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EVALUATION OF SOME CANDIDATE PROPULS1ON TECHNOLOGIES FOR MARS ASCENT

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A Mars ascent propulsion system trade study was conducted to determine I) what propulsion technologies allow a Mars sample return mission to be launched on a Delta III class launch vehicle, and 2) whether more exotic technologies., such as in-situ propellant production, allow major cost savings by enabling the use of a smaller launch vehicle or a direct return from the Martian surface to Earth without the need of a rendezvous in the Martian orbit. The results suggest that the mission can be accomplished using pressure-fed propulsion systems and storable propellants brought from Earth if a few key technologies are developed. No option considered allowed the use of a smaller launch vehicle or a direct return from the Martian surface to Earth.

Introduction

The primary scientific objectives of a Mars sample return (MSR) mission require the successful return of Martian rocks and soil samples to Earth where they can be examined in much more detail than is possible *in situ.* These samples could help answer the question of whether life exists or did exist on Mars. Finding evidence of life that originated independently from life on Earth would strongly suggest that it can develop anywhere in the universe given the right conditions, and that it probably has countless times. Beyond Earth, Mars is perhaps the best place in our solar system to begin this search. In short, an MSR mission could help answer one of the most fundamental questions humankind has asked itself: are we alone?

The purpose of this paper is to summarize Mars ascent propulsive options that would enable an 2004-2005 time frame MSR mission to succeed. A feasible Mars ascent system is one of the most challenging aspects in the design of an MSR mission. Options considered ranged from advanced storable propellant systems that use lightweight components, warm-gas pressurization systems, and composite propellant tanks to in-situ propellant production (ISPP) and pump-fed propulsion systems.

The investigation of Mars ascent propulsive options that is described in this paper was conducted by the Jet Propulsion Laboratory (JPL). The study began in January 1997 and was led by James French (JRF Engineering Services). A Mars Ascent workshop held in Pasadena, California in May 1997 summarized the results of this initial study. A follow-on study was performed by JPL using the inputs and conclusions drawn from the Mars Ascent workshop attendees. This paper represents a summary of the results of all of these studies.

Scope and Overview of the Study

This paper will attempt to focus on assumptions, results, and analyses which are key to making critical technology decisions that enable an MSR mission to succeed. Many of the technologies that are represented in this study are either new or immature. The new and/or immature technologies that are deemed critical for an MSR mission to succeed will require a significant investment in time and resources in order to reach technological maturity by 2004. This study has therefore attempted to compare the performance of propulsive options which represent similar levels of developmental risk.

The investigation into feasible Mars ascent system options was subject to two major assumptions. The first assumption was the use of a Boeing Delta III or Lockheed-Martin Atlas 2AR launch vehicle. This assumption resulted in a total injected mass limit of between approximately 2050 and 2310 kg which can be launched from Earth while satisfying the initial

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hyperbolic excess velocity requirements (-17.3 km^2/sec^2 and -9.0 km^2/sec^2, respectively) ofa range of possible MSR missions.' It should be stressed that the primary objective in evaluating ascent systems was minimizing the total injected mass from Earth and not necessarily in minimizing the ascent system mass.

The launch vehicle assumption also affected the shape of the ascent system due to the payload envelope constraints of the Delta III and Atlas 2AR. The JPL Mars Exploration Program Office indicated that the ascent system envelope could be an "inverted bowl" no larger than 1.2 m upper radius, 1.6 m lower radius, and 1 m high. Figure 1 illustrates the Mars ascent system envelope assumed in this study.

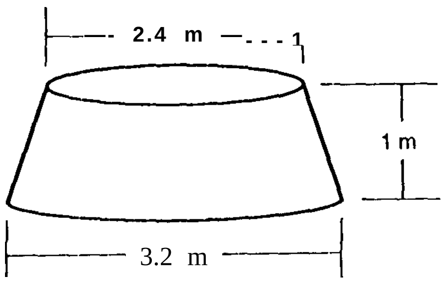

Figure 1. Mars Ascent System Envelope Assumed in this Study.

The wide and flat envelope constraint illustrated in Figure 1 had a significant impact on the design of the Mars ascent system. Instead of a "typical" two-stage launch system in which the second stage is on top of the first stage, the second stage on the Mars ascent system designed in this study was placed inside the first stage.

The second major assumption of this study was a $500 million (U.S. dollars) life cycle cost for the mission including launch vehicle. This cost constraint ruled out expensive launch vehicles such as the Lockheed-Martin Titan IV (~$350M) and forced a careful investigation into which technologies had the largest impact-to-dollar ratio in the overall design of an MSR mission. Since the cost of several of the technologies that were examined in this study were either unknown or not well known, the risk of technologies needed to be addressed. The risk level assumed for several of the technologies described in this paper represents the judgment of the authors.

Possible Mars Sample Return Mission Profile

Under the current NASA roadmap for the exploration of Mars, (wo separate spacecraft, an orbiter and a lander, are to be launched to Mars every twenty-six months.[2] The first pair of spacecraft, Mars Global Surveyor and Mars Pathfinder, was launched in November and December 1996, respectively. Another orbiter and lander will be launched in each of the 1999, 2001, and 2003 launch opportunities culminating with an MSR mission in 2005. Several mission timelines and configuration variations are possible for an MSR mission. The following section describes one possible MSR mission profile and configuration for the 2005 time frame launch opportunity. The MSR mission subsequently described actually launches in 2004 on a Type IV trajectory and returns a sample to Earth on a Type 11 trajectory in 2008. This mission profile assumes that all propellants required by the Mars ascent system are brought from Earth and no in-situ propellant production is necessary. Missions which use ISPP tend to favor the use of the more energetic Type I trajectory to Mars which allows a longer stay time on Mars to manufacture the required propellants.

Sunday November 14, 2004

As dusk settles on Kennedy Space Center in Florida, the MSR mission begins with a launch onboard a Boeing Delta III or Lockheed-Martin Atlas 2AR launch vehicle, Within the launch vehicle payload shroud arc a Mars Ascent System (MAS), ground system, and a lander. The MAS, ground system, and lander are encased in an aeroshell. Within the same launch vehicle payload shroud, stacked above or below the aeroshell, is an Earth Return Vehicle (ERV).

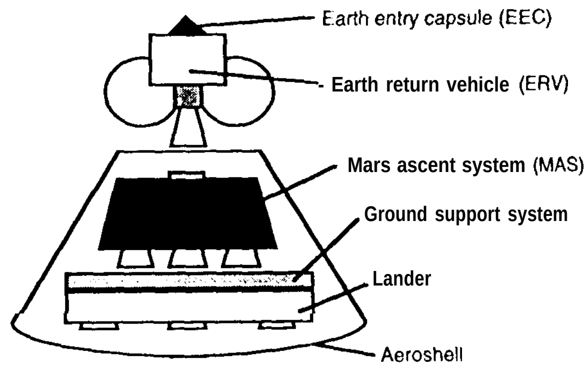

Figure 2. Mars Sample Return Schematic Configuration of Spacecraft Elements.

Tuesday February 6, 2007

Twenty-seven months after launch from Earth and over one and half orbits about the sun, the aeroshell and ERV arrive in the vicinity of Mars. The aeroshell (with the lander, ground system, and MAS inside) separates from the ERV and enters the Martian atmosphere at a speed of several kilometers per second, The aeroshell absorbs the heat generated by the atmospheric entry, slowing the aeroshell "package" down to a few hundred meters per second. The aeroshell is discarded and parachutes are deployed from the lander to further slow the lander, MAS, and ground system it is carrying. Guided by a beacon on the surface, the lander will effect a controlled propulsive landing within a few hundred meters of where one of the two previous Mars missions set down years earlier (Mars '01 or '03 lander). Meanwhile the ERV goes into a highly elliptical orbit about Mars. The ERV will circularize its orbit over the following fortnight through successive aerobraking maneuvers.

