Technologies Corporation (ORBITEC), Madison, WI, May 1990.

Sridhar, K.R.,and Foerstner, R., "Regenerative CO/O_2 Solid Oxide Fuel Cells for Mars Exploration", AIAA 98-0650, 36th Aerospace Sciences Meeting & Exhibit, January 12-15, 1998, Reno NV.

Sridhar, K. R., and Vaniman, B.T., "Oxygen Production on Mars Using Solid Oxide Electrolysis", Solid State Ionics, 93, pp.321-328, 1997.

Sullivan, T., Linne, D., Bryant, L., and Kennedy, K., "In-Situ-Produced Methane and Methane/Carbon Monoxide Mixtures for Return Propulsion From Mars," *Journal of Propulsion and Power*, Vol. 11, No. 5, pp. 1056-1062, September - October, 1995.

Zubrin, R.M., "Mars and Lunar Direct: Maximizing the Leverage of In-Situ Propellant Production to Develop a Coherent Architecture for the Space Exploration Initiative," AIAA Space Programs and Technologies Conf., Huntsville AL (Mar. 1992).

APPENDIX B

MARS INFORMATION DATA BASE

Abrego, Anita, "Project Hyreus Mars Sample Return Mission Utilizing In Situ Propellant Production: Final Report", NASA, 1994.

Acosta, Francisco Garcia, "Project Genesis Mars In Situ Technology Demonstrator Mission: Final Report", NASA, 1994.

Ad Astra, 1998, July/August.

"Ancient Life on Mars?", 1996, Summer.

Anderson, John E., Fester, Dale A., and Czysz, Paul M., "Evaluation of Long-Term Cryogenic Storage System Requirements", Martin Marietta Space Systems, 1990, pp. 1725-1731.

Arif, Hugh; Aydelott, John C.; Chato, David J., "Evaluation of Supercritical Cryogen Storage and Transfer Systems for Future NASA Missions", NASA, LRC, 28th Aerospace Sciences Meeting, 1990, January 8-11.

Armstrong, Elizabeth S., "Cooling of In-Situ Propellant Rocket Engines for Mars Mission", NASA, 1991.

Aviation Week & Space Technology, 1996, December 9.

Back, Dwight D.; Scaringe, Robert P.; Ramos, Charlie; Tapia, Per Eric, "Passive, Intermittent Cryocooler for Vibration-free and Load-Leveling Space Applications", Mainstream Engineering Corp., 31st Intersociety Energy Conversion Engineering Conference, Part 2, 1996, August 11-16, pp. 1349-1354.

Bailey, Stephen, IB and Weary, Dwayne, IB., "Artemis Boilerplate Lander - Flight Demonstration Project Proposal to the Office of Space Systems Development", NASA JSC, 1993, June 18.

Baillif, Faye F.; Bodepudi, Venu P.; Brewer, Robert A., "Cryogenic Insulation Selection and its Effects on Thermal Protection System Sizing for Reusable Launch Vehicles", Martin Marietta, 36th Structures, Structural Dynamics and Materials Conference, 1995, April 10-12.

Baird, R.S., Sridhar, K.R. and Gottmann, M., "2001 Mars In-Situ Oxygen Production Flight Demonstration", NASA JSC/University of Arizona, 35th Joint Propulsion Conference and Exhibit, 1999, June 20-24.

Bard, Steven and Jones, Jack A., "Regenerative Sorption Compressors for Cryogenic Refrigeration", JPL, 1990, pp.1357-1365.

Bard, Steven, "Development of a High-Performance Cryogenic Radiator with V-Groove Radiation Shields", JPL, 1987, May-June, pp. 193-197.

Benoit, A.; Caussigac, M.; Pujol, S., "New Types of Dilution Refrigerator and Space Applications", C.N.R.S., 20th International Conference on Low Temperature Physics, Part 3., 1993, August, pp. 48-53.

Berry, D.; Carrington, H.; Gully, W.J.; Kittel, P., "Two-stage Cryogenic Refrigerator for High Reliability Applications", Ball Aerospace Syst. Div., 1995 Cryogenic Materials Conference, 1995, July 17-21, pp. 1585-92.

"Beyond-the-Horizon Mars Technology Study", 1994, September 29.

Bhosri, Wisuwat; Cojanis, Philip; Gupta, Madhu; Khopkar, Manasi; Kiely, Aaron; Myers, Michael; Oxnevad, Knut; Sengupta, Anita; Sexton, Adam; Shaw, Don; Tellez, Joe; Tsuchiya, Takayuki and Wolford, Mark, "The Exploration of Mars: Crew Surface Activities", University of Southern California, 1999.

