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A Compilation of Lunar and Mars Exploration Strategies Utilizing Indigenous Propellants

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ABSTRACT

The use of propellants manufactured from indigenous space materials has the potential to significantly reduce the amount of mass required to be launched from the Earth's surface. The extent of the leverage, however, along with the cost for developing the infrastructure necessary to support such a process, is unclear. Many mission analyses have been performed that have attempted to quantify the potential benefits of in situ propellant utilization. Because the planning of future space missions includes many unknowns, the presentation of any single study on the use of in situ propellants is often met with critics' claims of the inaccuracy of assumptions or omission of infrastructure requirements. This paper responds to the critics by presenting the results of many such mission analyses in one format. Each mission analysis summarized in this paper used different assumptions and baseline mission scenarios. The conclusion from the studies is that the use of in situ produced propellants will provide significant reductions in Earth launch requirements. This result is consistent among all of the analyses regardless of the assumptions used to obtain the quantitative results. The determination of the best propellant combination and the amount of savings will become clearer and more apparent as the technology work progresses.

NOMENCLATURE

EOI	Earth Orbit Insertion
IMLEO	Initial Mass in Low Earth Orbit
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
LSB	Lunar Surface Base
LTV	Lunar Transfer Vehicle
MOI	Mars Orbit Insertion
TEI	Trans Earth Injection
TLI	Trans Lunar Injection
TMI	Trans Mars Injection

INTRODUCTION

As plans are developed to return men to the moon and continue on to Mars, one overwhelming obstacle stands in the way. That obstacle is cost. The most significant cost of such a program is that of operating a space transportation system to take men and equipment from earth to these extraterrestrial bodies. Since space transportation costs depend on the quantity of mass which must be transported, they can be reasonably represented by the mass of equipment and propellant required to accomplish the mission. This representative mass may be earth launch mass, the mass delivered to the destination, or, most commonly, the initial mass in low earth orbit (IMLEO). Numerous studies have considered methods of reducing a representative mass and, therefore, program costs. One method for reducing exploration mission IMLEO which has received significant consideration is indigenous propellant utilization.

Mission studies have shown that indigenous propellants may greatly reduce the mass required for space exploration and, hence, substantially reduce program costs. However, these mission studies are built on information which can only be estimated. Parameters related to raw material availability, propellant production, propellant performance, and optimum mission architectures are not accurately known. Thus, each analyst must make assumptions about the technology which will be available for the mission, and, invariably, there are differences from one study to the next. Analyses which have considered the use of indigenous propellants have covered a wide range of assumptions. For example, the missions range from a simple robotic sample return to ambitious twenty year manned exploration programs. Some analysts assume extraterrestrial propellant production will be readily available; others penalize the mission for propellant production equipment.

This report summarizes and presents key results of several of the most comprehensive studies of exploration missions using indigenous propellants. The variations in assumptions and techniques are stated for clarity and to demonstrate the range of analyses which have been performed. The objective is to present several analyses in one location with as uniform a format as is possible for easy comparison. The conclusion drawn from these comparisons is that, regardless of the initial assumptions which were made, indigenous propellants show promise for future space exploration programs. This collective result lends additional credibility to the conclusions of the individual studies.

DISCUSSION OF SUMMARIES

Each mission analysis taken from the literature has been summarized in a consistent manner. The summaries consist of a statement of the objectives, descriptions of the mission profile, vehicle, engine, and other assumptions, and a summation of the key results. The mission description includes a specification of the mission destination, mission duration, total number of missions, assumed use of aerobraking, and mission energy requirements. The description of the vehicle includes the number of stages, degree of reusability, and assumptions used for the calculation of the tanks and other dry masses. The engine description includes a list of propellant combinations and their specific impulses, and area ratios, chamber pressures, and thrust levels when available. Other significant assumptions used in the mission analysis may include orbital nodes other than LEO, and whether the additional infrastructure mass for the in situ propellant production is included in the final mass comparisons.

The results of each mission analysis are presented in two bar graphs, both with propellant combination on the independent axis. The first graph compares initial mass in LEO (IMLEO). Some of the mission analyses reviewed for this paper did not perform the mass calculations back to LEO, but rather stopped at some other significant node prior to which all mission options proceed identically. For these analyses, the quantity graphed in the first figure is the mass at that node (e.g. low lunar orbit, Mars orbit, or the Mars surface). For these figures, although the absolute numbers cannot be compared to the other analyses, the percent of mass savings is comparable because all masses will increase by similar multiples when transferred from LEO to the selected node if the same flight path is flown. Smaller mass requirements at the destination node can alternatively allow for a more energetic (faster) flight path to be flown with equivalent initial mass in LEO.

The second graph shows a quantity labelled effective I_{sp}. This quantity was developed to illustrate the benefits of in situ propulsion using engine comparisons. A typical mission analysis calculates a mission mass profile assuming the availability of propellants produced at the destination planet. These propellants often have modest specific impulses. Because they are available at the destination planet, however, the initial mass required in LEO is significantly reduced. The same mission, with the same initial mass in LEO, could be accomplished with all Earth propellants only if the Earth propellants were more efficient, i.e. if they had a higher specific impulse. The effective I_{sp} is therefore a measure of the specific impulse that would be necessary to accomplish the mission for the same initial mass with all propellants brought from the Earth.

An explanation of the calculation of effective I_{sp} follows:

Total Impulse, I, is equal to the integral of thrust, T, over time, t. This integral is equal to the total amount of propellant used, M_{prop}, times the exit velocity, u_e.

$$I = \int T \cdot dt = M_{PROP} \cdot u_e \tag{1}$$

Exit velocity, u_e, is calculated from the specific impulse, I_{sp}, times the gravitational acceleration at the Earth's surface, g_e. If various mission legs utilize different propulsion systems and exit velocities, the impulse for each mission segment is calculated separately and then summed together for a total mission impulse. Equation (1) can then be rewritten to calculate the average mission specific impulse.

$$I_{sp} = \frac{I}{(M_{PROP} \cdot g_e)} \tag{2}$$

In order to calculate the effective specific impulse necessary for all Earth propellants to provide the mission total impulse for the same initial mass, the mass of the earth-based propellants, $M_{prop(e)}$, is used in equation (2) in place of M_{prop}.

$$I_{sp(eff)} = \frac{I}{(M_{PROP(e)} \cdot g_e)} \quad (3)$$

(Note that for the baseline mission where all of the propellants are from Earth, equation (3) reduces to equation (2). The I_{sp} calculated in equation (3) will be the same as that used to calculate u_e in equation (1).)

Figure *i* parametrically shows the rise in effective specific impulse as the amount of in situ production is increased. The figure shows that as the amount of in situ propellant production approaches 100 percent, the effective specific impulse for the mission or portion of the mission approaches infinity. For example, the scale on the left shows an increase in effective specific impulse from a baseline of 475 seconds to 10,000 seconds when 95 percent of the propellants are produced in situ. The scale on the right shows that this is a 2000 percent increase in effective specific impulse.

For the mission analyses where a mass was calculated for the propellant processing infrastructure, this mass was considered part of the earth-based propellant mass in equation (3). For the studies that ended the mission mass analysis at the planet that provided the in situ propellants, and if an infrastructure mass was not included, then no effective I_{sp} graph is included. If the mass calculations were not carried back to LEO, then no Earth propellants were used at all, and the effective I_{sp} approaches infinity for that part of the mission, as can be seen in figure *i*.

Some of the analyses reviewed for this paper did not perform the calculations for the baseline propellants through to the same point as the in situ propellant options. For these cases, if enough information was provided, the mission mass calculations for the baseline propellants was repeated in order to provide a consistent comparison with the other mission analyses.

RESULTS OF MISSION ANALYSES

The celestial bodies that currently receive the most attention as potential sources of propellants are the Earth's moon, Mars, the martian moons, Phobos and Deimos, and the near-Earth asteroids. Because the moon, Mars, and Phobos and Deimos are considered to be targets for future manned missions and bases, only mission analyses studying these nodes were included in this paper.

