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Optimization of the Mars Ascent Vehicle for Human Space Exploration

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This paper presents an analysis of the ascent stage to be used in future human missions to Mars, commonly known as the Mars Ascent Vehicle. The focus of the analysis is to minimize the initial mass, which must be delivered to low-Earth orbit to have a fully functioning Mars Ascent Vehicle. At the end of the surface mission, the Mars Ascent Vehicle takes the astronauts from the Martian surface to the orbiting Earth Return Vehicle. Systems analysis tools are used to help identify the most influential mass drivers. The main focus of the analysis is on the optimization of the propulsion system, including the type of propellant, oxidizer-to-fuel ratio, type of feed system, nozzle area ratio, and initial thrust-to-weight ratio of the system. A full factorial analysis shows that oxygen/hydrogen, pump-fed engines with a two-stage cryocooler are the best option for a mission scenario for propellant brought from Earth. If in-situ resource utilization is used to produce the necessary propellant on the Martian surface, oxygen/methane pump-fed engines become the best option in terms of minimizing the initial mass in low Earth orbit.

Nomenclature

C^*	=	characteristic velocity, m/s
CD	=	drag coefficient
CL	=	lift coefficient
Cm	=	moment coefficient
Days@Temp	=	days at temperature, days
ΔV	=	change in velocity, m/s
HOV	=	heat of vaporization, kJ/kg
IMLEO	=	initial mass in low Earth orbit, kg
ISP	=	specific impulse, s
L/D	=	chamber length to diameter ratio
LOX	=	liquid oxygen
m_{Boiloff}	=	mass of boil-off propellant, kg
m_{fluid}	=	mass of propellant, kg
O/F	=	oxidizer-to-fuel ratio
$\boldsymbol{\phi} q$	=	net heat flux, kJ/m^2
$q_{\mathrm{allowable}}$	=	allowable heat, kJ
q_{Total}	=	total heat, kJ
$\%R$	=	percent residual propellant
R^2	=	correlation coefficient
#tanks	=	number of tanks
SA	=	external surface area of propellant tanks, m^2
$T_{\mathrm{Allowable}}$	=	boiling point of propellant at the tank storage pressure, K
T_{Loaded}	=	boiling point of propellant at ambient Earth pressure, K
T/Wi	=	initial thrust-to-weight ratio

I. Introduction

IN JANUARY 2004, a new vision for space exploration (VSE) was announced for NASA that would return humans to the moon by 2020 in preparation for human exploration of Mars. Until recently the VSE effort has been focused primarily on lunar exploration missions, including the implementation of the lunar architecture from the Exploration System Architecture Study [1], and very little analysis has been performed within NASA on architectures or elements for human Mars missions over the past decade. This paper attempts to address key issues in the design of a major element of a human Mars architecture, the ascent vehicle, in preparation for more comprehensive Mars architecture design efforts and eventual implementation of human Mars missions as a part of the VSE.

Many years before the VSE, NASA had conducted studies to determine a feasible approach for the human exploration of Mars. In July 1997, the Reference Mission of the NASA Mars Exploration Study Team [2] was released, which will be referred to as Design Reference Mission 1 (DRM1) in this paper. This reference mission laid the foundation for future Mars exploration studies. DRM1 set out to 1) challenge the notion that human exploration of Mars is a 30-year program that will cost hundreds of billions of dollars, 2) challenge traditional technical obstacles associated with sending humans to Mars, and 3) identify relevant technology development and investment opportunities [2]. A single outpost would be developed on the Martian surface with crews of six staying for 500–600 days per mission. A concept of operations was detailed that launched three vehicles in each of four opportunities. Cargo vehicles would deliver Earth return vehicles (ERV) to Mars orbit, and unfueled Mars ascent vehicles (MAV), in-situ resource utilization (ISRU) modules, nuclear power plants, liquid hydrogen, surface habitat/laboratories, and consumables would be delivered to the surface. The first ISRU module would generate the necessary propellant for the MAV before the launch of the first human mission. The first crew would leave Earth during the second launch opportunity, making use of a fast transfer trajectory (∼180 days), before landing in a surface habitat similar to the predeployed habitat/laboratory. A common descent stage would be used for both the cargo and crewed cases. The descent

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stage would be capable of landing approximately 65 tonnes of cargo on top.

The MAV, as described in DRM1, would be composed of an ascent stage and a crew capsule. The MAV would be delivered dry, and subsequently fueled by the ISRU module. The study estimated that about 5600 m/s of change in velocity (ΔV) would be required to rendezvous and dock with the waiting ERV. Approximately 26 tonnes of liquid oxygen and methane (LOX/CH_4) would be necessary. The MAV was equipped with two RL-10 class engines modified to burn LOX/CH_4 with a specific impulse (ISP) of 379 s. The ascent capsule, which is on top of the ascent stage, would not require a heat shield due to the low heating experienced during ascent.

Shortly afterward in June 1998, NASA released the Reference Mission Version 3.0 [3], which will be referred to as DRM3. The main goal of this document was to remanifest the payload elements so that two smaller 80 metric ton class launch vehicles could be used instead of the much larger 200 to 225 metric ton class vehicle of DRM1. The mass statements of DRM1 were refined, and the initial habitat lander was eliminated from the concept of operations. The descent stage would be able to deliver approximately 40 metric tons of cargo to the surface. Another major modification of DRM3 was that the MAV and the descent stage would share common engines and propellant feed systems, thereby eliminating the need for a separate ascent propulsion system [3]. This propulsion system would include four RL-10 class engines modified to burn LOX/CH_4 at an oxidizer-to-fuel (O/F) ratio of 3.5. Each engine would have a chamber pressure of 4137 kPa, nozzle area ratio of 400, thrust of 66,723 N, an average ISP of 379 s, and be capable of throttling and gimbaling. Approximately 39 metric tons of propellant would be required to accomplish the necessary 5,625 m/s of ΔV to rendezvous and dock with the ERV. As in DRM1, the plan was to deliver the MAV dry and produce the fuel using the ISRU modules delivered on the descent stage.

Finally, in September 1999, a number of documents summarizing the results of studies using a nuclear thermal rocket bimodal propulsion system were released [4]. These documents (which will be referred to as DRM4) included mass breakdowns of the various components for a human mission to Mars. A mass breakdown from DRM4 for the MAV was used to validate the model developed for this paper, and the results of this validation will be shown later.

An analysis of all of these previous studies indicates that mission mass is a key cost and risk driver. Entry, descent, and landing approaches require low Mars entry mass to ensure feasibility. Total mission mass is a key cost driver due to the high cost of launch. It is essential to develop systems that are optimized for reduced mass. This paper attempts to demonstrate tools, methods, and systems that achieve that goal. This paper builds on the results of previous studies by applying state-of-the-art integrated tools and methods to the optimization of the ascent vehicle propulsion system. Parameters traded include: the type of propellant, oxidizer-to-fuel ratio, type of feed system, nozzle area ratio, and initial thrust-to-weight ratio of the system. In addition, the previous DRMs all assumed that the technology for ISRU would be tested and available for a human mission to Mars. However, it is also of interest to determine if the mission would be possible in a bring-your-own-propellant (BYOP) mission scenario, in which the propellant is brought from Earth and stored for the duration of the mission. This paper explores the possibility of a BYOP mission with and without the use of cryocoolers to reduce or prevent the boil-off of cryogenic propellants. These analysis results are expected to inform future human Mars architecture design efforts as a part of the current VSE previously discussed.

