A STORABLE, HYBRID MARS ASCENT VEHICLE TECHNOLOGY DEMONSTRATOR FOR THE 2020 LAUNCH OPPORTUNITY.

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Introduction: A Mars Sample Return campaign is the desired next step of the Mars scientific community [1]. One of the most technologically challenging components of the current mission architecture is the Mars Ascent Vehicle (MAV). The MAV is expected to survive in the harsh Martian environment for at least one year in order to coordinate with other aspects of the campaign. This has forced the current solid rocket baseline design to include a substantial insulation and thermal control system.

The hybrid MAV is expected to be a game-changing technology for a Mars ascent system. The propellant combination was selected to survive in the Mars atmosphere with little to no thermal conditioning and it can offer approximately 20% mass savings over the solid baseline design. However, there is a reasonably high level risk involved with selecting a new system for what will become a flagship mission. A Phoenix-class technology demonstrator mission is proposed to bridge this technology gap and enable a Mars Sample Return campaign from a propulsion standpoint.

This technology demonstrator mission fits into the Concepts and Approaches for Mars Exploration mid-to longer-term goals in Challenge Area 2: Safe and Accurate Landing Capabilities, Mars Ascent, and Innovative Exploration Approaches. It is suggested for the 2020 timeframe.

Mars Ascent Vehicle Mission Challenges: Constraints are placed on the MAV by the Martian environment and the entry, descent and landing (EDL) system used. The MAV will have to endure daily temperature swings of up to about 100C with an annual maximum of about 24C and minimum of -111C. (These values were calculated using the NASA Ames Global Climate model for Holden Crater, which was the MSL candidate site with the greatest variation in temperature and is assumed to be representative of a MSR landing site.)

Current MSR architecture places the MAV in a sky-crane type lander along with a fetch rover to retrieve the samples. The EDL system constrains both geometric and mass limits on the system. The baseline design is a solid rocket. However, the thrust of solid rockets is dependent on their temperature. Additionally, any cracking of the fuel grain could lead to a catastrophic failure.

The Hybrid Rocket Design: A hybrid rocket burns a solid fuel and liquid oxidizer. The thrust is not dependent on temperature and minimally dependent on pressure. They have not been used for previous in-space missions due to the low regression rate of classical hybrid fuels. This generally led to the use of multi-port fuel grains, which have many associated disadvantages. However, the discovery of liquefying hybrid fuels has increased the regression rate by 3-4 times, enabling the use of simple, single port designs [2]. More over, these fuels enjoy an extremely low, weak, glass transition temperature and are expected to survive the Martian environment with little to no thermal conditioning.

Propellant combination. An aluminized, paraffin-based fuel and Nytrox (N_2O pressurized with gaseous O_2) oxidizer propellant combination is presented. The glass transition temperature of paraffin-based fuel has been estimated to be in the range of -110C to -130C. The fuel is predominately crystalline, therefore the transition is weak and the fuel is expected to survive departures below this temperature. Cracks in this system, while undesirable, would not cause catastrophic failure. The complete lack of oxidizer in the fuel grain prohibits burning down the depth of the crack. The paraffin-based fuel matrix will include strength additives and 40% aluminum particles (2 micron) by mass.

The oxidizer, Nytrox [3], is composed of refrigerated mixtures of nitrous oxide and oxygen. It combines the high vapor pressure of dissolved oxygen with the high density of refrigerated nitrous oxide to produce a safe, non-toxic, self-pressurizing oxidizer with high density and good performance. Performance increases with decreasing temperature, making this mixture ideal for the cold Martian environment. A key feature of the design is that most of the pressurant (O_2) can be burned efficiently, thus greatly improving the structural mass fraction and the delivered Isp of the system. The N_2O and O_2 would be stored separately up to several sols from launch.

A systems study of a two-stage MAV has been presented in [4] and [5]. The mass of the hybrid MAV will be just over 273 kg assuming a mass contingency of 40% on all structural components as is recommended for non-heritage designs. The MAV is designed to reach a 500 km, circular orbit with a total ΔV of 4,375 m/sec. The nominal Isp for both stages is just over 300 s. The payload is taken to be 36 kg comprised of a 5 kg orbiting sample (OS) plus 31 kg, which includes the OS interface and separation mechanisms, avionics, telecommunications, cabling, thermal control, structure, a reaction control system, and a 3 kg contingency.