The first two studies summarized consider the use of lunar produced propellants for lunar vicinity operations. The next paper extends this idea to using lunar propellants for a trip to Mars. The remaining summaries consider the use of Mars produced propellants for various Mars missions. First, a Mars sample return mission is summarized, then two manned Mars missions using Mars propellants for Mars ascent and Earth return. Finally, a study that assumes a more ambitious infrastructure is summarized, where propellants are manufactured at both Mars and Deimos.

Each summarized study is presented with the resultant graphs. The summaries are identified by the titles of the original papers from which they were taken, although other related papers by the same author(s) may have been used to present the complete analysis. All individual publications used are listed in the references.

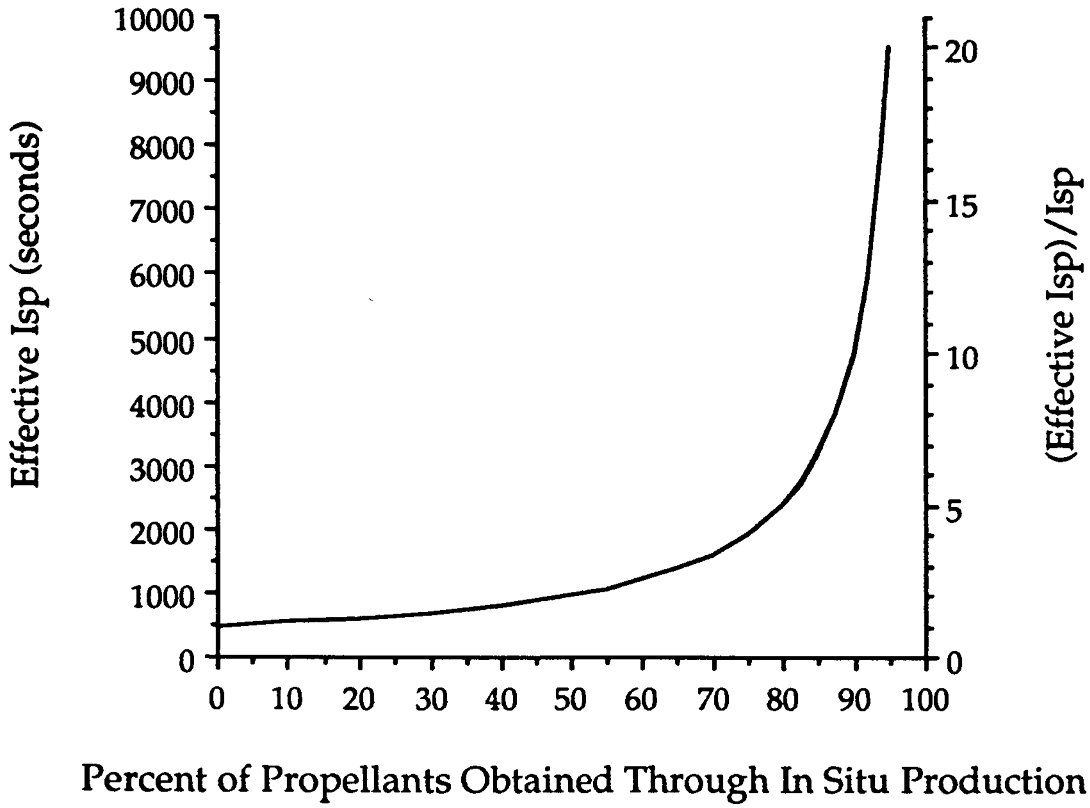

Figure i. - Example of increase in effective specific impulse with increasing use of in situ propellants

"Lunar Surface Base Propulsion System Study"[1]

This report proposes that cost of transportation supporting a permanent lunar base can be significantly reduced if lunar produced oxygen or lunar produced oxygen and fuel is used by the lander and earth return vehicle. Several potential lunar propellants were considered including oxygen, aluminum, silane produced with earth-supplied hydrogen and lunar silicon, and earth-supplied hydrogen metallized with lunar aluminum. In addition, the effect of operating an H_2/O_2 propulsion system at high mixture ratio was evaluated.

Mission. The missions considered were taken from a 1986 Johnson Space Center lunar base model. The model spans 20 years, beginning in 1995, and includes build-up and support of a permanently manned lunar base. A typical mission incorporated lunar orbit rendezvous and aerocapture upon earth return. The delta V for the outbound trip was 4170 m/s. Lunar descent and ascent delta V's were 2100 m/s each, and the earth return delta V was reduced to 1110 m/s by aerobraking in the earth's atmosphere. The portion of the mission for which indigenous propellants are used varies from none, to lander ascent/descent, to lander and return trip legs. Two round trips are required by the lander to supply the lunar transfer vehicle (LTV) with propellant for return to low earth orbit (LEO).

Vehicle. Two vehicles were utilized for each mission. The lunar transfer vehicle, based in LEO, traveled back and forth between LEO and low lunar orbit (LLO). The lunar ascent/ descent vehicle, based on the lunar surface, delivered payload to and from LLO. The maximum mission payload was 15.9 tonnes, which included a 6.9 tonnes capsule for manned missions.

Engines. The engines for the vehicles were designed and sized specifically for each propellant combination using the ELES-1984 computer code. H_2/O_2 systems are designed to operate at 13.8 MPa, 33.3 kN thrust, with a nozzle expansion ratio of 300:1. The other propellant systems operate at 6.9 MPa, 33.3 kN thrust, and a nozzle expansion ratio of 100:1. Propellant and performance specifications for the various engines are presented in table 1.

Other Assumptions. The cases presented here assumed the infrastructure required for propellant production and refueling was already in place at an established lunar base. These hardware masses ranged from 1 to 35 tonnes depending on the propellant being produced. It was assumed the emplacement cost of this mass would be amortized over the 20 year model so the costs are not specifically charged to any one mission. Each of the options which used lunar oxygen for earth return also delivered some amount of excess oxygen to LEO. However, most of the oxygen for the trans-lunar injection was earth supplied.

Results. The results in figures 1a and 1b are presented for a baseline H_2/O_2 case and six propulsion system options. The option numbers correspond to those in table 1. The results in figure 1a show that option 2, lunar Al/O_2, provided the greatest benefit compared to the baseline. It reduced IMLEO by 84 tonnes or 63 percent. Options 5 & 6 differed from the baseline and option 1 respectively by increasing the mixture ratio from 5.5 to 8.73. This attempt to make more use of the lunar oxygen was not beneficial. The maximum effective specific impulse (option 2 in figure 1b) is nearly 2000 seconds.

Table 1. Engine Propellants and Performance Estimates

Option	Baseline	1	2	3	4	5	6
Propellant* & Performance (Lander)	H_2/O_2 470 sec	$H_2/$ Lun O_2 470 sec	Lunar Al/O_2 260 sec	Lunar SiH_4/O_2 366 sec	Lunar $Al\text{-}H_2/O_2$ 400 sec	H_2/O_2 MR=8.73 421 sec	$H_2/$ Lun O_2 MR=8.73 421 sec
Propellant* & Performance (OTV)	H_2/O_2 470 sec	H_2/O_2 Lun O_2 Return 470 sec	H_2/O_2 Lun O_2 Return 470 sec	SiH_4/O_2 Lunar Prop. for Return 366 sec	$Al\text{-}H_2/O_2$ Lunar Prop. for Return 400 sec	H_2/O_2 MR=8.73 421 sec	H_2/O_2 MR=8.73 Lun O_2 Return 421 sec

* All hydrogen used is supplied from earth.

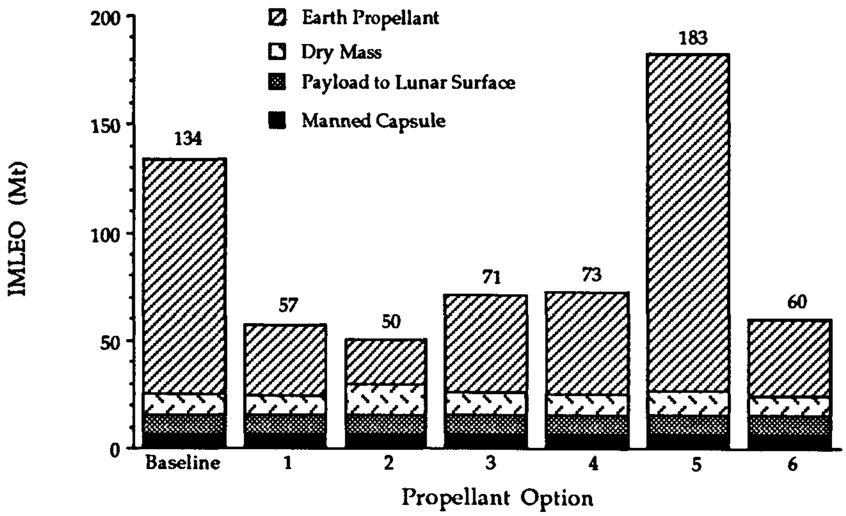

Figure 1a. - Initial Mass in low earth orbit supplied from earth for a single mission.