II. Systems Engineering Definition

A. Objectives

The main objective of this study is to minimize the initial mass in low Earth orbit (IMLEO) that must be delivered to have a fully functioning MAV by optimizing the propulsion system. The study focuses on a BYOP mission scenario, in which all necessary propellant is brought from Earth, but also explores the option of ISRU for the production of the required ascent propellant.

B. Concept of Operations

1. Bring-Your-Own-Propellant Case

During the first launch opportunity, a fully fueled MAV is delivered to the Martian surface. Before the subsequent launch opportunity, the system is checked to ensure no complications arose during transfer to Mars or landing. Approximately two years later, the astronauts arrive in a descent stage with a habitat during the second mission opportunity.

A second fully fueled MAV arrives on a cargo launch vehicle several months after the astronauts. This will serve as a redundant system. Five to six hundred days later, upon completion of the surface mission, the astronauts leave the surface habitat at the appropriate time to rendezvous with the orbiting ERV and enter the crew capsule mounted on top of the MAV. The main engines of the MAV ignite, separating from the descent stage, and the MAV follows the optimized trajectory to reach the appropriate orbit. The MAV performs any necessary orbit adjustments and then rendezvous and docks with the ERV, after which the crew transfers over to the ERV to complete the journey home. Depending on the on future exploration requirements, this mission could be repeated at multiple mission sites or be performed multiple times to the same site to support an eventual outpost.

2. In-Situ Resource Utilization Case

During the first mission opportunity, a dry MAV with an ISRU plant and hydrogen feedstock is delivered to the Martian surface. During the time until the subsequent launch opportunity, the ISRU plant produces the propellant necessary for the MAV to rendezvous and dock with the orbiting ERV. Upon completion of the propellant production, the ISRU plant can be used to produce caches of water, oxygen, or buffer gases for life support. Approximately two years later, the astronauts arrive during the second mission opportunity. A second dry MAV and ISRU plant arrives several months after the astronauts. The second ISRU plant begins production of propellant/caches. Approximately 500 days later, upon completion of the surface mission, the astronauts leave the surface habitat at the appropriate time to rendezvous with the orbiting ERV and enter the crew capsule mounted on top of the MAV. The main engines of the MAV ignite, separating from the descent stage, and the MAV follows the optimized trajectory to reach the appropriate circular orbit. The MAV then rendezvous and docks with the ERV, and the crew transfers over to the ERV to complete the journey home.

C. Quality Function Deployment

The quality function deployment (QFD) was used in this study to help identify which engineering characteristics have the greatest impact on the customer requirements. These engineering characteristics were selected based on their potential to impact the overall mission performance. For this study, the only requirement (shown on the left-hand side of the QFD) is to minimize the IMLEO of the MAV. The engineering characteristics are shown along the top of the QFD, and their impact on the customer requirement is rated on a nonlinear scale of 1–9, with 1 signifying a low correlation, 3 a medium correlation, and 9 a high correlation to the customer requirement. This nonlinear scale helps the key trades (the trades with the largest impact on the customer requirements) stand out among the other possible trades.

These correlations were determined based on a combination of expert elicitation and systems analysis. A QFD expert elicitation exercise was conducted with NASA engineers and Georgia Tech professors and students. Consensus correlation assessments were developed and entered into the QFD spread sheet. A top level sensitivity analysis was then performed using the MAV sizing model developed for this study to confirm that the selected parameters were indeed those to which the MAV ascent mass (and hence the IMLEO) were most sensitive.

As can be seen in Fig. 1, the most significant engineering characteristics from a mass standpoint are the chamber pressure, ΔV, engine type, nozzle area ratio, initial thrust-to-weight of the system (T/Wi), O/F ratio, propellant type, ISP, and whether the propellant is

			Propulsion										Power					Propellant				Mission
	Engine/Tank Support Structure Material	Thermal/Radiation/MMOD Protection	Chamber Pressure	Delta V	Engine Type (Pump or Pressure Fed)	Main Fuel Tanks and Feed/Fill/Drain System	Nozzle Area Ratio	Number of RCS Engines	Number of Main Engines	RCS Fuel Tanks and Feed/Fill/Drain System	Initial T/W of System	T/W of Main Engines	ISRU Production Power	Keep Alive Power	Control	Avionics	Environment	O/F Ratio	Propellant Type	Residuals	Specific Impulse	ISRU or BYOP
IMLEO, kg	3	1	9	9	9	3	9	1	3	1	9	3	3	1	1	1	1	9	9	1	9	9

Fig. 1 Quality function deployment for human mission to Mars.

brought to or produced on Mars. All of these engineering characteristics were optimized in this study with the exception of propellant source, which has been left for a future study.

D. Morphological Matrix

The QFD helped identify the engineering characteristics that have the largest impact on the customer requirements. The next systems tool used is a morphological matrix (shown in Fig. 2), which identifies the possible alternatives for each engineering characteristic under consideration. For example, the engine type can be either pressure-fed or pump-fed, and the baseline choice of pump-fed is highlighted in yellow. The ranges of values for chamber pressure and nozzle area ratio are dependent on the choice of engine type. The range of values of O/F ratio and the type of propellant tank insulation are dependent on the choice of propellant.

The baseline values are based on the propulsion system description and mass breakdown of DRM4 with the exception of propellant source. As mentioned in the introduction, the DRMs made use of ISRU modules to produce the MAV propellant. For the purpose of this study, the propellant is brought in all cases. If the various options for ISRU were included in the morphological matrix, the total number of combinations would increase dramatically from just over 18 million to over 2 billion combinations. A discussion on the potential benefits of ISRU is at the end of the paper, but a detailed investigation has been left for a future study.

Two other things to note about the morphological matrix (Fig. 2) are that the ISP and ΔV, although recognized as key trade areas, are not presented as characteristics with a range of alternatives. This is because the ISP will be determined based on the choice of propellant, engine type, chamber pressure, O/F ratio, and nozzle area ratio. In turn, the ΔV will be determined based on the initial system T/W ratio, the nozzle area ratio, and the ISP.

E. N^2 Simulation Model

The N^2 simulation model (shown in Fig. 3) presents the flow of information through the overall system model. Each bold box represents a different source of information or engineering tool, and the other boxes show the information that is passed between the tools. For instance, Excel will send the optimized ΔV to the exploration architecture model for in-space and Earth to orbit (EXAMINE), which will then send back the updated T/W and ISP to Excel. A description of the various simulation model tools follows.