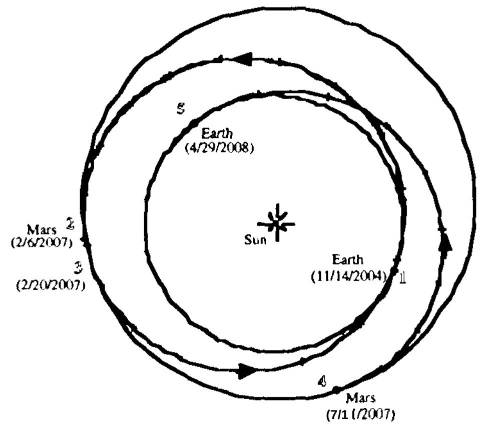

Figure 3. Trajectory Profile of a Potential Mars Sample Return Mission.

Tuesday February 20, 2007

During a two week stay on Mars, a sample of rocks, soil, and atmosphere is obtained from a cache stored by the '01/'03 mission rover. Once a sample is successfully obtained, the MAS launches from the surface and enters a 240 km circular orbit about the planet. An autonomous rendezvous occurs between the MAS and ERV in this 240 km circular orbit. The sample onboard the MAS is aseptically transferred over to an entry capsule onboard the ERV. After a successful transfer of the sample, the MAS separates from the ERV. The ERV remains in a low-Mars orbit while the MAS eventually falls back to Martian surface.

Saturday July 21, 2007

After five months in orbit about Mars, the ERV (with the sample contained in the entry capsule) begins its return trip to Earth using a Type II trajectory.

Tuesday April 29, 2008

Nine months after leaving Mars orbit and nearly three and half years after leaving Earth, the entry capsule carrying the precious samples from Mars separates from the ERV and lands at the Utah Test and Tracking Range, just southwest of the Great Salt Lake, in the United States. The entry capsule is brought to a quarantined facility where the samples brought from Mars are thoroughly investigated.

Ground Rules and Assumptions

The following section outlines the ground rules and assumptions that defined this study. Ground rules and assumptions concerning the Mars orbital rendezvous configuration are discussed first. A description of ground system assumptions follows. The section ends with an overview of potential trajectories that were investigated for a 2004/2005 time-frame MSR mission.

Mars Orbital Rendezvous Configuration Ground Rules and Assumptions

The MSR configuration assumed in this study was illustrated schematically in Figure 2. The MSR mission is comprised of five separate elements: the Mars Ascent System (MAS), the ground system, the Mars lander, the aeroshell, and the Earth Return Vehicle (ERV). This section describes each of these elements and concludes with a procedure for deriving the injected mass.

Mars Ascent System

The MAS was assumed to be a two-stage to low-Mars orbit (240 km circular) system. The original study of MAS options quickly determined that a single-stage to low-Mars orbit and a three-stage direct return to Earth configuration were not possible with the launch vehicle constraint. Both the single-stage to orbit and three-stage direct return to Earth configurations resulted in a much higher injected mass total than a Delta III or Atlas 2AR launch vehicle can provide. Although the two-stage to low-Mars orbit configuration adds the complexity of an autonomous rendezvous between the MAS and the ERV, it does significantly reduce the injected mass requirement and the possibility of back-contamination of Earth by Mars. The two-stage configuration assumed 2.15 km/s change in velocity ("Delta-V") requirement from each stage. Preliminary ascent trajectory calculations performed had indicated that a total of 4.3 km/s free-space equivalent change in velocity was required to achieve a low-Mars orbit. This 4.3 km/s value includes gravitational and drag losses, provided the ascent system has an initial thrust-to-weight ratio of approximately 2.5.

The JPL Mars Exploration Program Office estimated that the MAS payload would be 30 kg of ascent avionics, the sample, and the sample containment canister. System level estimates were made for the mass of structure, cabling, separation mechanisms, thermal control, and the thermal fairing.

The structural mass of the ascent system was assumed to be the trusses, links, and bolts that would hold the elements of the MAS together. Based on historical precedent, the structural mass of a given stage of the ascent system was assumed to be 5% of the total mass that stage had to support. For the first stage, this value was 5% of the sum of the wet mass of the first stage, the wet mass of the second stage, and the 30 kg payload. For the second stage, the structural mass was 59'. of the sum of the wet mass of the second stage and the 30 kg payload. An actual preliminary structural design and mass calculation was performed and indicated that this 50/. estimate was satisfactory for preliminary design purposes. The electronic and mechanical cabling mass of a given stage was assumed to be 10/. of the wet mass of that individual stage, again based on historical precedent.

Separation mechanisms were assumed for both stages of the ascent system. In order to minimize the overall mass of the ascent system, the bulk of the separation mechanisms between the first stage and the lander are located on the lander and the bulk of the separation mechanisms between the second stage and the first stage are located on the first stage. A flat 12.8 kg was estimated for the separation mechanisms on the first stage while a flat 2.4 kg was estimated for separation mechanisms on the second stage. Prior to the ascent, the MAS will remain securely held on top of the lander by these separation mechanisms. The thermal control mass of a given stage was assumed to be 1.3'% of the wet mass of that individual stage. This thermal control mass estimate is assumed to include insulation required by individual component elements but does not allow for special requirements such as storage of cryogens.

Finally, a 36 kg thermal fairing was assumed on the first stage. Preliminary est imates of heating loads and consideration of shock impingement indicated that a thermal fairing was necessary for a successful ascent. This 36 kg estimate does not include the mass of the insulation required to maintain thermal equilibrium within the MAS during it's surface stay on Mars. This insulation, which encases the thermal fairing, is jettisoned prior to ascent and is included in the ground system mass.

Ground System

The ground system is comprised of the equipment necessary to support the MAS prior to liftoff. Thermal control, power, and refrigeration systems are examples of what was assumed to constitute the ground system. The ground system mass is a function of the thermal control requirements of the propellants as well as whether or not the propellant are to be brought from Earth or produced in-situ on Mars. The actual mass estimates for the ground system of the ascent system are discussed in a subsequent section.

Lander

The purpose of the lander is to softly land the MAS within a few hundred meters of the Mars '01/'03 mission landing site. The lander will be directed to the correct location by a beacon onboard the 'O 1/'03 lander. The lander will deploy parachutes to slow its descent and utilize a monopropellant propulsion system for control during terminal descent. A rover onboard the Mars 'O 1/'03 mission will have collected a sample of Martian soil, rocks, and atmosphere in a cache. The MSR lander will have its own rover that will collect this sample cache and return it to the MAS. In the event the MSR mission does not reach the correct landing location or the 'O 1 /'03 rover was unable to collect a cache, the MSR rover will be able to collect a sample at the MSR mission landing site. The mass of the lander was assumed to be 0.316 times the mass of the total delivered mass (ascent system and ground system). The mass of the landing propellant was assumed to be 0.105 times the landed mass (ascent system, ground system, and lander). These numbers are typical of the results of past MSR mission studies.

Aeroshell

The purpose of the aeroshell is to slow down and protect the aeroshell payload (ascent system, ground system, and lander) during the high velocity entry into the Martian atmosphere. It is assumed that Mars Pathfinder technology will be applied to the design and construction of the aeroshell for this MSR mission. The mass of the aeroshell was assumed to be 0.215 times the mass of the aeroshell payload, based on Mars Pathfinder aeroshell-to-pay load performance.

