Technology Development and Design of a Hybrid Mars Ascent Vehicle Concept

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Abstract— **Hybrid propulsion has been investigated as an enhancing technology for a Mars Ascent Vehicle (MAV) concept as part of potential Mars Sample Return (MSR) because of its high specific impulse, restartability, and the ability to operate and survive at extremely low temperatures. A new wax-based hybrid fuel formulation has been developed that could withstand the harsh and variable Mars environment protected solely by a minimal layer of passive insulation. This formulation could provide substantial energy savings for a notional lander and is critical for rover mobility. Preliminary thermal cycle testing has determined that the formulation can survive the expected temperature extremes and lifetime thermal testing is currently underway. A complete preliminary design using this new fuel formulation combined with a low temperature oxidizer such as Mixed Oxides of Nitrogen (MON30) is presented. Several key features associated with a complete hybrid MAV concept are investigated to determine their mission suitability (e.g. Thrust Vector Control and restartable ignition options). Potential challenges along a path towards developing such a system are outlined and future work is suggested as a means of technology maturation. The hybrid design presented here was the lowest Gross Lift Off Mass (GLOM) result of a 2015 Jet Propulsion Laboratory (JPL) led MAV concept study [1].**

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1. INTRODUCTION

The Planetary Science Decadal Survey has identified Mars Sample Return as a high priority future step for Mars Science [2]. As currently envisioned, potential Mars Sample Return (MSR) would consist of three major tasks: 1) sample acquisition and caching, 2) launching the samples from the surface of Mars to orbit around Mars and 3) returning them to Earth. A Mars Ascent Vehicle (MAV) would complete the second task. The architecture trade space for such a vehicle is still open, with the two most commonly discussed options being a platform/fetch rover combination and a mobile MAV. The platform/fetch rover option utilizes a stationary MAV and a small rover which completes the round trip journey to fetch the samples and return them to the MAV [3]. The latter option has a larger rover capable of transporting the MAV to the cached samples [4].

The results of a yearlong (fiscal year 2015) study of a hybrid MAV concept are presented here. Previous studies have included all types of propulsion systems, including solids, liquids, hybrids, and gelled [5]. The JPL led 2015 study evaluated Two Stage To Orbit (TSTO) solids, and Single Stage To Orbit (SSTO) liquid and hybrid options with the goal of fitting within a mobile MAV configuration. This study had significant input from Marshall Space Flight Center (MSFC) as well as Langley Research Center (LaRC). In addition to the overall propulsion system study, development efforts were undertaken to mature the technologies that would be required for this hybrid propulsion system, some of which will be presented here.

Hybrid rockets typically utilize a solid fuel and liquid oxidizer. This propulsion option is particularly interesting for a MAV because of its high performance, low operating and survival temperatures and capability for multiple restarts. However, the propellant combination presented here is at a relatively low Technology Readiness Level (TRL). It is at low TRL because the fuel was developed specifically for this application (about 20 hotfire tests with

N_2O were conducted this year) and because the chosen oxidizer requires special handling. However, these will both be matured over the coming year. Hybrid propulsion has been the focus of multiple MAV studies in the past five years due to their low temperature survivability [6, 7] and potential for in situ resource utilization [8, 9].

Key Goals for FY15

Four key goals were identified and met during the FY2015 study.

1) Develop a hybrid fuel that is flexible enough to withstand the wide range of temperatures experienced on Mars. Paraffin-based fuel was initially considered; however, a new wax-based fuel (SP7) was developed by Space Propulsion Group, Inc. (SPG) to survive the 50 C to -100 C requirements. SP7 has been used in this design, but fuel specifics are left to another paper.

2) Advance the low Technology Readiness Level (TRL) of the propellant combination. This includes (most importantly) testing selected propellant combination at operating temperature (currently established as -20 °C). Thermal cycling of the fuel to survival temperatures is also included.

3) Evaluate multi-start ignition options and identify path forward.

4) Evaluate Thrust Vector Control (TVC) options and identify path forward for nozzle design.

The following sections summarize the current potential hybrid MAV design and describe progress towards achieving the goals described above, including relevant technology development. The paper includes an introduction to Mars Sample Return, the MAV and the FY2015 JPL study. Section 2 describes the baseline hybrid design. Section 3 describes the thermal cycle test results of the new fuel formulation. Section 4 includes the trade study and results of the ignition system. Similarly, Section 5 includes the trade study and down selection of the thrust vector control system. Key challenges and risks are discussed in Section 6. Potential avenues for future work are presented in Section 7. Finally, a conclusion and summary are included in Section 8.

