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Design Trade Space for a Mars Ascent Vehicle for a Mars Sample Return Mission

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Abstract

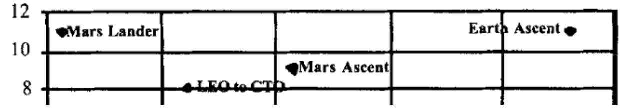
The design of an ascent vehicle for Mars sample return is one of the most challenging problems to be addressed for this type of mission. This paper identifies the spectrum of performance requirements that could be required of a Mars ascent vehicle for a sample return mission. With this understanding of performance requirements, an investigation of technology requirements is presented. These technology requirements are compared to past and existing technology in order to identify which are the lagging technologies and where development investment should be made. Mars ascent approaches which include storable propellants and in-situ production of propellants are considered. Several technology comparisons are performed to illustrate performance regions that are appropriate for different technologies. Several promising propulsion technologies are identified: miniature constant displacement pumps, bladder lined composite tankage, thin wall metal tankage and advanced propellants. Technologies that have been designed, built, tested and flown are emphasized. © 1999 Published by Elsevier Science Ltd. All rights reserved.

1. Introduction

The design of a Mars ascent vehicle for a sample return mission is a new and unique problem for the space exploration community. What makes this problem unique is the combination of performance and the small vehicle size required. Traditionally the only vehicles that have required high ΔV and high acceleration are Earth launch vehicles and no significant constraints have been placed upon their size. On the other end of the propulsion scale are propulsion systems for spacecraft. They are traditionally small-scale and they can possess the high ΔV required for a Mars ascent vehicle, but they did not possess the acceleration required for the problem. A Mars ascent vehicle requires both high ΔV and strong acceleration, like an Earth launch vehicle, but small size like a spacecraft propulsion system. Figure 1 provides a comparison of vehicle size and performance capabilities for various applications.

The authors have worked on a problem similar to the Mars ascent vehicle problem. In the late 1980's and early 90's the Strategic Defense Initiative Organization was developing space based interceptors for ballistic missile defense. The space based interceptor approach researched by Lawrence Livermore National Laboratory, LLNL, was called Brilliant Pebbles. Like a Mars ascent vehicle, these interceptors had to be small and required high acceleration and ΔV performance. The results and techniques of past research and development at LLNL towards the propulsion system of a space-based interceptor can be applied directly to the Mars ascent vehicle problem. This paper presents a systems analysis to identify the technologies that should be used for a Mars sample return mission.

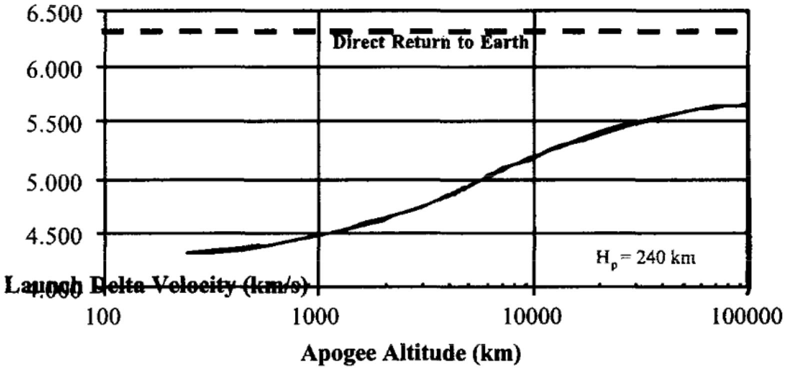
Figure 1, Typical Mission Requirements on Propulsion Systems



2. Ascent Performance Requirements

The results of this study are intended to span the range of options being considered for a Mars sample return mission. Previous mission studies[1,2] have identified ascent vehicle performance requirements. For the most part, these missions have used ascent vehicles to place a payload into a low circular orbit about Mars for eventual rendezvous and return to Earth. To make this study applicable to a broader range of mission studies, we will present results over a broad enough range to allow consideration of launch into highly elliptical orbits and even direct return to Earth. Figure 2 summarizes our survey of ΔV requirements for Mars ascent. Previous investigations of Mars ascent trajectories showed that a vehicle lift-off acceleration of ~10 m/s^2 was optimal. For this study we will always assume this lift-off acceleration.

