Hybrid Propulsion In-Situ Resource Utilization Test Facility Development

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Hybrid propulsion could be a potential game changing technology for several Mars applications, such as Mars Sample Return (MSR) and human exploration. A flexible hybrid test facility has been built at the Jet Propulsion Laboratory to provide data relevant to the design of such systems. This paper presents the motivations for such a system and its design. The facility is capable of testing 5 cm diameter fuel grains with gaseous oxygen and Mars *in situ* propellant production simulating oxidizer (varying mixtures of GO_2, CO_2 and CO). All currently planned tests utilize paraffin based fuels; however, alternative hybrid fuels may be used in the future. Variable length to outer diameter (L/D) ratios may also be tested to give insight on potential packaging constraints. The goal of this research is to enable the inclusion of hybrid propulsion systems in future mission design studies by determining the empirical constants in the regression rate equation for paraffin-based fuels with space storable and/or *in situ* oxidizers and to investigate the effect of L/D on combustion efficiency. Test results will be reported separately.

Nomenclature

a	=	Regression Rate Coefficient
G	=	Oxidizer Mass Flux [g/cm^{2}s]
Isp	=	Specific Impulse
n	=	Regression Rate Exponent
O/F	=	Oxidizer to Fuel Ratio
$\dot{r}$	=	Regression Rate [mm/s]

I. Introduction

HYBRID propulsion utilizing oxidizer produced *in situ*, on Mars could pave the way towards an achievable Mars Sample Return (MSR) campaign. It is also especially attractive for larger and longer, proposed human-scale missions. Hybrid rockets are made up of a solid fuel and liquid (or gaseous) oxidizer. For the potential MSR application, a hybrid Mars Ascent Vehicle (MAV) would allow the fuel to be transported to Mars in its dense, solid state. Several hybrid propellant combinations have been presented in the past for this application including a paraffin-based hybrid with a mixture of N_2O and O_2 augmented by CO_2 collected *in situ* (Ref. 1) as well as paraffin with liquid oxygen (Ref. 2). A related option: paraffin with compressed gaseous oxygen is suggested here. The oxidizer would be generated from CO_2 on the surface, reducing the required landed mass on Mars. The optimal oxidizer to fuel ratio allows at least 2/3 of the total hybrid propellant mass to be gathered while on the surface of Mars, making it more feasible to use the current state of the art Sky Crane Entry Descent and Landing (EDL) system. This is an especially attractive option to demonstrate the viability of *in situ* propellant production (ISPP) because it only requires one of the propellants to be made on Mars. Oxygen is easier to produce from the atmospheric constituents available on Mars than hydrocarbon fuels. Therefore, bringing the fuel allows an

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incremental approach to propellant production. The goal of achieving tangible progress toward MSR has been deemed the highest priority by the planetary science community for the period between 2013 and 2022 (Ref. 3). The capabilities necessary for the proposed MSR missions need to be developed immediately to ensure they are ready should any effort to return a sample commence (as early as the 2020's).

A Mars Simulating Hybrid In-situ Propulsion test facility has been designed and is currently being built at NASA's Jet Propulsion Laboratory (JPL). The test facility is capable of operating a 50 lbf (approximately 200 N) motor using paraffin-based fuel and a gaseous oxygen mixture from a simulated Mars in-situ production process. The first tests will be run with pure oxygen. However, the second phase of testing will introduce varying concentrations of CO_2 and CO into the system to determine their impact on performance. All tests will be conducted at ambient external pressure and temperatures. The facility was designed for flexibility. Many variables can be altered including fuel and oxidizer type, mass flow rate, length to diameter ratio of the fuel grain, oxidizer to fuel ratio, etc.