2. BASELINE HYBRID MAV

This 2015 study investigated solid, liquid and hybrid propulsion approaches parametrically, through six subcases. Each subcase had a different dry mass for telecommunications, avionics and a notional Orbiting Sample (OS). A point design for one of these subcases with a moderate mass payload (14 kg OS) is presented here, see Figure 1. This SSTO design takes advantage of the hybrid's ability to shutdown and restart. Most of the deltaV (nearly 80%) is imparted during the first burn, which is about 76 seconds long. After a little more than a 12-minute coast, a short second burn (10 seconds) is used to circularize the orbit in a near Hohmann transfer.

A hybrid MAV would utilize a newly developed high regression rate, wax-based fuel and Mixed Oxides of Nitrogen (MON-30) oxidizer. The high regression rate fuel enables the hybrid to be designed with a single, cylindrical port in the fuel grain. The mixture ratio of oxidizer to fuel is 4.56, producing a specific impulse of 317 s (assuming 95% efficiency). Similar and higher efficiencies have been demonstrated in large hybrid motors on Earth e.g. [10]. This design uses a pressure fed system with high pressure gaseous helium. The Gross Lift Off Mass (GLOM) of this point design is 219 kg with a total system length of 2.89 m.

The chamber pressure was selected to be 250 psia. Unlike solid rockets, a hybrid rocket's fuel regression rate is not a strong function of chamber pressure. Hybrids thus have the design flexibility to optimize chamber pressure with regard to overall system design. From a propellant combination standpoint, the only issue is to ensure that the chamber pressure stays above the supercritical pressure of the fuel. In this case, paraffin is used to provide a conservative lower bound of 97 psi [11]. The supercritical pressure of SP7 should be lower than that of paraffin.

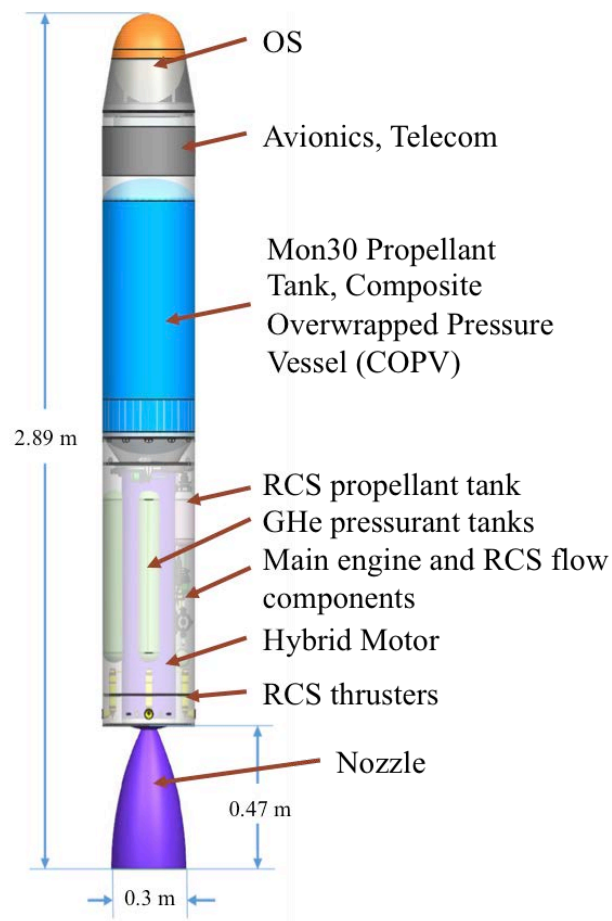

Figure 1. Baseline Hybrid MAV Design.

System optimization of chamber pressure (in either direction) is possible. For example, a chamber pressure of 160 psia lowers the total GLOM by 2.7% but increases stack height by 8 cm. Increasing the chamber pressure to 400 psia increases GLOM by 5.8%, but decreases stack height by 2 cm. A titanium combustion chamber with minimum manufacturing wall thickness can support a chamber pressure of at least 400 psi. Therefore, the decrease or increase in GLOM would primarily be due to the decreased or increased helium required by the oxidizer feed system. This variable could be tuned as the system design matures.

A cold gas Reaction Control System (RCS) operates off of separate gaseous N_2 tanks. The current design employs eight thrusters (four at 22 N and four at 5 N) predominately driven by dynamic pressure loading at the end of the first burn and coast period to maintain attitude control. There is potential to decrease the number of thrusters required to six, based on the results of a higher fidelity (6 degree of freedom (DOF) instead of 3 DOF) simulation. The thrusters would be located at the aft end of the combustion chamber, maximizing the moment arm and utilizing otherwise empty space around the motor. Having a completely separate GN_2 RCS sub-system produced the minimum GLOM in our study. This was traded against a dual use He RCS/pressurization system (utilizing a common tank).