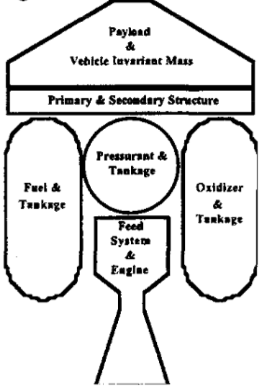
Figure 2, Range of Launch Delta Velocity



3. Ascent Vehicle Model

Figure 3 shows the mass accounting used in the paper to model a Mars ascent vehicle. The emphasis of the model is upon the propulsion-related components. Vehicle sub-systems that are relatively independent of vehicle weight, like avionics and payload, will be lumped into a single term call "P". "P" will be called the payload mass to orbit and is comprised of the cargo mass, M_c, and vehicle related invariant mass, M_k. For this study the ratio of structural mass to vehicle gross lift-off mass, M_s/M_o, is assumed to be 5%[3].

Figure 3, Major Weight Components of a Mars Ascent Vehicle



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[1] "Final-Report: Definition of Experimental Tests for A Manned Mars Excursion Module", January 1968, SD 67-755-2, Space Division, North American Rockwell Corporation

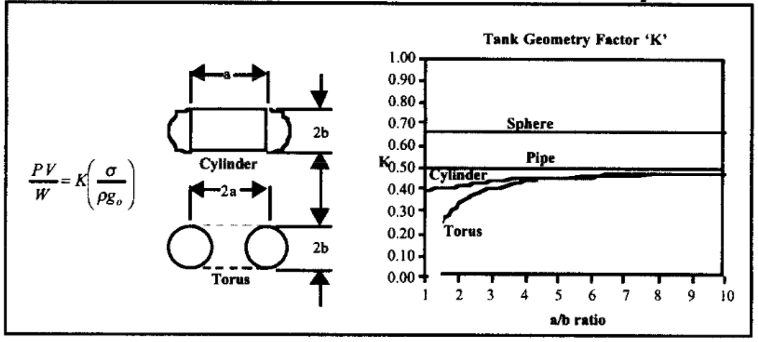
[2] "Mars Sample Return Mission – 1984 Study Report", September 1984, JPL D-1845, NASA Jet Propulsion Laboratory

[3] Guernsey, C., Thunnissen, D., Adler, M., French, J.,"Evaluation of Some Candidate Propulsion Technologies for Mars Ascent", 36th Aerospace Sciences Conf., AIAA 98-0651, Jan. 1998

4. Tankage Technology

The ratio of the tank burst pressure, P_b, times the tank volume, V, divided by the weight of the tank, W, is a common performance index for pressure vessels, P_bV/W. For tankage technology utilizing homogeneous materials and constant wall thickness the P_bV/W ratio is a constant. Figure 4 shows the derivation of P_bV/W. In this formulation σ is the yield stress of the tank material, ρ is the density of the tank material and g_o is the acceleration of gravity. K is an easily derived tank geometry factor[4] that allows many different tank shapes to be analyzed together with the same formulation.

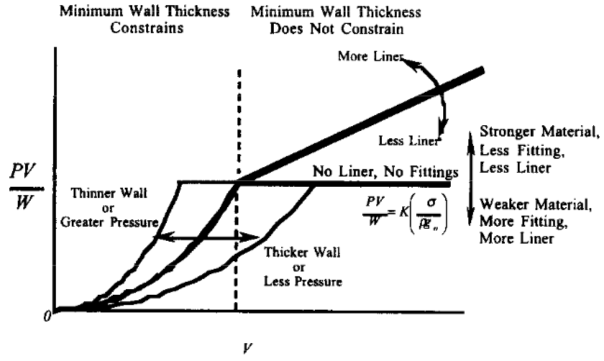
Figure 4, Ideal P_bV/W Models for Various Tank Shapes



In actual practice a tank wall cannot be made as thin as is desired. P_bV/W is reduced from the ideal when the tank wall must be thicker than is required by the stress limitations of the tank wall material. Figure 5 show a notional plot of P_bV/W to tank volume. As the volume of the tank is lowered, or as the minimum burst pressure of the tank is lowered, the wall thickness required becomes thinner. Eventually a manufacturing limitation prevents the wall thickness from becoming thinner. At this point the tank P_bV/W deviates from a constant value and begins to drop. This can be seen in figure 5. Much of the tankage to be used in a Mars Ascent vehicle will have small tank volumes and relatively low pressures. The manufacturing limitations concerning wall thickness will be important considerations when selecting the appropriate tankage technology.

