A Hybrid Mars Ascent Vehicle Design and FY 2016 Technology Development

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Abstract—**Hybrid propulsion is currently favored for a Mars Ascent Vehicle (MAV) concept from a thermal performance and Gross Lift Off Mass standpoint. However, it is at a relatively low level of maturity compared to conventional propulsion options. Technology development efforts are currently underway to bring hybrid propulsion to a technology readiness level that would enable its infusion into potential Mars Sample Return. A new propellant combination is being considered for this design that has excellent low temperature behavior. Preliminary results of two ground test campaigns are currently underway to characterize this propellant combination. Hotfire testing is being carried out in parallel at Parabilis Space Technologies and Space Propulsion Group. In addition to the new propellant combination, several other technologies are being pursued for a potential hybrid MAV: hypergolic ignition and Liquid Injection Thrust Vector Control. Both of these technologies have been applied in other rocket applications, e.g. liquid propulsion commonly uses hypergolic propellants and missiles, such as the Minuteman II, have used LITVC in the past. Hypergolic ignition, when oxidizer and fuel combust upon contact, is highly desirable for multiple starts required by the MAV concept. Therefore, testing at Penn State and Purdue is being completed in this area. An updated hybrid propulsion system design for a Mars Ascent Vehicle concept based on JPL's current understanding of potential Mars Sample Return requirements will be presented, leveraging the advances in technology development as well as updated understanding of how requirements may evolve.**

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1. INTRODUCTION

Potential Mars Sample Return (MSR) has been identified as the highest priority by the Planetary Science Decadal Survey [1]. Such a mission would be made up of three major tasks: 1) collecting and caching Martian samples, 2) launching the samples into Mars orbit (Mars Ascent Vehicle) and 3) returning the samples to Earth. The Mars Ascent portion of MSR is considered especially challenging since a launch has not yet been achieved from another planet. Therefore, design and technology development in this area has been ongoing at the Jet Propulsion Laboratory (JPL).

Many configurations and options for the MAV have been studied in the past. The MAV presented here is assumed to be in a mobile configuration, housed within a launch tube on a rover that can rendezvous with the samples on the surface of Mars. The rover is constrained to be Mars Science Laboratory (MSL) sized to fit within the current Entry Descent and Landing (EDL) technology capabilities of the Sky Crane lander system. The Mars environment presents the biggest challenge to the MAV design. The Mars surface temperatures encompassing the range of interesting landing sites are extremely harsh, with diurnal variations in excess of 100 C. The hybrid design takes advantage of low temperature performance to minimize active heating and maximize rover science return prior to MAV launch.

The design presented here is the result of three years of systems studies. A broad survey of potential propulsion options was considered in the first year (FY 2014) including two stage solids, Single Stage To Orbit (SSTO) liquids, SSTO monopropellants, SSTO hybrids and several two stage configurations of liquids, hybrids and combinations. In FY 2015, major mission drivers were identified and the options were narrowed down to only include the most promising options [2]. The hybrid propulsion option had the lowest Gross Lift Off Mass (GLOM) and the lowest storage temperature, translating to minimized power requirements. While it was at a comparatively low Technology Readiness Level (TRL), it showed the most promise. With concurrence from the Mars Program Office, focus shifted to the hybrid MAV design and technology development in FY 2016. Therefore, this past year has been spent refining the potential hybrid design.

Hybrid rockets consist of an oxidizer and fuel, which are stored physically separated as well as in different phases. This dramatically increases safety and prevents an inadvertent mixture of oxidizer and fuel. Typically, hybrid rockets consist of a solid fuel and liquid (or gaseous) oxidizer. In this application, the fuel is stored inside the combustion chamber as a cylinder with a single, cylindrical port to allow oxidizer to enter and combustion to occur. The single port fuel cylinder will be referred to in this paper as a fuel core, though it is often called a fuel grain in literature. This distinction is made to disambiguate solid propellant grains, which have both fuel and oxidizer in a premixed state from hybrid fuel cores, which are inert and only contain solid fuel (wax in this case).