Earth Return Vehicle

The purpose of the Earth Return Vehicle (ERV) is to aseptically obtain the sample containment canister (with the Martian rock, soil, and atmosphere sample inside) from the ascent system and return it successfully to Earth. The ERV is sent to Mars in the same launch vehicle as the aeroshell and aeroshell payload (see Figure 2). Unlike the aeroshell and aeroshell payload, the ERV does not descend to the Martian surface. The ERV inserts itself into a highly elliptical orbit about Mars and aerobrakes down to a low-circular orbit over a period of several weeks. Once the MAS has rendezvoused with the ERV and the sample containment canister is transferred, the ERV and MAS separate. The sample containment canister is placed in an Earth Entry Capsule (EEC). After several weeks in orbit around Mars, the ERV injects back to Earth. Upon arrival in the vicinity of Earth, the EEC is released and enters the Earth's atmosphere, landing at a quarantined facility. Meanwhile, the ERV continues on an altered trajectory towards a solar orbit. The wet mass of the ERV (and EEC) was assumed to be a flat 770 kg regardless of the type of ascent system, based on a preliminary design concept.

Resulting Injected Mass Derivation

The total injected mass from Earth can be derived from the ascent system mass through the assumptions stated in the previous paragraphs and the ground system masses discussed subsequently. The total injected mass from Earth is the value that determines whether a given MSR mission can be launched on a Delta II 1 or Atlas 2AR launch vehicle. Assuming a 644 kg MAS with a 94 kg ground system mass, the total delivered is 738 kg. The lander would have a mass of 233 kg (0.316 times the delivered mass). The mass of the landing propellant would be 102 kg (O. 105 times the landed mass). The aeroshell would have a mass of 231 kg (0.215 times the aeroshell payload). The mass of the Earth Return Vehicle (ERV) would be a flat 770 kg. The mass of the propellant for midcourse trajectory corrections by the ERV would be 83 kg (0.04 times the sum of the aeroshell payload and the ERV). Hence, the total injected mass from Earth would be 2157 kg (644 +94+233 + 102 + 231 + 770+ 83). In this case, the total injected mass is just over the 2050 kg launch vehicle limit for a Type 1 trajectory but under the 2310 kg launch vehicle limit for a Type IV trajectory.

Ground System Mass Assumptions

As stated in the introduction, the purpose of this study is to investigate Mars ascent propulsive options that would enable an affordable 2004-2005 time frame MSR mission. Essentially two missions scenarios were studied. The first mission scenario assumed that all the propellants required by the MAS for ascent were brought from Earth. This scenario is hereafter referred to as the "bring your own propellant" (BYOP) option. The second mission scenario assumed that some or all of the propellants required by the MAS for ascent were produced in-situ through the acquisition, compression, conversion, and liquefaction of the Martian atmosphere. This scenario is hereafter referred to as the "in-situ propellant production" (ISPP) option. Both Earth-storable and cryogenic temperature propellants were investigated. The ground portion of an ascent system is a function of the propellants used and whether or not the mission is BYOP-based or ISPP-based. This section begins with a discussion of assumptions concerning a typical Earth-storable BYOP mission, A discussion of the assumptions concerning a typical cryogenic BYOP mission follows. The section ends with a discussion of the assumptions concerning a typical cryogenic ISPP mission.

Earth-Storable BYOP Mission Assumptions

Earth-storable propellants that were assumed in this study included the fuels hydrazine, monomethyl hydrazine, and propane and the oxidizers nitrogen tetroxide, mixed oxides of nitrogen, and chlorine pentafluoride. Hydrazine and nitrogen tetroxide have operating and storage temperatures on the order of 293 K (20 "C). Monomethyl hydrazine, propane, mixed oxides of nitrogen, and chlorine pentafluoride have operating and storage temperatures of233 K (-40 "C) or less. Since the average temperature on Mars is on the order of 190 K (-83 'C), both the conventional Earth-storable propellants and the low-temperature Earth-storable propellants require thermal control. The ground system mass was estimated to be 107 kg for systems using conventional Earth-storables as one or both of the propellants. The ground system mass was estimated to be 94 kg for systems using low-temperature Earth-storables as both propellants. These two values were based on a fixed insulation mass and varying power requirements.

Cryogenic BYOP Mission Assumptions

Cryogenic BYOP propellants assumed in this study were liquid oxygen, liquid fluorine, liquid methane, and liquid carbon monoxide. All four of these propellants have operating and maintenance temperatures below I I 3 K (- 160 ‘C). The power system mass (m_{ps}) for BYOP cryogenic propellants (m,) was estimated to be

$$m_{ps} = 107\left(\frac{m_c}{500}\right)^{2/1} \quad (1)$$

where both m_{ps} and m_c are in kilograms. The refrigeration system mass (m_{rs}) for BYOP cryogenic propellants was estimated to be

$$m_{rs} = 105\left(\frac{m_c}{500}\right)^{2/3} \quad (2)$$

where m_{rs} is in kilograms. The total ground system mass for a cryogenic BYOP system is the sum of the power system mass and refrigeration system mass (sum of equations(1) and (2)).

Cryogenic ISPP Mission Assureptions

In-situ propellant production (ISPP) is an approach for converting the carbon dioxide that makes up approximately 95% of the Martian atmosphere to useable propellants, thus reducing the load of propellants that must be brought from Earth. The two most developed concepts for utilizing Martian carbon dioxide are the Sabatier/Electrolysis (WE) process and the Zirconia solid state electrolyte process. Although the S/E process is somewhat more mature than the Zirconia process, S/E requires that hydrogen be brought from Earth. Due to the enormous challenges of transporting hydrogen to Mars, only the Zirconia process was modeled in this study. A Zirconia ISPP system acquires and compresses the carbon dioxide in the Martian atmosphere and converts it to oxygen and carbon monoxide in a reactor. The oxygen is liquefied and stored while the carbon monoxide is either liquefied and stored also or discarded depending on the propellant combination used by the ascent system. An ISPP system for an MSR mission must be completely autonomous and requires a significant power source, photovoltaic or otherwise. Only photovoltaic systems were considered in this study.

The power required to run the Zirconia ISPP system (P_{ispp}) was estimated to be

$$P_{ispp} = 100 + 1050x \quad (3)$$

where x is the amount of cryogenic ISPP propellant to be produced (oxygen and/or carbon monoxide) in kg per day based on a 500 day stay time. P_{ispp} is measured in Watts. The liquefaction and storage power (P_{lm}) was estimated to be

$$P_{lm} = 175x^{2/3} + 526x \quad (4)$$

where P_{lm} is measured in Watts. The total ground system power was estimated to be

$$P_{gs} = P_{ispp} + P_{lm} + 100 \quad (5)$$

where P_{gs} is measured in Watts. Equations (6) and (7) describe the ISPP system (m_{ispp}) and liquefaction and storage mass (m_{lm}) in kilograms, respectively.

$$m_{ispp} = 16 + 33x \quad (6)$$

$$m_{lm} = 105x^{2/3} \quad (7)$$

The power system mass (m_{ps}) is assumed to be

$$m_{ps} = 0.0975P_{gs} \quad (8)$$

where m_{ps} is in kilograms. Finally, the total ground system mass for an ISPP cryogenic system (m_{total}) is assumed to be

$$m_{total} = m_{ispp} + m_{lm} + m_{ps} \quad (9)$$

where m_{total} is in kilograms.

