Advanced Storable Propulsion Technologies for Low-Cost Mars Sample Return

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Abstract

The propulsion requirements for ascent from the surface of Mars as part of a Mars sample return mission exceed the capabilities of current state-of-the-art chemical propulsion systems. The very high (4.6 km/s) free-space equivalent delta-V required to reach Mars orbit for rendezvous with a return is exceedingly challenging given the mass and envelope constraints on the ascent system. In particular, system studies[1] have shown very large leverage for a combination of advanced storable propulsion technologies which reduce the propulsion system dry mass.

This paper presents an overview of an ongoing R&D program at the Jet Propulsion Laboratory chartered to explore the feasibility of several promising chemical propulsion technologies. The specific technologies currently under study for the Mars Ascent Propulsion System (MAPS) are: 1) warm-gas pressurization systems for propellant tank pressurization, 2) low mass, high performance rocket engines using low-temperature propellants, 3) lightweight propellant and pressurant tankage, and 4) lightweight flow control components. Ongoing definition of the flight system configuration and performance are also being conducted to ensure that the technologies meet mission needs and in order to enable testing of integrated systems.

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Introduction

Under current planning[2], a series of Mars sample return missions will be conducted, with the first one launching in late 2004. This mission will recover samples of Martian rocks, regolith, and atmosphere collected by previous missions and return them to Earth. For this mission to meet the funding constraints of the NASA Mars Exploration Program (MEP), it is imperative that it use a launch vehicle no larger than a Delta III or Atlas 2AR. Failure to remain within the capabilities of these launch vehicles could make this mission fiscally untenable. To meet this objective, advances in the propulsion technology applicable to Mars ascent provides huge leverage.

The MEP has funded studies of potential MAPS propulsion technologies, the most recent being that documented in Reference 1. It was concluded that some concepts, such as the use of in-situ propellant production offered relatively little improvement considering the risk of such a development. Some concepts which had higher leverage, such as the use of liquid fluorine as an oxidizer, or of pump-fed rocket engines, were felt to represent excessive development cost and risk. This paper describes the set of mid-payoff / low risk technologies chosen for further development.

Warm-Gas Pressurization Systems

In a conventional spacecraft propulsion system, the propellant tanks are pressurized to about 20 bar with helium from a high-pressure (275 bar) supply tank so that propellants can be fed into the rocket engine(s). System studies have shown that the mass of the helium and storage tanks required for a conventional cold-gas pressurization system could approach 20% of the total system dry mass. For the Mars sample return mission, the propellants are consumed in just a few minutes which results in significant cooling of the pressurant gas in the supply tank, leading to very inefficient use of this gas, requiring more helium to be loaded. Analytic studies conducted in FY 97 show that about 35% of the dry mass of the pressurization system can be eliminated by the use of a warm-gas pressurization system of a type that does not require metallic diaphragms or bladders in the propellant tanks. Previous applications of warm gas pressurization have used diaphragms or bladders, increasing tank mass and eliminating much of the system benefit.

The warm-gas pressurization concept under study involves the use of a dilute mixture of hydrogen and oxygen added to the helium pressurant. This dilute mixture is non-flammable and non-detonable, but it can be reacted with a suitable catalyst to produce very hot helium with a small water vapor content. This small quantity of water is not a concern for a short-lived propulsion system such as that required for Mars ascent, although it would be a concern for longer-term applications due to formation of nitric acid in the oxidizer tank.

Designing a catalyst bed for essentially complete reaction of the hydrogen and oxygen is complicated by the fact that the catalyst will have to function with initial bed and pressurant temperatures as low as -40° C and with final pressurant gas mixtures as cold as -153° C. Subscale testing conducted during the summer of 1997 confirmed that practically complete reaction of the hydrogen and oxygen can be obtained with gas and hardware temperatures below -40° C. This is illustrated in Figure 1, which shows the temperature rise in the catalyst bed after a mixture of 95.5% helium (by volume), 3.0% hydrogen, and 1.5% oxygen began flowing through it. The steady state temperature rise of 350°C was within 1% of that computed assuming complete reaction.