Hybrid motor combustion stability depends on oxidizer mass flux (oxidizer mass flow rate divided by cross-sectional area of the fuel grain port) at end of burn, resulting in a minimum recommended thrust based on the fuel regression rate and fuel grain geometry. The high regression rate, wax based fuel presented here, would employ a single cylindrical port. For the purposes of our study, the thrust of the notional hybrid MAV has been set to this minimum to optimize GLOM and trajectory (5700 N in this reference case). Complete optimization of thrust/trajectory, nozzle expansion ratio, chamber pressure and stack height would be performed as this option matures.

3. PRELIMINARY THERMAL TESTING

One of the driving motivations for a Hybrid MAV is to provide a design that could inherently withstand cold and variable temperatures and therefore require significantly less electrical survival heating than other higher heritage propulsion systems. Table 1 compares the expected minimum allowable temperature for solid, bipropellant and hybrid propulsion systems. The allowable temperatures here assume a 10 °C margin on freezing (or tested capability in the case of the hybrid fuel). The bipropellant system being considered for the MAV concept uses Monomethyl Hydrazine (MMH) and MON25, since the oxidizer has similar temperature capability to the fuel. The hybrid fuel has been tested to temperatures down to -110 °C for the fuel, suggesting an allowable temperature of -100 °C for the fuel and -71 °C for MON-30 oxidizer. There is also potential to use MON-30 in the bipropellant case if lower temperature performance of the oxidizer becomes advantageous.

Table 1: Expected minimum temperature for different propulsion systems

Propulsion Type	Minimum Allowable Flight Temperature (°C)
Solid	- 40 °C
Bi-Propellant	-42 °C (fuel) / -45 °C (oxidizer)
Hybrid	- 100 °C (fuel) / -71 °C (oxidizer)

The lower allowable temperature of Hybrid rockets is important on Mars, where nighttime atmospheric temperatures for the ±30° latitude range are as low as -78 °C in the summer and -120 °C in the winter [12]. Martian winter is the worst-case cold (WCC) environment that a MAV would see throughout its lifetime. Hybrids are a compelling option for a MAV due to their capability for storage at low temperature and corresponding reduced survival energy requirement.

A MAV would need to survive more than just cold Martian winters. It would also need to withstand spacecraft integration and transportation on Earth, the launch environment, as well as a 6-10 month cruise to Mars followed by Entry, Descent, and Landing (EDL). Of all these phases, the worst-case hot (WCH) environment would likely be encountered on Earth or early in the cruise phase at a solar distance of about 1 A.U. The notional MAV could easily be exposed to ambient Earth temperatures of +30 to +40 °C. Cruise temperatures could be maintained below +40 °C without much difficulty, but some margin is needed so a temperature capability of +50 °C is desired.

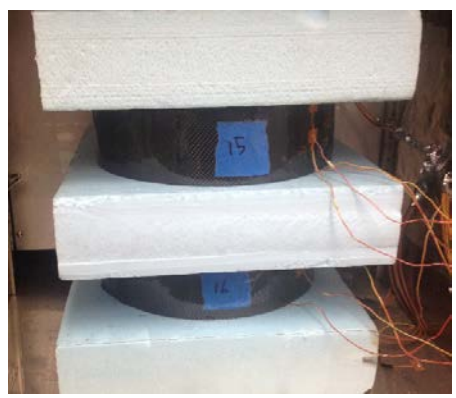

Figure 2. The Fuel Samples were sandwiched between Styrofoam insulation and were instrumented with thermocouples at the inside and outside diameter of the samples.

In order to test the temperature capabilities of the newly developed hybrid fuel, a preliminary testing program was performed at JPL in ambient (1 atm) pressure conditions. Two fuel samples were studied in these tests: one virgin SP7 sample and one metalized SP7 sample, with 20% aluminum by mass. The fuel samples were annular cylinders of the same diameter scale as the expected flight article. They

were sandwiched between Styrofoam insulation with thermocouples instrumenting the inner and outer diameter of the samples, as shown in Figure 2. There were a total of four thermocouples per sample: one at the centerline of the fuel and one at the edge of the fuel slice on both the interior and exterior surface of the grain. The insulation ensured that the primary thermal gradient was in the radial direction, and also made it simple to determine that the entire sample reached a cold temperature, since the inner diameter lags in temperature more than any other part of the sample.

