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A Two-Stage, Single Port Hybrid Propulsion System for a Mars Ascent Vehicle

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A systems study has been conducted in order to demonstrate the feasibility of a hybrid propulsion system for a Mars Ascent Vehicle (MAV) as part of a Mars Sample Return (MSR) campaign. A two-stage, single port, hybrid design is presented for the MAV and compared to a baseline two-stage solid design. The hybrid MAV uses a paraffin-based fuel and a mixture of nitrous oxide (N_2O) and oxygen (O_2) as the oxidizer. This propellant combination was selected because of its compatibility with the variable and low temperature Mars environment. Significant reduction in mass over the baseline design (up to almost 30%) is possible since the hybrid system enjoys a higher performance and does not require thermal conditioning. The hybrid MAV design meets the mass, volume and environmental requirements for the mission and allows the use of existing entry decent and landing (EDL) capabilities.

Nomenclature

Isp	=	specific impulse
O/F	=	oxidizer to fuel ratio

I. Introduction to the Mars Sample Return Campaign

A Mars Sample Return campaign has been identified as a high priority step in Mars science (Ref. 1). The current architecture has the campaign broken into three missions with the following objectives: collecting and caching samples, recovering and launching the samples into Mars orbit, and returning the samples to Earth. Of the tasks needing to be accomplished for the MSR campaign, the means for getting the samples into orbit around Mars, the Mars Ascent Vehicle (MAV), has been named the highest system technology risk (Ref. 2). This is because the low mass and long storage time coupled with the cold and highly variable environment greatly increase the risk.

The MAV will remain on the Martian surface for a year or more in order to coordinate with the other elements of the MSR campaign. Environmental conditions on Mars are a significant concern and have required designs from previous industry studies to include a substantial thermal control "igloo" (mass up to ~50 kg). Diurnal temperature variations have been determined to be about ±50 C with annual mean daily temperature extremes from -110 C to 25 C using data from the NASA Ames Research Center Mars Global Climate Model (ARC Mars GCM).

The MAV is required to return the 5 kg, 16 cm diameter sample container to a greater than 400 km circular orbit at an inclination angle of 30° ± 0.2°. Although there is no fundamental mass requirement, a gross lift off mass of less than 300 kg is a reasonable goal. There is a strong desire to reduce thermal conditioning of the MAV required throughout the mission in order to reduce both mass at Earth launch and Mars Entry Descent and Landing (EDL). Additionally, decreasing the required thermal conditioning also reduces system complexity. The payload is taken to be 36 kg comprised of a 5 kg orbiting sample (OS) plus 31 kg which includes the OS interface and separation

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mechanisms, avionics (Attitude Control System (ACS), Command & Data Handling (C&DH), power), telecomm, cabling, thermal control, structure, a reaction control system, and a 3 kg contingency (Richard Mattingly, JPL, private communication). Based on discussions with JPL personnel, the mass allocated for telecommunications is sufficient to include a system capable of transferring high fidelity engineering data in order to determine root causes of decreased performance or failure. The MAV is constrained by the EDL system to fit within a 3 m by 0.6 m envelope. For this study, calculations were done using a ΔV appropriate for a 500 km circular orbit at 30° ± 0.2° of inclination when launched from the surface of Mars at a latitude of 30° North. We have chosen to present this more difficult orbit as suggested in (Ref. 3) to demonstrate the hybrid capabilities.

II. MAV Mission Approach

The fundamental requirements outlined in the introduction are used for this study. They include information gathered through the literature and ongoing discussions with Mars and propulsion experts at JPL. The proposed design minimizes the need for thermal management by enabling the MAV propulsion system to survive a long period in the Mars environment without a thermal igloo and with minimal thermal conditioning. The approach is to utilize a two-stage hybrid rocket design burning a solid fuel with a mixture of nitrous oxide (N_2O) and oxygen (O_2): hereafter referred to as Nytrox (Ref. 4, Ref. 5). These propellants were chosen based on their ability to ensure MAV operation in the low temperature Mars environment. Hybrid rockets enjoy a reduced sensitivity to cracking, debonding, imperfections, and environmental temperature compared to solids. In contrast to a solid rocket, hybrid rocket thrust performance is insensitive to the propellant temperature alleviating the need for a system to warm the propellant prior to launch. Since the fuel and the oxidizer are separated, the system is inert, leading to enhanced safety. Hybrids can be throttled or stopped and restarted. They are non-toxic and non-hazardous to manufacture. They can be stored for long periods of time without the risk of decreased performance. The freezing point of nitrous oxide is approximately -90.8 C and the particular fuel formulation will be selected to ensure structural integrity and reliable ignition at temperatures that can approach the range of -100 C to -120 C.