Blamont, Jacques, "Piggybacking to Mars", 48th International Astronautical Congress, 1997, October 6-10.

Blevins, Elana, "Experimental Strategies in the Development of a Polyisocyanurate Foam Insulation with HCFC 141b Blowing Agent", Lockheed Martin Manned Space Systems., 2nd Aerospace Environmental Technology Conference, 1997, August 6-8, pp. 85-92.

Blevins, Elana; Sharpe, Jon, "A Water Blown Urethane Insulation for Use in Cryogenic Environments", Martin Marietta Manned Space Systems, Aerospace Environmental Technology Conference, 1994, August 10-11, pp. 201-206.

"Blueprint for the 21st Century: The Moon/Mars Initiative", NASA, 1989, November 6.

Bozek, John M. and Cataldo, Robert L., "Power Generation Technology Options for a Mars Mission", NASA LRC, Symposia on Space Nuclear Power Systems, 1994, January 9-13.

"Bringing Mars Home: Opportunities and Challenges Presented by the Mars Sample Return Mission", 1998, Fall.

Brown, Norman, "Mars Transit Vehicle Thermal Protection System: Issues, Options, and Trades", pp. 745-754.

Bruckner, A.P.; Nill, L.; Schubert, H.; Thill, B.; Warwick, R., "Mars Rover Sample Return Mission Utilizing In Situ Production fo the Return Propellants", 1993, June.

Budden, Nancy Ann and Duke, Michael B., ed., "HEDS-UP Mars Exploration Forum", Lunar and Planetary Institute, 1998, May 4-5.

"Building Solar Cells for Mars", *SpaceDaily*, 1999, May 19.

Caplan, David E., "Mars Sample Return Using In-Situ Propellant Production", NASA, JSC, 5th International Conference on Space '96, 1996, June 1-6, pp. 717-723.

Cerimele, M. and Connolly, J., "The Mars ISRU Sample Return Mission Study Report", 1994, October 1.

"Cold Facts", Newsletter of Cryogenic Society of America, Inc., 1998, Spring.

Collaudin, B., "Use of Mechanical Coolers for Space Application", Aerospatiale Cannes, 24th International Conference on Environmental Systems and 5th European Symposium on Space Environmental Control Systems, 1994, June 20-23.

Connolly, John, "MISR - Sampling Mars on a Budget", NASA JSC.

Cooke, Douglas R., Weary, Dwayne P. and Kaplan, David I., "Analysis of the Synthesis Group's Mars Exploration Architecture (Half of document)", Lunar and Mars Exploration Program Office, 1991, October 25.

Coons, S.C.; Curtis, R.M; McLain, C.; Williams, J.D.; Warwick, R.; Bruckner, A.P., "In Situ Propellant Production Strategies and Applications for a Low-Cost Mars Sample Return Mission", Univ. of Washington, 31st Joint Propulsion Conference, 1995, July 10-12.

Deveny, M., "Mirror Fusion Propulsion System- A Performance Comparison with Alternate Propulsion Systems for the Manned Mars Mission", McDonnell Douglas Aerospace Space Systems, 1993.

Dexter, Carol E.; Kos, Larry, "An Overview of Mars Vicinity Transportation Concepts for a Human Mars Mission", NASA, Marshall, 1998, October 26.

Drake, Bret G., ed., "Reference Mission Version 3.0 - Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team", 1998, June.

Duke, M.B., "Mars Surface Mission Workshop", Lunar and Planetary Institute, 1997, October 4-5.

Duke, Michael B. and Budden, Nancy Ann., "Mars Exploration Study Workshop II: report of a workshop sponsored by NASA Lyndon B. Johnson Space Center and held at the Ames Research Center", NASA, JSC, Mars Exploration Study Workshop, 1993, May 24-25.

Duke, Michael B., "Opportunities for ISRU Applications in the Mars Reference Mission", Lunar and Planetary Institute, Workshop on Using In Situ resources for Construction of Planetary Outposts, 1998, January 1, pp. 34-35.

Dyer, K., Acosta, J. and Yankura, George, "Low Mass Components for Mars Ascent Propulsion", VACCO Industries/JPL, 35th Joint Propulsion Conference and Exhibit, 1999, June 20-24.

Ehrlich, C.F., Jr., “TPS/Insulation System Concepts for Vehicles with Reusable Cryotanks”, Rockwell International, AIAA Space Programs and Technologies Conference, 1994, September 27-29.

“Experiment Lays Groundwork for 'Living Off the Land' On Mars”, JPL, 1999, March 25.