Temperature cycling of the samples was performed over an increasing range in order to determine the capabilities of the samples. First, three cycles were performed from +50 °C to -70 °C, followed by an inspection, three more cycles from +50 °C to -90 °C, followed by an inspection, and finally three more cycles from +50 °C to -110 °C. Chamber ramp rates were set as slow as possible (6 °C/hour) in order to make the temperature the only variable of interest by reducing gradient or ramp rate effects, and keep the cost of the test within reason. A 12 hour soak and a 3 to 5 °C overshoot were used at the temperature extremes of each cycle to ensure all of the low conductivity fuel reached the temperature extremes. A chart of the inner and outer diameter fuel temperature for the most severe temperature cycle (+50 °C to -110 °C) is shown in Figure 3. The anomaly in the middle of the third cycle was the result of a temporary power failure caused by a rare Southern California thunderstorm.

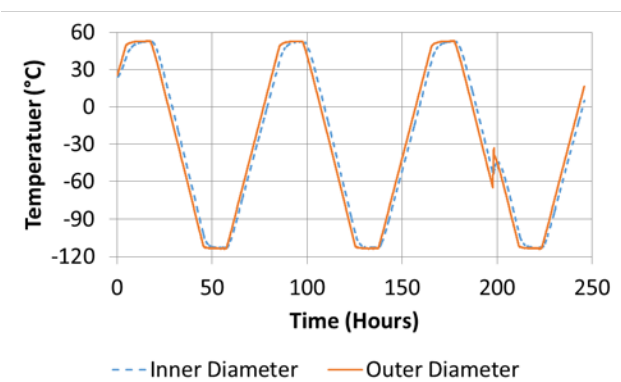

Figure 3. Temperature cycling profile for the most severe temperature cycle (+50 °C to -110 °C)

With the exception of a controller error during the +50 °C to -90 °C cycling that caused one of the samples to crack, all of the samples survived the temperature cycling. This particular instance led to an over test (e.g. unrealistically high temperature ramp rate). In all of the samples, delamination between fuel grain and insulation was observed. This delamination is shown in Figure 4 and is believed to be acceptable as long as oxidizer flow is not allowed through the gaps. It is believe that a small amount of delamination, as observed here, will be tolerable. This is a subject recommended for further study.

Figure 4. Delamination between fuel grain and insulation observed in both samples, which underwent the most severe temperature cycling (+50 °C to -110 °C).

Additional testing of SP7 samples, which is designed to test their ability to survive thermal gradients, temperature ramp rates, and fatigue cycling, is currently in progress at MSFC. Both aluminized and virgin SP7 samples will be exposed to approximately 200 Mars representative thermal cycles.

4. IGNITION

The ignition of a hybrid rocket is not as well modeled as a solid or liquid ignition systems. Conceptually, the ignition of a hybrid is fairly simple: get enough flame in the motor port to heat a portion of the surface and generate fuel vapor. The incoming oxidizer mixes with the fuel vapor and initiates motor combustion. This, in turn, heats up the rest of the motor port and the combustion is self-sustaining. It is quite simple in theory, but is one of the aspects of hybrid propulsion that has been studied the least in hybrid literature.

The ignition option space evaluated included various ignition options used to ignite liquid, solid and hybrid motors plus potential combinations of the options. Several of these options were determined to warrant further investigation and traded against each other.

The most conventional approach is to include multiple solid igniters (either pyrotechnic or pyrogen). These have been used many times for hybrid ignition (and reignition), including by the authors.

Electronically controlled propellant is a recently developed technology that shows great promise as an ignition (and a propulsion) system [13, 14]. Combustion occurs when an electrical current is run through the grain and quenches when electricity is no longer supplied (though there is a critical pressure above which the combustion is not easily quenched.)

Hybrid heater motors, a small hybrid lighting a larger hybrid, have been used to ignite a 250,000 lbf thrust hybrid, however they themselves need a restartable ignition source. This technique has not been proven with the desired fuel for this application.