Development of an in-situ propellant production process at JPL is occurring in tandem with this project and is paramount to the conceptual design of the MAV. In this process, CO_2 is collected at Mars ambient pressure using a vacuum pump and passed into a cryocooler where it is frozen. The frozen CO_2 is allowed to sublimate at a moderate pressure in an intermediate tank. A Solid Oxide Electrolysis (SOXE) process, developed by Ceramatec, Inc., is used to convert the CO_2 to O_2 and further pressurize the gaseous oxygen. Oxygen production via the SOXE process has been demonstrated on Earth at moderately low pressures. Tests at higher pressures, representative of what would be required for a MAV have been proposed. The size of the SOXE unit dictates the rate and amount of carbon dioxide converted into O_2. The electrolysis typically takes place at temperatures of about 800°C. The oxygen produced by this system is fairly pure, since it is collected at the positive electrode, while the CO and CO_2 are collected at the negative electrode. However, the efficiency of this system is still relatively low: roughly 25% of the CO_2 brought into the system is converted into O_2. The conversion process is also relatively power intensive, requiring approximately 350 W per sol in order to produce the necessary amount of oxygen within about one year. Therefore, it will be useful to evaluate the performance of the hybrid system at various oxygen concentrations. Progress in this research will be reported only as it directly applies to the MAV design. The focus of the paper will be on the hybrid propulsion testing. Allowing the oxidizer to have some "contaminants" (e.g. CO_2 and CO) could dramatically reduce the time or power required to produce the oxidizer.

II. Motivation

A Mars Sample Return campaign would be an exceptionally difficult challenge, requiring multiple missions to work in concert. The MAV propulsion system would be required to survive on the surface of Mars for an extended, currently undetermined, period of time under the harsh environmental conditions. Currently proposed propulsion systems (solid, liquid bi-propellant, gelled, monopropellant) require massive thermal conditioning systems most likely requiring both a passive CO_2 gap and heaters. The overall system designs presented to date (including the MAV, thermal igloo and erection systems) are pushing the capability of the state of the art EDL system: the Mars Science Laboratory-type Skycrane. The Skycrane can deliver nearly 1000 kg to the surface of Mars, with potentially 30% of that mass allocated to the MAV, giving a reasonable upper limit of about 300 kg.

It has been suggested that hybrid rocket propulsion could be an enabling technology for the MAV (Ref. 5). As mentioned previously, producing the oxidizer in situ translates into approximately a 2/3 reduction in propellant mass. However, it should be noted that this is at the expense of the hardware required to produce the oxidizer. CO_2 is collected from the Martian atmosphere.

A. Benefits of Hybrid Rocket Propulsion

Hybrid rockets have typically been passed over due to the disadvantages associated with multiport designs required by conventional, polymeric hybrid fuels (e.g. HTPB). Paraffin-based fuels enable single, cylindrical ports. Paraffin/GOx has relatively high performance, comparable to liquid bi-propellant engines. The specific impulse is plotted against oxidizer to fuel ratio in Fig. 1. This curve can be biased towards lower oxidizer to fuel ratios by adding aluminum the fuel grain. This could be

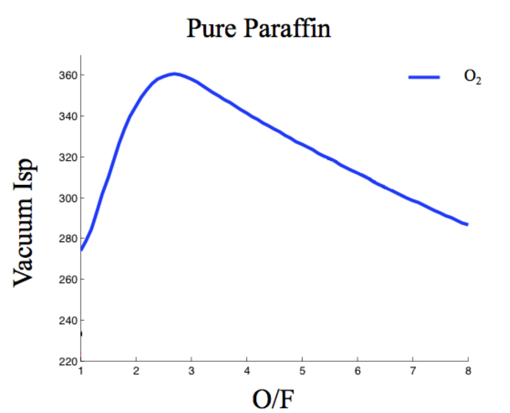

Figure 1: Vacuum Isp for pure paraffin with oxygen. Calculated using the open source software: Chemical Equilibrium with Applications (Ref. 4).

an added benefit because it reduces the amount of oxidizer that needs to be produced on Mars and adding aluminum particles to the fuel grain actually stabilizes the system (Ref. 6 describes the stabilizing mechanism applied to liquid and solid rockets). Hybrid combustion can be stopped and restarted enabling multiple, large ΔV burns. This could make a single stage to orbit hybrid MAV feasible if the payload is small enough. Hybrids can also be throttled to obtain precise thrust profiles in a simple manner because the burn rate of the fuel is dependent on the oxidizer mass flux (Eqn. 1, described in the following section). An overview of the benefits of the hybrid system versus the state of the art is presented in Table 1 and a more complete discussion is available in Ref. 7. It should be noted that the low temperature performance of the paraffin-based fuel is a predicted value and requires confirmation. It is based on the low glass transition temperature of the predominately crystalline wax (Ref. 8). The oxygen will be stored above its critical point as a supercritical fluid, and therefore should have little impact on storage temperature limits.