III. Low Temperature Fuels

A. Martian Environment

Prolonged exposure to the Martian environment will be the main obstacle for the MAV to overcome. The four candidate landing sites for the Mars Science Lab (MSL) were used as representative of possible landing locations for the MSR campaign. Environmental conditions at the launch site were derived from a recent run (2009) of the NASA Ames Research Center Mars Global Climate Model. The launch site is represented by the candidate MSL site with the largest annual temperature variation (Holden Crater) where the yearly minimum and maximum are -111 C and 24 C, respectively. The current state of the ARC Mars GCM is described in Ref. 6. It should be noted that the MSL candidate site nearest the 30° North requirement currently being pursued by NASA (Mawrth Vallis) only varies between -100 C and 5 C, so the site examined here is the worst case. The average temperature over a Martian year is about -60 C.

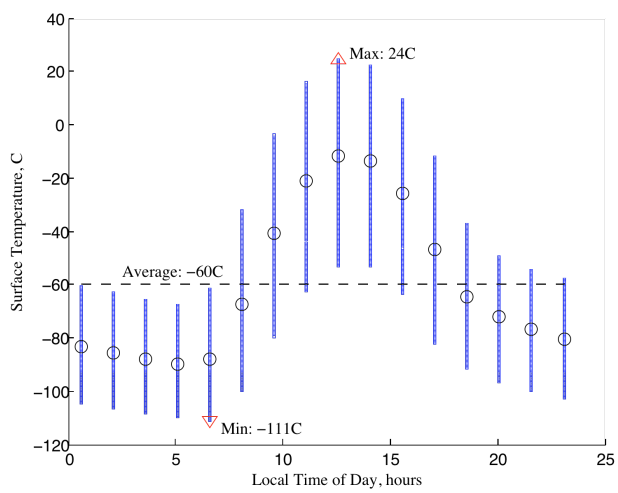

Figure 1. Temperature Profile at Holden Crater. *Holden Crater is the candidate MSL site with the most temperature variation. It is located at 26°S, 325°E, -1.9 km. Data from the ARC Mars GCM.*

Figure 1 shows temperature data at the Holden Crater site over one Martian year. The temperature range, averaged by time of day over the course of a year, is -90 C to -10 C, represented by the circles in the plot. This is the temperature range we used in the oxidizer tank design, assuming that, if necessary, it would be protected from seasonal temperature extremes by a thin layer of passive insulation. The blue lines show the range of temperatures for that time of day over a year.

Pure paraffin wax has a low glass transition temperature of about -108 C, much lower than the glass transition temperature of conventional polymeric fuels (i.e. for HTPB the glass transition occurs at -70 C) making it a viable candidate for the Martian environment. The glass transition temperature of the baseline paraffin-based fuel considered here is estimated to be even lower, in the range of -110 C to -130 C. Moreover since paraffins are highly crystalline, the effect of glass transition on material properties is significantly less severe than for amorphous polymers. It is anticipated that the favorable glass transition behavior of paraffin-based fuels will lead to formulations that can be safely used at the low ambient temperatures that exist at the desired landing site on Mars. However, this statement must be supported by structural testing at low temperatures to ensure that the selected fuel can withstand the thermal stressed imposed by initiation of combustion at low temperatures. A major benefit of paraffin-based fuels is that they can be optimized for the requirements of the mission. The current systems studies revealed the need for increased yield strength and a decreased regression rate compared to pure paraffin. This opens the door to using paraffin with additives or a different fuel entirely.