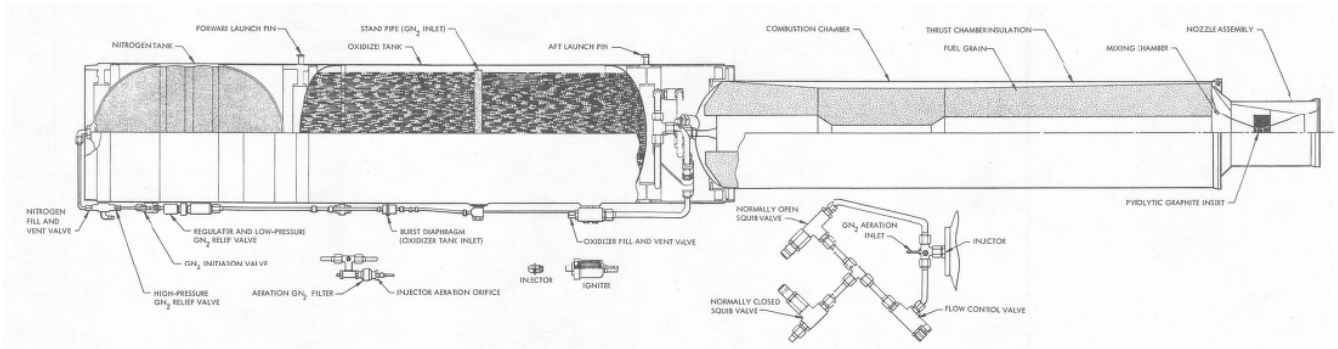

Figure 5. Sandpiper hybrid propulsion system (PMMA/MON25) [20]

A possible hybrid heater motor ignition source is an electric Arc Discharge Igniter [15], which uses the properties of printed Acrylonitrile Butadiene Styrene (ABS) to conduct electricity and form gaseous fuel and hot spot, which ignites with the oxidizer.

Hypergolic ignition is another hybrid ignition option, which has been getting some attention over the years [16, 17, 18]. Three hypergolic options exist. The first is to mix an additive into the fuel grain that is hypergolic with the oxidizer (MON30). The alternatives are to have a hypergolic fuel heater motor or a hypergolic liquid combination injected into motor. Both of these later options require pressure vessels and plumbing outside of the main combustion chamber, so are not ideal.

A first light technique could be a wired bag igniter, which has been used on ground-based solids, with the ignition voltage from the rover thru the nozzle and less penetrations thru the motor case. This could be used as a first ignition on any of the other igniter schemes, but a secondary system would be required for the circularization burn.

Hybrids lag other propulsion systems in flight experience, but they have been demonstrated in many applications, with one sharing similarities to the MAV concept. The Sandpiper target drone was developed in the 1960s. It had MON25 as the oxidizer, high and low thrust settings, a long burn time and was tested over a wide range of environmental conditions [19, 20, 21]. The Sandpiper is a major data point for this report. It used a small solid rocket motor to ignite the hybrid grain. See Figure 5.

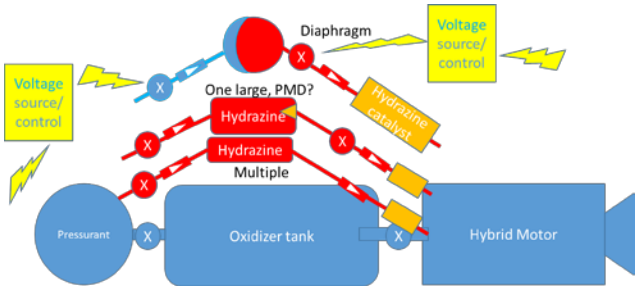

Figure 6. Hypergolic ignition system concept schematic

For the purpose of trying to ignite a MAV, it is assumed that both the surface of the fuel grain as well as the incoming oxidizer are cold. As such, ignition energies that may work at room temperature may be insufficient for this application. However, Sandpiper had tested motor ignition at – 65 °F (-54 °C). So, the low temperature data is used as the basis for this analysis and to size the ignition devices.

An analysis of the various ignition systems has been performed. For example, a hypergolic liquid ignition system could look similar to Figure 6. The components were roughly sized based on the Sandpiper ignition system.

The calculated TVC system masses were input into the trade study. In the trade study, the criteria were chosen to have the following weights: Mass: 40%, Volumetric Density: 25%, Complexity: 20% and Programmatic and Schedule Risk: 15%. Results of trade study in ranked order in Table 2. Hypergolic fuel additives came out as the top choice. However, it should be noted that this is a relatively low TRL option and would require development.

Table 2: Ignition Trade Study Results

Category	Features	Total Score
Hypergolic	Hypergolic additives in the fuel react with the oxidizer	4.7
Multiple Solid Igniters	Pyrotechnic or pyrogen	4.2
Pyrogen	Solid barrier/pulse igniter	4.2
Solid	Electrically controlled solid	3.9
Pyrotechnic	$BKNO_3$	3.6
Hypergolic	Hypergolic liquid injection igniter – requires additional liquid	3.3
Electric	Electric arc discharge igniter	3.0

The igniter trade study led to the following recommendations:

1. Ongoing testing of the SP7 fuel (currently at Purdue) must quantify the amount of heat needed to ignite the non-hypergolic hybrid fuel. Sizing studies done here were based on Sandpiper experience with PMMA/MON25 and Mg, which may or may not be enough energy with the current fuel/oxidizer combination at the low temperature range. Sizing of the various options could then be adjusted.