P_bV/W modeling is also a good performance metric for comparing technologies which do not use homogenous materials. Composite tankage falls into this category. Fluid compatibility issues and tank wall permeability requirements often require the use of a liner material in the tank. The weight of the tank liner makes the tank heavier and makes the P_bV/W ratio deviate from a constant value. Figure 5 provides a notation description of how the P_bV/W ratio changes with the application of liners. In addition, figure 5 describes how the tank P_bV/W ratio changes with the introduction of fittings and other necessary hardware to the tank design.

Figure 5, Factors which cause variation of P_bV/W with Tank Volume



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[4] Humble, R.W., Henry, G.N., Larson W.J.,"Space Propulsion Analysis and Design",McGraw-Hill, Inc., 1995

LLNL has been interested in tankage technology because of its key role in the performance of regenerative fuel cell systems[5] and because of its application to high performance propulsion systems for space based interceptors[6]. These applications cover the entire spectrum of tankage technologies required for a Mars ascent vehicle. For these other applications, the tanks tend to have small volumes, like those required by a Mars ascent vehicle, and they required the highest P_bV/W ratio achievable. Tanks for regenerative fuel cell systems tend to be high-pressure (>5 MPa) composite tanks. This type of tank would be required for a Mars ascent vehicle if its propulsion system utilizes a pressurized propellant feed system. Reference 6 has found that tanks for space based interceptors should to be low-pressure (~350 KPa) thin wall metal tanks. This type of tank would be required for a Mars ascent vehicle using a pumped propulsion system.

Table 1 presents the parameterization of various tankage technologies. Using the P_bV/W method described above, figure 6 provides a comparison of these technologies in both the high and low-pressure categories. One experimental technology is included in this comparison to illustrate advance options available to a Mars ascent vehicle design. Carbon Fiber in a Poly Dicyclo Pentadiene[7] matrix (DCPD) is a thermoset polymer that uses a new ring-opening metathasis catalyst. It is relatively low cost and is suitable to many applications that require inert tank wall materials.

Table 1, Parameterization of Tankage Technology

Technology	Minimum Wall Thickness	Material Density	Material Ult. Strength	Liner Material Wall Thickness	Liner Material Density
	mm	Kg/m^3	GPa	mm	Kg/m^3
LLNL Demonstrated ASTRID Ti Metal Tank	.20	4430	1.17	N/A	N/A
Typical CC/Al Lined Tank	2.0	1600	2.5	1.52	2700
LLNL Demonstrated DOE/Ford CC/Bladder Lined Tank	1.0	1600	2.5	.25	1000
Proposed CC / DCPD Tank	1.0	1600	2.5	N/A	N/A