III. Simulation Model Tools

A. Aerodynamic Analysis

The Aerodynamic Preliminary Analysis System (APAS) [5] is actually a combination of three programs. The first, APAS executive, is where the desired geometry is input and the analysis run is set up. The second program, Unified Distributed Panel code, computes the subsonic and supersonic aerodynamic analysis using linear theory

Characteristics			Alternatives								No. of Options
			1	2	3	4	5	6	7	8	
1	**Body**										
1.1	Primary Structure Material		Aluminum	Titanium	Composite	Hybrid					4
2	**Propellant Tanks**										
2.1	Main Prop Tank Material		Graphite Wrap-Al	Al 7075	Al-Li 8090	Titanium					4
2.2	Main Prop Tank Insulation		None	MLI	Cryocooler	Combination					4
2.2.1	Main Prop Tank Cryocooler		None	1-Stage *	2-Stage						3
3	**Main Propulsion System**										
3.1	Engine Type		Pressure-fed	Pump-fed							2
3.2.1	Chamber Pressure, kPa	(Pump-Fed)	3,447	4,137	4,629	5,220	5,811	6,402	6,993	7,584	8
3.2.2	Chamber Pressure, kPa	(Pressure-Fed)	1,034	1,108	1,182	1,293	1,330	1,404	1,477	1,551	8
3.3	Propellant		O2/H2	O2/CH4	O2/Ethanol	NTO/MMH					4
3.4	Propellant Source		Brought	ISRU	Combination						3
3.5.1	Oxidizer/Fuel Ratio	(O2/H2)	5.000	5.214	5.429	5.750	5.857	6.071	6.286	6.500	8
3.5.2	Oxidizer/Fuel Ratio	(O2/CH4)	3.000	3.086	3.171	3.257	3.343	3.429	3.514	3.600	8
3.5.3	Oxidizer/Fuel Ratio	(O2/Ethanol)	1.600	1.686	1.771	1.900	1.943	2.029	2.114	2.200	8
3.5.4	Oxidizer/Fuel Ratio	(NTO/MMH)	1.500	1.571	1.643	1.750	1.786	1.857	1.929	2.000	8
3.6.1	Nozzle Area Ratio	(Pump-Fed)	20	74	129	183	237	291	346	400	8
3.6.2	Nozzle Area Ratio	(Pressure-Fed)	20	75	129	183	237	291	346	400	8
3.7	Initial System T/W Ratio		1.20	1.45	1.54	1.71	1.89	2.06	2.23	2.40	8
									No. of Combinations:		18,874,368

Baseline * O2/H2

Fig. 2 Morphological matrix for bring-your-own-propellant mission scenario, with baseline values highlighted in yellow.

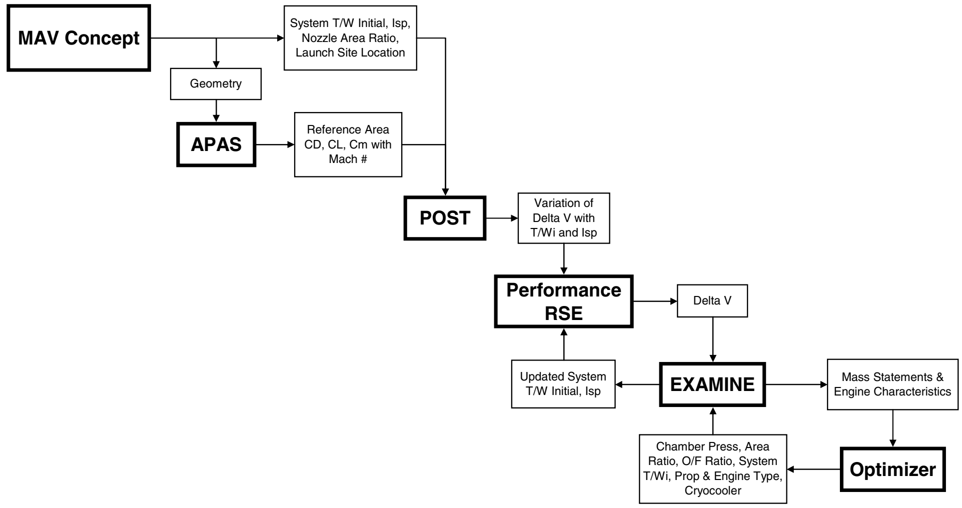

Fig. 3 N^2 diagram for the study. Information flows between the MAV concept and the five engineering tools: APAS, POST, Excel, EXAMINE, and JMP Optimizer.

and calculates the wave and base drag using modified flat plate theory. The third program, Hypersonic Arbitrary Body program, computes the hypersonic aerodynamics based on impact theory. These three programs are used to perform the full aerodynamic analysis for a given geometry. From this analysis, the drag, lift, and moment coefficients are obtained as they vary with Mach number and angle of attack.

There is no clear description of the geometry of the MAV as described in the design reference missions. The geometry input for APAS, shown in Fig. 4, was based on the MAV depiction of DRM1. The crew capsule sits on top of the MAV, which is composed of propellant tanks held in place by a tank support structure, and the four engines protruding from underneath the tanks. The geometry was scaled so that the diameter of the crew capsule is sufficient to locate six crew in a side-by-side arrangement (5.5 m), and the tanks and engines were approximated as a single cylinder.

B. Performance Analysis

The Program to Optimize Simulated Trajectories (POST) is described in the user manual as a generalized point mass, discrete parameter targeting and optimization program which provides the capability to target and optimize point mass trajectories for a powered or unpowered vehicle near an arbitrary rotating, oblate planet [6]. POST was used in this study to determine the amount of ΔV necessary to reach orbit while following an optimized trajectory for a range of initial system thrust-to-weight ratios and ISP values.

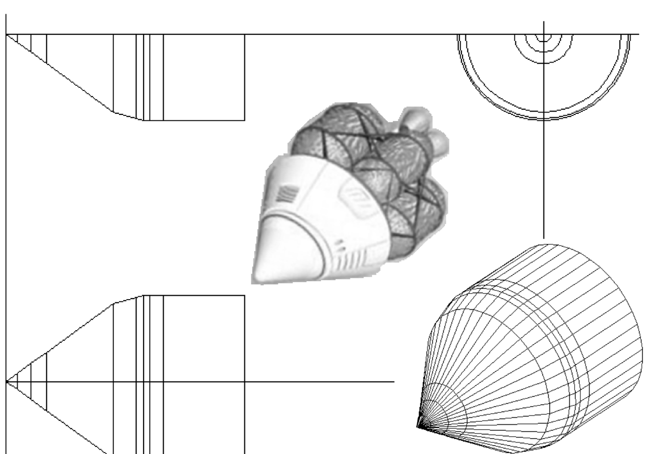

Fig. 4 Baseline Geometry for MAV from DRM1 and corresponding input for APAS.

The performance model used in this study was for a single-stage MAV to take off vertically from the equator and travel to a circular, polar orbit of 100 km altitude. The aerodynamic data from the APAS results were included, along with atmospheric density and pressure tables and a speed of sound table for Mars. The acceleration was limited to 4 times the gravity of Earth. The optimizer targeted the minimum mass consumed during ascent with a final relative velocity of 3.507 km/s, a flight-path angle of 0 deg, and an inclination of 90 deg. The initial system T/W and ISP were varied to capture their effects on the required ΔV. Table 1 shows the ranges of values for T/Wi and ISP that were run and their corresponding ΔV.

The velocity losses due to drag losses, gravity losses, thrust vector losses, and atmospheric thrust losses were all recorded. Their values for a sample trajectory from above ($T/Wi = 1.8$, ISP $= 400$ s), are shown in Table 2. The atmospheric thrust losses, which will increase with increasing engine exit area and/or atmospheric pressure, were found to be approximately 18 m/s in the worst case scenario. Because this loss is so small when compared with the total ΔV required to attain orbit, the effect of changing exit area on the required ΔV was considered to be insignificant, and was thus ignored in this study.