Possible MSR Missions and Trajectories

The Mars ’01 and ’03 landers will have rovers that will collect and cache high-quality samples. The first MSR mission will go to one of those two sites, the one with the scientifically best sample, to retrieve and return the sample. Two sample collection missions are important to the program strategy both for science and engineering redundancy. The first MSR mission is to launch in the 2004/2005 opportunity, and return in 2008. Tables 1 and 2 list all the low-energy, non-gravity-assist trajectories available in this time-frame.

Table 1. List of Investigated Low-Energy Non-Gravity Assist Earth-Mars Outbound Trajectories.

Type	Earth (departure)	Mars (arrival)	C_3 (km/s)	Ins ΔV (km/s)
IV	1 1/14/2004	216/2007	9.0	1.23
III	12/21/2004	12/18/2006	11.9	2.44
II	8/7/2005	7/1 9/2006	17.9	1.26
I	8/20/2005	3/26/2006	17.8	1.12

“Type” is the numbering of the local energy minima in order of trip time. The trajectories are listed in order of increasing departure date, which is also decreasing arrival date and decreasing trip time. C_3 is the square of hyperbolic excess velocity in km^2/s^2 (optimizing the combination of injection C_3 and insertion ΔV for the best date-definition ofa launch period would increase the C_3 somewhat). Ins AV is the required change in velocity to insert to an 8-hour elliptical Mars orbit in km/s.

Table 2. List *of Investiqated Low-Energy Non-Gravity Assist Mars-Earth Inbound Trajectories.*

Type	Mars (departure>	Earth (arrival)	Dep ΔV (km/s)
IV	4/8/2006	916/2008	1.98
III	4/14/2006	6122/2008	2.00
II	7/21/2007	4/29/2008	2.38
I	7/30/2007	2129/2008	2.72

Dep ΔV is the required change in velocity in km/s to depart from a low-Mars orbit and inject back to Earth.

An Earth-to-Mars trajectory is combined with a later Mars-to-Earth trajectory for a candidate sample return mission trajectory set. The best mission performance will be for combinations that minimize the launch C_3, the insertion ΔV for the return orbiter, and the departure AV for the return orbiter. These considerations eliminate the outbound Type 111 in favor of the outbound Type IV, the outbound Type II in favor of the outbound type I, and the inbound Type I in favor of the inbound type 1[.

There is a great deal to do between Mars arrival and Mars departure, including aerobraking to a low orbit, fetching the sample, transferring the sample to the ascent system, ascent and orbital rendezvous, another sample transfer to the orbiter, and departure from Mars orbit. The time to do all of this, under the best of circumstances, is estimated to be on the order of five weeks. If in addition propellant must be manufactured on the surface, then much more time is needed, 500 days was assumed to be required for propellant production in this study.

This gives the Type IV outbound and Type 11 inbound as the most favorable BYOP option, with over five months at Mars to complete the operations. For ISPP, the best option is the Type I outbound and the Type II inbound with 16 months at Mars. Another combination that almost works for a BYOP mission is the Type I outbound and the Type III or IV inbound. However, those both leave less than five weeks at Mars. If we take the slightly later Type III outbound and delay it with some cost in departure AV, a workable combination is possible. This Type III * trajectory departs from Mars on 5/3/2006 and arrives at Earth 6/19/2008 with a departure ΔV of only 2.03 km/s. Much more delay results in a rapidly increasing penalties.

This adds another BYOP option: the Type I outbound and the Type III* inbound. [n comparing this option with (he Type IV outbound and Type II inbound, it is apparent that there are competing factors. The Type IV C_3 is less than the Type I, but the orbit insertion and departure AVS for that case are greater than for the Type I departure. Interestingly, these competing factors nearly cancel in evaluating the injection capability of the launch vehicle and the mass of the return orbiter, so that the two BYOP options are nearly equal in performance. However the Type I/Type III* option has far less time at Mars, leaving the Type IV/Type II as the preferable BYOP option. It would be possible to wait for a lower velocity return opportunity one year later with the return in mid to late 2010, but that does not fit the programmatic constraints.

In comparing the best BYOP option with the best ISPP option, we see that the ISPP option is penalized with a higher injection C_3, and so lower mass available for the system. This is a characteristic of this particular opportunity. Later opportunities have lower penalties for *long* stay times, though those stay times are on the order of 12 to 14 months and would incur additional mass for significantly faster propellant production.

Propulsion System Descriptions and Key Assumptions

System Configurations

The configurations assumed for Mars ascent propulsion systems in this study were made as consistent as possible between different propulsion concepts to try to provide a level playing field for comparison of different propellant combinations and feed system concepts. In all cases, the main engine masses were scaled to provide an initial thrust-to-Mars weight of approximately 2.5 at liftoff. All systems used two oxidizer tanks and two fuel tanks on each of the two stages to allow the center-of-mass of the system to be maintained on the centerline of the system during ascent. The first stage was assumed to use four main engines (partially off-modulated to effect pitch and yaw control) and two roll control thrusters, while the second stages were assumed to use one main engine and four thrusters canted so as to permit pitch, yaw, and roll control by on-pulsing of the thrusters. The main engines on each stage were assumed to be of the same design and thrust level, in order to reduce development cost.

For cases in which one or both of the propellants was cryogenic, it was assumed that separate pressurant supplies would be used for the fuel and oxidizer, while for Earth-storable propellants a common pressurant system was assumed. This was motivated by the fact that pressurant tanks integral with the cryogenic propellant tanks by providing intimate thermal contact between the pressurant and the propellant are more efficient. This is generally not desirable for Earth-storable propellants. It is also not desirable in the case of cryogenic propellants (liquid oxygen for example) which are to be produced on the Martian surface, since this would require active cooling of these pressurant tanks during the Earth-Mars transit.

The first-stage schematics shown in Figures 4 and 5 illustrate these two types of pressurizat ion system as well as other features of the propulsion system design common to all options **studied** (except the hybrid system described subsequently).

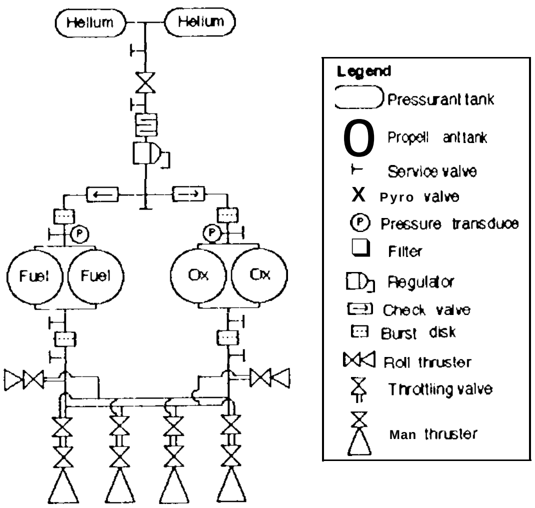

Figure 4. Typical Mars Ascent Propulsion System Schematic for Earth-Storable Propellants.