This paper is not intended to endorse a specific configuration or choice of MAV, but rather to establish an example for consideration with other aspects of potential MSR. Additionally, it presents an ongoing technology development program to bring the hybrid option to a TRL that will enable its consideration for the MAV in the future. This paper builds on past work for Mars Ascent Vehicle design and technology development. Section 2 describes the current conceptual hybrid design and Section 3 presents the propellant combination developed and tested specifically for Mars storage temperatures. Section 4 discusses hypergolic ignition to facilitate multiple motor starts. Key challenges and risks that remain for this design are discussed in Section 6. Potential plans for a terrestrial demonstrator are presented in Section 7. Section 8 provides a summary.

2. Conceptual Hybrid Mars Ascent Vehicle Design

The hybrid MAV delivers approximately 800,000 N-s of impulse at near 7000 N of thrust. The propellant combination is the wax-based SP7 fuel, developed for this application by Space Propulsion Group, with Mixed Oxides of Nitrogen (MON) oxidizer. The propellant selection will be discussed in more detail in section 3.

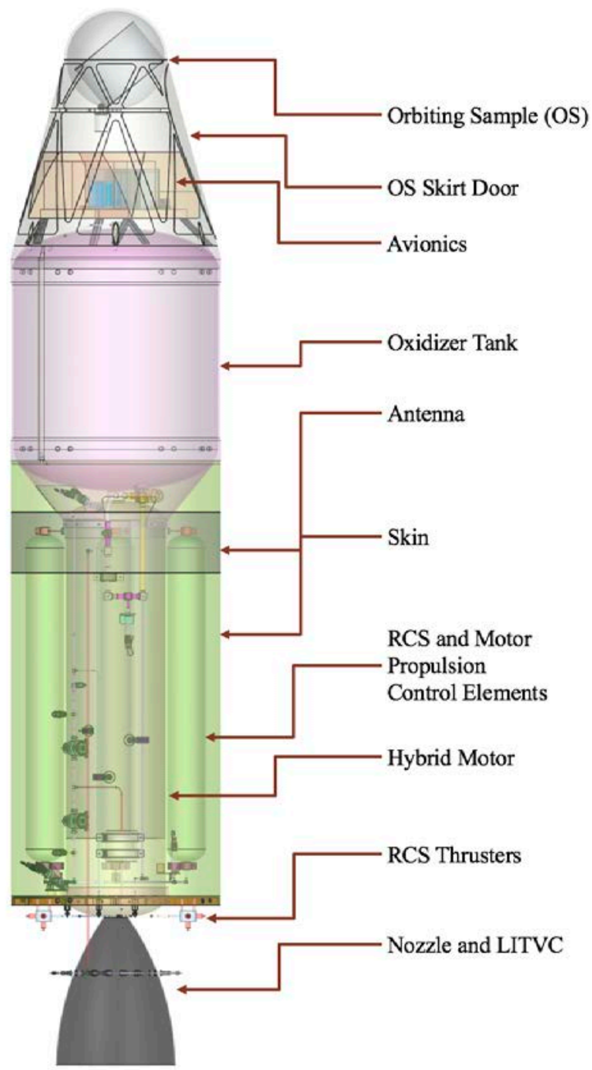

Figure 1. Pressure Fed Hybrid Mars Ascent Vehicle Concept

The hybrid Mars Ascent Vehicle concept is presented in Figure 1. This design has evolved substantially over the course of this work. The current iteration has a GLOM of 346 kg, an outer diameter of 57 cm (22.4 inches) and length of 2.9 m. For previous iterations see References [3,4]. There has been a steady mass increase due to increase payload size, margin policy and higher orbit. The current target orbit is circular at 479 km and 92.7° inclination. This orbit requires substantially more propellant to achieve than previous targets. However, it represents a good bounding case.