The ΔV information gathered from POST was fit into a response surface equation (RSE) using Excel. The RSE was used in the sizing analysis of the MAV in EXAMINE (as discussed in the next section). As the full factorial analysis is run, the ISP and T/Wi are incorporated into the RSE and the corresponding ΔV required is obtained for use in the performance calculations in EXAMINE. The R^2 value of the response surface equation is 0.9997. Figure 5 shows a plot of the RSE. By using an RSE, all of the necessary runs could be carried out in POST before running EXAMINE, rather than having to feed information back and forth between the programs.

In addition to the Mars ascent ΔV, the MAV is responsible for providing the ΔV required to arrive in the appropriate orbit to rendezvous and dock with the awaiting ERV. DRM4 assumes that the ERV will be in a 250 km by one sol orbit, which translates to a 250 km × 33793 km elliptical orbit for Mars, as shown in Fig. 6. The first burn places the MAV on a Hohmann transfer ellipse to a 250 km altitude. At the apoapsis of the ellipse, the second burn provides the energy to stay in the ERV's orbit. The two burns total 1.271 km/s of ΔV. This additional ΔV was added to the ΔV found through the response surface equation for use in the performance calculations within EXAMINE.

C. Mass and Sizing Analysis

EXAMINE is an architectural study tool used in engineering to help size launch and in-space vehicles [7]. The program was originally created at NASA Langley Research Center to help with the lunar architecture studies related to the new VSE, but has been

Table 1 ΔV output from the Program to Optimize Simulated Trajectories with varying T/Wi and specific impulse

		Initial system thrust-to-weight ratio						
		1.2	1.4	1.6	1.8	2	2.2	2.4
ISP, s	330	4506.92	4327.09	4243.86	4203.48	4190.73	4195.38	4211.83
	365	4521.53	4329.07	4236.66	4189.86	4171.02	4169.64	4181.24
	400	4536.94	4332.50	4234.54	4180.37	4156.31	4151.09	4158.18
	435	4558.87	4337.49	4229.93	4173.68	4146.00	4137.16	4140.64
	470	4578.62	4343.82	4230.71	4169.42	4138.42	4126.20	4127.06

Table 2 Program to Optimize Simulated Trajectories output for velocity losses for sample trajectory of initial system thrust-to-weight ratio of 1.8 and specific impulse of 400 s

	Relative velocity losses, m/s				
ΔV ideal, m/s 4180	Drag	Gravity	Thrust vector	Atmospheric thrust	Total
% of ΔV ideal	42.9 1.0%	510.3 12.2%	106.8 2.6%	15.7 0.4%	673.5 16.1%

modified throughout the years to address various other engineering problems. For this study, EXAMINE was used to model one segment of the Mars architecture, the MAV. EXAMINE is capable of modeling all aspects of a space vehicle, including body, habitat, propulsion system, protection system, avionics, thermal control system, crew accommodations, and landing gear.

1. EXAMINE Sizing Model

EXAMINE sizes an engine based on the following user inputs: number of engines (including engines out), thrust per engine, engine type, whether the engine has gimbals or is throttleable, propellant tank pressure, if the engine is regeneratively cooled, the steady-state power requirements for engine control, engine valves, and tank valves, the propellant combination, the amount of propellant burned, the oxidizer-to-fuel ratio, the oxidizer and fuel vapor pressures, the chamber pressure, area ratio, percent bell nozzle, chamber contraction ratio, and chamber L/D.

The ideal characteristic velocity (C^*) and ISP are determined based on curve fits of the results of the combustion calculations from NASA John H. Glenn Research Center's chemical equilibrium code (for a variety of propellant combinations) that are embedded in the model. Both the ideal C^* and ISP are dependent on the propellant combination, O/F ratio, and combustion chamber pressure, and the ideal ISP is also a function of the nozzle area ratio. The ideal C^* and ideal ISP are then multiplied by the appropriate efficiencies, which are also based on the propellant combination. The ISP is used to determine the fuel and oxidizer mass flow rates, which are in turn used to calculate the engine throat area. The chamber contraction ratio and L/D are used with the engine throat area to size the engine thrust chamber. The chamber thickness is a function of the chamber diameter and pressure. The nozzle is sized based on the area ratio, the engine percent bell nozzle, and the engine throat diameter. If the main engines are pump fed, fuel and oxidizer pumps are also sized. The total engine mass is then the sum of the turbopump assembly mass (if pump fed), gimbal mass, the engine thrust chamber, nozzle, lines and valves, and other components.

The ΔV information from the RSE (combined with the additional ΔV required to reach the orbit of the ERV) is used in the rocket equation along with the calculated ISP to determine the propellant mass. The propellant tanks are sized according to the propellant mass and volume, including the additional volume for the propellant that will boil off during the mission. The calculation of the boil-off propellant is described in the following section. The propellant tank diameters are restricted by the payload shroud diameter. The engines are sized to maintain the user-input initial system T/W, taking into account any decrease in propellant mass due to boil-off. Finally, the structure is sized to support the tanks and engines. The resulting mass ratio is fed into the ΔV calculation again, and the process continues until a closed solution is found, provided one exists.

2. EXAMINE Boil-Off Model

One major concern for a bring-your-own-propellant mission scenario is the boil-off of cryogenic propellants. The sun's rays heat the propellant tanks, and even with multilayer insulation, heat leaks into the propellant through struts, penetrations, and through the

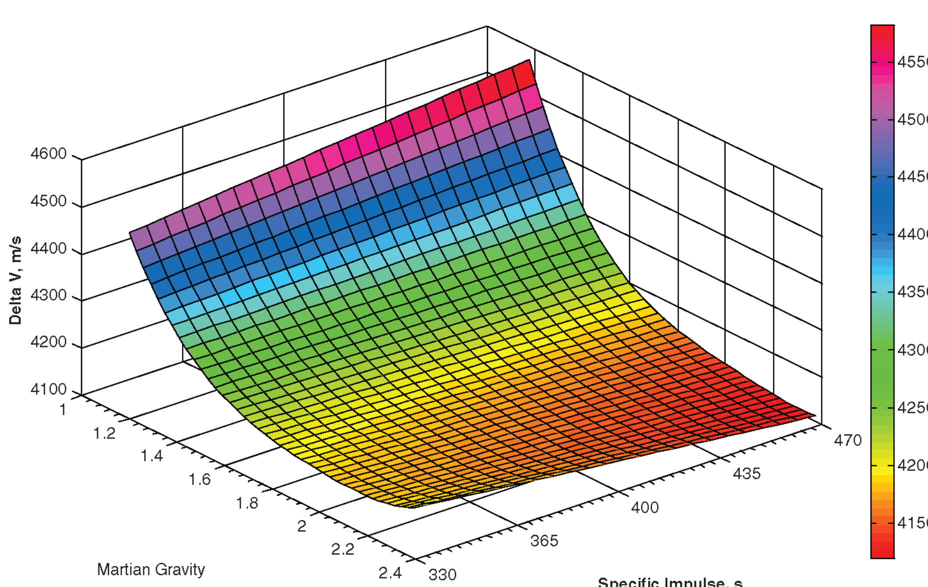

Fig. 5 Plot of RSE of ΔV output from POST.