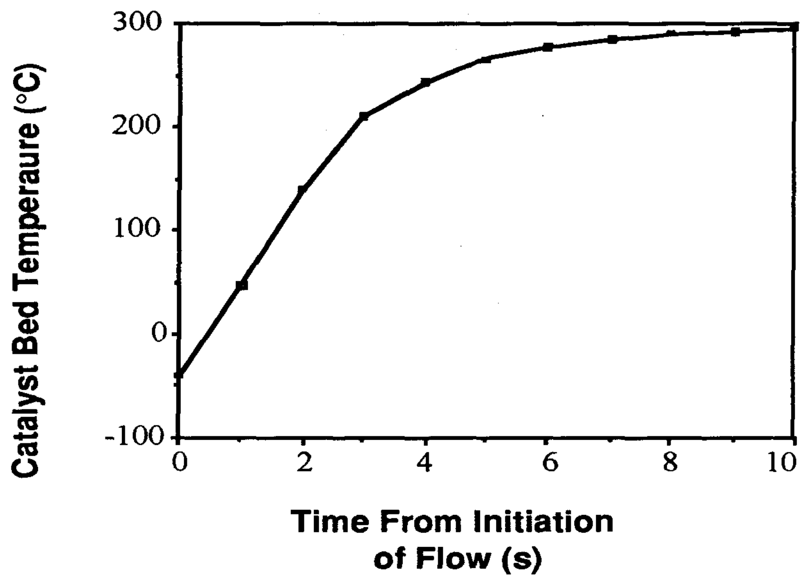

Figure 1 - Reaction of -40°C Gas Mixture

Testing is continuing in order to address a range of potential catalyst materials, flowrates, and temperatures as well as to provide quantitative measurements of the percent reaction completion. These data are the fundamental information required before optimum designs of a flight catalyst bed can begin.

Another potential technical issue which is under investigation is the temperature above which the pressurant gas might have undesirable interactions with the propellants. This temperature will probably be determined by the threshold for thermal decomposition of the fuel and/or the interaction of the propellant vapors with tank materials and any residual hydrogen or oxygen which may remain after passing through the catalyst bed. Laboratory experiments are being prepared to evaluate interactions of warm, possibly incompletely reacted, pressurant gas with the propellant vapors using data on the percent reaction completion obtained from catalyst testing.

Heat exchange with the propellant tank walls and propellant is also a significant factor in determining the effectiveness of a warm gas pressurization system. A preliminary modeling effort indicates that the heat losses to the propellant tank walls and propellant, while significant, will not eliminate the benefit offered by this technology. More detailed analytic efforts will be conducted as the effort progresses, and they will be calibrated against experimental data. Since both heat transfer and propellant vapor interactions are expected to be scale-dependent, a complete feasibility demonstration will require full-scale testing. The first such full-scale testing will be conducted at the NASA White Sands Test Facility this summer.

Rocket Engines Using Low Temperature Propellants

Conventional storable (i.e., non-cryogenic) bipropellants used in spacecraft propulsion have freezing points of - 11 °C or higher. This high freezing point is also often accompanied by severe performance decreases at propellant temperatures below about 0 °C. This places significant thermal control requirements on the ascent propulsion system while on the surface of Mars. These thermal control requirements in turn increase the demand for radioisotope heat sources and/or electrical power while on the surface. The present effort is focused on exploring the feasibility of using low temperature propellants which have freezing points below - 40 °C, which is just slightly above the mean diurnal temperature on Mars at the latitudes of interest. The propellant combination presently under consideration is Monomethyl Hydrazine (MMH) fuel and an oxidizer which consists of 75% by weight nitrogen tetroxide and 25% by weight nitrogen monoxide. This oxidizer is referred to as Mixed Oxides of Nitrogen-25 (MON-25).

MMH and MON-25 have been considered for use in a number of past systems, but no reports of testing at operating temperatures as low as - 40 °C were found in the literature or by personal contacts with participants in those programs. While it was known that these propellants will ignite hypergolically (i.e., spontaneously on contact) at room temperature, this had never been demonstrated at -40 °C. In addition, several key physical properties of MON-25 are not available in the current literature. A knowledge of these parameters (especially viscosity), as well as of propellant reactivity and droplet vaporization rates is needed to design a high-performance bipropellant rocket engine using these propellants at low temperatures.

An experimental facility to investigate hypergolic ignition and physical properties of MON-25 and MMH at - 40 °C was completed in FY 97. The viscosity of MON-25 was determined and the experimental results are shown in Figure 2. These results imply that the viscosity of MMH, which is given in the literature as approximately 6 centipoise at -40° C, will be the limiting factor in obtaining turbulent injector passage flow with acceptable pressure drop.