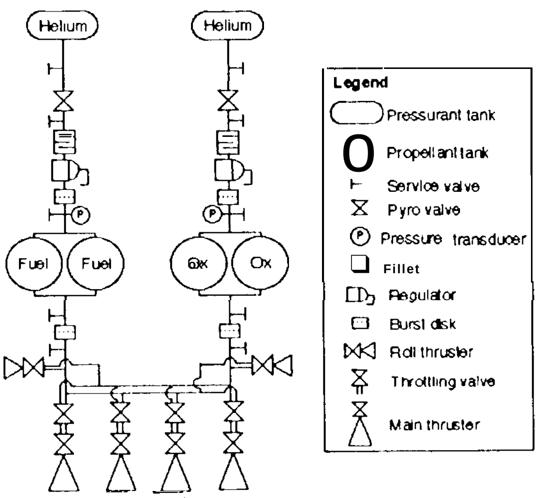

*Figure 5. **Typical** Mars Ascent Propulsion System Schematic for Cryogenic Propellants.*

The pressurant tanks are isolated from the pressurization system(s) by normally closed pyrotechnic valves, while the propellant tanks are isolated from the pressurization systems and the liquid propellant feed systems by burst disks. Opening of the pyrotechnic valve will result in rupture of the burst disks, leading to pressurization of the propellant tanks and priming of the propellant feed lines. Service valves are provided where necessary for servicing or functional testing of the system. Throttling valves are provided for partially off-modulating the first stage main engines by limiting propellant flow to what can pass through a calibrated orifice. Pump-fed systems were qualitatively similar, except that the engine flow control might have to be downstream of the pump assembly. All pressure-fed systems assumed a propellant tank operating pressure of 300 psia, while pump-fed options assumed the propellant tanks were maintained at 50 psia in order to prevent cavitation in the pumps. It has been pointed out that a viable propulsion system using a cryogenic propellant also requires provisions for venting and/or cryocooling which were not included in these schematics; this is particularly true for cryogenic propellants to be brought from Earth. Therefore, it is likely that the present study may overestimate the performance of cryogenic systems to some degree.

Nine propellant combinations were considered in this study, and some had unique technology assumptions associated with them. The candidate propellant systems considered in this paper are:

1. *Baseline storable:* this system used the conventional storable propellant combination of nitrogen tetroxide (NTO) and monomethyl hydrazine (MMH). It also used component masses based on existing off-the shelf hardware designs and used main engine mass and performance data based on slight modifications of the existing Kaiser Marquardt R-40B rocket engine.

2. *Advanced storable:* this system used the storable propellant combination mixed oxides of Nitrogen-25 (MON-25) and MMH to enable low storage and operational temperatures (down to -40 "C), thus reducing power requirements during surface operations. In addition, the pressure-fed version of this system assumed the use of lightweight composite over-wrapped Pressure vessel (COPV) technology for the propellant tanks and of a warm-gas pressurization system. In the warm gas pressurization system small amounts of hydrogen and oxygen (at stoichiometric ratios) are added to the helium pressurant gas; the gas mixture is reacted in a catalyst bed located downstream of the pressure regulator. This reduced pressurant and storage mass

by approximately 30% compared to the baseline storable case.

3. LO_2/C_3H_8: This is a propellant combination which has received a great deal of attention for use in a Zirconia-cell-based ISPP mission because the propane (which would be brought from earth) is not a cryogen and would not require the highly insulated cryogenic tanks needed for the storage of liquid oxygen or for cryogenic fuels such as methane. Propane is also denser than methane, reducing tank volume and mass.

4. LO_2/N_2H_4: This combination was considered because there is currently work ongoing at two rocket engine companies to develop engines using these propellants and because the hydrazine fuel is even denser than the propane discussed above. One negative attribute is the relatively high (-275 K) freezing point of hydrazine.

5. LO_2/CH_4: Liquid oxygen and methane are logical propellant choices for ISPP missions based on the Sabatier/Electrolysis (WE) process, since both propellants can be produced *on* Mars. As discussed previously, this process was not explicitly dealt with in this study because the S/E process requires transporting a significant amount of hydrogen from Earth, which introduces enormous thermal control challenges and significant packaging concerns. However, this study did consider the scenario in which the methane was brought from Earth for either BYOP or ISPP mission scenarios. This combination was impacted by the fact that the methane is a low-density cryogen, requiring large insulated tanks.

6. LO_2/CO: This propellant combination has the merit that both the oxidizer and fuel can in principle be manufactured from the Martian atmosphere using a Zirconia-cell-based ISPP process. Unfortunately, it is penalized by low performance and the fact that both propellants are cryogenic, requiring heavily insulated propellant tanks. In addition, at pressures of interest as a propellant, carbon monoxide has a liquid temperature range of only about 6 °C. While this was not specifically addressed in this study, it would likely greatly complicate the thermal design of an ascent system using these propellants.

7. ClF_5/N_2H_4: This propellant combination is Earth-storable, and offers both higher specific impulse and higher density than conventional storable propellants. It does suffer from the high freezing point of hydrazine, and there is no current U.S. source of production for the oxidizer, but because of it's consideration for SDI systems in the 1980s there is some technology base on which to build.

8. LF_2/N_2H_4: This combination was actively studied until the early 1980s. It provides extremely high specific impulse combined with high density, but suffers from significant safety concerns related to the extreme reactivity of the fluorine oxidizer. The requirement for cryogenic insulation on the oxidizer tank also reduced system performance in this application below that which could be obtained in a deep-space application.

9. $LO_2/HTPB/Al$: One option considered was the use ofa hybrid rocket using liquid oxygen as oxidizer and a solid fuel comprised of hydroxyl terminated polybutadeine (HTPB) with a 16% aluminum loading. To meet the envelope requirements of the Mars ascent propulsion system, it would be necessary to develop a short hybrid rocket motor; it is possible such a development could take the form of the radial flow design.' A blowdown monopropellant hydrazine system was assumed to be used for attitude control during ascent. The inert mass of the hybrid motor was assumed to be 13% of the fuel it contained, consistent with solid rocket motors in this size class.

Component masses

As discussed above, all of the systems except the baseline storable used lightweight flow control devices assumed to be based on advancements made for SDI systems in the 1980s. The mass of components developed for those systems were used as the basis of component mass estimates for a baseline system design with a main engine thrust of 500 lbf; for systems which required significantly higher or lower thrust levels, the component masses were assumed to scale with the square root of thrust level. These differing thrust level requirements were a consequence of the variation in liftoff mass between the various options considered. Table 3 lists these reference component masses.

ble 3. Component Referen Masses Assume

Component	Mass (kg)
Engine throttling valve	1.20
Pressurant reactor	1.00
Roll thruster	0.55
TVC thruster	2.00
Pressure regulator	0.50
Check valve	0.02
Burst disk	0.20
Service valve	0.01
Pyrotechnic valve (NC)	0.20
Pressure transducer	0.27
Gas filter	0.20
Lines	1.10
Fittings	1.00
Primary battery	0.72

[t is worth noting that a primary battery was included on all second stages to provide a power source during ascent.

Main engine performance

With two except ions, the performance of the various propellant combinations was assessed using the rigorous JANNAF procedure.[4] For all cases, a reference main engine thrust level of500 lbf and a parabolic wall profile were assumed; no effort was made to optimize the nozzle design for each propellant combination. One-dimensional performance calculations incorporating chemical kinetic effects were used to determine the optimum mixture ratio for each propellant combination. The performance of the LO_2/CO combination was evaluated by Ms. Diane Linne of Lewis Research Center using the same methodology. In all cases, the theoretical performance (including kinetic, two-dimensional, and boundary layer losses) were reduced by 2% to account for an assumed 98'% combustion efficiency (i.e., vaporization and mixing efficiency). It should be remembered that there may be practical limitations imposed by chamber cooling or two-phase flow effects which will prevent the performances computed using this procedure from being attained in a practical rocket engine design. However, this procedure should allow reasonable comparison between propellant combinations for a system study such as this one.