Figure 6, Low-Pressure & High-Pressure Tankage Technology Comparison

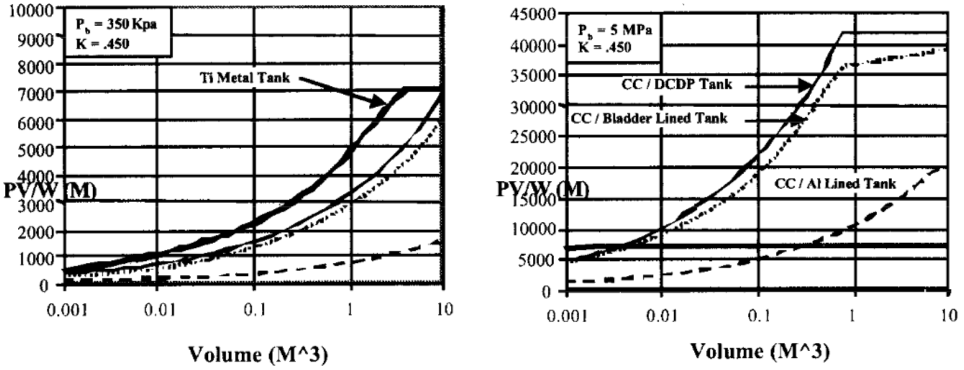

Figure 6 clearly shows, that for low-pressure tanks with volumes below 10 m^3, Ti thin wall metal tanks have superior P_bV/W performance. The figure also shows, that for high-pressure tanks above 0.05 m^3, composite liner-less tanks or bladder lined tanks have the best performance. This type of technology comparison can be expanded to include different pressures and technologies, but the results presented in this paper show a general trend.

This type of analysis directed LLNL in the development of tankage technology for regenerative fuel cell systems and for space-based interceptor propulsion systems. In the case of regenerative fuel cells systems, LLNL developed bladder lined tanks which have demonstrated a P_bV/W of 40,000 m. For space-based interceptors, LLNL has

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[5] Mitlitsky, F., Myers, W.B., Weisberg, A.H.,"Regenerative Fuel Cell Systems", Energy & Fuels, Vol. 12, No. 1, pp.56-71, 1998

[6] Whitehead, J.C., "A Lightweight Pumped Hydrazine Maneuvering Vehicle", Proceedings of the 1992 JANNAF Propulsion Meeting (CPIA Publication 580), Vol. I, pp.89, 1992, also UCRL-JC-109568

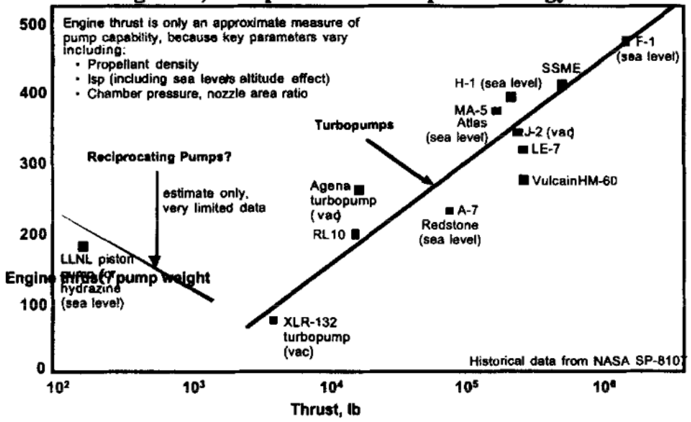
[7] Discussions of unpublished results with Ron Humble, KB Sciences, (719)333-6181

demonstrated deep-drawn, thin wall, aluminum tanks and welded, thin wall, titanium tanks. Welded titanium tanks, designed and built at LLNL, have been flight qualified and flown in a flight demonstration[8].

3. Pump Technology

Small pump-fed engines would be ideal for Mars ascent vehicles[9]. Figure 7 is a plot of pump performance versus thrust level. The graph illustrates the limitation of scaling down turbopumps. Reciprocating pumps are preferred to turbopumps for low thrust levels up to ~5000 lb_f. A 100 lbf thrust pump fed hydrazine system has been demonstrated in flight[8].

Figure 7, Comparison of Pump Technology



4. Estimate of the Stage Hardware Mass Fraction

A parameterized estimate of the stage hardware mass fraction was developed to facilitate the comparison of concepts. The following is the formulation that was used:

$$\text{Stage Hardware Mass Fraction} = \beta = \frac{(M_s + M_e) + M_l + M_t}{(M_s + M_e) + M_l + M_t + M_p}$$

$$(M_s + M_e) = \text{Mass of Engines and Stage Structure} = \left[\left(\frac{M_s}{M_o}\right) + \left(\frac{T}{W}\right)_{Veh} \Big/ \left(\frac{T}{W}\right)_e\right] \frac{\mu}{\mu - 1} M_p$$

$$M_i = \text{Residual Propellant} = \left[\varepsilon_{ox} \frac{OF}{(OF+1)} + \varepsilon_{fu} \frac{1}{(OF+1)}\right] M_p$$