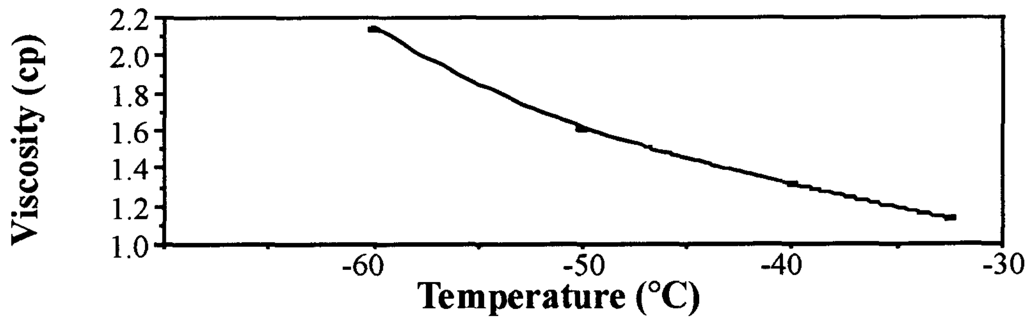

Figure 2 - Measured Viscosity of MON-25

Using the same experimental facility, hypergolic ignition of MON-25 and MMH was demonstrated at -40° C. Ignition occurred sometime during the second video frame after a jet of MON-25 impinged on a pool of MMH. This implies a delay time of 30 to 60 ms, which would be of concern if experienced in an actual rocket engine. However, it is expected that the ignition delay will be shorter in an impingingjet or sheet configuration due to the absence of a significant pool of fuel which tends to transfer heat away from the initial reaction zone and retard the initial reactions which lead to ignition.

This effort will continue with testing of a single injector element to: 1) determine whether a stable, repeatable propellant stream can be established with the highly viscous MMH fuel, 2) determine ignition delay times under conditions more typical of an actual rocket engine, and 3) search for chamber pressure or injector design constraints which may be required to prevent separation of the impinging propellant streams as a result of liquid-phase or gas-phase reactions of the hypergolic propellants.

Lightweight rocket engine development will require the use of lightweight combustion chamber materials and injector designs. This effort will build on industry experiences gained in support of the Strategic Defense Initiatives as well as new and innovative materials concepts such as nanolayered composite materials. A Request for Proposal has been issued which will lead to selection of one or more rocket engine manufacturers to use these fundamental data to demonstrate performance of a 35-lbf rocket engine operating on these propellants at - 40 °C and begin advanced development of an injector for a lightweight engine using these propellants for use as an attitude control thruster on a Mars ascent system. Assuming this advanced development work meets with success, development of a 600-lbf main engine required for the first stage of the Mars Ascent Propulsion System (MAPS) will be initiated in future years.

Lightweight Propellant and Pressurant Tankage

Conventional propellant tanks for spacecraft are typically machined from titanium forgings, while conventional pressurant storage tanks are either titanium or a metallic liner (of titanium, stainless steel, or aluminum) overwrapped with a high-strength carbon composite. Tankage fabricated using these conventional technologies would be the largest single contributor to the dry mass of a Mars ascent propulsion system. Twenty to thirty percent of the mass of high-pressure Composite Overwrapped Pressure Vessels (COPVs) used in conventional pressurant tanks is in the metallic liners which are required for acceptably low gas permeability. Attempts to apply COPV technology to low pressure propellant tanks have been hampered by the limitations of conventional liner fabrication technologies; a minimum thickness metallic liner can typically support a large fraction of the pressure load and thus the benefit of the composite overwrap is greatly reduced. Also, the composite overwrap has a ply thickness which is greater than that needed for structural performance in low pressure tanks which usually has resulted in excessive mass. The goal of this effort is to explore new liner fabrication technologies, fibers, and matrix materials which could produce factor of two reductions in propellant tank mass and reductions of 10 to 20 percent in pressurant tank mass.

Replacement of metal liners with low-permeability polymeric materials is one potential technology for reducing tank mass. Testing of candidate polymeric liner materials for propellant compatibility was performed last summer. Materials tested included polybenzoxazole (PBO), polyethylenenaphthalate, polyphenylenesulfide, polycarbonate, polypropylene, and polyetheretherketone. All of these materials were found to have limited compatibility with the oxidizer, but several candidates were shown to have acceptable compatibility with the fuel. Some fluorocarbon polymers such as Teflon have excellent oxidizer and fuel compatibility characteristics, put are far too permeable to be used as a tank liner. The study of polymeric tank liners is now focused on technologies which may allow thin-film metalization of the inside and outside surfaces of fluorocarbon liners. These dual metalization layers separated by a fairly low-permeability, propellant compatible, polymer should prove more forgiving of minor localized porosity or cracking of the metal than schemes which rely on a single metallic layer to prevent permeation into the overwrap. It is also hoped that if a truly amorphous metalized layer can be produced it would exhibit resistance to cyclic fatigue superior to that of conventional machined metal liners.

Under contract to JPL, Metal Surfaces, Inc. demonstrated metalization of the inside of a PFA Teflon test bottle by electroless plating during the summer of 97. Three stages were used in the metalization process: 1) a palladium "assist" layer was followed by 8000 angstroms of copper, 2) this was followed by an additional 8000 angstroms of nickel, and 3) 4000 angstroms of gold was applied to provide an inner layer with excellent propellant compatibility. This is a critical process for forming a seamless inner metallic permeation barrier on the inside surface of a polymeric liner. Planar samples are currently being fabricated for permeability testing and microstructure examination, as well as cyclic fatigue testing. Additional samples which use magnetron sputtering to apply a thin layer of ultrapure aluminum to the polymer are also being prepared.