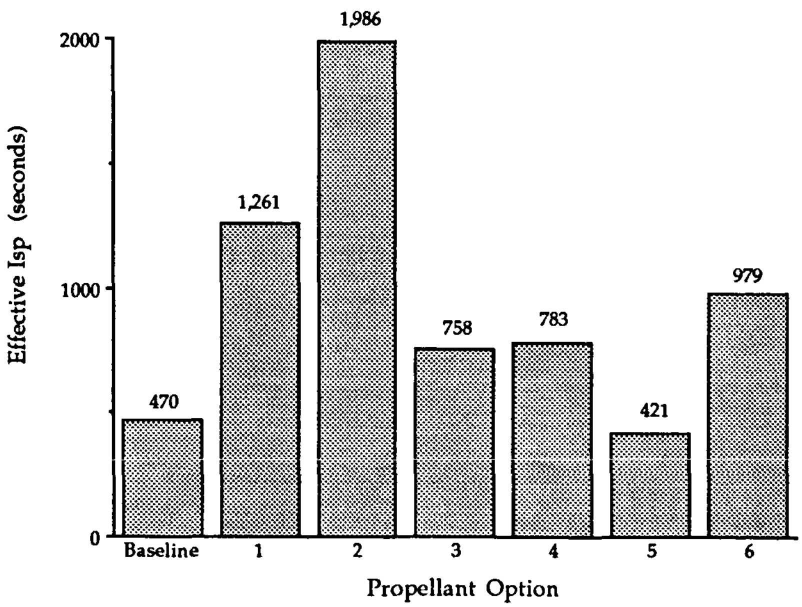

Figure 1b. - Effective Isp for baseline and six propellant options of reference 1.

"Lunar Base Spacecraft Propulsion with Lunar Propellants"[2]

This paper proposes a variety of propellants which could be produced from lunar resources for use in lunar ascent/descent missions. The required propellant masses to perform the missions are calculated for each of the propellant options. The results suggest that the propellant mass which must be delivered from low earth orbit (LEO) to low lunar orbit (LLO) for the ascent/descent vehicle operations can be greatly reduced or eliminated.

Mission. The mission presented here is representative of several manned lunar ascent/descent missions evaluated in the paper. Two ascent/descent vehicles are used. One is an unmanned tanker that delivers propellant to a LLO node and returns to the LSB empty. The other vehicle lifts a manned capsule from the LSB to LLO, fills up on the propellant delivered by the unmanned tanker, and then returns to the LSB with an additional payload picked up at the LLO node. The mission is derived from a NASA Johnson Space Center mission model for a permanently manned lunar base. The model initiates manned missions in 2003 and increases frequency to approximately three manned lunar missions per year by 2006. Lunar oxygen is assumed to be available in 2008 for propellant usage.

Vehicle. The manned vehicle is a single stage lander based on the lunar surface. The manned capsule is sized at 6.8 tonnes, and the vehicle delivers an additional 10.4 tonnes payload from LLO to the lunar surface. The propellant tanker vehicle is also a single stage vehicle sized to deliver the required propellant in one trip from the surface to LLO. A structural coefficient of 10 percent ($m_s/(m_s + m_{prop}) = 0.1$) was assumed for both vehicles.

Engine. Engine performance for each of the potential propellant combinations was estimated using a one-dimensional equilibrium computer code. The engines operate at a chamber pressure of 6.9 MPa and an expansion ratio of 50:1. The propellants, mixture ratios, and specific impulse are listed in table 2.

Other Assumptions. No calculations of low earth orbit to LLO transfer requirements were performed. For the aluminum/oxygen propellant, a small (2 percent of the mass of aluminum) amount of helium, supplied from earth, was used to transport the aluminum to the thrust chamber. The LLO node depot was not included in the calculations.

Results. The results are presented in two bar graphs. (We repeated the author's calculations to produce the baseline H_2/O_2 masses). The graph in figure 2a indicates the total dry mass and propellant which must be delivered to LLO from LEO for a single ascent/descent mission. The tanker vehicle requirements (propellant and dry masses) are included in these charts. The value of lunar propellants is indicated by a maximum reduction in mass delivered from LEO of 39 percent when lunar oxygen is used. The graph in figure 2b charts the cumulative impact of the lunar propellants for five missions with re-usable vehicles. By re-using the vehicles, the heavy tanks of the all lunar propellant vehicles are spread out over five missions. The maximum advantage of indigenous propellants is increased to 54 percent, but this cumulative benefit is maximized for all-lunar propellant cases with Al/O_2 or S/O_2. It was not possible to calculate a meaningful effective I_{sp} because the mass analysis was not carried back to LEO.

Table 2. Engine propellants and performance estimates for reference 2.

Option	Baseline	1	2	3
Propellant	H_2/O_2 All Earth Supplied	H_2/O_2 Lunar O_2	Al/2% He/O_2 He from Earth	Sulfur/O_2 All Lunar Supplied
Mixture Ratio	6.0	6.0	3.0	1.0
Specific Impulse (sec)	456.5	456.5	276.1	244.5

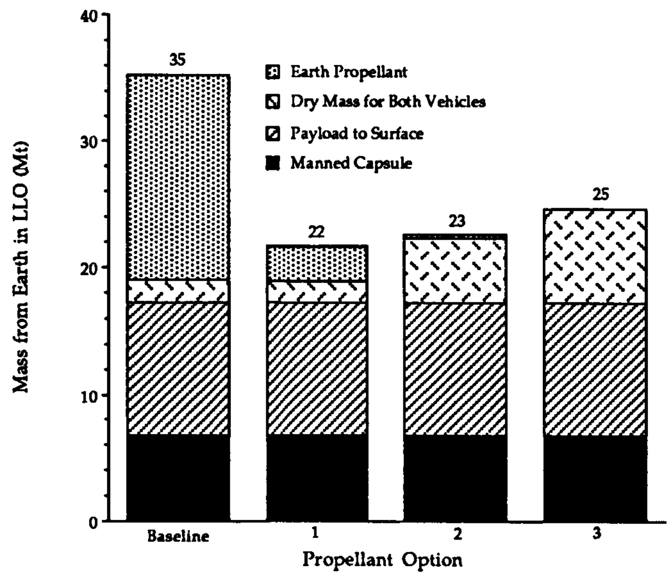

Figure 2a. - Propellant and vehicle dry mass delivered to LLO from LEO for single mission.

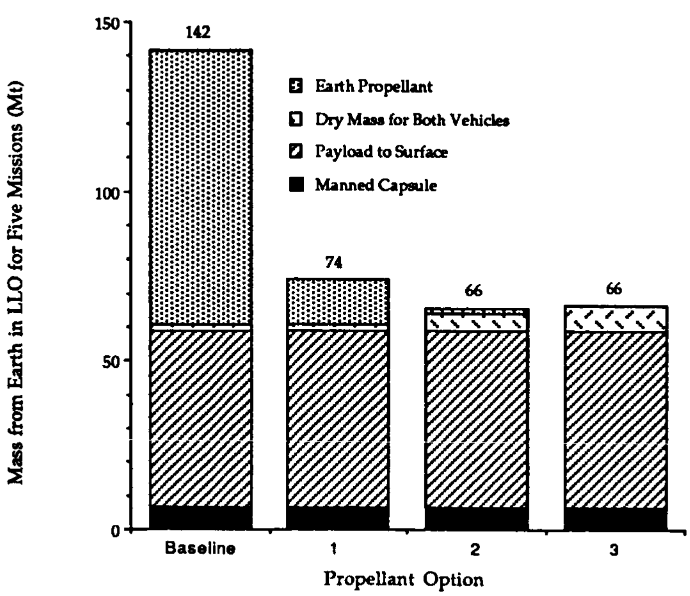

Figure 2b. - Cumulative mass in LLO from LEO for five missions with reusable vehicles.