$$M_t = \text{Mass of Tankage} = \left[\left(\frac{M_t}{V}\right)_{ox} \frac{1}{\eta_{ox}\rho_{ox}} \frac{OF}{(OF+1)} + \left(\frac{M_t}{V}\right)_{fu} \frac{1}{\eta_{fu}\rho_{fu}} \frac{1}{(OF+1)}\right] M_p$$

$$\mu = \frac{M_o}{M_f} = e^{\Delta V / I_{sp} g_o} \qquad \left(\frac{M_t}{V}\right) = SF\, P_b \Big/ \left(\frac{P_b V}{W}\right) g_o$$

Where:	M_o	is the initial mass
	M_f	is the final mass
	M_p	is the propellant mass
	M_s/M_o	is the ratio of structural mass to the gross mass of the vehicle or stage
	T/W_{veh}	is the initial thrust to weight of the vehicle or stage
	T/W_e	is the installed thrust to weight of the engine, pump and feed system

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[8] Whitehead, J.C., Pittenger, L.C., Colella, N.J., "Design and Flight Testing of a Reciprocating Pump Fed Rocket", AIAA 94-3031, 30th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 1994

[9] Whitehead, J.C., "Propulsion Engineering Study for Small-Scale Mars Missions", Report prepared for the Advanced Projects Branch, Space Projects Division, NASA Ames Research Center, September 1995

ε	is the percent of residual propellant mass
OF	is the oxidizer to fuel ratio
M_t/V	is the volume specific mass of a tank
η	is the volume usage ratio
ρ	is the density of the propellant
SF	is the safety factor applied to the tankage

3. Comparison of Concepts

A number of configurations have been suggested for a Mars ascent vehicle. A comparative analysis was performed for various propellant combinations and propellant feed technologies. The figure of merit used for this analysis was the ratio of the vehicle gross lift-off mass, M_o, to the payload mass, P. Note that lower values of this figure of merit have better performance. Table 2 provides a list of the configurations that were investigated and the parameters used to specify each configuration. In all cases, a 0.5% residual propellant fraction was assumed. For pressurized feed systems an initial ullage of 25% was assumed to account for pressurization. For pumped systems, an initial ullage of 5% was assumed. The propellant volume effectiveness of the solid propellant configuration was assumed to be 90%. The thrust to weight ratio of the engine, pump and feed system was assumed to be 30. Using the tankage information from the analysis described above, tanks were assumed to have a minimum mass per volume of 15 kg/m^3 and maximum P_bV/W of 25,000 meters with a design safety factor of 1.5.

The last column of table 2 shows the M_o/P figure of merit. In all cases, it is assumed that the vehicle was a single stage vehicle with 4500 m/s of ΔV and a lift-off thrust to mass of 10 m/s^2. Note that in all cases, pumped versions of like propellant combinations were superior. Monopropellant N2H4 and the solid have significantly lower performance. Traditionally it is thought that higher density propellants would be the good performers, but this analysis showed that when the best performing options are compared to each other, this trend does not hold. MON-25 / MMH appears to be very promising and should be investigated further.

Table 2, Parameters Assumptions & Mo/P Comparison of Various MAV Configurations

Technology	Density Oxidizer Kg/m^3	Density Fuel Kg/m^3	OF	Density Average Kg/m^3	Stage HW Mass Fraction	I_{sp} Sec.	M_o/P
90% H2O2 Pumped / RP Pumped	1372	810	7.5	1268	11.33%	298	8.79
90% H2O2 Pressurized / RP Pressurized	1372	810	7.5	1268	11.71%	298	9.09
Solid	N/A	1800	0	1800	11.97%	295	9.65
N2O4 Pumped / N2H4 Pumped	1440	878	2.25	1203	11.60%	315	7.57
N2O4 Pressurized / N2H4 Pressurized	1440	878	2.25	1203	12.00%	315	7.80
MON-25 Pumped/ MMH Pumped	1350	878	2.1	1150	11.72%	320	7.30
MON-25 Pressurized/ MMH Pressurized	1350	878	2.1	1150	13.34%	320	8.28
N2H4 Pumped	N/A	1010	0	1010	10.96%	235	27.70
N2H4 Pressurized	N/A	1010	0	1010	11.44%	235	32.42