“Extraction of Atmospheric Water: Mars In Situ Resource Utilization (Abstract only)”, University of Washington, 1998, April 30.

Finn, John E., Kliss, Mark H. and Sridhar, K.R., “ISRU Technologies for Mars Life Support”, NASA Ames/University of Arizona, 1999.

Flynn, Thomas M., “Method for Making and Storing Cryogenic Monopropellant Containing Methane and Oxygen.”, 1998, September 8.

Frahmand, T.; Razani, A., “Thermal Analysis of a Multi-Tank Cryogenic Storage System for Space Applications”, Univ. of New Mexico, 29th Intersociety Energy Conversion Engineering Conference, Part 2, 1994, August 7-11, pp. 700-705.

French, J.R., “Concepts for In-Situ Resource Utilization on Mars - A Personal Perspective”, JRF Engineering Services, 1995, July, pp. 311-313.

Frisbee, R., “Mass and Power Estimates for Martian In-Situ Propellant Production Systems”, 1986.

Gaseor, Thomas, “Advanced Refrigerator/Freezer Technology Development Technology Assessment”, NASA, 1996.

Gefert, L., et al., “Options for the Human Exploration of Mars Using”, 1999.

Gibson, Jr., Everett K., McKay, David S., Thomas-Kerpta, Kathie and Romanek, Christopher S., “The Case for Relic Life on Mars,” *Scientific American*, Vol. 277, 1997, December, pp. 58-65.

Goroshin, S., Higgins, A.J. and Lee, J.H.S., “Powdered Magnesium-Carbon Dioxide Propulsion Concepts for Mars Missions”, Combustion Technologies International/McGill University, 35th Joint Propulsion Conference and Exhibit, 1999, June 20-24.

Green, S.T., Deffenbaugh, D.M. and Miller, M.A., “A Comparison of Five ISPP Systems for a Mars Sample Return Mission”, Southwest Research Institute, 35th Joint Propulsion Conference and Exhibit, 1999, June 20-24.

Grover, M.R, et al., “Ares Explore: A Study of Human Mars Exploration Alternatives Using In Situ Propellant Production and Current Technology”, University of Washington, The Case for Mars VI Conference, University of Colorado at Boulder, 1996, July 17.

Guernsey, C., Thunnissen, D. and Adler, M./French, J., "Evaluation of Some Candidate Propulsion Technologies for Mars Ascent", California Institute of Technology/JRF Engineering Services, 36th Aerospace Sciences Meeting and Exhibit, 1998, January 12-15.

Gulick, V.C., ed., "Mars 2005 Sample Return Workshop", NASA Ames Research Center, 1996, March 25-27.

Hickey, G.S.; Braun, D.; Wen, L. and Eisen, H., "Integrated Lightweight Structure and Thermal Insulation for Mars Rover", JPL Division 35, 1995.

Hickman, J.M., "Solar Electric Propulsion for Mars Transport Vehicles", NASA, 1991.

Hirata, Christopher; Greenham, Jane; Brown, Nathan; Shannon, Derek and Burke, James D., "A New Plan for Sending Humans to Mars: The Mars Society Mission", California Institute of Technology/JPL.

Hoffman, Stephen J. and Kaplan, David L., ed., "Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team", NASA JSC, 1997, July.

Hoffman, Stephen J. and Kaplan, David L., ed., "Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team (Draft)", NASA JSC, 1997, March.

Imbert, M.D.; Faure, A.; Ravex, A.; Seyfert, M.P.; Duwelz, A., "Evaluation Study of Mechanical Refrigeration Systems for Application to Space Missions", Air Liquide/Societe Europeene de Propulsion.

"In-Situ Materials Utilization".

"The International Exploration of Mars", The Case for Mars VI Conference, University of Colorado at Boulder, 1996, July 17.

Jacobs, Mark K.; Deal, Steven V.; German, Darla J.; Stancati, Michael L., "An Assessment of In Situ Propellant Production for Low Cost Mars Sample Return", Science Applications International Corp., 25th International Conference on Environmental Systems, 1995, July 10-13.

Jones, J.A.; Britcliffe, M., "Sorption Compressor/Mechanical Expander Hybrid Refrigeration", pp. 21-26.

Joosten, Kent; George, Jeff; Condon, Gerald; Hoffman, Stephen J., "Continuing Development of the NASA Human Mars Mission Design", NASA, JSC, 6th International Conference and Exposition on Engineering Construction and Operations in Space, 1998, April 26-30, pp. 563-570.