The payload is called the Orbiting Sample (OS). The OS protects and contains Martian samples and is housed at the nose of the vehicle. The OS is currently assumed to be loaded into the MAV through the OS skirt door. The OS and upper structure of the MAV have an 18 kg mass allocation in this design. The avionics and temperature sensitive elements are housed just below the payload. This section of the MAV must be thermally insulated and controlled to remain above -40 C.

Continuing aft along the rocket, the oxidizer tank is shown in pink. The oxidizer tank is a lightweight Composite Overwrap Pressure Vessel (COPV) to reduce mass. The composite combustion chamber or hybrid motor is shown in blue, directly below the oxidizer tank, in Figure 1. The fuel core would be cast first and then the insulator and load bearing structure would be wrapped directly onto the fuel core. The propulsion components and pressurant/RCS helium tanks encircle the motor case. Mounts for the components will be wrapped directly into the case as it is being built up. A trade is ongoing to determine if Titanium might be a better material for the motor case. Drivers in this trade include ease of mounting components CTE of the case material compared to SP7 and mass.

The aft deck is shown in orange in Figure 1. It is designed to support the MAV once erected on the launch tube. Fill and drain valves, the pyro valve and the RCS thrusters are mounted to the base of the aft deck for late access and maximum moment arm respectively. Thrust vector control is achieved through injecting liquid oxidizer into the expanding, supersonic portion of the nozzle. This is called Liquid Injection Thrust Vector Control (LITVC). Four ports, separated at 90°, direct oxidizer into the nozzle for this purpose. Each port has two parallel valves enabling two discrete thrust deflections.

One of the benefits of the hybrid propulsion system is that its performance is only very weekly dependent on chamber pressure. This allows the use of chamber pressure as a design variable. A chamber pressure of 250 psi has been selected as a compromise between the use of pressurant and achieved specific impulse (nozzle size). This variable will be optimized with performance and nozzle geometry. The nozzle area ratio remains unchanged from previous iterations of the hybrid design at 40:1. The length of the nozzle is directly related to the chamber pressure and could be further optimized if the pressure is changed.

A major driver for this design is geometry. Since a mobile MAV was assumed as the most challenging case, the hybrid concept must fit into a MSL sized rover. This packaging constraint led to the desire for a high volumetric packing of the fuel. The fuel core is designed as a simple cylinder with a single, circular, center perforation. The outer diameter of the motor case is 28.5 cm. The inner to outer diameter ratio (a/b) of fuel core is three. A more typical a/b ratio is 2 for paraffin-based fuel. However, the increased strength of the SP7 enabled this more efficient fuel loading of 88%. The integrity of a 7 cm fuel grain with a/b = 3 was demonstrated at Space Propulsion Group this year. Demonstration in larger scale motors will be completed shortly.

The length to outer diameter ratio of the fuel core (L/D) is also three. The length of the fuel core, coupled with the oxidizer mass flow rate, determines the oxidizer to fuel ratio of the hybrid. An L/D of three does not give much length for the oxidizer and the fuel to mix, therefore length for a post combustion chamber is included in the design. The total length required of a stable, higher performance motor will be determined through the ongoing hotfire test campaign.

The entire rocket will be heated to -20 C immediately prior to launch. A predetermined launch temperature sets the oxidizer surface tension and viscosity, which are crucial for injector design. The preheat temperature was set because it is the highest temperature (plus margin) that the MAV is predicted to experience while on the surface of Mars. It guarantees that cooling of the MAV will not be necessary. Current design iterations suggest that this design temperature may be depressed further.

3. PROPELLANT COMBINATION

New propellant combination

A novel propellant combination has been developed for this application and was discussed in References [3-5]. The predicted low temperature behavior of paraffin [6] led to the consideration of other wax-based hybrid fuels. However, desire to allow the MAV to survive launch and the Mars environment with only passive insulation led to challenging temperature requirements, currently set to -67 C to 40 C (non-operational allowable flight temperatures). A small, passive CO_2 gap should provide sufficient insulation to damp out most of the diurnal temperature variations and keep the MAV within this range.