Pressure-fed systems assumed a combustion chamber pressure of 200 psia and a nozzle area ratio of 100:1. This area ratio was about the maximum which could be configured within the envelope constraints of the Mars ascent system and resulted in near-ideal expansion to the Mars surface atmospheric pressure of approximately 8 mbar. For pump-fed systems, the chamber pressure was increased to 1000 psia and the nozzle area ratio was increased to 250:1. The nozzle length was constrained to be approximately the same in all cases.. All pump-fed cases were assumed to use a gas generator cycle to drive the pumps. For the "conventional" pump technology (described in the next section), it was assumed that the effective specific impulse was reduced an additional 2% by this gas generator flow; for the "advanced" pump technologies, it was assumed that the effective specific impulse was reduced by 5% because these pumps, while potentially very lightweight, are not terribly efficient.

The exceptions to this performance assessment procedure were the baseline storable system and the hybrid. The mixture ratio and specific impulse values used in the baseline storable system were those that have been demonstrated in preliminary testing of upgraded versions of the Kaiser Marquardt R-40B, and are lower than would be produced by the computational procedure used for other cases due to limits on combustion chamber temperature. The performance used for the hybrid was 90% of the theoretical ideal performance from a one-dimensional calculation which assumed perfect chemical equilibrium during expansion; the performance of an actual hybrid would likely be lower than this due to two-phase flow losses in the nozzle.

Table 4 provides the optimum mixture ratio, specific impulses (lbf•sec/lbm), and average propellant density (kg/m^3) for each of the cases analyzed,

Table 4. Performance of Propellant Combinations.

Propellants	Notes	MR	I_{sp}	p
NTO/MMH	pressure-fed	1.65	307	1151
	pumps (c)	n/c		
	pumps (a)	n/c		
MON-25/MMH	pressure-fed	2.1	323	1149
	pumps (c)	2.3	336	1160
	pumps (a)	2.3	326	1160
LO_2/C_3H_8	pressure-fed	3.0	338	923
	pumps (c)	3.0	359	923
	pumps (a)	3.0	348	923
LO_2/N_2H_4	pressure-fed	0.8	346	1064
	pumps (c)	n/c		
	pumps (a)	n/c		
LO_2/CH_4	pressure-fed	3.1	346	811
	pumps (c)	2.9	359	799
	pumps (a)	2.9	348	799
LO_2/CO	pressure-fed	0.55	252	891
	pumps (c)	0.55	269	891
	pumps (a)	0.55	261	891
ClF_5/N_2H_4	pressure-fed	2.7	332	1472
	pumps (c)	n/c		
	pumps (a)	n/c		
LF_2/N_2H_4	pressure-fed	1.9	384	1284
	pumps (c)	n/c		
	pumps (a)	n/c		
LO_2/HTPB/Al	pressure-fed	1.08	328	—
	pumps (c)	n/c		
	pumps (a)	O/c		

pumps: c = conventional pumps, a = advanced pumps

Notes that the rows with n/c listed signify that these cases were not considered in this study. The average propellant densities are significant in that propellants having lower density will require larger, heavier tanks and pressurization systems. Therefore, it is possible for a propellant combination which delivers high specific impulse to have poor system performance if it has low average density. This is particularly true for pressure-fed cryogenic systems due to the low temperature of the pressurant gas in the propellant tank ullage.

Main engine mass

To estimate main engine mass at the thrust level appropriate to each case studied, it was necessary to scale engine mass with thrust level for pump-fed engines, "conventional" pump-fed engines, and "advanced" pump-fed engines. Jim Glass of the Rocketdyne Division of Boeing North American was able to provide us with a self-consistent set of engine masses for pressure-fed and "conventional" pump-fed engines. These were used in this study, with the understanding that they are very approximate, but would provide a common basis for comparison of the different vehicle concepts. The "advanced' pump-fed engines were assumed to be based on positive-displacement pump concepts.[5] The pump and gas generator masses were assumed to be 1.1 % of the engine thrust, a figure provided to us by Dr. John Whitehead of Lawrence Livermore National Laboratory. This mass was added to the mass of a conventional pressure-fed engine with a 1000 psia chamber pressure and a nozzle area ratio of 250:1, as given by the Rocketdyne correlations.

Figure 6 shows the engine mass scaling results.

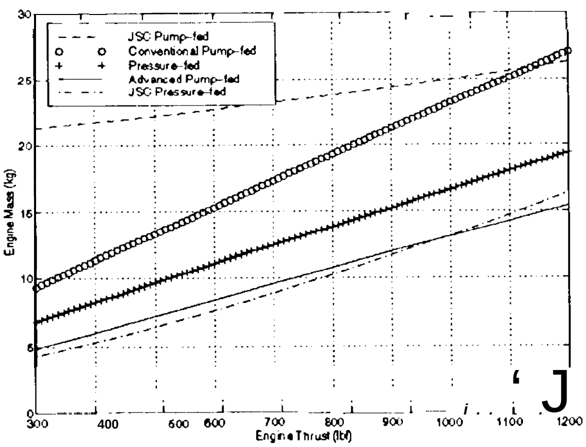

Figure 6. Engine Mass as a Function of Engine Thrust.

As might be expected, the Rocketdyne mass estimates showed an increase in engine mass when the engines were pump-fed. In large systems, this increase can easily be outweighed by savings in tank and pressurization system mass, but it was not clear this would be the case with a small system such as the Mars ascent propulsion system. There is also considerable uncertainty as to the adaptability of conventional turbopump technology to such small sizes. In contrast, the advanced pumped engine mass is actually lower than that of a pressure-fed engine because the pump and gas generator mass is more than offset by reductions in the dimensions and mass of the combustion chamber enabled by the five times higher combustion chamber pressure. The authors do not claim that the engine masses used herein are necessarily attainable in the real world, but they do provide a common basis for comparison.

For comparison, Figure 6 also shows pressure-fed and pump-fed engine masses obtained using correlations developed at Johnson Spaceflight Center (JSC); these correlations were provided to the authors by Mr. Jerry Sanders of JSC. While the JSC correlations provide somewhat lower masses for pressure-fed engines, the pump-fed cases are more massive than either of the assumptions considered in this study, particularly for low thrust levels. It is clear that there is a good deal of uncertainty in estimates of engine masses for which no definite design has been generated.

Pressurant and Provellant Tank Masses

Most of the cases considered in this study assumed COPV pressure vessels for pressurant storage with figure-of-merit (burst pressure times volume divided by Earth weight) of 1.2 x 10^6 inches. The three exceptions to this were cases where cryogenic propellants were to be brought from Earth to Mars (i.e., cases where we used liquid fluorine and liquid methane). In these cases it was assumed that the pressurant tanks were constructed of titanium so that they could be built integral to the propellant tank, reducing pressurant storage mass. Due to the lower figure-of-merit of the titanium pressure vessels (500,000 inches), the pressurant storage mass was almost indistinguishable from similar cases using a separate COPV pressurant tank.

The pressurant tanks for storable propellants were sized assuming a nearly adiabatic (polytropic coefficient of 1.5) expansion of the helium in the pressurant tank during ascent. The final propellant tank ullage gas temperatures were assumed to be the mass-averaged temperature of the gas supplied from the pressurant tank and the initial ullage gas (after adiabatic compression of the initial gas to the regulated tank pressure). For cryogenic propellants, the ullage gas temperature was assumed to be constant at the normal boiling point of the propellant. For the warm gas pressurization system, the pressurant was assumed to be warmed by 150 'C by the catalytic reaction before reaching the propellant tanks.