Table 1: Comparison of Hybrid and Conventional Propulsion Systems. *Predicted value, requires empirical confirmation.

		Paraffin/GO_2 Hybrid	Conventional Propulsion Systems		
			Solid	**Liquid (MMH/NTO)**	**Monopropellant**
Isp (performance)		~330 s	285 s	324 s	240 s
Restart capability		Yes, multiple	N/A	Yes, multiple	Yes, multiple
Throttling		Simple, 10:1	None	Complex, 3:1	Simple, 10:1
Low temperature storage & operation		< -100 C*	- 40 C	+13 C	+ 13 C
Safety	Toxicity	Nontoxic	Toxic	Toxic	Toxic
Safety	System Complexity	Moderate	Low	High	Moderate
Safety	Recurring Cost	Low	Low	High	Moderate

B. Regression Rate Determination

The hybrid fuel regression rate ($\dot{r}$) can be found from Eq. (1), where G is the oxidizer mass flux (the oxidizer mass flow rate divided by the cross sectional area of the motor port) and a and n are empirically derived constants dependent on the propellant combination.

$$\dot{r} = a\, G^{n} \tag{1}$$

The coefficient, a, can be tailored by mixing additives with the pure paraffin. Reducing the regression rate by half (or more) is easily achievable. The more important variable to be determined is the exponent, n. The shift in oxidizer to fuel ratio (O/F = $\frac{\dot{m}_{ox}}{\dot{m}_f}$) over the course of the burn depends on the exponent, which is typically between 0.5 and 0.8. As the port opens up, the oxidizer mass flux decreases and the burning area of the fuel increases. These two components balance when the exponent equals 0.5, as is the case for N_2O, which does not see a shift in O/F ratio. Efforts to predict this relationship without the need for testing are underway (Ref. 9) and data from this facility will be used to validate the effort.

III. Driving Requirements

Combustion in hybrid rockets occurs in the boundary layer where vaporized fuel mixes with the oxidizer stream. Combustible mixtures may not occur throughout the length of the fuel grain depending on the L/D. This is a fundamentally different process from solid rockets. Fig. 2 is a graphical depiction of the combustion regions. In the two dimensional diagram, the boundary layers are shown converging at the aft end of the fuel grain. Hybrid motors are further complicated because the boundary layer profile continuously evolves as the fuel regresses. Empirical results have typically shown best behavior in the length to outer diameter range of 6-12. However, many in space packaging constraints drive the motors to lower L/D's in order to be competitive with more conventional solid or liquid propulsion systems.

The MAV system would be constrained geometrically by the landing system and by the chosen platform. It is likely that the Skycrane landing system would be used for such a mission. The currently proposed baseline has a platform based MAV with a fetch rover; however, it would be more desirable to have a mobile MAV. The mobile system would further constrain the propulsion system to fit on a rover as well as within the EDL constraints. The mobile platform would be more challenging for an in situ propellant production system because not only would the full volume of the MAV be required to fit, but the additional components necessary to produce the oxidizer would also have to be accommodated. The mobility system would also require power to rove, reducing the available power for propellant production.

The challenges imposed by the Martian system are significant. NASA Ames Research Center's Mars Global Model (Ref. 11) was used to determine an average annual temperature of -60° C with diurnal variations of up to about 100° C at the most extreme of the four final candidate MSL sites. As future missions select landing sites, the actual temperature requirements can be determined. It is believed that using paraffin would give a substantial advantage in the thermal conditioning system due to its predicted exceptionally low glass transition temperature. However, the thermal stresses on the fuel grain should be evaluated and tested. It is likely that some sort of stress relieving device will be necessary to ensure the fuel does not debond from the insulator and case. However, even if a debonding event did occur, it would not be catastrophic since there is no available oxidizer outside the central port.