2. The top choice from the trade is incorporating additives into the fuel to make it hypergolic with the selected oxidizer (MON). This is the simplest concept from an operational standpoint. Hypergolic fuel testing and development should be undertaken in FY2016.

3. The uncertainty in the development of a successful hypergolic additive leads to the desire to study electrically controlled solids as an alternative.

4. Other options for hybrid ignition of a notional MAV are not as risky, such as pulsed solids (without any electrically controlled propellant) or hypergolic fluid injection. There are development tests that could be done in conjunction or in series with the ongoing hybrid testing.

5. Thrust Vector Control

The thrust vector control option space was evaluated and Liquid Injection Thrust Vector Control (LITVC) was recommended for the hybrid MAV concept application. In this case, oxidizer would be injected into the nozzle expansion area (downstream of the throat). This would deflect the flow exiting the nozzle and can provide a thrust vector of ± 5 degrees. LITVC takes advantage of the fact that the hybrid already has a liquid on board and has the added benefit, that there are no moving seals (e.g. flex seal or trapped ball). LITVC also leverages the tank and pressurization components of the hybrid propulsion system.

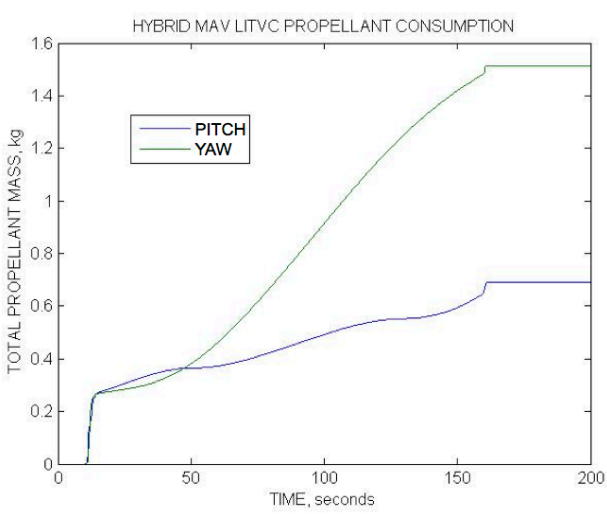

Figure 7: LITVC Propellant Consumption of the Hybrid MAV concept

A first order analysis of LITVC propellant consumption was performed based on N_2O_4 LITVC specific impulse data [22] applied to an example GNC simulation case of the MAV ascent trajectory. The propellant consumed by each axis, labeled "Pitch" and "Yaw", is shown in Figure 7 for the example case.

The Sandpiper hybrid (PMMA+Mg/MON25) reports also documented nozzle erosion information. For the final configurations tested and used in flights, it had minimal (nearly zero) nozzle erosion over a large range of conditions tested [20]. Early conceptual MAV hybrid fuel options included aluminum, which may have matched the Sandpiper data more closely. If minimal nozzle erosion can be achieved with a metalized fuel, as suggested by the Sandpiper data, it may be included in future hybrid MAV designs. The conventional, Sandpiper nozzle is shown in Figure 8.

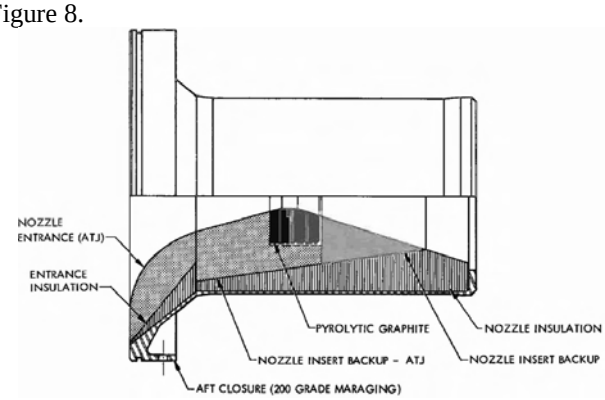

Figure 8: Sandpiper Nozzle [20]

6. Key Challenges and Risks

The inherent advantages and potential for high performance of the hybrid propulsion option must be traded against its comparatively low TRL. Risks associated with this technology are discussed in this section.

A new fuel formulation was developed to survive the Mars environment and testing of this fuel is still underway. Hotfire testing of the desired propellant combination had just started in the final months of FY15. About 20 tests of the fuel were completed with a different oxidizer to determine the regression rate compared to paraffin. No issues with combustion, stability or otherwise were observed. This risk is represented by "MON" in Figure 9. This risk can be retired with testing of the desired propellant combination in FY16.

Any MAV is likely to be a geometrically constrained system. The final design must fit into an EDL system to land on the surface Mars. The hybrid design offers some flexibility in packaging; however, there are also several constraints. For example, the fuel grain needs to be sized to give the proper oxidizer to fuel ratio and to allow for proper mixing to occur. This risk is represented by "PACK" in Figure 9. This risk can be addressed through design.