Propellant tank technologies were assumed to differ between storable propellants, which used conventional titanium pressure vessels, cryogenic propellants, which used advanced insulation over cryoformed stainless steel pressure vessels, and the advanced storable system, which was assumed to use an advanced COPV propel lanttank with halfofthe mass of an equivalent titanium tank. The use of stainless steel pressure vessels in place of titanium is essential for liquid oxygen; it might be possible to relax this assumption for some of the other cryogens considered. All titanium and steel pressure vessels were assumed to have a minimum design wall thickness of 0.015" and a machining tolerance of 0.003". The maximum allowable operating stresses were assumed to be 87.4 ksi and 150 ksi for the titanium and steel, respectively. The mass of insulation required for the cryogenic tanks was provided by Mr. Rick Hunter of Oceaneering Space Systems. The insulation was designed to limit liquid oxygen boiloff to between 0.1 and 0.3 kg/day as the tank size increases from approximately 0.045 to 0.22 m'. This boiloff was then reliquified by the cryogenic cooling system. The resulting propellant tank masses are shown if Figure 7.

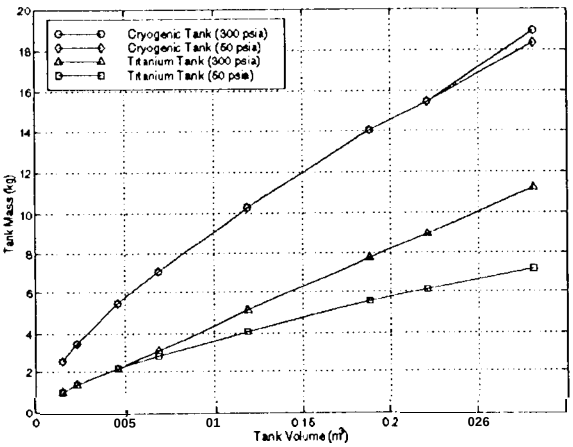

Figure 7. Propellant Tank Mass as a Function of Volume.

It is readily apparent that the cryogenic tanks tend to be heavier than the uninsulated tanks used in the storable systems. In addition, due to the higher allowable stress of the stainless steel liners used for the cryogens, the minimum design wall thickness of 0.015" severely limits the tank mass savings attainable in pump-fed systems.

Results and Discussion

Given the large number of diverse cases considered in this study, it is helpful to compare results with an eye toward the relative developmental cost and risk associated with them. For example, while it may in fact be true that an advanced pump-fed system using cryogenic propellants and lSPP may provide superior performance to a storable pressure-fed system, the costs and risks associated with the former system greatly exceed those of the later. The ordering of the following paragraphs are based on the following relative risk assessments: 1) ISPP-based systems have higher risk than BYOP systems, 2) pump-fed systems have higher risk than pressure-fed systems, and 3) "advanced" pump-fed systems have higher risk than "conventional" pump-fed systems. Within each category, system options are also presented in order of risk, based on the subjective ranking of the authors; the options are presented with increasing risk as one progresses from the left to right in Figures 8 through 13.

In addition to risk considerations, another qualitative factor which should be remembered when reviewing these results is the error margin of this study. One issue is that the error likely to be present in the results is larger for the less technically mature options; the other issue is that errors in system studies are almost always in the form of underestimation of system masses and overest imat ion of system performance. Therefore, even though 25% dry mass contingency has been used in estimating the system launch masses, it is desirable to show some level of launch vehicle margin, with the size of the desired margin growing with the uncertainty in the estimates. For a relatively mature design concept such as the pressure-fed storable systems, launch vehicle margins of as little as 5'% might be acceptable, while for an ISPP-based, pump-fed cryogenic system launch vehicle margins should probably be more like 25%. Again, these are the opinions of the authors, and we recognize that there is room for legitimate dissent. An alternative approach would be to build larger dry mass margins for the less mature concepts, but the authors feel that this subjective evaluation should not be build into the results.

To further complicate matters, the injection capability of the launch vehicle is not a fixed number, but varies with the trajectory chosen. As described earlier in this paper, the BYOP missions are able to use a Type IV trajectory to Mars with a Type II return trajectory; this yields a launch vehicle injection capability of approximately 2310 kg. ISPP-based missions, because of their requirement for a surface stay time of approximately 500 days, must use a Type I trajectory from Earth to Mars; this yields a lower launch vehicle injection capability of2050 kg.

BYOP Systems

Figure 8 compares the total injected mass requirements for options considered which use BYOP propellants in pressure-fed ascent system designs.

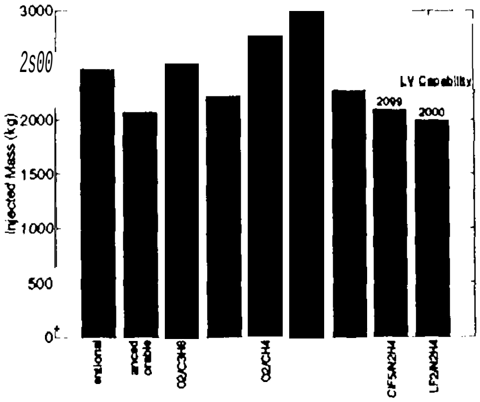

Figure 8. Injected Muss Requirements for Pressure-Fed BYOP Systems.

The best performer of the group is LF_2/N_2H_4, due to it's high specific impulse and density. However, due to it's cryogenic nature and the extreme safety concerns over the use of liquid fluorine, the authors consider it to be the highest risk option in this group. The second best performer is the advanced storable system, which is also judged to be one of the lower risk options, given that there is no requirement to store cryogens while the system is enclosed in an aeroshell during transit to Mars. The ClF_5/N_2H_4 system also shows good performance, but is also considered to be a high-risk option due to safety concerns similar to those of liquid fluorine and the absence of a production capability for the oxidizer. The LO_2/N_2H_4 and hybrid systems also appear feasible, but with reduced margins and higher risk than the advanced storable system. The systems which use liquid oxygen with lower-density liquid fuels such as methane do not appear to be feasible.

Figure 9 compares the options for which "conventional" pump-fed systems were used with propellants brought from Earth.

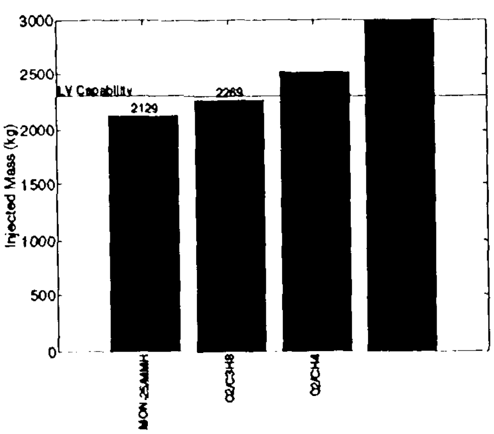

Figure 9. injected Mass Requirements for "Conventional" Pump-Fed BYOP Systems.

A number of propellant combinations, especially those using fluorinated oxidizers or hydrazine fuel, were not considered for pump-fed applications due to a risk stack-up which the authors found increasingly untenable; they could be considered if necessary. Of the options considered, only the advanced storable propellant system of MON-25/MMH and the LO_2/C_3H_8 system fell within the launch vehicle constraint. Comparison with Figure 8 shows that the additional complexity of the pump-fed system actually reduced launch vehicle margin for the MON-25/MMH; this is in part because the pump-fed option did not assume the use of lightweight COPV propellant tanks or warm gas pressurization assumed in the advanced pressure-fed system. The LO_2/C_3H_8 system did show a significant mass reduction with the use of pumps, although the launch vehicle margin is still quite low.

At the highest risk levels considered for BYOP systems, Figure 10 compares options using the "advanced' pump-fed technologies.