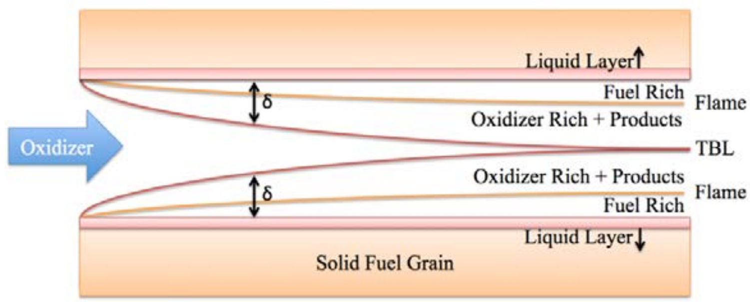

Figure 2: Combustion regions in a hybrid rocket, Ref. 10.

IV. System Design and Capabilities

A robust, self-contained system has been developed at JPL to research several of the open questions in hybrid propulsion for Mars. The completed facility is shown in Figure 3. The system can deliver a maximum of 222 N (50 lbf) of thrust. The tests will be limited to no more than ten seconds in duration. This limit is based on the fuel grain diameter of two inches (5 cm), the fuel regression rate and system safety requirements. The oxidizer mass flow rate and the chamber pressure (maximum expected operating pressure is 1.72 MPa) will be measured during the burn. The mass of the fuel grain before and after the test will be used to determine a time-averaged regression rate, characterizing performance. NASA Standards for Oxidizer Safety (Ref. 12) have been adopted for the design of the system.

Figure 3: Hybrid test facility set up.

A. System Overview

The hybrid test setup is shown in Figure 4. Many safety precautions have been taken including complete venting of the oxidizer line if the regulator were to fail while open. If power is removed from the system, all valves fail closed and combustion cannot be sustained. All of the components have been sourced such that the oxidizer can be gaseous oxygen (GOX, as shown here) or a Mars in situ produced propellant simulant. The flow is controlled with a full-bore ball valve (SM2). The oxidizer mass flow is controlled at the injector and measured with a venturi based on ISO standards (Ref. 13) and fabricated for this application. A purge system is included to quench the combustion on demand. This feature will be used in nominal operation as part of the shutdown procedures.

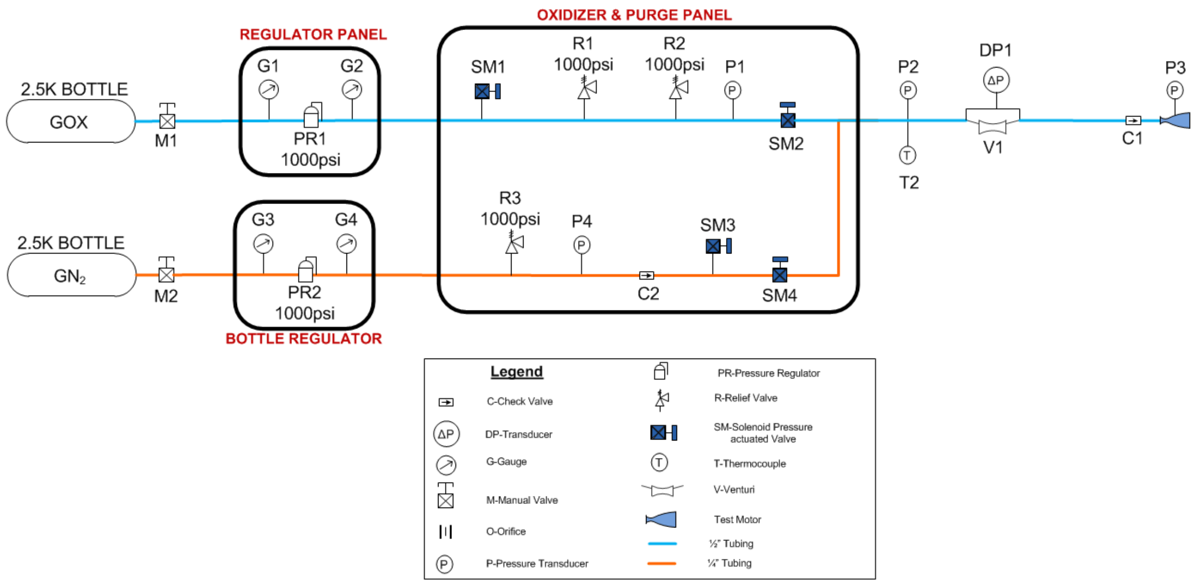

Figure 4: Overview of the Hybrid Facility.