Nozzle erosion, survivability and TVC are a challenge to most propulsion systems, including the hybrid MAV concept. Hybrids have many options because they have a liquid on board that could be used for regenerative cooling of the nozzle or LITVC. The current design takes advantage of LITVC, but does not assume regenerative cooling. Therefore nozzle erosion with a conventional nozzle, such as the Sandpiper nozzle presented in Figure 8, is still a concern. Testing will be required to confirm the performance of both the nozzle and TVC system. This risk is identified by "TVC" in Figure 9.

The propellant combination will then need to be validated for use in the Mars environment (represented by "ME" in Figure 9). Additional risks due to the low and variable conditions include CTE mismatch of fuel and liner/case. Heat soak back during the ~12 minute coast before second start has the potential to melt some of the fuel. A preliminary thermal analysis at JPL predicted temperatures well below the fuel melting temperature, however, a complete study for the final design will be required, including test data to anchor the analysis.

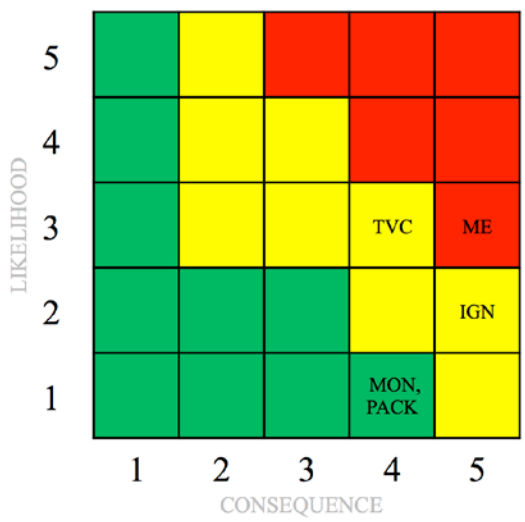

Figure 9: Risk Chart for a Hybrid MAV concept. Risks include the Mars Environment (ME), a Nozzle with thrust vector control (TVC), Ignition (IGN), low TRL of the propellant combination (MON) and packaging (PACK).

7. FUTURE WORK

There is a great deal of work that still needs to be completed before a hybrid MAV concept could be implemented. All of the relevant technology is at a relatively low TRL.

The new fuel, SP7, needs to be completely characterized. This includes determining all the material properties and temperature sensitivities, which is already in progress. A long-term test has been initiated at MSFC (100 day, 200 cycle) including EDL (one cycle) and seasonal tests (winter x 50, spring/fall x 100, summer x 50). This will provide confidence that the samples can survive the expected temperatures on Mars.

Results of the preliminary thermal cycling tests at JPL suggested that debonding between the fuel grain and case can occur during deep thermal cycling. A complete study of the best way to deal with the mismatch in coefficients of thermal expansion (CTE) between the insulator/case and fuel grain is necessary. Unlike a solid motor, it is believed that the combustion will not be negatively affected as long as oxidizer is not actively flowing through these potential gaps. This assumption, and the best means to mitigate gaps and oxidizer flow through this region, should be determined.

Initial hotfire testing with SP7 and both N_2O and MON3 oxidizers was conducted this year. This gave relative performance compared to paraffin fuel in the N_2O case and proof-of-concept that SP7 can be used with MON in the latter case. A compete regression rate versus oxidizer mass flux curve needs to be determined through testing with MON. This will enable accurate modeling and final ballistic design of the motor. This testing is planned for FY 16.

Ongoing testing of the SP7 fuels must quantify the amount of heat needed to ignite the hybrid. Sizing studies done at MSFC were based on Sandpiper experience with PMMA+Mg/MON25, which may or may not be enough energy with the current fuel/oxidizer combination, especially if it is to be ignited at low temperature. Sizing of the various options could then be adjusted. The top several candidates for ignition should be tested. Hypergolic ignition using additives in the fuel grain (the top rated restartable ignition option) can be tested on a small scale initially.

The current design assumes an eroding nozzle with a LITVC system. A LITVC nozzle for this hybrid configuration needs to be developed and tested with the proposed propellant combination to ensure performance can be achieved and that the throat erosion is within expectations. As mentioned previously, there is potential for regenerative nozzle using oxidizer if erosion becomes a problem. However, this will add mass and complexity to the system and although it has been done, there are risks associated with using the oxidizer to cool a nozzle made out of a potential fuel. Future work includes testing nozzle configurations with the current oxidizer/fuel configurations, analytical modeling and trimming the weight of the nozzle configuration.