"If We're Going to Mars, Why Stop at the Moon?"[3]

This paper proposes that for the expanded exploration plan, manned missions to Mars can benefit significantly from using the established lunar base as a refueling depot. Lunar oxygen is used for the lunar to Mars and Earth return trips, and for all lunar vicinity operations. The paper also investigates added benefits obtained when using a lunar fuel for all lunar vicinity operations.

Mission. The mission considered is a manned Mars mission of the sprint class. Typical mission duration is 390 days, with 150 days for the trip to Mars, 30 days on the surface, and 210 days for the return trip to Earth. The baseline mission leaves directly from LEO for Mars. The missions that use in situ resources travel from LEO to low lunar orbit (LLO), refuel, and then travel to Mars. The paper considered two trajectories from the moon to Mars: direct from LLO to Mars, and LLO with an Earth-swing-by to Mars. Only results from the former will be included in this paper, because this was the lower energy option. Aerobraking was assumed at both Mars and the Earth. The breakdown in the delta V requirements is given in table 3 for a departure in February or March, 2016.

Vehicle. A multi-stage, expendable vehicle is used with stages being discarded after each major mission leg (with the exception of the trans lunar injection tanks being refilled at LLO for the trans Mars injection burn). Tank masses were calculated as four percent of the propellant mass. The aerobrake masses were calculated as 9.5 percent of the braked mass. The manned capsule, which is the only component that completes the entire LEO-Mars-LEO round trip, was approximately 62.7 tonnes.

Engine. An oxygen/hydrogen engine at a mixture ratio of 6 was assumed for all stages. For the Earth departure stage (trans Mars injection for the baseline or trans lunar injection for lunar staging), an I_{sp} of 475 sec was assumed. An I_{sp} of 485 seconds was assumed for all other stages.

Other Assumptions. This study assumed the infrastructure required for propellant production and refueling was already in place at an established lunar base. This infrastructure consisted of a lunar oxygen production facility and a tanker vehicle to carry the surface produced propellants to low lunar orbit. The tanker uses lunar oxygen and either hydrogen brought from Earth by the Mars vehicle, or silane (SiH4) produced on the lunar surface.

Results. The IMLEO and effective I_{sp} results are presented for three cases in figure 3. The first case is the baseline case where all of the oxygen and hydrogen are brought from Earth, and the vehicle travels from LEO directly to Mars. The second case is the in situ option, where the vehicle travels from LEO to lunar orbit and picks up enough oxygen for the remainder of the trip (the tanker uses hydrogen brought from Earth). The third case is the same as the second, except that the tanker uses silane produced on the lunar surface. Figure 3a shows that the use of lunar oxygen provides a 22 percent reduction in IMLEO, and the use of lunar silane fuel for the tanker provides an additional 14 percent reduction. Figure 3b shows that the use of lunar oxygen and silane can more than double the effective specific impulse of the Earth-supplied propellants.

Table 3. - Delta V (km/sec) requirements for Feb./March, 2016 departure to Mars

	Trans Lunar Injection	Lunar Orbit Insertion	Trans Mars Injection	Mars Orbit Insertion	Trans Earth Injection	Earth Orbit Insertion	TOTAL
Baseline	0.0	0.0	3.63	0.0	4.92	0.29	8.84
Earth-Moon-Mars	3.2	1.1	1.72	0.0	4.92	0.29	11.23

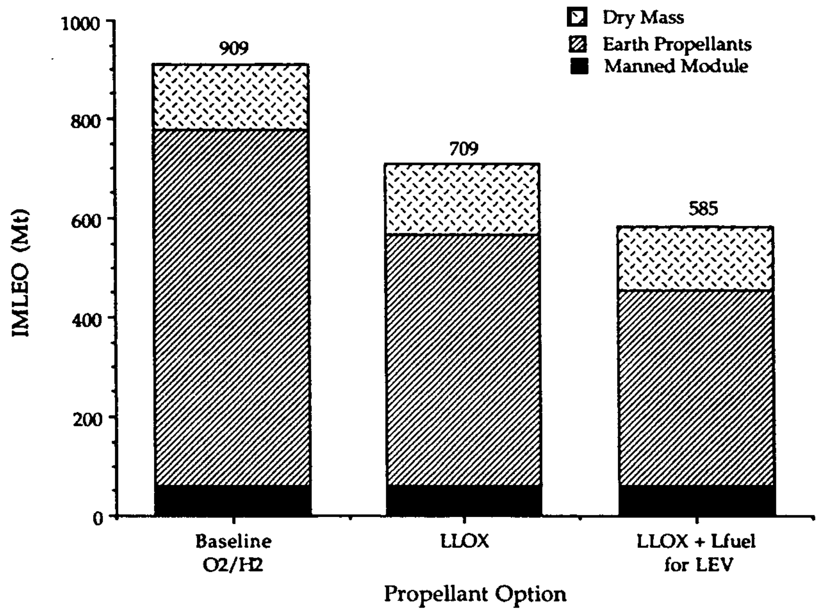

Figure 3a. - Initial mass in LEO for a manned Mars mission using lunar propellants

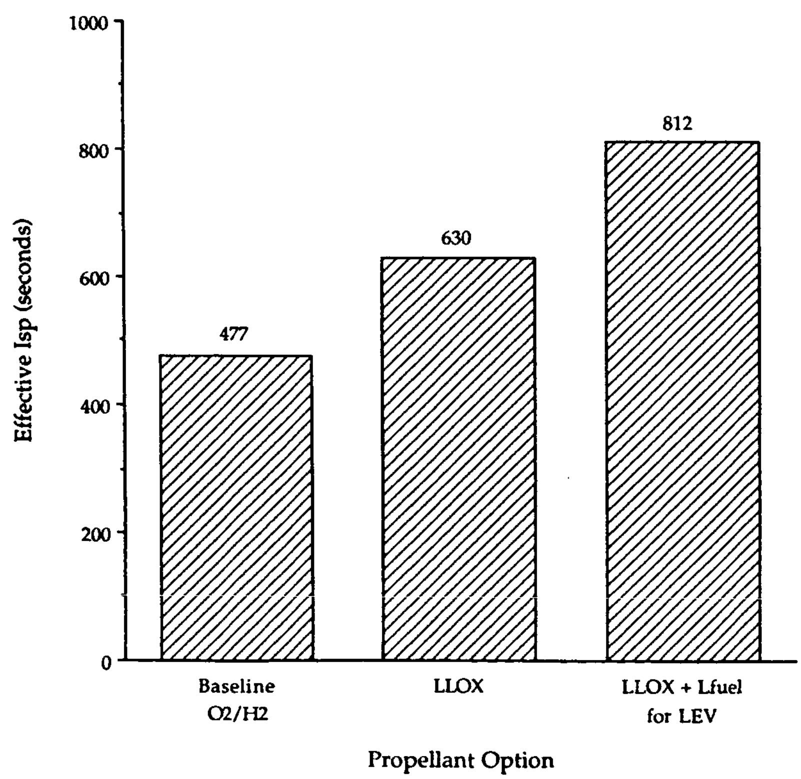

Figure 3b. - Effective Isp for manned Mars mission using lunar propellants

"In Situ Propellant Production for Improved Sample Return Mission Performance"[4]

This paper proposes that a Mars Sample Return mission using a direct entry/direct return trajectory can be enabled in one shuttle launch if return propellants are produced on the surface of Mars. In this scenario, a small processing plant is carried to the surface in lieu of return propellant. This plant produces either oxygen, or oxygen and methane for a direct return to Earth.

Mission. The objective of the mission is to journey to Mars and return a 1 kg sample to Earth orbit, where it will be recovered by the space shuttle. Although the paper investigated the sample return mission for all of the possible mission modes, only the direct entry/direct return mission mode will be summarized here. This demonstrates how in situ propellants can enable the mission in one shuttle launch, while eliminating the need for an autonomous Mars orbit rendezvous. Both a 1990 and a 1994 mission opportunity are evaluated; these opportunities are representative of a poorer and better mission opportunity in terms of energy requirements. The delta V schedule and mission times are listed in tables 4a and 4b.