The instrumentation in the system is minimal, with plans to expand capability in the future. The current system enables monitoring of the inlet line pressures (P1 and P4), calculation of the oxidizer mass flow (P2, T2 and DP1) and the measurement of the chamber pressure. The latter will be the main performance metric. Future plans include expanding capabilities to measure the regression rate for more precise empirical constant determination. (instead of calculating it) and to take temperature measurements on the outside of the combustion chamber case. The heavyweight design may prohibit useful data for the external temperature measurements. However, the potential of heat soak back, especially from the nozzle area, will eventually need to be answered if long periods in between burns are desired (e.g. coasts). The data is taken at 1 kHz using a LabVIEW setup.

B. Combustion Chamber

The combustion chamber is shown in Figure 5. It was designed to be modular, so that the main tube can be removed and replaced with shorter tubes (or longer if desired). Three main tubes have already been fabricated, allowing for L/D's of 2, 4, and 6 to be tested. The longest of the three is shown below. These L/D's are all on the short end of the spectrum for what has been suggested to give optimal performance. However, they were selected to minimize the length of the motor, which is believed to be a constraint on the final design to compete with solid motor designs and accommodate current Entry Descent and Landing (EDL) technology for Mars applications. It is also likely to be an issue for other in space propulsion applications. It is anticipated that mixing in the combustion chamber may need to be enhanced as it is shortened and modifications to the design will be made as necessary.

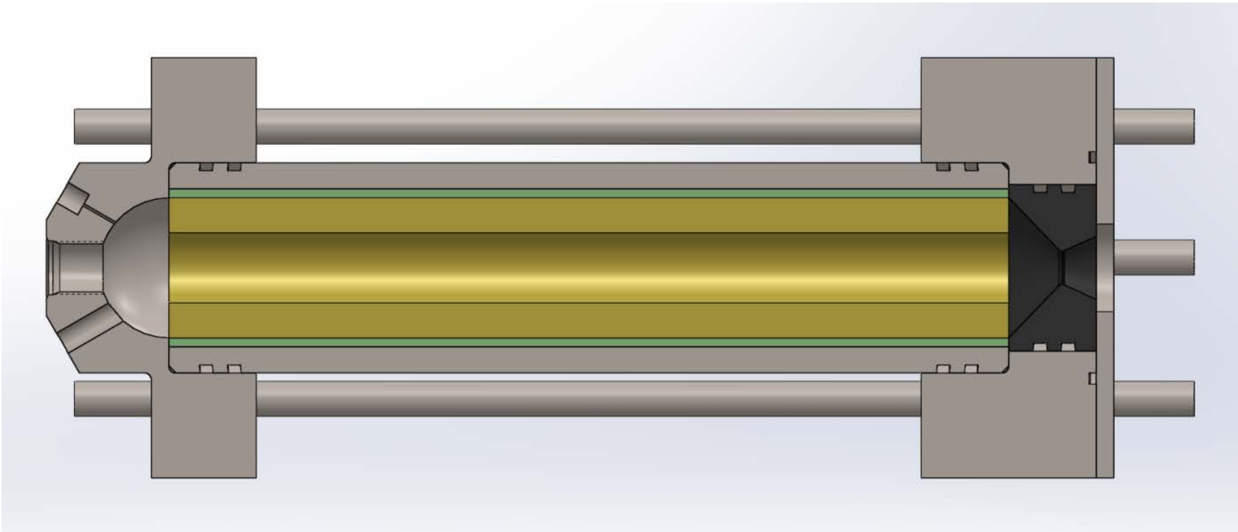

Figure 5: Combustion chamber design. The support structure is made of stainless steel and shown in grey. The insulator is a high temperature composite, shown in green and the nozzle is graphite, colored black above. The fuel grain is shown in gold.

The fuel grain is spun cast, neat paraffin, $C_{32}H_{66}$, with black dye (0.5% by mass). The blackener is added to prevent heat from radiating through the fuel grain during the burn. It has an outer diameter of five centimeters, an inner diameter of 2.5 cm and a L/D of 6. Fuel grains of this diameter can easily be cut to length by hand. Four fuel grains have been completed by graduate students at Stanford University and transferred to JPL. JPL is currently working towards the capability to spin cast fuel grains on lab.