B. Thermal Vacuum Testing

Thermal/vacuum testing of pure paraffin (worst case behavior since paraffin used for hybrid fuel generally has strength additives) was conducted using NASA Ames test facilities. Tests of paraffin under roughing pump vacuum (but without thermal control) confirmed that there was no noticeable outgassing below Mars pressures. Further tests were conducted which involved putting a paraffin brick into a refrigerator. The preliminary test took two paraffin bricks, one with embedded thermocouples and the other without, to almost -80 C (the minimum temperature attainable by the system) at the maximum cooling rate. A second test was conducted to ensure that pure paraffin could survive an average daily ΔT of about 100 C as is expected on Mars. The paraffin brick was subjected to the equivalent of two sols at a linear rate of temperature change (or about 8 C per hour from +20 C to nearly -80 C). It survived this conservative test without trouble. A third test was conducted to more accurately represent a Martian day. Since the density of the Martian atmosphere is so low, the temperature changes rapidly with sun exposure. Data from the ARC Mars GCM for Holden Crater was averaged for the same time of day over a year to provide a representative daily temperature profile for the MAV, represented by circles in Figure 1. Cooling and heating rates were programmed to match these averaged rates as closely as possible. Note that the refrigerator was not capable of attaining the minimum average daily temperature on Mars (approximately -90 C), however it was cooled as far as possible. The pure paraffin is able to withstand the cooling rates expected on Mars and showed no signs of cracking whatsoever upon visual inspection.

IV. Nytrox

Nytrox (Ref. 4, Ref. 5) is composed of refrigerated mixtures of nitrous oxide and oxygen. It combines the high vapor pressure of dissolved oxygen (O_2) with the high density of refrigerated nitrous oxide (N_2O) to produce a safe, non-toxic, self-pressurizing oxidizer with high density and good performance. Note that in the mixture, the oxygen is the volatile component, which serves as the pressurizing agent whereas the N_2O is less volatile with the primary function of densifying the mixture. As increasing amounts of oxygen are dissolved in the liquid nitrous oxide, the vapor pressure varies from about 2 to 12 MPa. The percent of dissolved oxygen and the vapor pressure of the mixture are determined by the selected refrigeration temperature. At a selected self-pressurization level (or selected pressure), the densities of the mixtures are significantly higher than the densities of the pure substances. Also, for Nytrox at a given temperature, the liquid density is not sensitive to the system pressure as long as the pressure is not close to the critical value at that temperature. This feature gives the designer the flexibility of selecting the system pressure without affecting the liquid oxidizer density significantly.

The Nytrox mixture has many advantages over its individual components. It has improved Isp performance compared to N_2O. A Nytrox system is also much safer than a pure N_2O system because the vapor phase of the Nytrox system has a large O_2 concentration. A typical Nytrox system with 70% oxygen in the vapor phase requires 10,000 times more ignition energy compared to the pure N_2O and therefore it is impossible to cause N_2O decomposition with any practical ignition source that exists in the tank. Nytrox is capable of partial self-pressurization at high densities and it is not a deep cryogen like liquid oxygen, enabling the use of composite pressure vessels. Nytrox temperatures can be perfectly matched to the surface temperature on Mars eliminating the need for any temperature conditioning for the oxidizer. The design presented here uses Nytrox60 or Nytox at -60C, which is the average annual temperature on Mars.

A key feature of our design is that most of the pressurant (O_2) can be burned efficiently, thus greatly improving the structural mass fraction and the delivered Isp of the system. When all the liquid has been expelled, it is possible to achieve gas phase combustion with this oxidizer. In equilibrium, Nytrox is predominately N_2O in the liquid phase and O_2 dominates in the gas phase (Ref. 5). During the expulsion of liquid the motor operates at an O/F ratio of 3.7. When the oxidizer flow from the tank transitions from liquid to gas the O/F drops to 2.2 which happens to be close to the value required for maximum Isp with an oxidizer that is predominately oxygen. Combustion of the pressurant in a rocket system has been considered before: see, for example, Ref 7.

V. MAV Design Results

The design enables each component of the MAV propulsion system to survive an extended period of time (more than one year) on the surface of Mars without thermal control. Table 1 gives the temperature requirements anticipated for the selected propellants during both storage and operation of the propulsion system. It is not expected that any temperature conditioning will be required, either for the oxidizer or the fuel during the long storage period or for the orbital launch operation. However, it may be necessary to include a small amount of passive insulation to decrease the diurnal temperature variation felt by the system. The maximum stagnation temperature encountered by the MAV during launch is expected not to exceed 200 C (Ref. 8).

Table 1. Temperature requirements for selected propellants.