Vehicle. The vehicle stages were taken from the Sample Return Mission model as developed by JPL at the time that the analysis was performed (1978). The sample is returned to Earth orbit in an Earth Orbiting Capsule which masses 30 kg dry and 55 to 58 kg wet. The propellant tank fraction is 0.16 times the propellant mass. The Mars ascent and return vehicle is a three stage vehicle for the direct return options that utilize oxygen and methane.

Engine. The oxygen/methane engines were assumed to achieve a specific impulse of 342 seconds at a mixture ratio of 3.4. The mass of each engine was assumed to be 75 kg. The baseline case used solids for Mars ascent and Earth return with a specific impulse of 285 seconds.

Other Assumptions. Two production plants were sized: one to produce oxygen only, and one to produce oxygen and methane (the authors acknowledged that because of the scarcity of water at Mars, methane production may be difficult unless a quantity of hydrogen is brought from Earth). The oxygen plant massed 690 kg and the oxygen & methane plant massed 900 kg. Both were sized for a production rate of 10 kg/day.

Results. Figure 4 shows the results of the mission analyses for three cases. The first is the baseline solid propellant case. The second case is for oxygen production with Earth methane. The third case is oxygen and methane production at Mars. Both the 1990 and 1994 opportunities are shown. Figure 4a shows the required injected mass and the Shuttle/IUS capability for each opportunity. Figure 4b shows the effective specific impulse for the Mars ascent and Earth return portions of the mission. For the in situ cases, the masses of the processing plants were included in the Earth propellant mass. Although figure 4a shows that the production of propellants at Mars can reduce IMLEO by over 60 percent, the more significant result is that the sample return mission can now be accomplished with a single shuttle launch. Also, the production of oxygen at Mars can nearly triple the effective specific impulse of the Earth-supplied propellants.

Table 4a. - Delta V (km/sec) requirements for three stage Mars ascent/Earth return

	1st Stage	2nd Stage	3rd Stage
1990	2130	2130	2381
1994	2130	2130	2197

Table 4b. - Mission dates and flight times for Mars Sample Return Mission

Launch	Flight Time	Arrive	Stay Time	Leave	Flight Time	Arrive Earth	Total Time
8/31/90	400	10/5/91	291	7/22/92	288	5/6/93	979
10/14/94	308	8/18/95	342	7/25/96	343	7/3/97	993

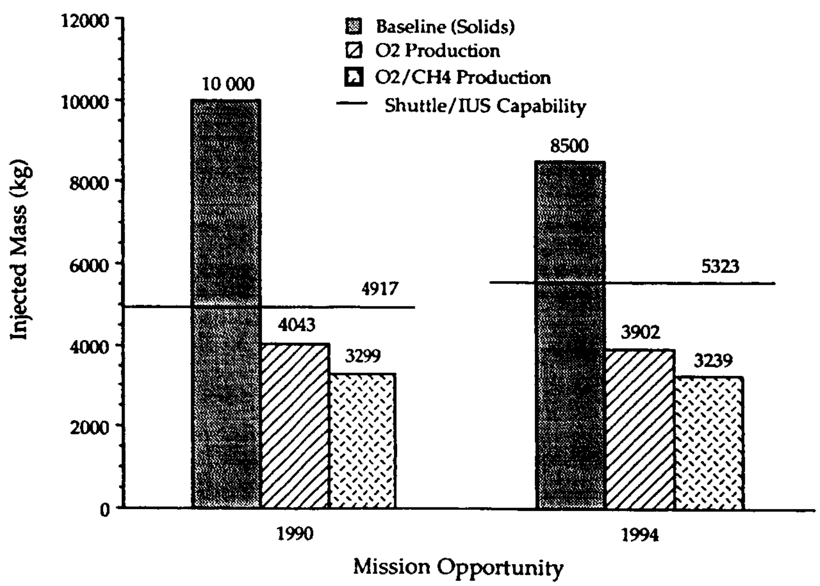

Figure 4a. - Injected mass to Mars for a direct entry/direct return Mars Sample Return mission using Mars propellants

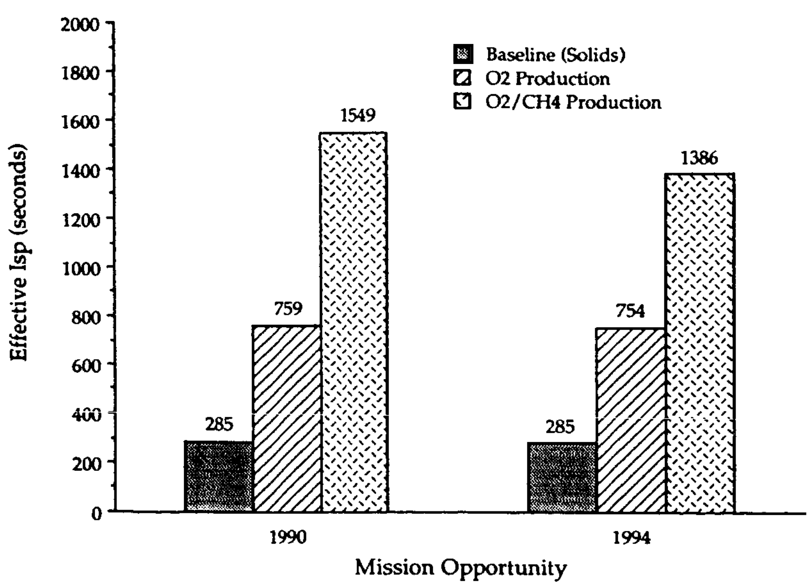

Figure 4b. - Effective Isp for direct entry/direct return Mars Sample Return mission using Mars propellants

"Rocket Propellants from Martian Resources"[5]

This paper proposes that the production of propellants on the surface of Mars will be essential to reduce Earth launch requirements. Oxygen is the oxidizer of choice, with both methane and carbon monoxide considered for the in situ fuel.

Mission. The mission considered was a manned Mars mission, with in situ propellants used for the Mars ascent to orbit or ascent with a direct return to Earth. The mission was of the low energy class, with a round trip of approximately three years and stay times of at least one year. Delta V's of 4600 m/s for ascent to orbit and 6600 m/s for direct return were assumed. The delta V for the direct return mission was based on an injection energy of 5.8 km^2/sec^2. The payload delivered to Mars orbit or to an Earth return trajectory was a 10,000 kg manned capsule.

Vehicle. A one stage ascent vehicle was assumed unless the initial-to-final mass ratio exceeded six. For those cases, a two stage ascent vehicle was used with an even split of the delta V requirement between the two stages. For the analysis presented, the O_2/H_2 baseline used one stage for both missions; the O_2/CH_4 used one stage for the ascent to orbit, and two stages for the direct return; the O_2/CO used two stages for both missions.

Engine. The specific impulses that were assumed for the analysis were 440 seconds for the O_2/H_2, 360 seconds for the O_2/CH_4, and 260 seconds for the O_2/CO.

Other Assumptions. The in situ production system mass was included in the mass-to-surface calculations. The O_2/CH_4 system was sized for an oxygen production rate of 85 kg/day (ascent to orbit) and 221 kg/day (direct return). The estimated mass and power requirements of these systems were 1550 kg at 15 kW_e, and 2900 kg at 39 kW_e, respectively. The O_2/CO system estimated mass was 1700 kg and 3200 kg for the ascent to orbit and direct return missions, respectively. The mass of the power system was not included in the mass to surface calculations.

Results. Figure 5a shows the mass that must be delivered to the Mars surface for the three different propellant options. (The values for the O_2/H_2 all Earth propellants were calculated based on information contained in the original analysis.) Because the mission to deliver this mass to the Mars surface would be identical for all cases, these masses indicate the benefits that would be achieved in reducing Earth launch mass. The figure shows that production of methane and oxygen at Mars reduces the Mars landed mass by 40 percent. Production of carbon monoxide and oxygen reduces the landed mass by 50 to 60 percent. Figure 5b shows the effective I_{sp} for the Mars ascent and Earth return portion of the mission. Because this mission does not consider the trip from Earth to Mars, the Earth-supplied propellant mass is reduced to only the mass of the Mars production system, and effective I_{sp} is very high. Figure 5b indicates that Earth-supplied propellants would need to have an effective I_{sp} nearly 30 times greater to accomplish this mission with the same initial mass.