A new fuel, called SP7, was developed by the Space Propulsion Group for this application. This wax-based fuel is higher strength, has even more favorable low temperature performance, has an elevated melting temperature, near 100 C, and burns more slowly compared to paraffin. The MAV application actually benefits from lower thrust than would be provided from paraffin fuel. SP7 has been shown to burn about 60-70% as fast as paraffin with N_2O [4]. The regression rate of the fuel ($\dot{r}$) with MON3 is currently being refined; however, it seems to parallel the SP7/N2O regression rate fairly well. Most importantly, the n value in Equation 1, appears to be near 0.5.

$$\dot{r} = aG_{ox}^{n} \quad \textbf{(1)}$$

Where a and n are empirically derived constants and G_{ox} is the oxidizer mass flux a.k.a. the mass flow rate of oxidizer divided by the cross sectional area of the combustion chamber. This equation takes advantage of several approximations and is averaged over time. Having an exponent near 0.5 means there is very little shift in Oxidizer to Fuel ratio (O/F) over the course of the burn, which enables operation at near optimal O/F and increases performance. The target, average delivered Isp is 314 s with a 40:1 nozzle. This assumes a 95% efficiency.

Motor testing at two subcontractors was carried out to verify predicted internal ballistics and scaling laws. SPG has been firing small, 7 cm, subscale diameter fuel cores and Parabilis is testing 26 cm, full scale diameter fuel cores, see Figure 2. Eight successful tests of the 7 cm motor were completed at SPG and three very short tests of the 26 cm motor were completed at Parabilis. SPG provided the fuel cores for the full scale testing since they have more experience working with the fuel. (SPG developed it.) The initial burn rate models will be updated as more data is collected to reflect a burn rate information compiled from these tests.

Figure 2: Hotfire testing at SPG, 7 cm diameter motor (top) and Parabilis 26 cm diameter motor (bottom).

The selected oxidizer is MON30 which is comprised of 70% N_2O_4 and 30% NO. The NO depresses the freezing temperature of the MON to about -80 C [7]. It is possible that some active heating may be necessary to keep the oxidizer from freezing. This requirement is expected to derive thermal constrains on the hybrid propulsion system. The nitrogen tetroxide (N_2O_4 a.k.a. NTO) base is a commonly used, space storable oxidizer that is hypergolic with monomethyl hydrazine. Some amount of NO is typically used to reduce its corrosiveness. Higher percentages of NO, typically up to 25% have been used in liquid bipropellant rockets.

Initial models, based on small scale test data collected this year, indicate that the motor should be able to achieve and maintain a high specific impulse (Isp) throughout the burn even though there will be some decrease in thrust due nozzle regression and fuel area change, see Figure 3. As described above, an n exponent near 0.5 minimizes the O/F shift and associated loss in performance. A nozzle material combination that appears to have low erosion properties at the peak ISP and O/F has been identified. This material combination will be tested shortly.