	Storage Min T	Storage Max T	Operation Min T	Operation Max T
Oxidizer	-90.8 C (freezing point)	-10 C (tank volume limit)	-90 C	-20 C
Fuel	-110 C to -130 C (weak glass transition)	~50 C	-100 C (targeted)	Determined by oxidizer max T
Pressurant	None	25 C (tank pressure limit)	None	25 C (tank pressure limit)

A pressure fed hybrid rocket propulsion system based on Nytrox60 (Nytrox at -60 C) oxidizer and paraffin-based fuel with 40% aluminum loading by mass has been selected for both stages. The hybrid design with pressurant combustion maintains both a performance advantage and reduced complexity compared to a small pump fed system. The addition of aluminum both increases the performance of the system and decreases the optimum O/F ratio. This allows a larger percentage of the propellant to be the high-density fuel. Nytrox60 was chosen because the average temperature on Mars is approximately -60 C and performance increases inversely with temperature for this oxidizer due to an increase in oxygen concentration (Ref. 5).

A simple process for Nytrox formation shall be implemented. Initially only N_2O will be stored in the oxidizer tanks until a few sols prior to launch. The oxidizer tank volume is selected for a maximum storage temperature of -10 C at which the density of N_2O is 953 kg/m^3. A few sols prior to launch, gaseous oxygen will be transferred from the pressurization tanks to the main oxidizer tanks in order to make equilibrium Nytrox. It is predicted that this short time would be sufficient to form the equilibrium mixture in the main oxidizer tanks.

The fuel for the proposed design is composed of 60% paraffin-based matrix and 40% aluminum powder (2 micron). The mean molecular composition of the grade of paraffin used is $C_{32}H_{66}$ (melting point 60 C). Additives are included to tailor the yield strength and regression rate to the desired levels. The c* efficiency and nozzle efficiencies are assumed to be 96% and 97% (a low value assumed to account for two phase flow losses), respectively. Gas phase combustion of 80% of the oxygen pressurant is assumed. The gas phase combustion contribution to the total impulse is around 4% for both stages. Since the gas phase combustion occurs at the end of the burn when the rocket is the lightest, this seemingly small contribution to the impulse makes a significant and positive impact on the ΔV.

There is a considerable degree of uncertainty in the aerodynamic drag that will occur during the first stage launch due to wide variations in atmospheric density that can occur during the course of the Martian year and uncertainty in both the drag coefficient and the effective cross section of the MAV since its shape will be largely determined by length and cross-section constraints imposed by the Mars lander (Ref. 8, Ref. 9). Given the present uncertainty in drag losses, the variety of alternative lander sites still under consideration, and the still unknown amount of thrust vectoring that may be required, the ΔV used in the preliminary design is taken to be a relatively conservative 4,375 m/sec, for a 500 km orbit.

In the baseline design of a two stage solid MAV (Ref. 3) shown in Table 2, a modified (stretched) ATK Star 17A (Isp = 287 s, gross mass = 126.0 kg, empty mass = 14.0 kg) was used for the first stage and a commercial off the shelf (COTS) Star 13A (Isp = 287 s, gross mass = 38.0 kg, empty mass = 5.00 kg.) is used for the second stage. The specifications for the Star motors were found on ATK's website. In order to more directly compare the solid system with the hybrid options, the two-stage solid is designed to take a 36 kg payload to a 500 km orbit. Though these are slightly different requirements than those presented in Ref. 3, the result agrees well with the previous design. The gross mass of the Star 17A is scaled up to 215.6 kg, keeping the propellant mass and structural masses in the same proportion in order to meet the required ΔV of 4375 m/sec. This is consistent with the stretched Star 17A referred to by Ref. 3 and would essentially be a brand new motor. A COTS Star 17A plus Star 13A will not make the mission.

Two detailed hybrid MAV designs are presented and compared to a baseline solid rocket system in Table 2. The first assumes a conservative structural coefficient with a mass contingency of 40% on all structural components. The second design reduces the contingency to 20% and reduces the gross lift off mass accordingly. Structural coefficients and contingencies of the hybrid rocket designs are derived from a detailed mass analysis using composite materials for the tanks and motor cases. In all three cases, a 4 kg payload faring is assigned to the first stage mass. The payload faring is assumed to separate from the MAV after the first stage burn.