Kaloupis, P.; Nolan, P. and Cutler, A., "Martian Resource Utilization I. Plant Design and Transportation Selection Criteria", 1992, pp. 343-375.

Kaplan, D.I., “Mars ISRU Sample Return - A Mission Description”, NASA, JSC, 27th Lunar and Planetary Science Conference, 1996, March 18-22.

Kaplan, D.I.; Ratliff, J.E.; Baird, R.S.; Sanders, G.B.; Johnson, K.R.; Karlmann, P.B.; Juanero, K.J.; Baraona, C.R.; Landis, G.A.; Jenkins, P.P. and Scheiman, D.A., “In-Situ Propellant Production on Mars: The First Flight Demonstration”, NASA, JSC/JPL/NASA, Glenn/Ohio Aerospace Institute, 30th Lunar and Planetary Science Conference, 1999, March 15-19.

Kaplan, Dave, “Construction and Operation of a Mars In Situ Propellant Production Unit”, NASA JSC.
Kaplan, David, “HEDS Strategy for Robotic Mars Missions”, NASA JSC.

Knoll, Richard H.; Stochl, Robert J.; Sanabria, Rafael, “A Review of Candidate Multilayer Insulation Systems for Potential Use on Wet-Launched LH2 Tankage for the Space Exploration Initiative Lunar Missions”, 1991.

Kuznetsov, G.V.; Rudzinskaya, N.V., “Numerical Estimation of the Efficiency of Thermal Insulation for the Main Elements of an Aerospace Vehicle”, Res. Institute of Appl. Math & Mech., 1996, March-April, pp. 263-8.

Landis, G., “Footsteps to Mars: An Incremental Approach to Mars Exploration”, 1995.

Landis, Geoffrey A. and Jenkins, Phillip P., “Dust Accumulation and Removal Technology (DART) Experiment on the Mars 2001 Surveyor Lander”, NASA Glenn Research Center, 1998.

Landis, Geoffrey A., “Solar Cell Selection for Mars”, NASA, LRC - Ohio Aerospace Institute, 2nd World Conference on Photovoltaic Solar Energy Conversion, 1998, July 6-10.

Ledord, Otto C., Jr., “An Overview of Spacecraft Cryocooler Development in the USA”, Consultant, 24th International Conference on Environmental Systems and 5th European Symposium on Space Environmental Control Systems, 1994, June 20-23.

Levenduski, R. and Scarlotti, R., “Joule-Thomson Cryocooler for Space Applications”, Ball Aerospace Systems Division, 1996, pp. 859-866.

Linne, Diane L., “A Rocket Engine for Mars Sample Return Using In Situ Propellants”, NASA, 1997.

Linne, Diane L., “Carbon Monoxide and Oxygen Combustion Experiments- A Demonstration of Mars In Situ Propellants”, NASA, 1991.

Linne, Diane L., “Experimental Evaluation of the Ignition Process of Carbon Monoxide and Oxygen in a Rocket Engine”, NASA Lewis Research Center, 32nd Joint Propulsion Conference, 1996, July 1-3.

Linne, Diane L.; Roncace, James and Groth, Mary F., “Mars In Situ Propellants Carbon Monoxide and Oxygen Ignition Experiments”, NASA, 1990.

Littman, Franklin D., “Mars Power System Concept Definition Study”, NASA, 1994.

Mankins, J., “A Fresh Look at Space Solar Power: New Architectures, Concepts and Technologies”, 48th International Astronautical Congress, 1997, October 6-10.

Mankins, J., “Technology and Mars Exploration”, AIAA Space Programs and Technologies Conference, 1990, September 25-28.

Mankins, John C., “Technologies for Future Exploration”, NASA, 43rd Annual AAS National Conference, 1996, December 9-11.

“The Many Faces of Mars: A Portfolio of Global Surveyor Images”, 1998, Spring.

“Mars Atmospheric Resources”.

“Mars Exploration: A Tour of Past, Present, and Future Exploration”, NSSDC.

“Mars Sample Return”, Lockheed-Martin Corp.

“Mars will Require a Hardier Breed of Pioneer”, 1996, March.

Martin, James J.; Smith, James W., “Cryogenic Testing of a Foam-Mulitlayer Insulation Concept in a Simulated Orbit Hold Environment”, NASA, Marshall, 1993.

“MISR Goals, Objectives Capabilities, Constraints, and Top-Level Requirements (Final Draft)”, 1995, January 6.

Mendell, Wendell W., “A Mission Design for International Manned Mars Mission: (From the 1991 International Space University (ISU) Design Project)”, NASA JSC.

“A Messenger from Mars?”, *Final Frontier*, 1996, December, p. 24.