A quick analysis suggested that it might be beneficial to use an electrical pump to pressurize the oxidizer (also considered for the liquid bi-propellant case in the JPL FY2015 study). A complete trade should be undertaken to determine if the pump would enhance the design enough to outweigh the risk associated with relatively low pump TRL.

8. SUMMARY

A conceptual design for a hybrid Mars Ascent Vehicle is presented as a potentially promising alternative to conventional propulsion systems. It was the lowest GLOM result from the JPL FY2015 study; however, it is also at the

lowest relative TRL. A great deal of technology development was undertaken as part of this year's effort. Some of the highlights are presented here, however, many of the details are left to future papers.

A new fuel was developed to survive operations on Earth, launch, transit to Mars and extended periods in the Martian environment. Thermal cycling of the fuel (both aluminized and virgin) over a range of +50 °C to -110 °C was successfully completed at JPL. The fuel grains themselves did not crack, however, some debonding between the grain and the insulator/case was observed. Debonding in a hybrid motor is not catastrophic (as it is for a solid motor) because little or no oxidizer exists at the grain boundary making a burn through unlikely. However, further testing to confirm this assumption will be required.

Trade studies to determine the best options for ignition and TVC were conducted. Hypergolic additives to the fuel grain were deemed the most favorable ignition technique and development in this area is recommended. Liquid injection thrust vector control was selected for future work. It has many benefits for the hybrid MAV concept, since the Mars environment presents a substantial challenge to many of the more common TVC systems (e.g. flex seals cannot operate at low temperatures).

While several technological challenges remain, significant development and risk mitigation has already been accomplished in this short time period. No insurmountable or significant performance degradations have been identified. Hybrid propulsion is a very promising option for a notional Mars Ascent Vehicle. The benefits that make it favorable for Mars could also make hybrid propulsion promising for other in-space applications.

ACKNOWLEDGEMENTS

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BIOGRAPHY

Ashley Chandler Karp *earned a Ph.D. in Aeronautics and Astronautics from Stanford University in 2012. She also has 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 and Sample Containment). She is the PI for JPL's Hybrid Propulsion Test Facility. She also works on the Mars 2020 propulsion system. She has also been involved with many mission concept studies. She is the newly elected Vice-Chair of the AIAA Hybrid Rocket Propulsion Technical Committee.*

Matthew Redmond *has been at the Jet Propulsion Laboratory for two years where he is a thermal engineer within the Thermal Technology and Fluid Systems Group. He is currently supporting the Mars 2020 rover design and MAV technology development, but worked on the Low Density Supersonic Decelerator (LDSD) project during his first year at JPL. Prior to JPL, Matthew received a B.S. and M.S in Mechanical Engineering from Georgia Tech, and performed microchannel research in Turkey as a Fulbright student grantee for one academic year.*

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David Vaughan *Group Supervisor for Propulsion and Flight Fluid Systems, Jet Propulsion Laboratory (JPL). Prior to working at JPL, David worked for Lockheed Martin for 21 years. He was involved in the External Tank program, Shuttle System Main Engine development and return to flight, Main Propulsion Test Article, Aluminum Lithium weld development and qualification for super lightweight tank, cryogenic propellant densification, Alternate turbo-pump development and engine and pressurization system*

modeling. Before coming to USA he worked for British Aerospace and Marconi Space and Defense systems on various spacecraft and weapon systems. David holds a MSc in High Speed Aeronautics from the von Karman Institute for Fluid Dynamics, Brussels, Belgium and a BSc in Mechanical Engineering from Bath University, England

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.*

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'.*

Dale Jackson *received a B.S. in Aerospace Engineering from the University of Tennessee, Knoxville in 2008. He has been with MSFC for 9 years. He has been working in the area of solid rocket motor design and ballistic analysis for the last 7 years. He has been the solid rocket motor igniter designer at MSFC for the last 5 years and has been operating the Hybrid Fuel and Inert Propellant mixing and casting lab since 2013.*

David Young *received a Ph.D. in Electrical Engineering from Louisiana State University, in 1997, where we later taught as an adjunct in the College of Engineering. He is a Thrust Vector Control Systems Engineer at NASA's Marshall Space Flight Center, providing engineering support services as a specialty subcontractor, and is an employee of Raytheon. He has supported the Constellation program and currently supports the Space Launch System and Commercial Crew programs. He has also supported electromechanical actuator development and has served as a mentor in the NASA Propulsion Academy at Marshall. Prior to this, David was the PI for a NASA/EPSCoR funded research project studying active control methods of unstable shock wave-boundary layer interaction, at Louisiana State University, where he began his research career studying computer architectures based on artificial neural networks. David is a member of IEEE and AIAA.*