Table 2. System Comparison. *ΔV = 4,375 m/sec, 500 km orbit. The structural coefficient is defined as one minus the stage propellant mass fraction.*

MAV System	Two Stage Solid	40% Margin Hybrid	20% Margin Hybrid
Stage 1 Mass (kg)	215.6	145.89	128.86
Stage 1 Structural Coefficient	0.170	0.189	0.167
Stage 1 Initial Nozzle Area Ratio	60.7	45	45
Stage 1 Average Nozzle Area Ratio	53.2	40.1	40.0
Stage 1 Maximum Chamber Pressure (MPa)	4.83	1.72	1.72
Stage 1 Delivered Specific Impulse (m/sec)	2813	2952	2951
Stage 1 ΔV (m/sec)	2530	1675	1675
Stage 2 Mass (kg)	86.2	91.37	83.57
Stage 2 Structural Coefficient	0.175	0.169	0.147
Stage 2 Initial Nozzle Area Ratio	49.8	50	50
Stage 2 Average Nozzle Area Ratio	41.0	41.9	41.7
Stage 2 Maximum Chamber Pressure (MPa)	6.45	1.38	1.38
Stage 2 Delivered Specific Impulse (m/sec)	2813	2977	2976
Stage 2 ΔV (m/sec)	1845	2700	2700
Total Ideal ΔV (m/sec)	4375	4375	4375
Maximum Diameter (m)	0.442	0.541	0.524
Vehicle Length (m)	2.56	3.829	3.747
Payload Mass (kg)	36	36	36
MAV Gross Lift Off Mass (kg)	301.8	273.3	248.4
Igloo Mass (kg)	50	0	0
System Total Mass (kg)	351.8	273.3	248.4

Table 2 illustrates that if the orbital requirement becomes 500 km as suggested in Ref. 3, a scaled up solid with the same performance figures will exceed the 300 kg gross lift off mass goal. It is important to note that the Nytrox60/Paraffin-based hybrid has a significant advantage (9.4-17.7% in gross lift off mass) over the solid due to its superior Isp performance. Also, neither of the hybrid designs requires a thermal igloo, which leads to an additional mass savings of up to 50 kg. The structural mass fractions of the hybrid stages are comparable to the mass fractions of the solids due to their comparatively low chamber pressures.

VI. Conclusions

The MAV is a complex new problem that cannot be simply solved with existing technologies. The two key technologies: liquefying hybrid fuels and Nytrox oxidizers, can be combined to design high energy, operationally flexible, low cost, safe, non-toxic, and environmentally friendly hybrid propulsion systems that can be stored and operated at the temperatures encountered on Mars. Hybrids are ideal for mission flexibility since they can be actively throttled (by a simple valve action) with a relatively small impact on the system performance. Easy throttling and shut down capability of hybrid systems is an important virtue especially for planetary ascent missions for which the propulsion system operation takes place following a long storage period in the spaceship and on the planet surface under adverse environmental conditions. It is expected that the technology developed for the hybrid MAV could also be used for ascent vehicles on other non-terrestrial bodies (the moon, etc.) and for in-space propulsion.

The hybrid MAV concept presented here can perform the mission using a less massive system than the current baseline solid design. The high performance (20% mass margin) hybrid design reduces the propulsion system mass by 17.7% and the total mass by 29.2%, since it does not require a thermal igloo. If a more conservative mass margin is desired, the mass savings are still significant. The 40% margin hybrid MAV enjoys a 9.4% reduction in the propulsion system mass and 22.3% mass reduction overall.

Further development of the hybrid MAV concept could prove to be an enabling technology for the Mars Sample Return campaign. Validation of fuel structural properties down to the minimum expected temperatures will be crucial in the final design of the MAV. Compared to previous studies, this design will drastically reduce the mass required for thermal management. In addition to the obvious advantages of a less massive system, by meeting the size and mass constraints given for the MAV, this design will allow for the use of heritage EDL capabilities thus decreasing complexity of the system. The hybrid rocket approach can be the basis for a system with increased safety, reduced cost, reduced complexity, increased design flexibility, and increased performance; one that should be able to compete favorably with conventional solid or liquid systems. We hope that this research will be a catalyst for new ideas for planetary ascent missions and in-space propulsion.

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