3. Staging Considerations

The analysis presented in the last section assumes a single stage to orbit (SSTO) configuration for the Mars ascent vehicle. Many studies have assumed a two stage to orbit (TSTO) or even a three stage to orbit configuration. Historically, staging Earth ascent vehicles has been very beneficial; however, these vehicles are large in comparison to Mars ascent vehicles. For Earth ascent vehicles the stage hardware mass fraction of each stage will not vary much between stages of different sizes. For a Mars ascent vehicle, the stages are already very small and the stage hardware mass fraction, β, is sensitive to the variation in stage size. Figure 8 shows the stage hardware mass fraction as a function of the stage size. This figure assumes that the nominal tank volume for a Mars ascent vehicle is ~0.1 m^3 and that the M_t/V of the tank does not change significantly. This is a conservative estimate in favor of smaller stages since additional minimum wall thickness limitations are not being applied for tanks below the nominal case. If we assume that the ascent ΔV is split evenly between first and second stages of a vehicle[10], the stage hardware mass fraction of each stage will suffer an increase of about 30%. Because of this increase in the stage hardware mass fraction and the added mass of staging hardware, it is not clear that a Mars ascent vehicle will benefit from staging the vehicle to orbit.

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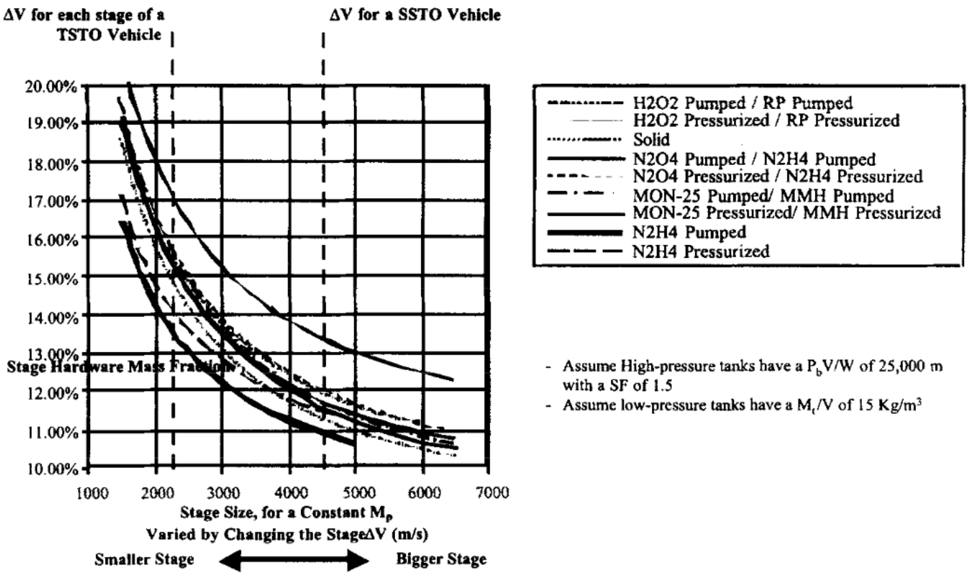
[10] Thomson, W. T., "Introduction to Space Dynamics", Dover, 1986, pp. 246-248

Figure 8, Stage Propellant Mass Fraction as a Function of Stage ΔV



- Assume High-pressure tanks have a P_bV/W of 25,000 m with a SF of 1.5
- Assume low-pressure tanks have a M_t/V of 15 Kg/m^3

Table 3 shows the M_o/P performance of each of the configurations listed in table 2 for SSTO and TSTO options. For TSTO cases, the stage hardware mass fraction has been adjusted according to figure 8. The mass from stage separation hardware has been added to the stages for the TSTO cases. A three-percent mass fraction was added to the first stage and a one-percent mass fraction was added to the second stage. To add this change to the stage hardware mass fraction the following formula was used:

$$\beta_s = \frac{\beta + \phi}{1 + \phi}$$

Where: β is the stage hardware mass fraction that does not include stage separation hardware mass

β_s is the stage hardware mass fraction adjusted to include stage separation hardware mass

ϕ is the ratio of stage separation hardware mass to the stage initial gross mass

Note in table 3 that the difference in M_o/P between the SSTO and TSTO options becomes small for the better performing configurations. The added risk, complexity and cost of a TSTO option is probably not justified given the small performance gain achieved. Also, because the estimate of β in this analysis held M_t/V for the tanks constant, a more detailed analysis may show that there exists even less advantage to TSTO options.