Meyer, Thomas R. and McKay, Christopher P., “Using the Resources of Mars for Human Settlement”, Boulder Center for Science and Policy/ NASA, Ames, 1996, pp. 393-442.

Morgenthaler, George W., “Some Realistic Barriers to International Mars Human Exploration Now: Work to be Done - New Approaches”, University of Colorado at Boulder, 1996 AIAA Space Programs and Technologies Conference, 1996, September 24-26.

“More Findings About Life on the Red Planet”, *Science News*, Vol. 157”, 1997, February 8, p. 87.

Mueller, P.J. and Batty, J.C./Zubrin, R.M., "Hydrogen Storage System for a Mars Sample Return Mission: Analysis of a Non-Venting Approach", Utah State University/Pioneer Astronautics, 32nd Joint Propulsion Conference, 1996, July 1-3.

Nakanishi, Shigeo, "Space Environmental Considerations for a Long-Term Cryogenic Storage Vessel", Cryogenic Fluid Management Technology Workshop, pp. 175-191.

"NASA's Mars Design Reference Mission", 1993.

Niebur, C.S.; Arvidson, R.E.; Bowman, J., "A Mission Model for the 2001 Mars Rover/Athena Payload", Washington Univ., Saint Louis, 29th Lunar and Planetary Science Conference, 1998, March 16-20.

Noever, David A.; Smith, David D.; Sibille, Laurent; Brown, Scott C.; Cronise, Raymond J. and Lehoczky, Sandor L., "High Performance Materials Applications to Moon/Mars Missions and Bases", NASA Marshall/Universities Space Research Association/Southern Research Institute.

Pelaccio, Dennis, et al., "Engine System Assessment Study Using Martian Propellants, Final Report", Science Applications International Corp., 1990.

Preiss, Bruce Kenneth, "A Quantitative Figure-of-Merit Approach for Optimization of an Unmanned Mars Sample Return Mission", Univ. of Arizona, 1991, pp. 334.

Preiss, Bruce; Pan, Thomas; Ramohalli, Kumar, "Further Applications of a Figure-of-Merit in Space Missions", NASA, Space Engineering Research Center, 27th Joint Propulsion Conference, 1991, June 24-26.

Preiss, Bruce; Pan, Tom; Ramohalli, Kumar, "Oxygen Production on Mars: A Figure-of-Merit Approach", p. 33.

"Project Ares Acquire: Mars Sample Return Mission Utilizing In Situ Propellant Production", University of Washington.

"Pumping Up the Gas", *SpaceDaily*, 1999, February 22.

Radebaugh, R.; Haruyama, T.; Mitsui, T.; Yamafuji, K., "Advances in Cryocoolers", Nat. Institute of Stand. & Technol., 16th International Cryogenic Engineering Conference/International Cryogenic Materials Conference, 1997, pp. 33-44.

Ramohalli, Kumar and Lewis, John, "A Survey of Technology Advances in In-Situ Resource Utilization for Economical Space Missions", Univ. of Arizona, 46th International Astronautical Congress, 1995, October 2-6.

Ramohalli, Kumar; Brod, Stephen; Colvin, James; Crockett, Richard; Seymour, Nathaneal, "Recent Advances in Mars ISRU Technologies", Univ. of Arizona, 31st Joint Propulsion Conference, 1995, July 10-12.

Ramohalli, Kumar; Marcozzi, Massimilano, “A Quantitative Methodology for Mission Architecture and Figure-of-Merit with ISRU and Comparisons with a Baseline (Abstract only)”, Univ. of Arizona, 1997, January 1.

Ramohalli, Kumar; Marcozzi, Massimiliano, “Cost Reductions through ISRU - Recent Developments”, Univ. of Arizona, 48th International Astronautical Congress, 1997, October 6-10.

Ramohalli, Kumar; Preiss, Bruce, “Quantitative Figure-of-Merit for Space Missions: Importance of the Life-Cycle Costs”, Univ. of Arizona, 1992, pp. 29-41.

Reichert, M., “Cost Benefits for Future Space Programs by Using Lunar and Martian Propellants”, Deutsche Forschungsanstalt fuer Luft- und Raumfahrt (DLR), 1996.

“Reusable Cryogenic Foam Insulation Has Space and Earth-Bound Uses”, NASA, Langley/Lockheed California/National Bureau of Standards, 1990, September 10, p. 4.

Roberts, Barney B. and Theillier, Francis, “Strategic Approach for Surface Systems and Operations for an International Mars Mission”, 1993, October.