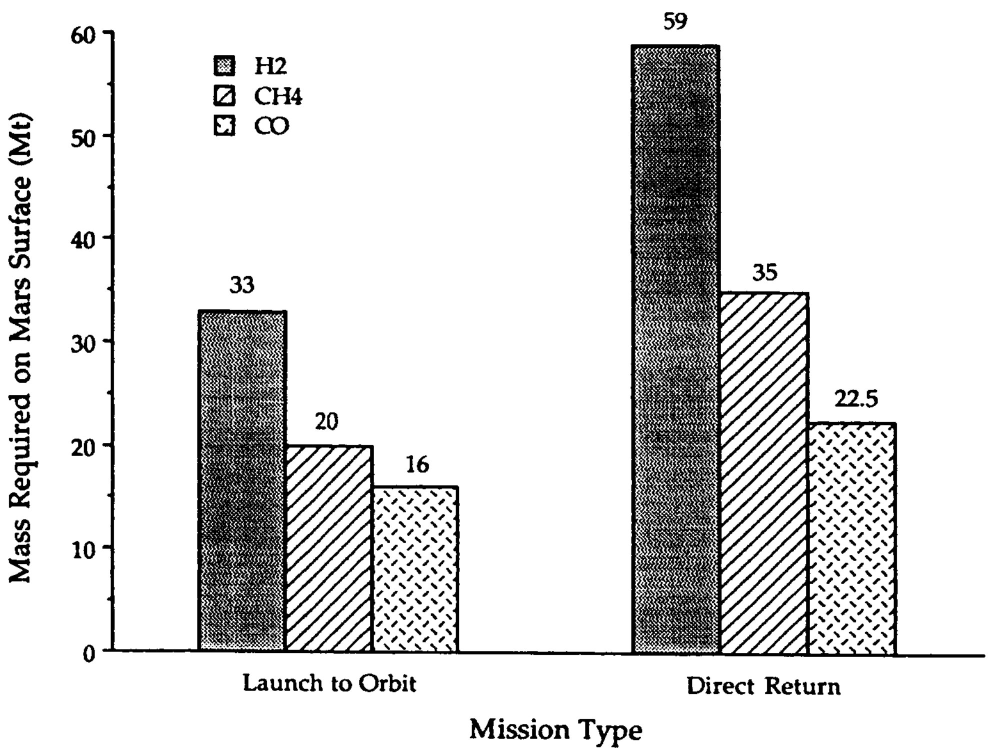

Figure 5a. - Mass required delivered to surface of Mars for Earth return using Mars propellants

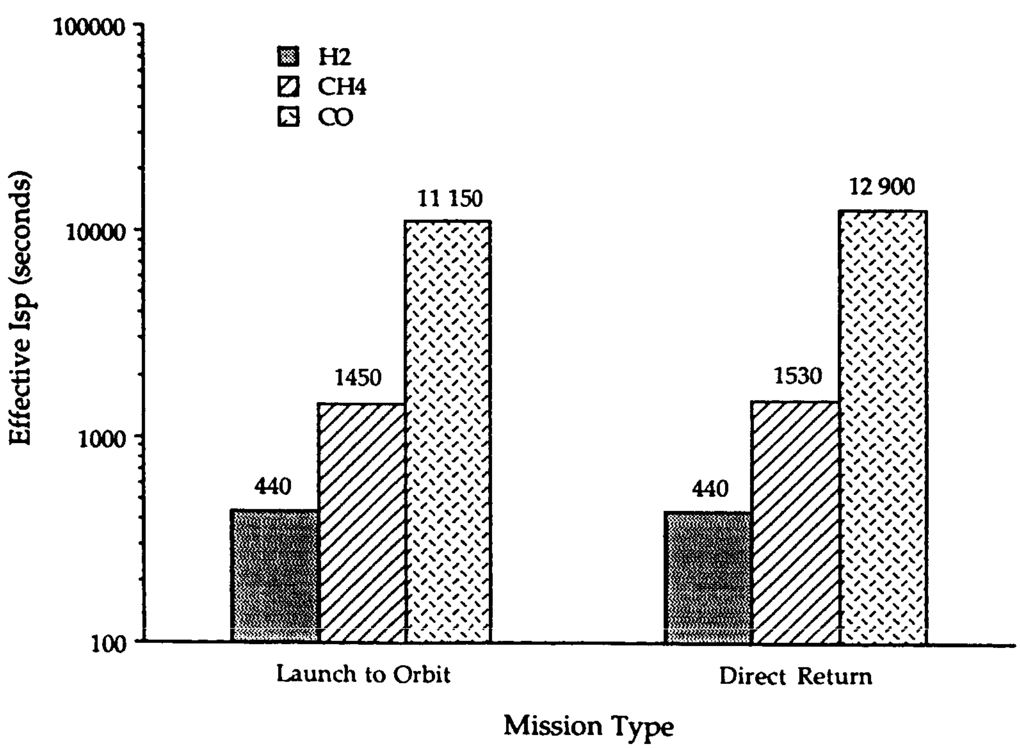

Figure 5b. - Effective Isp for Mars launch and Earth return missions using Mars propellants

"Mars Direct"[6]

This concept proposes that repeated manned missions to Mars can begin in the year 1999 if the return propellants are produced at Mars. In this scenario, hydrogen would be sent to Mars with an automated processing plant. The hydrogen would then be combined with the atmospheric carbon dioxide to produce oxygen and methane propellants.

Mission. Each manned mission to Mars is preceded by a companion unmanned mission that delivers the hydrogen, automated processing plant and power supply, and Earth return vehicle. The paper begins with the unmanned mission leaving Earth in December, 1996, with an injection energy (C_3) of 10 km^2/sec^2 and a trip time of 230 days. The manned mission leaves in January, 1999, with a C_3 of 15 km^2/sec^2, and requires 190 days for the outbound trip to the prepared site where the astronauts spend 500 days. The Earth return vehicle is then used for the 180-day return trip. Aerobraking is assumed at Mars and Earth, and also for Mars landing. The delta V for the upper stage, which is ignited sub-orbital and then propels the manned craft to Mars, is 6436 m/s. The Mars lander delta V is 563 m/s. The delta V for Mars launch and Earth return is 6300 m/s.

Vehicle. The manned habitat used outbound and on the surface masses 28 tonnes. The aerobrake mass is assumed to be 15 percent of the braked mass. The Mars launch vehicle is a two stage vehicle with an inert mass of 12 percent and a tank mass of 10 percent of the propellant mass. The manned capsule for Earth return masses approximately 12 tonnes.

Engine. Oxygen and methane at a mixture ratio of 3.6 and a specific impulse of 373 seconds are used for the Mars launch and Earth return trip. The two-stage vehicle uses a 90 kN engine, with eight engines on the first stage and one engine on the second stage. The Mars lander uses an oxygen/hydrogen engine with a specific impulse of 450 seconds.

Other Assumptions. The production plant that is sent out on the unmanned mission uses a 100 kW_e power source and can produce all of the propellant for the return trip in 310 days operating at 50 percent power levels. Additionally, 12 tonnes of methane and oxygen are available for a pressurized ground rover and unpressurized light truck. To enable the unmanned and manned missions to each be launched in one launch, a heavy lift launch vehicle is assumed with an upper stage capability of 47,200 kg to an injection energy (C_3) of 15 km^2/sec^2.

Results. Figure 6a shows the breakdown of the total mass required on the surface of Mars with and without in situ propellant production. Without propellant production, approximately 100 tons of methane and oxygen must be landed on the surface for return to Earth. The Mars Direct scenario replaces this requirement with 5.8 tons of hydrogen and a 5.6 ton processing and power plant to decrease the required mass delivered to the surface by 60 percent. The larger surface mass required without in situ propellants would also require heavier landing vehicles and heavier upper stages for the trans-Mars injection. Figure 6b shows the effective specific impulse of the entire mission compared to that of the oxygen/hydrogen upper stage, which is the maximum specific impulse that could be achieved with all Earth propellants. The figure shows that the production of oxygen and methane at Mars nearly doubles the effective specific impulse.