Table 3, Results of MAV Staging Configuration Analysis

		TSTO					SSTO		
Technology	Δ in M_o/P for SSTO	β_{nom}	β_1	β_2	μ	M_o/P	β	μ	M_o/P
90% H2O2 Pumped / RP Pumped	9.7%	15.02%	17.50%	15.87%	2.161	7.929	11.33%	4 669	8.7857
90% H2O2 Pressurized / RP Pressurized	11.5%	15.38%	17.84%	16.21%	2.161	8.050	11.71%	4 669	9.092
Solid	13.8%	15.57%	18.03%	16.41%	2.178	8.324	11.97%	4 742	9.652
N2O4 Pumped / N2H4 Pumped	6.1%	15.51%	17.97%	16.35%	2.073	7.106	11.60%	4 296	7.570
N2O4 Pressurized / N2H4 Pressurized	7.6%	15.88%	18.33%	16.71%	2.073	7.210	12.00%	4 296	7.803
MON-25 Pumped/ MMH Pumped	5.4%	15.69%	18.15%	16.53%	2.049	6.908	11.72%	4 199	7.301
MON-25 Pressurized/ MMH Pressurized	11.5%	17.17%	19.59%	17.99%	2.049	7.328	13.34%	4 199	8.277
N2H4 Pumped	47.5%	13.76%	16.27%	14.62%	2.656	14.527	10.96%	7 057	27.697
N2H4 Pressurized	53.8%	14.22%	16.71%	15.07%	2.656	14.969	11.44%	7 057	32.419

3. Conclusions

A number of system studies were described. The results from these studies were included in a vehicle trade study that compares the performance of various configurations.

The conclusions from the tankage system study indicated:

- Manufacturing limitations on the minimum wall thickness is a critical factor in choosing a tank technology for a Mars ascent vehicle.
- Tank pressure and tank volume determines the best technology. Two category of tankage were investigated, low-pressure tanks, P_b =350 KPa, and high-pressure tanks, P_b = 5 MPa. Thin wall metal tanks are the best design choice for low-pressure tanks with volumes less than 10 m^3. Liner-less, or bladder-lined, carbon composite tanks are the best design choice for high pressure tanks with volumes greater than .05 m^3.
- LLNL has developed and demonstrated both thin-walled metal tank designs and bladder-lined carbon composite designs.

The following are the conclusions from the pump system study:

- Turbo-pumps have not been produced at the size required for a Mars ascent vehicle. An extrapolation of the historical trend for the performance of turbo-pumps shows that constant-displacement pumps may have better performance at the sizes appropriate for a Mars ascent vehicle.
- The simplicity of constant-displacement pumps to design and operate may make this pump technology the preferred low cost option.

The following are the conclusions from the vehicle trade study:

- Configurations that had pumped propulsion systems out performed options with pressure feed systems.
- Mon-25 / MMH is a very promising propellant combination.
- A two stage Mars ascent vehicle provides only a small advantage in performance over a single stage Mars ascent vehicle. Because a single stage vehicle presents less risk, less complexity and lower cost to a Mars sample return mission, it is felt that a single stage vehicle may be preferred.
- More detailed analysis is warranted for the trade off between a single stage or a two stage vehicle. Assumptions and simplifications used in the presented analysis favored a two stage vehicles, yet the two stage option showed only minor performance improvement over the single stage option for the best propellants. A more detailed analysis may confirm the authors' suspicions that a single stage vehicle is the best approach for a Mars ascent vehicle.

3. Acknowledgement

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