Romero, M.; Bourrieau, J.; Mandeville, J.C.; Fernandez-Maillard, A., “Main Environmental Hazards for Mars Missions”, 1993, September.

Roncace, Elizabeth A., “In-Situ Propellant Rocket Engines for Mars Mission Ascent Vehicle”, NASA, 1991.

Sanders, J.B., “Integrated Propulsion and ISRU Propellant Production Systems for Mars Sample Return”, NASA, JSC, 31st Joint Propulsion Conference, 1995, July 10-12.

Schlif, L., “The MBB-Vacuum Super Insulation”, Oak Ridge National Lab, 1993 Non-fluorocarbon Insulation, Refrigeration and Air-conditioning Technology Workshop, 1993, September 27-29.

Seybold, Calina C.; Fowler, Wallace T., “Human Mars Exploration - Student Designed Missions and the Current NASA Reference Mission”, Univ. of Texas at Austin, AAS/AIAA Space Flight Mechanics Meeting, 1998, February 9-11, pp. 537-550.

“Several Drawings and Photographs”.

Shafirovich, E. Ya. And Goldshleger, U.I., “Comparison of Potential Fuels for CO2 Using Martian Rockets”, Institute of Structural Macrokinetics, 32nd Joint Propulsion Conference, 1996, July 1-3.

“Smaller, Cheaper, Faster Missions to the Moon and Mars”, NASA Space Engineering Research Center, University of Arizona, 1993, February 18-19.

Smith Joseph L., Jr., “Cryogenic Refrigeration for Space Exploration, A Challenge for the Future”, MIT, 18th International Congress of Refrigeration, 1991.
"The Space Solar Power Option," *Aerospace America*, 1997, May, p. 30.

Sridhar, K.R. and Foerstner, R., “Regenerative CO/O2 Solid Oxide Fuel Cells for Mars Exploration”, University of Arizona, 36th Aerospace Sciences Meeting and Exhibit, 1998, January 12-15.

Sridhar, K.R., “Mars Sample Return Mission with In-Situ Resource Utilization”, University of Arizona, 1995, November-December.

Stancati, Michael L.; Jacobs, Mark K.; Niehoff, John C., “Programmatic Evaluation of Technologies for Mars Propellant Manufacture”, Science Applications International Corp., 34th Joint Propulsion Conference, 1998, July 13-15.

Stoker, C.R.; McKay, C.P.; Haberle, R.M. and Andersen, D.T., “Science Strategy for Human Exploration of Mars”, 1990.

Sullivan, Thomas A. and McKay, David S., “Using Space Resources”, Mission Science and Technology Office Solar System Exploration Division, JSC, 1991.

Sullivan, Thomas A., “ISRU Approaches to Mars Sample Return Missions”, 1993, September.

Sullivan, Thomas A.; Linne, Diane; Bryant, Lee; Kennedy, Kriss, “ISRU-Produced Methane and Methane/Carbon Monoxide Mixtures for Return Propulsion from Mars”, NASA, JSC/NASA, LRC, 30th Joint Propulsion Conference, 1994, June 27-29.

“Surveyor: Final Engineering Report”, Volume 1, Summary, Hughes Aircraft Company, 1968, June.

Tillotson, B., “Mars Transfer Vehicle Using Regolith as Propellant”, 1994.

Tupper, M.L; Pulley, A.L.; Johnson, E.F.; Munshi, N.A.; Fabian, P.E.; O'Leary, J., “A Spray-On Cryogenic Insulating Material for NASA Which Prevents the Formation of Liquid Oxygen on Liquid Hydrogen Vent Lines”, Composite Technology Development, NASA Stennis, Advances in Cryogenic Engineering Engineering Materials, 1993, July 12-16, pp. 1177-1182.

“An Unmanned Mission to Mars with Sample Return and In-Situ Resource Utilization PDR-2 Report: Project MARVIN”, Exploratory Technologies, 1994.

Wadel, Mary A., “Benefits of In Situ Propellant Utilization for a Mars Sample Return Mission”, 1993, June.

Wadel, Mary F., “Propulsion Systems Using In Situ Propellants for a Mars Ascent Vehicle”, NASA, 199.

Walberg, G., "How Shall We Go to Mars? A Review of Mission Scenarios", 1993.

Wallace, R.A., "Low Cost Mars Sample Return Study", JPL, 1995, February 13.

Ward, Mark, "Hop, Skip and Jump", 1997, December.

Weaver, David B. and Duke, Michael B., "Mars Exploration Strategies: A Reference Program and Comparison of Alternative Architectures", NASA JSC, 1993.