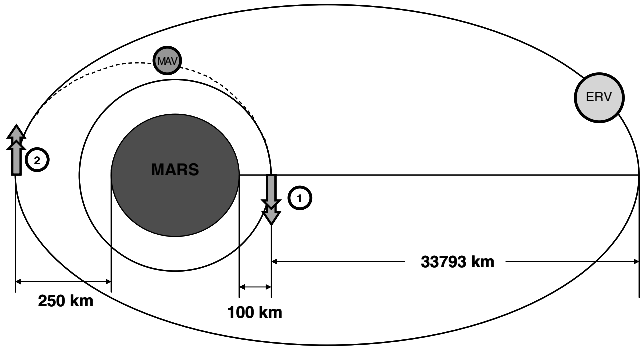

Fig. 6 Depiction of the extra burns required to transfer to the ERV's orbit.

multilayer insulation (MLI) itself. When the net heat flux causes the total heat added to the propellant to exceed the allowable heat absorbed, the propellant will begin to boil off. Depending on the properties of the propellant, the boil-off can be partially or fully mitigated through use of cryocoolers and/or active loading.

The total fluid boil-off is determined by Eq. (1)

$$m_{\text{Boiloff}} = (q_{\text{Total}} - q_{\text{Allowable}}) * (1 + \%R/100)/\text{HOV} * \#\text{tanks} \tag{1}$$

where q_{Total} is the total heat added to the propellant, $q_{\text{Allowable}}$ is the allowable heat absorbed by the propellant, $\%R$ is the percent residual propellant, HOV is the fluid heat of vaporization of the propellant, and #tanks is the number of propellant tanks. The total heat added is determined by Eq. (2)

$$q_{\text{Total}} = \boldsymbol{\phi} q * \text{SA} * \text{Days@Temp} * \text{Time/Day} \tag{2}$$

where net heat flux $\boldsymbol{\phi} q$ is the sum of all heat leaks (with an added margin of 50% for hydrogen and 30% for methane or oxygen), SA is the external surface area of the propellant tanks, and Days@Temp and Time/Day are based on the concept of operations. For this study, Days@Temp was set to 1560 days at 210 K. There are approximately 26 months in between launch opportunities, followed by 180 days for the astronauts to travel to Mars, and a 600 day surface mission for a total of 1560 days. The average surface temperature of Mars, 210 K, was assumed to be the environmental temperature for this study because the MAV will be on the surface of Mars for the majority of the mission, and the MAV tanks were assumed to be exposed to this temperature 24 h per day. The net heat flux goes to zero for methane and oxygen if a one-stage cryocooler is used, but a two-stage cryocooler is necessary to reduce the net heat flux to zero for hydrogen because of its much lower boiling point.

The allowable heat absorbed is given by Eq. (3)

$$q_{\text{Allowable}} = m_{\text{Fluid}} * \text{factor} * (T_{\text{Allowable}} - T_{\text{Loaded}}) \tag{3}$$

where m_{Fluid} is the mass of the propellant, factor is dependent on the choice of propellant, $T_{\text{Allowable}}$ is the boiling point of the propellant at the propellant tank storage pressure, and T_{Loaded} is the boiling point at ambient Earth pressure. Active loading takes advantage of the fact that, at higher pressures, the boiling point of the propellant is increased. All propellants were assumed to be loaded at 101.35 kPa. If the engine is pump fed, the tank storage pressure was assumed to be 275.8 kPa. Otherwise, for pressure-fed cases, the propellant tank pressure was assumed to be 2413.2 kPa. The exception is nitrogen tetroxide/monomethyl hydrazine (NTO/MMH), which was always loaded at 2158.1 kPa. The boiling points of the propellant at loaded and stored pressures were found using curve fits of data from the National Institute of Standards and Technology's reference fluid thermodynamic and transport properties database. Because the stored pressure is higher than the loaded pressure, the boiling point of the propellant is increased, and the onset of boil-off is delayed.

3. *Key Modeling Assumptions*

For this study, the baseline MAV as described in DRM4 was modeled, updating assumptions where necessary. The total payload mass assumed for this study was 5805 kg: 5115 kg for the crew capsule, 600 kg for the crew itself, and 90 kg of science payload. The payload attachment structure was assumed to be 2% of the mass of the payload. The engines were sized to allow gimbaling and throttling, and four engines were assumed in all cases. The average external environmental temperature was assumed to be 210 K, the average temperature on the surface of Mars, because the MAV would spend the majority of its time there. The propellant tanks were modeled as Al 7075 for all pump-fed cases, and graphite-wrapped aluminum tanks for all pressure-fed cases. The propellant tank pressurization system was gaseous helium, unless the propellant was hydrogen, in which case an autogenous pressurization system was used. The tank maximum storage pressure was 275.8 kPa for pump-fed and 2,413 kPa for pressure-fed. Two oxidizer and two fuel tanks were used, except for hydrogen, which used four fuel tanks. The tank diameters were dependent on the number of tanks and the limiting diameter of the payload shroud (9.1 m based on the latest decisions by NASA [8]). Initially boil-off sensitivity studies were run and, based on those results, 50 layers of MLI were used for either H_2 or CH_4, and 10 layers for either ethanol or MMH. A sensitivity analysis to the number of MLI layers for each propellant tank is considered beyond the scope of this analysis. For MMH, a heater was added to prevent the propellant from freezing. The reaction control system (RCS) fuel and oxidizer was assumed to be stored with the main propellant. The power source was assumed to be outside of the MAV, so the only power related mass is the power management and distribution system, which is sized based on the vehicle peak power consumption. The majority of the control and avionics and all of the environmental control and life support systems and crew accommodations were assumed to be contained within the crew capsule (payload). Landing gear and any necessary parachutes were assumed to be part of the descent stage (which was not part of this study). Two percent residuals were assumed for both the main and RCS propellants. Eight percent and 5% ullage were assumed for the main and RCS propellants respectfully. A 30% growth margin was assumed for the dry mass.

D. Optimization Analysis

The results of the full factorial analysis run in EXAMINE were used in JMP, a statistics software developed by SAS Institute, Inc. JMP was used to fit the EXAMINE outputs of chamber pressure, nozzle area ratio, O/F ratio, and system T/Wi into response surface equations. Different RSEs were developed for each combination of propellant type, engine type, and choice of cryocooler. These RSEs were used to determine the ideal settings for the variables to minimize dry mass, IMLEO, or the mass of the MAV at ascent. With this input, EXAMINE computed the corresponding ISP and engine T/W.

IV. Results

A. Sensitivity Analysis

A sensitivity analysis was performed on the variables of the morphological matrix to identify their impact on the key figure of merit, IMLEO, and to determine which ranges of the variables were appropriate for use in the full factorial analysis. As one variable was changed along its range of values, all other variables were held at their baseline values (as identified in the morphological matrix of Fig. 2). Based on the results of the sensitivity analysis, the ranges for the variables for the various propellant combinations and engine types were chosen. The ranges chosen are displayed in Table 3 . The values omitted from the ranges were eliminated because they resulted in very high values of IMLEO.