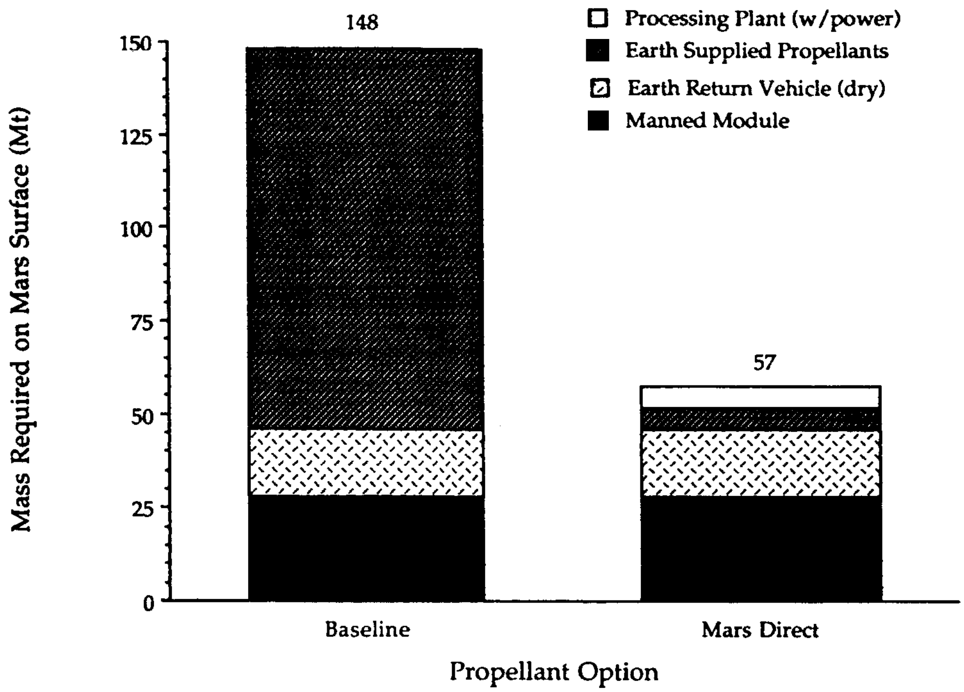

Figure 6a. - Mass required delivered to Mars surface with and without methane/oxygen production at Mars

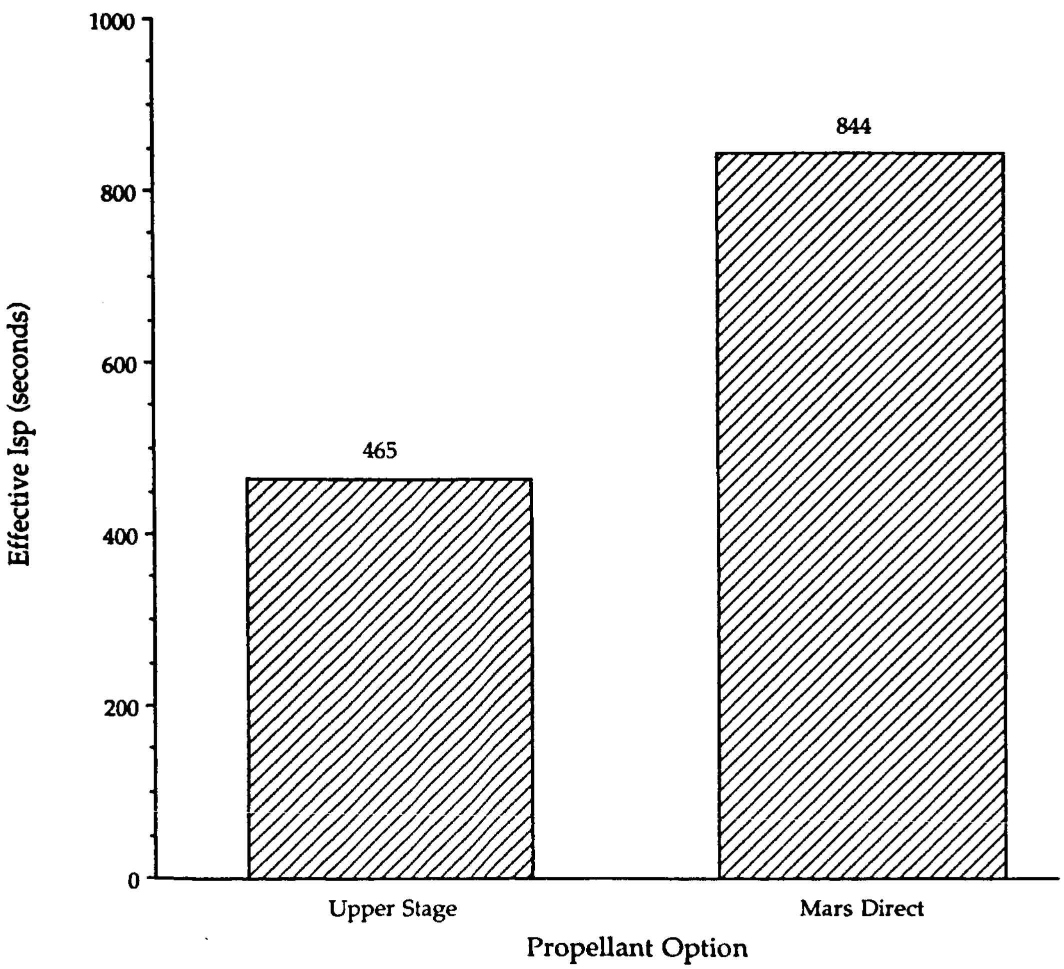

Figure 6b. - Effective Isp for "Mars Direct" option using Mars methane/oxygen propellants

"A Get Started Approach For Resource Processing"[7]

This paper proposes that propellant processing equipment be delivered to Deimos and Mars incrementally, with full production capability reached after five missions. Subsequent missions can then take full advantage of the oxygen and methane produced for Mars ascent/descent and for Earth return.

Mission. The mission profile consists of manned missions to Mars of the conjunction class with 330 to 550 day stay times. Aerobraking was assumed at both Mars and Earth. Five Deimos processing plants provide sufficient propellant (oxygen/methane) for all Mars orbital operations, for Mars descent, and for Earth return. The Mars ascent vehicles receive their oxygen/methane on the surface from five more processing plants (the hydrogen for the methane is obtained from ice previously delivered from Deimos). The delta V schedule that was used for the mission analysis is listed in table 7a.

Vehicle. Two shuttle vehicles, two Mars excursion vehicles, and an Earth vehicle are used for various parts of the mission. Table 7b lists the vehicle types, their uses, and their dry masses. All dry masses used in the analysis are approximately 15 percent of the propellant mass.

Engine. The baseline case uses an O_2/H_2 engine with a 461 second I_{sp} for the trans Mars injection, an N_2O_4/MMH engine with a 354 second I_{sp} for the Mars excursion vehicle, and an O_2/H_2 engine with a 482 second I_{sp} for the trans Earth injection. The in situ propellant mission scenario uses the same engine for the trans Mars injection, and an O_2/CH_4 engine with a 365 second I_{sp} for the Mars excursion vehicles and the trans Earth injection. The mixture ratio of the oxygen/methane engine is 3.4.

Other Assumptions. It was assumed that the soil on Deimos contains 10 percent water, and that a fuel depot is in a Mars parking orbit. The total time for propellant production is the 2.2 year interval between conjunction class opportunities. Each Deimos processing plant produces 44 tonnes of oxygen and methane and 13 tonnes of ice per duty cycle, masses 1.5 tonnes, and requires 13 kW_e and 4 kW_t of power. The Mars processing plants produce a total of 82 tonnes of propellant every 2.2 years. Each plant masses 2.5 tonnes, and requires 20 kW_e (day), 3 kW_e (night), and 4.4 kW_t of power.