Wickman, J.H., "In-Situ Mars Rocket and Jet Engines Burning Carbon Dioxide", Wickman Spacecraft & Propulsion Co., 35th Joint Propulsion Conference and Exhibit, 1999, June 20-24.

Wittman, David A., "Cryogenic Insulation and TPS System Demonstration for RLV Reusable Composite Hydrogen Tank", Rockwell Space Systems, 37th Annual Structures, Structural Dynamics and Materials Conference, 1996, April 15-17, pp. 925-934.

"Workshop on Research for Space Exploration: Physical Sciences and Process Technology", NASA Lewis Research Center, 1998, May.

Zubrin, R. and Price, S., "Mars Sample Return Mission Utilizing In-Situ Propellant Production", Lockheed-Martin Corp., 1995, March 31.

Zubrin, R., "A Comparison of Methods for the Mars Sample Return Mission", 32nd Joint Propulsion Conference, 1996, July 1-3.

Zubrin, R., "Missions to Mars and the Moons of Jupiter and Saturn Utilizing Nuclear Thermal Rockets with Indigenous Propellants", 28th Aerospace Sciences Meeting, 1990, January 8-11.

Zubrin, Robert and Baker, David, "Humans to Mars in 1999!", Martin Marietta Astronautics.

Zubrin, Robert M.; Baker, David A. and Gwynne, Owen, "Mars Direct: A Simple, Robust, and Cost Effective Architecture for the Space Exploration Initiative", Martin Marietta Astronautics/NASA Ames, 1991.

Zubrin, Robert, "Athena: A Possible First Step in a Program of Human Mars Exploration", Pioneer Astronautics, 32nd Joint Propulsion Conference, 1996, July 1-3.

Zubrin, Robert, "Methods for Achieving Long Range Mobility on Mars", Martin Marietta Astronautics, 1992.

Zubrin, Robert, "The Economic Viability of Mars Colonization", Lockheed Martin Astronautics.

Zubrin, Robert; Price, Steve; Mason, Larry and Clark, Larry, "Report on the Construction and Operation of a Mars In-Situ Propellant Production Plant", Martin Marietta Astronautics, 30th Joint Propulsion Conference, 1994,

APPENDIX C

MISSION MODEL WORKSHEETS

Mission Category: Scientific Exploration and Research

Mission/Submission Scope?		# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Past/Current Life on Mars – search for evidence of past life, geology of the planet, ice at poles or permafrost (tools, sample boxes, life support, rover, sample rocks/dust, measure seismic activity)		2/Robotic 2/Robotic 2/Robotic	1-5 days 1 day 3-7 days	4000 km 500 km 10,000 km	Minutes Hours Minutes	300 300 300	Ballistic Flight Ground Ballistic Flight
		Robotic	Infinite	Infinite	N/A	50	Ground
Meteorology – study/characterize atmosphere, dust storms, other weather phenomena (temperate, pressure, wind velocity, solar radiation, humidity)	Delivery Vehicle Recovery Vehicle Sounding Rocket	Robotic Robotic Robotic	1 day 1 day < day	10,000 km 10,000 km ? altitude	Minutes Minutes Minutes	10 10 2	Ballistic Flight Ballistic Flight Ballistic Flight
Astronomy – any orbiting systems supplied from Earth - any ground-based systems located at base, so no requirement for transport							
Solar Monitoring – located at base, so no need for transport							
Other Science – study meteorites, characterize poles		2/Robotic 2/Robotic 2/Robotic	1-5 days 1 day 3-7 days	4000 km 500 km 10,000 km	Minutes Hours Minutes	200 50 200	Ballistic Flight Ground Ballistic Flight
Mars Moon Exploration (landing equipment, tools similar to the search for life/geology mission)		3/Robotic	1 week	Moon Orbits	Hours	100	Flight vehicle
Mission to Asteroid Belt		3/Robotic	Months	Asteroid Belt	Hours	100	Flight vehicle

Mission Category: Commercial Exploration

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Identify Resources – similar to past/current life and geology missions (look for water, mineral and metal deposits, fuels and other valuable resources)	**2/Robotic 2/Robotic 2/Robotic**	**1-5 days 1 day 3-7 days**	**4000 km 500 km 10,000 km**	**Minutes Hours Minutes**	**300 300 300**	**Ballistic flight Ground Ballistic**
	Robotic	**Infinite**	**Infinite**	**N/A**		**Ground**