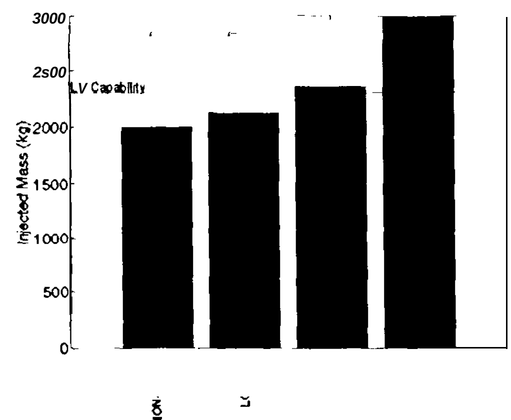

Figure 10. Injected Mass Requirements for "Advanced" Pump-Fed BYOP Systems.

The best performance is obtained with the MON-25/MMH system; this performance would be further enhanced if the advanced pump-fed system were used in conjunction with the lightweight COPV propellant tanks used in the advanced storable pressure-fed system. Among the cryogens, only LO_2/C_3H_8 fell within the launch vehicle capability. Even with the most optimistic assumptions considered for the advanced pump-fed system, the LO_2/CH_4 system does not appear to be feasible for the BYOP mission; this is primarily due to the cryogenic nature of both propellants and the very low density of liquid methane.

ISPP Systems

The lowest-risk ISPP options are the pressure-fed systems compared in Figure 11.

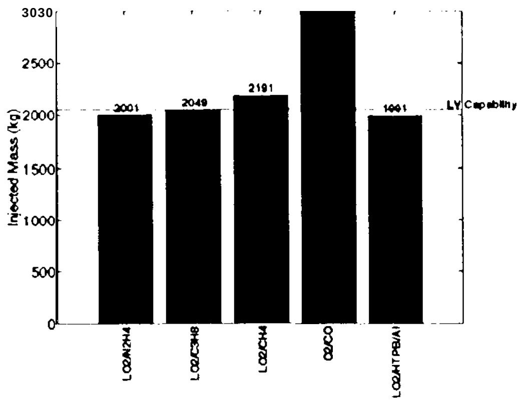

Figure 11. Injected Mass Requirements for Pressure-Fed ISPP Systems.

The best overall performance is predicted for the hybrid system, but due to the uncertainties associated with it the authors consider it to be the highest risk of these options. LO_2/N_2H_4, at the low end or the risk spectrum, is a close second and would therefore be a more reasonable choice for selection as a candidate flight System. The LO_2/C_3H_8, LO_2/CH_4, and LO_2/CO pressure-fed systems do not appear to be viable.

Figure 12 shows the results for those ISPP options for which "conventional" pump-fed engines were considered.

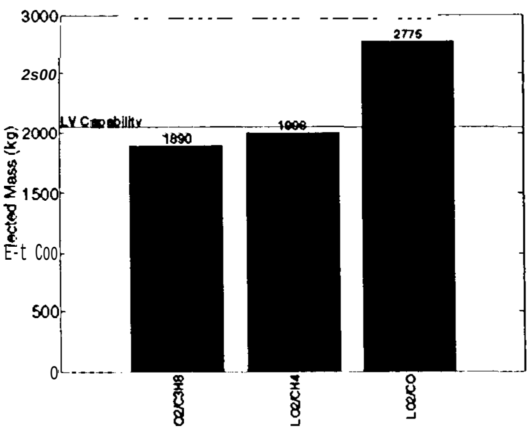

Figure 12. Injected Mass Requirements for "Conventional" Pump-Fed ISPP Systems.

By reducing the pressurant requirements by a factor of six, the masses of all of these cryogenic options were decreased significantly. Note that the LO_2/N_2H_4 propellant combination was not considered in a pump-fed system; it would presumably offer lower launch mass than LO_2/C_3H_8 if it were. This is the lowest level of technology where it appears that the LO_2/CH_4 propellant combination might be a viable candidate for an ISPP mission, and the launch vehicle margin of 2.5% hardly seems adequate given the uncertainties involved.

Finally, the results for advanced pump-fed ISPP systems are shown in Figure 13.

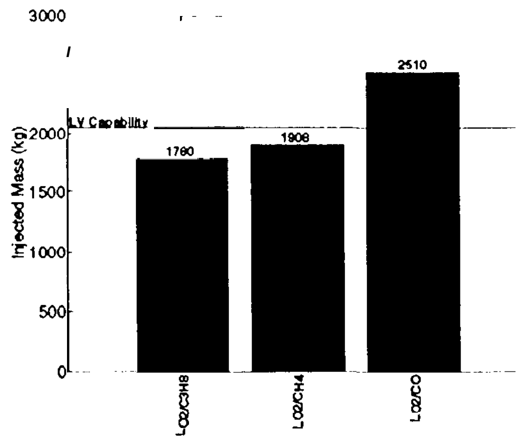

Figure 13. Injected Mass Requirements for "Advanced" Pump-Fed ISPP Systems.

The best performance of the systems considered is provided by LO_2/C_3H_8, which shows a launch vehicle margin of about 13%. The only combination which still fails to show margin is LO_2/CO. The combination of low performance, cryogenic tankage, and low density is too much of a burden for this propellant combination to become attractive, even under the most optimistic assumptions.

Conclusions

The present results strongly suggest that advanced storable propellant technologies in which propellants are brought from Earth are the logical choice for robotic Mars sample return missions in the size class considered in this study. The advanced storable system is capable of doing the mission with adequate launch vehicle margins (approximately 10%, or 240 kg) with relatively modest investments in lightweight tankage, warm gas pressurization, low temperature propellants, and lightweight components. The complexity of pump-fed engines and ISPP systems is not required, and the problems of cryogenic fluid storage are avoided,

From the present results. it appears that a robotic mission using ISPP and an advanced pump-fed liquid oxygen/propane propulsion system might provide slightly greater launch vehicle margin (by around 30 kg) than is available using an advanced storable system. However, this increased margin is offset by increased risk, including higher risk of mass growth. Launch vehicle margin is actually reduced by using ISPP in conjunction with a pressure-fed ascent propulsion system; an advanced pump-fed engine technology is required to show launch vehicle margin increases over the advanced storable system. In fact, given the technical uncertainties associated with this technology, it might not prove feasible to carry out such a mission using ISPP.

The present study differs from past studies, which have shown significantly higher leverage for ISPP, primarily in the small size of the payload (30 kg) and in considering the trajectories which are available in the time frame of interest. If the IS PP-based missions were able to use the lower-energy Type IV trajectory from Earth to Mars, significantly larger margins would be available. However, use of ISPP would still not enable use of a smaller launch vehicle, direct return from the Martian surface, or use of a single stage Mars ascent propulsion system even if such trajectories were compatible with the surface stay time requirements of ISPP systems and reasonable mission durations.

For missions requiring significantly larger payloads to be returned from Mars, it is likely that ISPP-based systems would prove to have higher leverage than for the small systems considered in this study. This would be particularly true omissions (such as a human expedition to Mars) which would probably use high-energy, fast trajectories even if propellants for the return flight were carried from Earth.

Recommendat ions

The authors recommend that the technologies required for the advanced storable propulsion system be aggressively pursued to support near-term Mars sample return missions. Indeed, such work is presently ongoing at the Jet Propulsion Laboratory.

With a view toward larger future missions, and perhaps eventual human expeditions, the authors also advocate continuing R&D activities in ISPP and related propulsion technologies. However, any decision to use these technologies on near-term sample return missions would appear to be premature.

Acknowledgments

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