Results. Figure 7a compares the mass required in LEO for two propellant options. The first option is the baseline case and uses all propellants from Earth. The second option is the ISP (in situ propellants) option, and it uses Earth oxygen/hydrogen propellants for the trans Mars leg, and Deimos or Mars produced oxygen/methane for all other legs. The figure shows a 25 percent reduction in mass in LEO for the ISP option. Figure 7b shows the effective I_{sp} for both propellant options. For the ISP option, one Deimos and one Mars processing plant are included in the Earth propellant mass. The ISP option provides a 40 percent increase in the effective specific impulse of the Earth-supplied propellants.

Table 7a. - Delta V requirements for manned Mars mission

Mission Leg	Delta V (m/s)
Trans Mars Injection	3700
Mars Descent	1300
Mars Ascent	5200
Mars Orbit to/from Deimos	800
Trans Earth Injection	1300 - 2000

Table 7b. - Summary of mission vehicles

Vehicle	#	Mission Use	Dry Mass (kg)
Shuttle	2	Transport O2/CH4 and ice from Deimos to fuel Depot Transport crew from fuel depot to Deimos and back Transport ice from fuel depot to Mars surface	1000
Mars Excursion Vehicle	2	Transport crew and payload (25 - 44 t) to surface Transport crew from surface to Mars parking orbit	3000
Earth Return Vehicle	1	Return crew to Earth orbit	9000

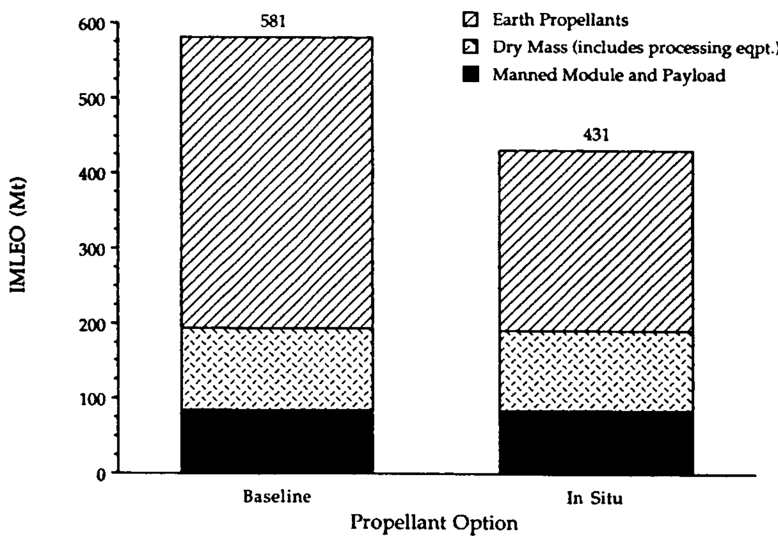

Figure 7a. - Initial mass in LEO for a manned Mars mission using Mars/Deimos propellants

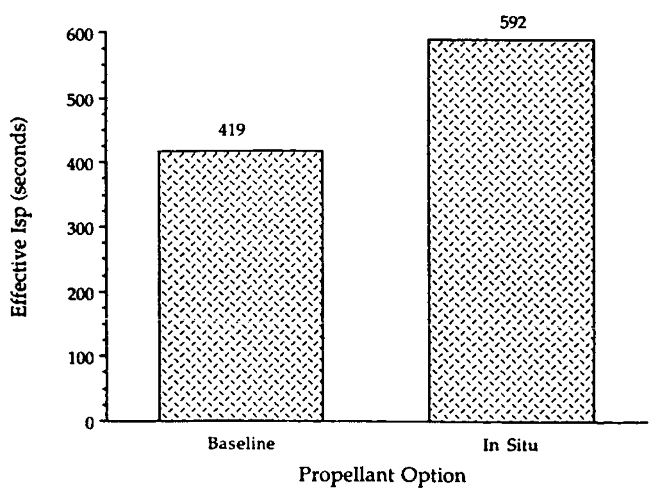

Figure 7b. - Effective Isp for a manned Mars mission using Mars/Deimos propellants

CONCLUSIONS

Several papers from the literature that studied the potential of in situ propellants were summarized. Because of the many unknowns involved with the planning of future space missions, the mission analyses used many different assumptions. The baseline mission, mission energy requirements, vehicle mass sizing parameters, infrastructure requirements, and potential engine performance were different for most of the studies. Despite these differences, the results from each separate effort are consistent. The use of in situ propellants from the moon and Mars can reduce the amount of mass required from the Earth's surface by 22 to 63 percent.

In addition to the mission mass values presented, an effective specific impulse for the propellant from earth was calculated. To accomplish the same missions with the same initial masses in low Earth orbit as a vehicle that uses in situ propellants, a vehicle that uses all propellants from the Earth would have to deliver an I_{sp} of 592 to 12,000 seconds to match the effective I_{sp} of the various indigenous propellant options. The effective I_{sp} parameter can be used to make comparisons between indigenous propellant options and other advanced propulsion concepts.

These collective results indicate that, although it is difficult to know with certainty the exact technologies that will be available for future lunar and Mars missions, the use of in situ propellants promises to offer substantial benefits.

REFERENCES

1a. Teeter, R.R., and Crabb, T.M., "Lunar Surface Base Propulsion System Study," NAS9-17468, February, 1987.

1b. Teeter, R.R., Crabb, T.M., Rice, E.E., and Johnson, L.R., "OTV Technology for Lunar Missions," AIAA-87-2072, June, 1987.

2. Wickman, J.H., Oberth, A.E., and Mockenhaupt, J.D., "Lunar Base Spacecraft Propulsion with Lunar Propellants," AIAA-86-1763, June, 1986.

3a. Repic, E., Jones, R., Skruch, G., and Kehrbaum, J., "If We're Going to Mars, Why Stop at the Moon?" in The Case for Mars IV, June, 1990.

3b. Rockwell International, "To the Moon and Beyond: Lunar/Mars Exploration," Independent Research and Development Project 90220, July, 1990.

4a. Stancati, M.L., Niehoff, J.C., Wells, W.C., Feingold, H., and Ash, R.L., "In Situ Propellant Production for Improved Sample Return Mission Performance," AAS 79-177, 1979.

4b. Stancati, M.L., Niehoff, J.C., and Wells, W.C., "Mars Automated Refueling System Final Report and Presentation," SAI 1-120-196-S7 (JPL Contract No. 955096), October, 1978.

5. French, J.R., "Rocket Propellants from Martian Resources," Journal of the British Interplanetary Society, Vol. 42, pp. 167-70, 1989.

6a. Zubrin, R.M. and Baker, D.A., "Humans to Mars in 1999!," in Aerospace America, August, 1990.

6b. Zubrin, R.M., Baker, D.A., and Gwynne, O., "Mars Direct: A Coherent Architecture for the Space Exploration Initiative," AIAA 91-2333, June, 1991.

6c. Baker, D.A., and Zubrin, R.M., "Mars Direct: Combining Near-Term Technologies to Achieve a Two Launch Manned Mars Mission," Journal of British Interplanetary Society, November, 1990.

7. Giudici, B., "A Get Started Approach for Resource Processing," AAS 87-262, in The Case For Mars III: Strategies for Exploration - Technical, July, 1987.

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13. ABSTRACT ***(Maximum 200 words)***

The use of propellants manufactured from indigenous space materials has the potential to significantly reduce the amount of mass required to be launched from the Earth's surface. The extent of the leverage, however, along with the cost for developing the infrastructure necessary to support such a process, is unclear. Many mission analyses have been performed that have attempted to quantify the potential benefits of in situ propellant utilization. Because the planning of future space missions includes many unknowns, the presentation of any single study on the use of in situ propellants is often met with critics' claims of the inaccuracy of assumptions or omission of infrastructure requirements. This paper responds to the critics by presenting the results of many such mission analyses in one format. Each mission analysis summarized in this paper used different assumptions and baseline mission scenarios. The conclusion from the studies is that the use of in situ produced propellants will provide significant reductions in Earth launch requirements. This result is consistent among all of the analyses regardless of the assumptions used to obtain the quantitative results. The determination of the best propellant combination and the amount of savings will become clearer and more apparent as the technology work progresses.

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