C. Ignition

A custom ignition system has been developed for this system. NASA Standard Initiators (NSI's) with a bridgewire are loaded with a small amount (less than 600 mg) of a pyrotechnic mixture. The mixture is utilized to vaporize some of the fuel and initiate combustion. Combustion is self-sustaining after this initial heat input into the system. The port for the installation of the igniter is shown in the bottom left of Fig. 5. The igniter is shown in Fig. 6. It works in two stages. The burnwire is used to ignite the small amount of loose, initiation powder. The initiation mixture ignites the slower burning output grain, which in turn starts the hybrid motor. Many different initiation schemes are possible and it is not suggested that this method is any better than others. However, it is relatively simple and utilized familiar materials.

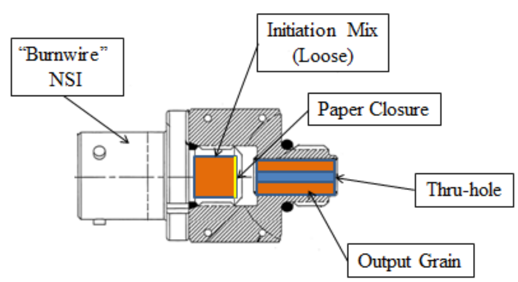

Figure 6: Two stage igniter for the hybrid motor. Pyrotechnic material is shown in orange.

D. Development and Testing Schedule

Development of this facility is planned to take place over two years, with a fully operational system able to complete preliminary tests near the end of the first year (shown in Fig. 7). The facility development is currently on schedule, with tests expected to commence in July 2014. The first iteration of the facility has been completely assembled and cold flows have been conducted. The bulk of the tests will be conducted next year (FY15) and presented separately.

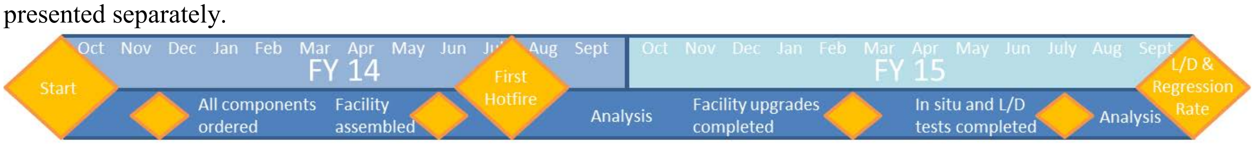

Figure 7: Two year development, test, and analysis schedule

E. Analysis Plans

Several variables are being tested using this facility. It is anticipated that the addition of CO and CO_2 to the gaseous O_2 will decrease the combustion efficiency. Once several tests have been completed, the chamber pressures and regression rate of the fuel can be compared. Chamber pressure gives insight into the combustion efficiency and will highlight any instabilities. The regression rate data will allow the MAV system to be modeled more accurately. Additionally, the effect of the length to diameter ratio of the fuel grain will be investigated. With this data, the possibility of utilizing the suggested propellant combination for a Mars Ascent Vehicle can be confirmed or an alternate can be researched.

V. Conclusion

Hybrid propulsion presents many benefits specifically for Mars based propulsion. The most convincing advantage is the capability to harvest some of the propellant *in situ*. However, several tests to quantify the performance of the propellant combination and the packaging constraints are necessary for hybrid propulsion to be a feasible alternative. Motivations for and the design of a test facility to retire risk of using hybrid motors for Mars applications has been presented. It is a flexible system, which is capable of testing multiple geometries and oxidizers to determine optimal performance for space propulsion (specifically Mars) applications. It is a necessary step toward the adoption of hybrid propulsion systems into trade studies for these applications, as it begins to clarify some of the performance parameters.

Acknowledgments

The research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The authors would like to thank the Jet Propulsion Laboratory for funding this research through its internal Research and Technology Development program. We also thank Elizabeth Jens and Jonah Zimmerman of Stanford University for the hard work they put into casting fuel grains for the first set of tests and Richard Webster, Alex Luna and William Gavid for assistance developing reliable igniters for this application.

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