B. Full Factorial Analysis

A full factorial analysis was run on the values of the variables selected after the sensitivity analysis was performed. A partial set of results are presented in Table 4. The two-stage cryocooler, pump-fed

Table 3 Ranges of values of variables run in the full factorial analysis

		Pump fed				Pressure fed			
O_2/H_2	Chamber pressure, kPa	5792	6389	6987	7584	1344	1413	1482	1551
	Area ratio	200	266.67	333.33	400	50	73.33	96.67	120
	O/F ratio	5.86	6.07	6.29	6.5	5.86	6.07	6.29	6.5
	T/W	1.40	1.60	1.80	2.00	1.3	1.47	1.63	1.8
	Cryocooler	None	1-stage	2-stage		None	1-stage	2-stage	
CH_4/O_2	Chamber pressure, kPa	5792	6389	6987	7584	1344	1413	1482	1551
	Area ratio	200	266.67	333.33	400	50	73.33	96.67	120
	O/F ratio	3.34	3.43	3.51	3.6	3.34	3.43	3.51	3.6
	T/W	1.40	1.60	1.80	2.00	1.3	1.47	1.63	1.8
	Cryocooler	None	1-stage			None	1-stage		
O_2/Ethanol	Chamber pressure, kPa	5792	6389	6987	7584	1344	1413	1482	1551
	Area ratio	200	266.67	333.33	400	40	63.33	86.67	110
	O/F ratio	1.86	1.94	2.03	2.11	1.77	1.86	1.94	2.03
	T/W	1.4	1.60	1.80	2.00	1.2	1.33	1.47	1.6
	Cryocooler	None	1-stage			None	1-stage		
Nitrogen tetroxide/ monomethyl hydrazine	Chamber pressure, kPa	5792	6389	6987	7584	1344	1413	1482	1551
	Area ratio	100	166.67	233.33	300	40	63.33	86.67	110
	O/F ratio	1.79	1.86	1.93	2	1.79	1.86	1.93	2
	T/W	1.4	1.60	1.80	2.00	1.2	1.33	1.47	1.6
	Cryocooler	None				None			

Table 4 Results from full factorial analysis with lowest initial mass in low Earth orbit, with and without cryocoolers

Cryocooler choice	Propellant	Chamber pressure, kPa	Area ratio	O/F ratio	T/Wi	Isp_{Vac}, s	Engine T/W_{Vac} (uninstalled)	Dry mass, kg	Initial mass in low-Earth orbit, kg
2-stage	O_2/H_2	7584[a]	400	6.50	1.80	469.0	38.32	8,657	51,047
	O_2/H_2	1551[b]	96.67	6.50	1.63	448.2	29.67	27,123	134,344
1-stage	O_2/H_2	7584[a]	400	6.50	1.80	469.0	38.32	8,792	51,825
	O_2/CH_4	7584[a]	400	3.60	1.60	377.2	40.85	8,056	69,328
	O_2/Ethanol	7584[a]	400	1.94	1.60	354.4	41.03	7,783	74,864
	O_2/H_2	1551[b]	96.67	6.50	1.63	448.2	29.77	26,601	132,210
	O_2/CH_4	1551[b]	96.67	3.60	1.63	356.1	30.32	20,944	151,185
	O_2/Ethanol	1551[b]	86.67	1.86	1.47	333.5	19.23	26,686	211,647
None	O_2/CH_4	7584[a]	400	3.60	1.60	377.2	40.82	8,194	72,906
	O_2/Ethanol	7584[a]	400	1.94	1.60	354.4	40.97	7,927	78,598
	Nitrogen tetroxide/ monomethyl hydrazine	7584[a]	300	2.00	1.60	341.1	44.39	13,178	111,399
	O_2/CH_4	1551[b]	96.67	3.60	1.63	356.1	30.33	20,912	151,006
	Nitrogen tetroxide/ monomethyl hydrazine	1551[b]	63.33	2.00	1.47	324.8	24.12	16,902	151,327
	O_2/Ethanol	1551[b]	86.67	1.86	1.47	333.5	19.25	26,631	211,290

[a]Pump-fed
[b]Pressure-fed

O_2/H_2 cases resulted in the lowest IMLEO out of all 4080 cases run. The top one-stage cryocooler case also used O_2/H_2, however, if no cryocooler was used, no solution existed for the O_2/H_2 propellant combination. The best no-cryocooler case was the pump-fed O_2/CH_4, which even without a cryocooler resulted in a slightly better IMLEO than the baseline case.

The pressure-fed cases all resulted in higher IMLEO and lower engine T/W than the pump-fed cases. Interestingly, the overall best pressure-fed case was O_2/H_2 with a one-stage cryocooler, not the two-stage cryocooler. This can be explained by the fact that in the pressure-fed cases, the tank pressure is much higher, making the boiling point of the propellant higher. By taking advantage of active loading, the total boil-off is reduced to zero, and so having a two-stage cryocooler instead of a one-stage cryocooler results in extra dry weight and no additional benefit. A similar result is seen when comparing the O_2/CH_4 and O_2/Ethanol one-stage cryocooler, pressure-fed cases with the no-cryocooler pressure-fed cases.

In both pump and pressure-fed cases, the highest chamber pressure always resulted in the lowest IMLEO. The original range of values for the pump-fed chamber pressure was restricted to 7584 kPa because the engine is assumed to use an expander cycle. Although gas generator and staged combustion cycles are normally used for launch vehicles, they are so much more complex and difficult to start and manage that their use was assumed to be restricted to Earth launch vehicles. For this mission to Mars, a highly reliable system is desired for the MAV engines because the engines will not be used until ~1500 days into the mission, and launch control will be limited due to the distance from Earth. The pressure-fed chamber pressure range was restricted to 1551 kPa because the propellant tanks have to be at a higher pressure than the combustion chamber. High propellant tank pressures result in much higher propellant tank masses.

The area ratio for the top pump-fed cases also ran up against the upper bound of the range run for this study. The nozzle area ratio for pump-fed cases had been restricted to 400 for two reasons. First, an area ratio of 400 is already 40% more than RL10B-2, which has the highest area ratio amongst the current RL-10 derivatives. Secondly, as the area ratio increases, packaging of the MAV becomes more of a concern because of the increasing engine nozzle length with exit area. Additionally, the O/F ratio limit was reached for all propellant combinations except O_2/Ethanol. The limit was placed on the O/F ratios to try to avoid the high combustion temperatures associated with higher O/F ratios. The behavior of the O_2/Ethanol cases followed the trend shown in the sensitivity study, with the lowest IMLEO resulting at midrange values of the O/F ratio.

