A Comparison of Preliminary Design Concepts for Liquid, Solid and Hybrid Propelled Mars Ascent Vehicles Using In-Situ Propellants

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ABSTRACT

With a mission to Mars no longer merely an idea of science fiction, it is not too early to determine the technology requirements that will ultimately make it possible for humans to establish a long term outpost on Mars. One key aspect is the development of a reliable, reusable launch vehicle to shuttle astronauts between the Mars surface and low Martian orbit. This preliminary design study serves to provide an in depth comparison of liquid, solid, and hybrid propulsion concepts for a long-term In-Situ Mars Ascent Vehicle (IMAV) which relies only on propellants which can be harvested from the Mars atmosphere or soil. Because of the low Δv, a Single Stage to Orbit (SSTO) launch vehicle can be used to carry the crew plus cargo from the Martian surface back to the command module. Theoretical chemical equilibrium calculations have been performed to determine the optimum in-situ propellant combination for each propulsion type. The approach we took in performing a comparison of the possible design configurations contained several steps. First, we identified a baseline configuration against which we compared our design. The baseline configuration we chose was the Mars Ascent Vehicle (MAV) outlined in the current Mars Reference Mission. The second step was the addition of several constraints not specified in the baseline configuration, but which have been deemed important for this analysis. One significant constraint was that only non-hydrogen containing fuels were considered. Finally, we compared three different design alternatives to the baseline configuration. The areas of comparison were performance, safety, and feasibility. Based on these evaluation criteria, we have recommended a liquid propulsion system using CO/O_2 propellants as the most favorable configuration for the development of a long-term, Mars ascent vehicle.

INTRODUCTION

Currently the Space Exploration Initiative (SEI) offers an ambitious plan that includes the human exploration of Mars. A manned mission to Mars will impose huge burdens on financial and technological resources. One technology that may relive some of these burdens is In-Situ Resource Utilization (ISRU) technology. ISRU is the use of materials at the site of an interplanetary mission for the production of rocket propellant or life support products. With the ability to reduce Earth launch mass and decrease cost due to a reduction in the number of required launches, in-situ propellants are being recognized as the most viable option for sending humans to Mars [1].

ISRU also holds the key to establishing permanent outposts on the moon and Mars. Development of such an outpost will require maximizing the resources available on the surface and/or within the atmosphere of these bodies. Data from previous unmanned Mars missions has shown that a wealth of resources are available on the surface of Mars that are suitable for use as chemical energy sources for sustained, long-term manned presence on Mars. One critical chemical energy requirement that must be addressed is a rocket propulsion system since human explorers will have to be periodically transported from the surface of Mars to low Martian orbit.

In this investigation we analyzed a variety of possible in-situ propellant combinations for the development of a long-term In-Situ Mars Ascent Vehicle (IMAV). The propellant combinations were then configured into the appropriate rocket propulsion class: solid, liquid or hybrid. An optimum in-situ propellant combination choice was then chosen for each of the three propulsion classes. Each of the three candidate propulsion systems were then compared against the baseline Mars Ascent Vehicle detailed in the current Mars Reference Mission [1]. The evaluation criteria consisted of areas such as performance, safety, and feasibility.

APPROACH

We began by identifying a baseline configuration against which to compare new design configurations for the IMAV. We chose the Mars Ascent Vehicle (MAV) [1] as our baseline using the published results of the engine analysis as a comparison for our design. Next, we recognized several mission constraints not specified in the baseline configuration. As described below, a key constraint was that our launch vehicle would utilize a hydrogen-free propulsion system. These constraints provide a guideline for the evaluation criteria, which included performance, vehicle size, and weight. Finally we compared three different propulsion alternatives (i.e., solid, liquid or hybrid) using a theoretical chemical equilibrium computer code [2]. The raw materials necessary to produce propellant for each of the three propulsion system alternatives for the IMAV are available from resources within the atmosphere or soil of Mars.

BASELINE CONFIGURATION

To compare the performance of our three alternative IMAV propulsion systems, the Mars Ascent Vehicle (MAV) [1] was chosen as a baseline. The MAV consists of a single common descent stage that delivers all hardware systems to the surface of Mars including the habitats, ascent vehicle, propellant production plant, and other surface cargo. As outlined in the Mars Reference Mission [1], the lander consists of four subsystems. These subsystems include a structure which contains payload and all other elements, a parachute to assist in the slow down of descent, a propulsion system to slow the lander prior to landing, and a surface mobility system.