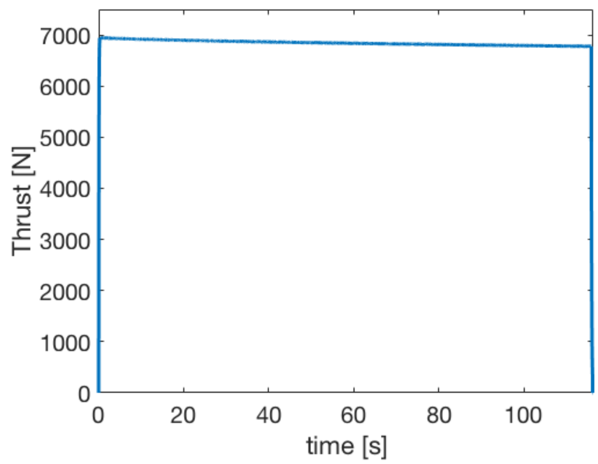

Figure 3. Motor Parameter Estimates

Another advantage of the hybrid propulsion system is its insensitivity to bulk temperature. Combustion theory predicts that solid rocket propellants are sensitive to the propellant mean bulk temperature as the heat has to penetrate the propellant to get it to burn. This has been included in development tests of most solid rockets, testing the high and lowest temperature limit of the motor. Hybrid combustion theory predicts that the bulk temperature of the fuel has a minimal influence on the performance, since the regression rate is driven by convection/radiation at the surface of the fuel. Last year, SPG completed testing with N_2O and SP7 chilled (between -40 C and -20 C) and, as expected, there were no noticeable changes in regression rate. The temperature is expected to have an impact on injector design. This lead to the requirement for a preheat temperature, described previously.

Thermal cycling for 5-7 cm thick slices of SP7 fuel core was completed at MSFC. The SP7 had one layer of fiberglass and two wraps of carbon fiber to simulate potential insulator and case material. Over 200 cycles were achieved, representing 50 diurnal cycles for each season and one Entry Descent and Landing (EDL) cycle. Spring and Fall were combined into a single profile. The most challenging thermal case was a single EDL cycle, which exposed the samples to +55 C temperatures then dropped the chamber to -115 C. The EDL and winter cycles were analyzed and presented in [5]. The spring/fall and summer cycles have now been analyzed and will be published this summer. The final results of the entire campaign were very promising. Samples were removed and inspected in between each season of cycling and only minor de-bonding was observed due to the expected coefficient of thermal expansion mismatch between the wax and the insulator/case. No large axial or radial cracking was evident. Alternative insulation to minimize the de-bonding is being investigated.

4. IGNITION

Previous studies suggested that hypergolic ignition would be the best option for the MAV [3]. Ignition of a hypergolic additive in a wax has been demonstrated previously with nitric acid [8]. Solid additives to the fuel are being investigated to find a candidate material that combusts upon contact with the MON. The current design assumes conventional ignition via a solid pyrogen for the first burn and hypergolic ignition for the second burn. SP7 (and wax in general) appears to act as an inhibitor for hypergolic reaction. Fully encapsulating the additive in SP7 protects and isolates it, thus facilitating ground handling and storage, but precluding hypergolic ignition for the first burn. The first burn exposes the additive for second and subsequent burns, making the hypergolic ignition possible. However, depending on the performance of the additives and ground handing/storage constraints, it may be possible to utilize hypergolic ignition for both burns.

Two universities were funded to complete hypergolic testing of solid materials with MON3 oxidizer: Pennsylvania State University and Purdue University. Each university did a survey of potential hypergolic options for the MAV. The following were drivers for the hypergolic additives:

1. It was assumed that additive's reactivity with MON3 would correlate directly with reactivity with MON30. The difference between MON3 and MON30 is the percentage of NO added to the N_2O_4. In general, NO should be more reactive, so it is believed that if the additive reacts with MON3, it should also react with the MON30.

2. The additive must be solid

3. It must be hypergolic, not just reactive, with MON3 (ignition time as short as possible, target of less than 100 ms).

4. The additive should survive being heated to at least 100 C, to meet the desire to mix it with melted SP7. However, alternative processing methods may exist if this is not possible.

Both Penn State and Purdue were successful in identifying multiple solid additives that are hypergolic with MON3 through droplet testing. Penn State discovered four hypergolic additives that reacted in less than 100 ms and Purdue discovered six. In each case, a droplet of MON3 was released onto a small amount of the loose hypergolic material. The front runners for each university had ignition delays of less than 10 ms, see Figure 4. The top several candidate materials from each test campaign were then mixed with SP7. A hypergolic reaction has been achieved with high one of the materials mixed with SP7 so far. The loading levels for this reaction are higher than would be desired for uniform distribution throughout long sections of the fuel grain, but could be reasonable for short sections. Additionally, lower concentrations have not yet been tested. These tests will be completed shortly.

Figure 4. Top hypergolic candidate with MON3 discovered at Penn State (left) and Purdue (right).