Mission Category: Terraforming

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
Terraform Planet **Small Scale**	**Robotic 2/Robotic Robotic**	**Infinite 3-7 days < day**	**Infinite 10,000 km ? altitude**	**NA Minutes Minutes**	**50 100 20**	**Ground Ballistic Sounding**
Large Scale	**Robotic Robotic**	**Infinite Weeks**	**Infinite 10,000+ km**	**NA NA**	**500 10**	**Ground Long-Duration Atmosphere Vehicle**

Mission Category: Infrastructure Construction

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Survey Mission – transport people to site for survey purposes (carry life support equipment, return samples, survey tools)	3	1-5 days	10,000 km	Minutes	300	Ballistic
Construction Equipment – bulldozer, soil mover, digger, drill, portable habitat Construction Equipment Transport Equipment	1 1	< 1 day 1-2 days			_ _	Ground Ground
Heavy Lift Air Vehicle	2/Robotic	< 1 day			10,000	Balloon

Mission Category: Agriculture/ Farming

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
No requirement for additional transportation system.						

Mission Category: Manufacturing/Industry

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
No requirement for additional transportation system.						

Mission Category: Resource Mining

Mission/Submission Scope?		# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Resource Transport – move raw materials from mine site to processing site	Rover Flatbed Truck	2/Robotic 1	< 1 day 1-2 days	350 km		10,000 500	Ground Ground
Heavy – Lift Air Transport	Large	Robotic	< 1 day			10,000	Long-duration Air vehicle
Ballistic Launch Trajectory (catapult) – move raw materials from mine site to processing site		Robotic	< 1 day	?		100	Ballistic
★Explosives - Al/O_2, other propellant combinations							
Excavation equipment – drilling, rock crushing, loaders	Excavating Equipment	1	< 1 day				Ground

Mission Category: Weather/Environmental

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
Deploy/maintain/update weather stations 10 – 10,000 stations Call NOAA office UW – what we could need	Robotic Robotic	Indefinite days	5000 km 5000+ km		5 station	Ground Ballistic
Satellite Launch – satellites supplied from Earth	Robotic	< 1 day	Orbit	Minutes	50-100	Launch vehicle

Mission Category: Communication and Navigation Service

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
GPS Satellites – supplied and launched from Earth	**Robotic**				**50**	**Launch Vehicle**
Telecommunication Satellites – supplied and launched from Earth	**Robotic**				**50**	**Launch Vehicle**

Mission Category: Surveying/Mapping

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Satellites – supplied and launched from Earth						
Mapping/Surveying					**30**	**Long-Duration Air Vehicle**

Mission Category: Personal Transportation

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Car/Van **Electric/ISRU**	1-6	< 1 day	160 km	5 Hours	100	Rover
Bus	40	< 1 day	300 km	6 Hours	100	Bus
All-Terrain vehicle	1-2	< 1 day	50 km	5 Hours	200	Small rover
Motorized sled	1	< 1 day	50 km	1 Hour	50	Rover
Ballistic Vehicle Personal Transport		< 1 day	10,000 km	Minutes	100	Ballistic

Mission Category: Package/Mail Delivery

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
Cargo delivery	Robotic	1 week	10,000 km	DAYS	2,000 kg	Ground
Emergency	Robotic	< 1 day	VARIED	Minutes	10 kg	Ballistic

Mission Category: Government Activity

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
Emergency Response (medical supplies, life support) Large Small	6 6	< 1 day < 1 day	10,000 1,000	Minutes Minutes	100 10	Ballistic Ballistic
Law Enforcement (use personal transportation vehicles)	Robotic	Infinite	Infinite	NA	50	Ground

Mission Category: Launch/Space Transport

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Orbital vehicle **Passenger** **Cargo**					**1,000** **1,000**	**Launch Vehicle** **Launch Vehicle**

Mission Category: Auxiliary Power

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
Base emergency back-up power. **Portable power source.**						

Mission Category: Life Support

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
See resource mining transport and cargo transport. **Rest of equipment on the base site.** **No requirement for additional transportation system.**						

Mission Category: Waste/Trash Management

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATION	DISTANCE FROM BASE (KM)	TRAVEL TIME	PAYLOAD MASS (KG)	VEHICLE TYPE REQUIRED
Reprocess on site or shipped via cargo transport						
High-level nuclear waste transport (Iron Ore transport)	2		10,000+ km			Ground

Mission Category: Health Care/Maintenance

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
No additional transport required.						

Mission Category: Virtual Travel Market

Mission/Submission Scope?	# OF CREW/ ROBOTIC	MISSION DURATI ON	DISTANCE FROM BASE (KM)	TRAVE L TIME	PAYLOA D MASS (KG)	VEHICLE TYPE REQUIRED
No additional transport required. Use existing rover to mount video/control system.						