In addition to investigation of polymeric liners, several potential technologies for forming very thin metallic liners on expendable mandrels are being pursued. These technologies could also prove viable for metalizing the outside surfaces of polymeric liners. Preparation of thin test coupons of ion sputtered titanium, wire-arc sprayed aluminum, vacuum arc vapor deposited titanium and/or aluminum, electroless gold / electroformed nickel and electroformed aluminum are currently being investigated. These samples will be tested for permeability and microstructure, and will be subjected to cyclic fatigue testing. Extension of the technology used to form beverage cans from very thin metal foils is also being explored. The extension of existing techniques for chem-milling wrought aluminum liners to wall thickness below 0.005" is being investigated as a lower-risk "fall back" technology.

If one or more candidate liner fabrication methods looks promising, demonstrations of fabrications of tank liners will be followed by demonstration of prototype tank fabrication. In the propellant tank application, the use of PBO fibers in the overwrap will be investigated. In a minimum-lay-up application, the lower density of the PBO fibers may make them a superior material to the conventional T-1000 carbon fibers in spite of their lower strength. Fibers which might allow thinner plies are also being considered. Full-size prototype tanks will be fabricated for compatibility tests and use in breadboard propulsion system functional testing later this year.

Lightweight Components

This portion of the effort is just getting underway, and is less well defined than the others. In the past, there has been little effort expended to reduce the mass of propulsion flow control devices because for very large spacecraft their mass is a tiny fraction of the spacecraft mass. Large and probably unnecessary factors-of-safety have traditionally been levied on these components to avoid the costs of detailed analysis and test required to lower the margins. In addition, the drive to reduce fabrication costs has often led to non-optimal designs. For the Mars ascent propulsion system, and for many future missions where reduced mass is necessary to minimize overall system costs, these luxuries simply can not be afforded. Therefore this effort is concentrating first on applying state-of-the-art analysis and fabrication methods to the reduction of component mass.

There are also a number of new technologies, such as the use of magnetoconstrictive or shape-memory materials which might be applied to reduce feed system component mass. These are under evaluation, as are design practices developed to miniaturize propulsion components for Strategic Defense Initiative programs.

Configuration and Structure

Figure 3 illustrates the latest version of the MAPS configuration under study. It is a two-stage design which uses somewhat different structural concepts on each stage. The second stage used the aerodynamic fairing required during flight through the Martian atmosphere as a major structural element, with propellant tanks and other hardware suspended below it. It is connected to the first stage by a central core structure which transmits loads to the first stage. The stage separation is executed by firing a linear shaped charge wrapped around the circumference of the core structure at the first to second stage interface. (or to be more succinct: A linear separation device along the circumference of the central core structure is used for stage separation).

The first stage is attached to the landing system at four hard points which are located directly under the four propellant tanks. This allows all of the vertical load produced by the tanks during Earth launch and Mars entry to be supported by the lander structure, reducing mass of the ascent system. The central core structure carries second stage loads and a portion of the lateral loads produced by the first stage tanks into the composite honeycomb base plate at the bottom of the first stage. This plate transmits these loads to the hard points on the landing system.

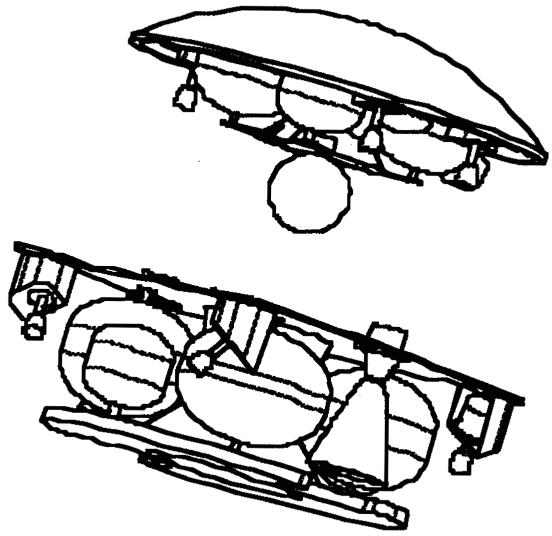

Figure 3 - MAPS Configuration Concept

Conclusion

A combination of advanced storable propulsion technologies promises to enable a Mars sample return mission to be performed on a launch vehicle compatible with the funding profiles constraints of the Mars Exploration Program. These technologies are being pursued by the Jet Propulsion Laboratory in conjunction with several industrial partners.

Acknowledgment

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