Another interesting result that can be drawn from Table 4 is that the pressure-fed cases had lower engine T/W (from 19 to 30) due to their heavy engines, compared with the lighter pump-fed engines that had an engine T/W ranging from 38 to 44. Consequently, the pressure-fed

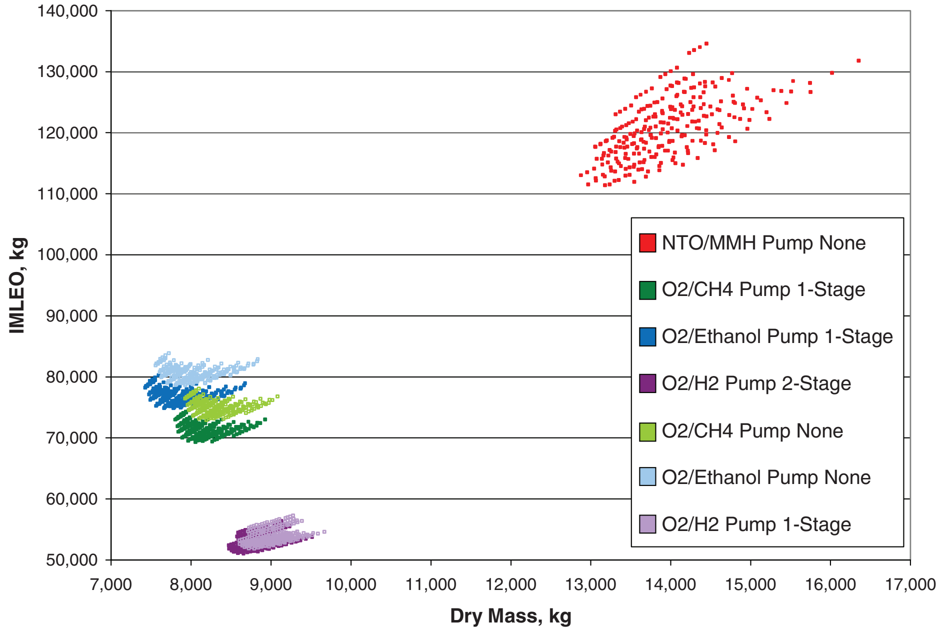

Fig. 7 IMLEO versus dry mass for all pump-fed cases.

cases all favored lower system T/Wi values with a range of 1.4–1.6, and the pump-fed cases ranged from 1.6 to 1.8 system T/Wi.

Dry mass is directly related to development and production costs, whereas IMLEO is directly related to the launch costs. Reducing mass is also important to reduce the risk of entry, descent, and landing. Depending on which set of costs is more of a concern, the top choice for propellant and engine type may change. Table 4 shows that the lowest dry mass is actually provided by the case using pump-fed O_2/Ethanol engines with a one-stage cryocooler. This case has 10% less dry mass than the case with the lowest IMLEO, but 47% more IMLEO. The pump-fed O_2/Ethanol case with no cryocooler has 8% less dry mass and 54% more IMLEO than the case with the lowest IMLEO, but it would avoid the cost of developing a cryocooler. The O_2/CH_4 pump-fed cases also have lower dry masses and slightly higher IMLEO than the O_2/H_2 pump-fed cases.

Figures 7 and 8 show the dry mass and IMLEO for all of the converged cases of the full factorial analysis. The scales of the two graphs are different to assist with viewing the data. Figure 7 makes it obvious that any pump-fed NTO/MMH mission would be very costly, both in terms of development and production and launch costs. The best pump-fed cases in terms of development and production costs are the O_2/Ethanol, and the O_2/H_2 are the cheapest to launch. Adding a cryocooler tends to reduce overall costs for the pump-fed cases. Figure 8 shows that the NTO/MMH pressure-fed missions come in cheapest in terms of development and production costs, and a close third place in terms of launch costs. The O_2/H_2 systems had the lowest IMLEO mass and thus the lowest launch costs. The addition of a cryocooler has little effect on the IMLEO and dry masses for the pressure-fed cases (because active loading has already reduced any boil-off mass to almost zero), so if a pressure-fed engine is chosen, it is probably not worth the development of a cryocooler system for the mission.

C. Optimization Results

The results from the full factorial analysis were put into JMP to create response surface equations for the effects of chamber pressure, nozzle area ratio, O/F ratio, and system T/Wi on the dry, IMLEO,

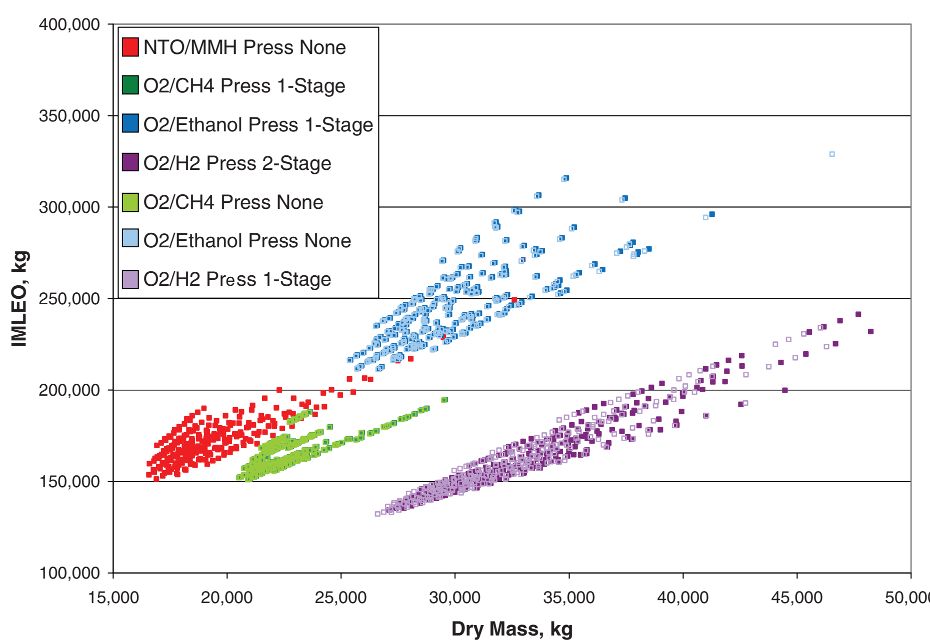

Fig. 8 IMLEO versus dry mass for all pressure-fed cases.

Table 5 Optimization results from JMP

Cryo choice	Propellant	Chamber press, kPa	Area ratio	O/F ratio	System T/Wi	Isp_{Vac}, s	Engine T/W_{Vac}	Initial mass in low-Earth orbit, kg
2-stage	O_2/H_2	7584[a]	380	6.500	1.81	480	43.35	50897
	O_2/H_2	1551[b]	90	6.500	1.64	450	63.68	128677
1-stage	O_2/H_2	7584[a]	380	6.500	1.81	480	43.41	51671
	O_2/CH_4	7584[a]	383	3.600	1.73	393	41.93	69179
	O_2/Ethanol	7584[a]	345	1.964	1.72	366	48.06	74673
	O_2/H_2	1551[b]	90	6.500	1.64	450	63.69	126995
	O_2/CH_4	1551[b]	89	3.577	1.58	359	69.02	149683
	O_2/Ethanol	1551[b]	81	1.881	1.40	336	54.00	200960
None	O_2/CH_4	7584[a]	384	3.600	1.72	393	41.91	72810
	O_2/Ethanol	7584[a]	346	1.965	1.71	366	48.04	78451
	Nitrogen tetroxide/ monomethyl hydrazine	7584[a]	247	2.000	1.73	343	70.15	110463
	Nitrogen tetroxide/ monomethyl hydrazine	1551[b]	71	2.000	1.44	327	56.94	149110
	O_2/CH_4	1551[b]	89	3.577	1.58	359	69.03	149514
	O_2/Ethanol	1551[b]	80	1.886	1.39	336	54.11	199976