The assumption that addives that react with MON3 also react with MON30 currently being confirmed through a second set of tests with MON25 to begin shortly at Purdue. MON25 is being used as a simulant for MON30 because is readily available, while MON30 needs to be custom made. Trends between MON3 and MON25/MON30 are expected to be similar since the latter just adds a slightly higher concentration of NO. Testing of the final additive will be completed with the correct oxidizer once selected to confirm the performance.

In parallel to the effort to identify hypergolic additives to the fuel, an alternative delivery method is being investigated through a SBIR. Instead of proposing to mix the solid hypergolic material into the SP7, the hypergolic additive could be introduced into the oxidizer directly, entrained by an inert gas [9]. The inert pressurant would also protect the particles from oxidation and hydrolysis during storage, both of which are issues with many of the candidate materials. The major challenge thus far has been caused by the binder (SP7) encapsulating the particles and inhibiting the reaction. This option would bypass that challenge completely. The focus of the study is on the injection of particles, not on hypergolic candidate materials. Candidates from Penn State and Purdue could be used with this injection mechanism.

5. Thrust Vector Control

Thrust vector control is necessary for guidance, navigation and control of the MAV. Liquid injection Thrust Vector Control was recommended by a study done at MSFC [4]. This option is thought to be lower mass than alternatives such as a trapped ball nozzle. LITVC has been used in missile designs such as for the second stage of the Minuteman II.

Proposed Design

An ascent trajectory simulation run has shown the MAV system can be stable with a low frequency control of the LITVC valves. The need for TVC is highest immediately after launch, where an angle of ±2° is required. After a relatively short period of time (on the order of 10 seconds), demand decreases to ±1° or less for the remainder of the flight. Valve types, actuator, injection combinations and operating schemes in order to inject the liquid are being examined. Eight, fast acting (<5 ms) valves are being assumed to act in pairs around the nozzle to create these deflections. Control and necessary cooling of these valves is currently being determined. A current allocation of 2.9 kg of oxidizer is being made for the LITVC. Further refinements in the control system and updates to match the hardware and injection characteristics should lead to lower oxidizer usage.

Technology Development

A technology development contract to design LITVC capable nozzles was carried out with Whittinghill Aerospace. Whittinghill is providing LITVC test data for rubber based hybrid motors with nitrous oxide to anchor a model for the SP7/MON design. They have predicted side Isp and estimated performance and oxidizer usage.

MSFC has also been developing a CFD model of a liquid injection thrust vector system. Initial runs were made with high pressure gas to simulate liquid injection. This was due to complications in getting the correct property tables for the MON. Also, there were runs made with liquid oxygen as a proxy oxidizer until the MON input table was ready. The reaction chemistry is now working and a bell nozzle is replacing the initial conical nozzle used in development. A series of input combinations will be run to get a further understanding of the trade space for the optimization of the LITVC system.

6. Key Challenges and Risks

Technology Readiness Level

The propellant combination is still at a relatively low TRL level. Eight successful tests with SP7/MON3 have been carried out so far. However, only a little more than half of the oxidizer mass flux regime has been investigated so far. The entire operating regime must be demonstrated to verify the burn rate. Additionally, the full scale tests were not run long enough to provide reliable regression rate data. Stable motor combustion for the full burn duration, at the 28.5 cm scale is required. This will be completed in FY 2017.

Hypergolic Ignition

Multiple ignitions present a challenge to the hybrid MAV. Research carried out this year resulted in several attractive, candidate hypergolic materials. The testing at Penn State resulted in four candidate materials and Purdue discovered six. Of these candidate materials, only one overlapped between the two Universities and two candidates rose to the top as front runners with less than 10 ms ignition delay (measured with a high speed camera). One of these candidates has been tested successfully in SP7. Future testing at Purdue in their 5 cm motor will determine their behavior in a more realistic environment.