Technology Demonstrator Mission: A Phoenix-sized mission is proposed as a technology demonstrator for the hybrid MAV. A stationary platform will simulate the MSR conditions while reducing cost as compared to a mobile system. The MAV could be left in stow position for an extended period of time to confirm its capability to be stored without thermal conditioning. Once the desired length of time has elapsed, the MAV would be erected and launched into orbit around Mars. This demonstration would substantially reduce the risk associated with the MAV within the MSR architecture.

Scalability. While it would be desirable to test a full scale MAV, it is possible to test a subscale version for demonstration purposes. The main reason to scale the propulsion system would be to utilize the entry decent and landing (EDL) system used by the Phoenix mission. Using a previously tested EDL system reduces cost and risk associated with the mission. The Phoenix mission had a useful landed mass of 167 kg [6] with a science payload of 59 kg. Phoenix's science payload was not pushing the capabilities of the landing system and it is reasonable to assume that a larger payload could be achieved. Due to the uncertainty in the maximum landed mass, several options for a scaled down MAV are presented.

The preferred way to scale the MAV would be to reduce the payload of the system. If the payload is decreased by 20 kg (from 36 kg to 16 kg), a smaller (131 kg) two-stage hybrid MAV could still reach a 500 km. This assumes a 10% growth in the structural mass fraction from that required by the original design and still incorporates a 40% mass margin on all non-propellant masses. It can be imagined that the size of the MAV will continue to decrease as subsystems are removed from the payload. Designs with lower system masses will be considered. However, there is a point where the structural mass will take up too much of the system. This scaled design retains roughly the same stage mass ratios of the original design.

It is highly desirable to test a two stage MAV since stage separation and second stage ignition are important failure modes. However, if the payload is reduced to 1 kg (including only minimal communication capabilities), a single stage system that is approximately the size of the second stage of the scaled two-stage system can make the orbit. This option would still be able to confirm the ability of the hybrid to survive and launch in the harsh environmental conditions, but would not test the system reliability.

	Hybrid MAV [5]	Scaled Two-Stage MAV*	Scaled Single Stage MAV
Stage 1 Mass (kg)	145.89	77.2	51.02
Stage 1 Structural Fraction	0.189	0.208	0.215
Stage 1 Delivered Specific Impulse (s)	301	301	301
Stage 2 Mass (kg)	91.37	52.9	-
Stage 2 Structural Fraction	0.169	0.186	-
Stage 2 Delivered Specific Impulse (s)	303	303	-
Total Ideal ΔV (m/s)	4,375	4,375	4,375
Payload Mass (kg)	36	16	1
MAV Gross Lift Off Mass (kg)	273.3	130.1	52.02
Erector Mass (kg)	15	12	10
Total System Mass (kg)	288.3	142.1	62.02

Table 1: Hybrid Mars Ascent Vehicle Designs. The structural mass coefficient can be found by dividing the structural mass by the mass of the stage. *Desired design.

Cost. The mission is expected to be within the cost envelope of the Phoenix mission and lead to substantial cost savings for an eventual MSR campaign. The inherent safety of hybrid propulsion systems makes their development less costly than their solid or liquid counterparts. The EDL technology will be borrowed from a previously successful design. The choice to make this demonstrator a static platform greatly reduces the complications once on Mars. It is expected that the largest percentage of the cost will come from the launch vehicle.

References: [1] Committee on the Planetary Science Decadal Survey, National Research Council (2011) "Vision and Voyages for Planetary Science in the Decade 2013-2022." [2] Karabeyoglu, A., Cantwell, B.J and Altman, D. (2001) AIAA 2001-4503. [3] Karabeyoglu, M. A. (2009) AIAA 2009-4966. [4] Chandler, A. A., Cantwell, B. J., Hubbard, G. S. and Karabeyoglu, A. (2011) Acta Astronautica 69 1066-1072. [5] Chandler, A. A., Cantwell, B. J., Hubbard, G. S. and Karabeyoglu, A. (2010) AIAA 2010-6635. [6] Braun, R. D. and Manning, R. M. (2006) JPL TRS 05-3869.