[a]Pump-fed
[b]Pressure-fed

and ascent masses. The ascent mass is the mass of the MAV directly before takeoff from the Martian surface after completion of the surface stay. The second-order response surface, which approximates the design space, can then be used to determine the optimum values for the parameters of interest. When the IMLEO mass is optimized, the best values for the MAV are achieved with a chamber pressure of 7584 kPa, a nozzle area ratio of 380, an O/F ratio of 6.5, and a system T/Wi of 1.81, providing a total IMLEO of 50,897 kg. Optimizing for the lowest ascent mass typically results in nearly the same settings for the MAV, especially if cryocoolers are used. This is because the IMLEO is the gross mass plus crew capsule (including any propellant that will later boil off) whereas the ascent mass is the gross mass plus crew capsule, crew, and science payload, subtracting any mass related to propellant boil-off. If cryocoolers are used, reducing the boil-off to zero or close to zero kilograms, the difference between the ascent and IMLEO masses is only 690 kg, or the mass of the crew and science payload. The best values for IMLEO, however, are not the same as the best values for minimizing the dry mass. When the dry mass is optimized, the best values for the MAV are achieved with a chamber pressure of 7584 kPa, a nozzle area ratio of 202, and O/F ratio of 6.5, and a system T/Wi of 1.68 for a total dry mass of 8473 kg and total IMLEO of 52,000 kg. This is a 2% increase in IMLEO for a 1.7% reduction in dry mass. When these cases were input into the EXAMINE model to see how well it correlated with the approximate JMP RSE results, the differences were less than 1%. The JMP results are summarized in Table 5.

D. Benefits of In Situ Resource Utilization

All of the propulsion systems explored for this MAV optimization study were based on a bring-your-own-propellant mission scenario. However, there is much interest in the possibility of producing the propellant for the MAV on the surface of the planet using an ISRU module. This would eliminate the need to carry and store the propellant during any delay in low Earth orbit or during the long trip from Earth to Mars. As a part of this study, a model was developed to carry out a preliminary analysis of the mass-saving benefits of ISRU. The results using this model were found to correlate well with that of [9]. In both models 52,446 kg of oxygen and methane were produced at an O/F ratio of 3.5 over the course of 500 days. Both models used methane pyrolysis and reverse water gas shift to produce the necessary oxygen and methane. A hydrogen feedstock is required for these processes.

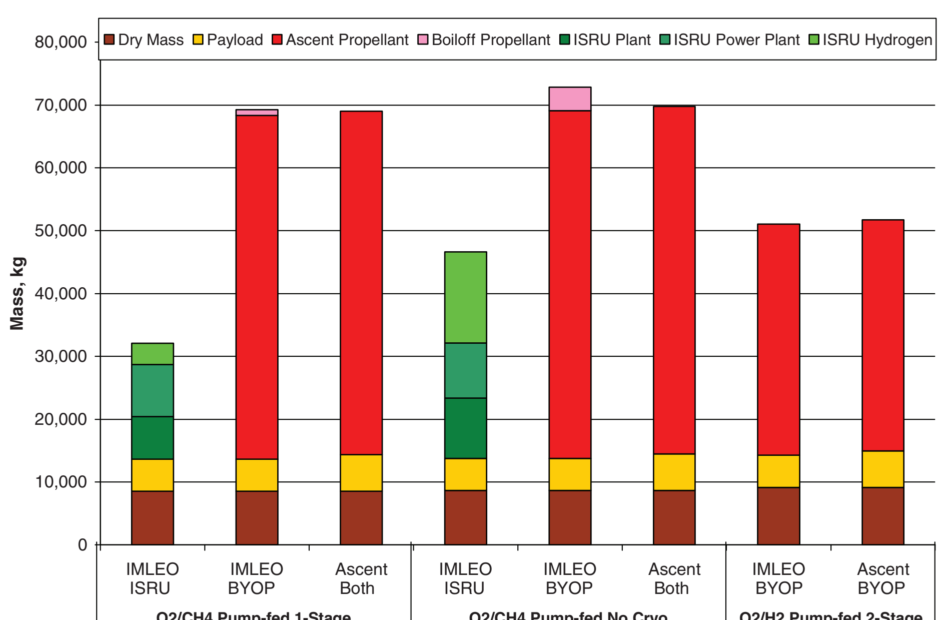

Fig. 9 Comparison of IMLEO and Ascent masses for top O_2/CH_4 and O_2/H_2 options.

The ISRU model was run for a variety of O/F ratios over a range of propellant mass required. It was assumed that 500 days were available for the propellant production, which would leave a one-month margin between the arrival of the ISRU module at Mars (during the first launch opportunity) and the departure of the astronauts from Earth (during the second launch opportunity). The model was run with and without cryocoolers. Without cryocoolers, the hydrogen feedstock was assumed to boil off at a rate of 3% per month. The mass of the propellant produced by the ISRU module included extra propellant to account for the boil-off that would be experienced after propellant production and before departure. When cryocoolers were used, a two-stage cryocooler was assumed for the hydrogen feedstock. The propellant produced was assumed to be stored directly on the MAV, so that the only additional storage tank required was for the hydrogen feedstock. The ranges of values for the propellant required came from the results of the full factorial analysis performed for this study.

Figure 9 illustrates the possible benefits of using ISRU. The best case from the optimization of the full factorial results is shown on the far right: O_2/H_2, pump-fed with a two-stage cryocooler. The IMLEO for this case is 51,000 kg. The ascent mass is 690 kg more (adding the crew and science payload) because there is no boil-off with a 2-stage cryocooler. The other two cases displayed are the best cases for O_2/CH_4, pump-fed engines, with and without a cryocooler. For the cryocooler case, the IMLEO drops from 70,000 kg to 31,000 kg if ISRU is used. This is a 56% savings in IMLEO mass. For the case without cryocoolers, if ISRU is used the IMLEO drops from 72,000 kg to 47,000 kg, a 35% savings. In both O_2/CH_4 cases, if ISRU is used the IMLEO is lower than the best case O_2/H_2 case. These considerable savings warrant further investigation into ISRU technology.

V. Conclusions

This paper presented a study of the MAV's propulsion system for a bring-your-own-propellant human mission to Mars. Four propellant combinations using two engine types and ranges of values for combustion chamber pressure, nozzle area ratio, oxidizer-to-fuel ratio, initial system thrust-to-weight ratio, and choice of cryocoolers were investigated in a full factorial analysis. The results indicate that for a bring-your-own-propellant mission scenario, the best choice is an oxygen/hydrogen pump-fed system with a two-stage cryocooler. Although pump-fed systems provide the lowest mass, they are also generally not as reliable as pressure-fed systems. Additional analysis beyond the scope of this paper is required to determine the relative system reliability. The use of in-situ resource utilization was also explored, and the results show that significant mass savings could be realized by bringing the supplies to produce propellant on Mars rather than bringing the propellant itself. In fact, oxygen/methane pump-fed engines using in-situ resources offer significant savings in initial mass in low Earth orbit as compared with the best choice for a bring-your-own-propellant mission. Further research is recommended in the area of in-situ resource utilization.

Within the full factorial analysis, the optimum designs resulted at the extremes of the ranges analyzed for chamber pressure, area ratio, and oxidizer-to-fuel ratio. The ranges used in this study already push the current limits on these engineering parameters. Technology improvements in these areas may lead to a more optimum solution at a higher chamber pressure, area ratio, or oxidizer-to-fuel ratio.

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