The MAV allows the crew from the surface to launch back into orbit to rendezvous with the Earth Return Vehicle (ERV). This vehicle consists of the crew ascent capsule and the ascent propulsion system. The vehicle will use propellant made by the propellant production plant that was delivered by the surface lander. The crew is returned to orbit via the capsule using the ascent propulsion system fuelled by propellants derived from Martian atmosphere and soil. The MAV utilizes LOX/CH_4 produced by the propellant production plant on the surface of Mars. The use of methane was made possible by the transportation of hydrogen from Earth. Producing the fuel on Mars benefits the mission by allowing more equipment to Mars [3]. The current MAV vehicle was used as a baseline against which we compared our alternative IMAV designs.

ADDITIONAL CONSTRAINTS

We have determined important variables that help define characteristics of an optimum IMAV system. For instance, the velocity needed to ascend to a low Martian orbit is related to the ratio of the mass of the propellant to the overall mass of the vehicle:

$$\ln R = \frac{U_{esc} + g_{mars} t_B}{g_o I_{sp}}$$

$$\frac{M_p}{M_o} = 1 - \frac{1}{R}$$

So to conserve mass, it is important to minimize propellant, which is relative to the minimum velocity need to escape Mars' gravitational force near the surface. For our theoretical missions, we allow for the mass of four astronauts and an additional amount for equipment and Martian test samples.

The attention towards having a SSTO also addresses the mass conservation issue. The addition of multiple fuel tanks simply increases the mass. Another benefit of a SSTO is the simplification of the ascent stage, which offers savings in controls and additional nozzles/plumbing.

The overall dry mass of the IMAV must be minimized since it will most-likely be manufactured on earth and transported to Low Earth Orbit (LEO) using whatever earth-to-LEO launch system that will be in use in the future. Specifically, to avoid assembly of the vehicle in LEO, the weight of the empty vehicle must be less than the maximum payload capacity of the most powerful LEO launch vehicle available. As a first approach, the current Space Shuttle fleet has been chosen as the system for launching the fully assembled IMAV into LEO.

Since the current Space Shuttle fleet has been chosen as the launching platform, an additional constraint that can't be overlooked is the size of the IMAV. Specifically, the IMAV must fit inside the current Space Shuttle cargo bay under the assumption that one of current fleet will be carrying the vehicle. Only one launch from Earth will be necessary to transport the complete IMAV to lower Earth orbit. Thus, the IMAV must comfortably fit into a 15-ft wide x 15-ft high x 60-ft long volume. This constraint is significant to the analysis since it impacts the size of the fuel and oxidizer tanks.

As seen from the analysis of the Martian atmosphere, there is a sparse amount of available hydrogen most commonly used high performance liquid fuels, such as hydrogen and hydrocarbons.

For repeated launches from the Mars surface, it is not cost effective or reasonable to continually transport hydrogen from Earth to manufacture these high performance fuels. Therefore, hydrogen was eliminated as an element in the in-situ propellants compared in this study.

MARS ENVIRONMENT

In order to produce in-situ propellants, we need to know what resources are available. Previous unmanned missions have provided invaluable information regarding the surface and atmosphere of Mars such as NASA's Viking missions and the Mars Pathfinder.

Mars has a very thin atmosphere [4], only about 1% as dense as on Earth [5], consisting of mainly carbon dioxide and some other common gases as shown in Table 1.

Gas	Concentration (%)
Carbon Dioxide	95.32
Nitrogen	2.7
Argon	1.6
Oxygen	0.13
Carbon Monoxide	0.07
Water Vapor	0.03
Neon	0.00025

Table 1. Martian atmospheric constituents [1].

The Martian atmosphere contains less than 1% of the water vapor found in our air, but even this amount can condense forming clouds very high in the atmosphere. At the Viking Lander 2 site, a thin layer of winter frost covered the ground each winter. There is geographical evidence that in the past, in a higher-pressure environment, water flowed on the planet surface [5].

As discussed in the previous section, due to the sparse amount of available hydrogen in the Mars atmosphere, most commonly used high performance liquid fuels (e.g. molecular hydrogen and hydrocarbons) were eliminated as possible in-situ propellants.