Testing with MON3 instead of MON30

The test results presented here used MON3 instead of the design oxidizer: MON30. The Mars design case presented in Section 2 uses MON30. The development path uses MON3 for initial testing. Transition to MON30 testing will take place in 2019. MON3 can be more easily procured than MON30, which must be custom made. Additionally, MON3 can be used at ambient conditions on Earth, while MON30 would require temperature control to cool the system. Testing with MON30 during the initial technology development phase is prohibitive from a cost standpoint.

Initial testing with MON3 not only reduces costs, but presents a solution for the MAV if a Radioisotope Thermoelectric Generator (RTG) were to be adopted to power the mission instead of solar panels. No decision has been made favoring either power source, so the more challenging option: the solar panels, is being assumed for this study. The driver for using MON30 is its low freezing point and therefore compatibility with the Martian environment. The abundant waste heat from a RTG would mitigate this challenge and enable the use of MON3.

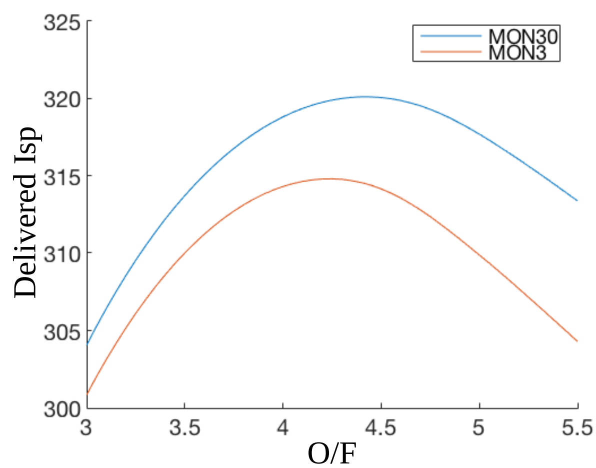

Figure 5. Motor performance with MON3 vs MON30

The increased performance of MON30 as compared to MON3 is shown in Figure 5. This is caused by the increased concentration of the more reactive NO. It has the additional benefit of decreased freezing temperature, described previously. The figure also shows that the optimal O/F ratio increases from MON3 to MON30. This means there will be slight differences in a system design with MON3 to that with MON30. Table 1 shows a comparison of hybrid MAV parameters when using MON3 instead of MON30.

	Changes when moving from MON30 to MON 3
GLOM	0.58%
Thrust	0.44%
Isp	-0.35%
Useable Prop	0.73%
Average O/F	-5.56%
Fuel Core OD	-0.70%
Fuel Core L/D	4.83%
Motor Length	2.66%
Motor Mass	1.35%
Loaded Ox*	-0.09%
Loaded Fuel	4.63%
Ox Tank Length	-1.52%
Loaded He†	-1.26%

Table 1. Hybrid MAV concept with MON3 compared to MON30

MON with high concentrations of NO, such as MON30 has been considered challenging for operation in a liquid bipropellant engine. MON30 has a considerably higher vapor pressure than MON3 at ambient conditions. Injector design to combine both fuel and oxidizer in the liquid phase through impinging pairs is quite difficult, since the oxidizer can flash to vapor if it is not chilled. This phenomenon is actually beneficial for the hybrid design, where mixing occurs in the gas phase. Injector design typically focuses on atomizing and/or vaporizing the oxidizer to the greatest extent possible.

7. Future Work & Terrestrial Demo

Future Work

The finalized design in this paper was presented for a Point of Departure Review in December, 2016. A specification for a Mars MAV design is being developed to enable design and testing of a flight-like motor with MON3 based on this work. This is the full scale testing described earlier.

Purdue will continue work with hypergolic additives. They have capability to test with MON25 and will test the best candidates identified in the first round of testing. This includes those discovered by Penn State. They will also mix the additives into fuel cores and test them in their 5 cm hybrid motor to achieve a more realistic test configuration.

Terrestrial Demonstrator

A terrestrial demonstrator is currently in the planning stages for an Earth-based launch of a representative hybrid MAV before the end of the decade. In FY 2017 several vendors are expected to compete to build a MAV terrestrial demonstrator. The meaning of "representative" is still being defined, since not all Mars conditions and/or parameters can be matched from an Earth-based launch. The design will use MON3 instead of MON30 to minimize costs. This demonstrator would be an overall system test that would increase the TRL of hybrid propulsion and potentially enable its selection for a Mars application.

8. Summary

A hybrid propulsion concept for a MAV was presented. Great strides have been made in the hybrid propulsion technology necessary to achieve this design. Major technology developments include: hotfire testing of the propellant combination, determining hypergolic additives for ignition and completing a preliminary LITVC design. This design is being further refined and will potentially be used as a basis for design of a terrestrial MAV demonstrator. The terrestrial demo has a target launch date of 2019.

Acknowledgements

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* Includes LITVC allocation

† Includes RCS allocation

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BIOGRAPHY

Ashley Chandler Karp *earned a Ph.D. in Aeronautics and Astronautics from Stanford University in 2012. She also received a B.A. in Astrophysics, Physics and Political Science from the University of California, Berkeley in 2005. She is currently a Propulsion Engineer at JPL. She is heavily involved in Mars Sample Return technology (especially the Mars Ascent Vehicle). She is the PI for JPL's Hybrid Propulsion Test Facility. She also works on the Mars 2020 propulsion system and has been involved with many mission concept studies. She is the Vice-Chair of the AIAA Hybrid Rocket Propulsion Technical Committee.*

Barry Nakazono *received a B.S. in Engineering from the California Institute of Technology in 1977. He has spent the last 25 years at JPL where he has been responsible for delivery of the Cassini Main Engine, MER cruise stage propulsion subsystems, DAWN xenon feed system, and SMAP propulsion subsystem. He started his career as a jet propulsion engineer at Boeing Aircraft Company and then moved to Hughes Aircraft Company where he learned to design, fabricate, and test spacecraft; five years in spacecraft propulsion and eight in system engineering.*

Robert Shotwell *received a B.S. in Aerospace Engineering (Magna Cum Laude) from Texas A&M University in 1995 and a MS in Astronautics (with Honors) from USC in 2003. He has been with JPL for more than 23 years. He is currently the Mars Program Chief Engineer for the Mars Exploration Program Office at JPL. Prior to this role, he served as the Program Engineer for the Advanced Optical Systems Program Office at JPL, preceded by serving as the Project Systems Engineer for the Phoenix Lander. He has led numerous micro & small sat efforts at JPL, as well as served in the Propulsion (both chemical and EP groups) developing, delivering and testing FHW for Mars Pathfinder, Deep Space 1, SIRTF and TelStar 8. He spent two years serving as the facility manager for the EP labs, as well as performing technology development and hardware testing for both JPL and external customers.*

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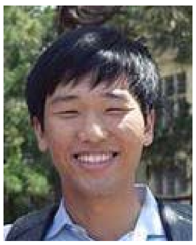

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Erich Brandeau *is a mechanical engineer in the Entry, Descent, and Landing and Formulation group at NASA's Jet Propulsion Laboratory. He worked on NASA's Low Density Supersonic Decelerator project after joint JPL in 2012 and is now working on the next Mars rover. He holds a B.S. degree in Mechanical Engineering the from Massachusetts Institute of Technology.*

George Story *received a B.S. in Aerospace Engineering from North Carolina State in 1988. He is a solid propulsion systems engineer at NASA's Marshall Space Flight Center, before that he was a Martin Marietta/Lockheed Martin. Over the years, he has contributed to multiple solid projects including support of the Space Shuttle Booster Separation Motor and Reusable Solid Rocket Motor, and hybrid rocket programs including Joint IRAD hybrid program and Hybrid Propulsion Demonstration Program. George is a AIAA senior member and a former HRTC chairman and one of many of the committee members that contributed to the HTRC's book 'Fundamentals of Hybrid Rocket Combustion and Propulsion'.*