The Martian surface is reported to be a type of iron-rich clay that contains a highly oxidizing substance that releases oxygen when wet. Silicon Dioxide and ferric oxide are the main constituents of the soil as shown in Table 2.

Composition of Martian Samples	Weight % (Approx.)
Silicon Dioxide (SiO_2)	44
Ferric Oxide (Fe_2O_3)	19
Sulfite (SO_3)	8.5
Magnesium Oxide (MgO)	8.4
Aluminum Oxide (Al_2O_3)	5.5
Calcium Oxide (CaO)	5.3
Titanium Dioxide (TiO_2)	0.9
Chlorine (Cl)	0.75
Potassium Oxide (K_20)	<0.3

Table 2. Typical constituents in Martian Soil [6].

From experiments carried aboard the Viking Landers, iron-rich smectite clays, magnesium sulfate, iron oxides, and reactive oxidizing agents of unknown chemistry were found on the Martian surface [5]. Smectites are unique materials which have the property of expanding when they contact water, and contracting when dry. Other soil components include silicate minerals, oxides (mostly iron), and some calcium carbonate [5]. The surface contains no organic molecules that were detectable at the parts-per-billion level .

Carbon dioxide, the major component of the atmosphere, freezes at each polar cap covering each hemisphere with snow that evaporates in the spring. The ice caps appear to have a layered structure forming alternating layers of ice with varying concentrations of dark dust [4].

This layering process may also be a result of the wide range of climates Mars experiences due to its orbit. The seasonal changes in the volume of the polar caps are responsible for changing the global atmospheric pressure approximately 25% [4].

The presence of carbon dioxide provides a variety of possible propellant configurations. Options incorporating the plentiful supply of available carbon dioxide include producing methane as the fuel and oxygen as the oxidizer, or producing carbon monoxide as the fuel and oxygen as the oxidizer. If methane and oxygen were to be used it would be necessary to bring some payload from Earth. While the use of carbon monoxide and oxygen would require no Earth recourses, and could be completely manufactured on Mars using the most abundant and readily available Martian resource, carbon dioxide.

The average temperature on the surface is -64^{o} C with a range from -140^{o} C at the winter pole, to 27^{o} C on the day side during summer [4].

Mars' significantly elliptical orbit has a major influence on its climate. One of which is the variation of about 30 C at the subsolar point between aphelion, when Mars is at the point farthest from the Sun, and perihelion, when Mars is at the point in its orbit where it is closest to the Sun [4].

The average pressure on the surface of Mars is only about 0.00069 atm. Although it varies greatly with altitude from almost 0.00888 atm in the deepest basins, to about 0.000987 atm at the top of Olympus Mons, the largest mountain in the Solar System rising 24 km above the surrounding plain [5]. On occasion the entire planet can undergo very strong winds and vast dust storms for months.

Environmental effects due to climate and extreme surface conditions are important to consider for propellant storage, production, and performance purposes. For example with solid rocket fuel the following relationship demonstrates how burning rate and chamber pressure are extremely sensitive to the initial temperature:

$$\Pi_r = \frac{1}{\dot{r}} \frac{\partial \dot{r}}{\partial T_i}\Big|_{P_c}$$

where Π_r is the sensitivity coefficient of burning rate and is measured in terms of [% change / ^{o}C].

Energy expended to store the fuel and oxidizer can be minimized. Any necessary precautions can be taken to protect equipment from extreme or hazardous environmental conditions.

Mars' thin atmosphere produces a small greenhouse effect but it is only enough to raise the surface temperature by 5^{o}C, much less than what we see on Earth.

The research has exposed a variety of elements from Martian atmosphere and soil that are applicable for use as raw materials for rocket propellants. Possible compounds that could undergo some chemical processes and be stored as propellants for the IMAV for launching astronauts from the surface of Mars to orbit are:

- Magnesium and Carbon Dioxide
- Aluminum and Oxygen
- Carbon Monoxide and Oxygen
- Aluminum and Carbon Dioxide
- Magnesium and Carbon Dioxide
- Aluminum, Carbon Monoxide and Oxygen
- Magnesium, Carbon Monoxide and Oxygen

As shown in Figure 1, these propellants can be incorporated into a variety of solid, liquid and/or hybrid rocket propulsion systems.

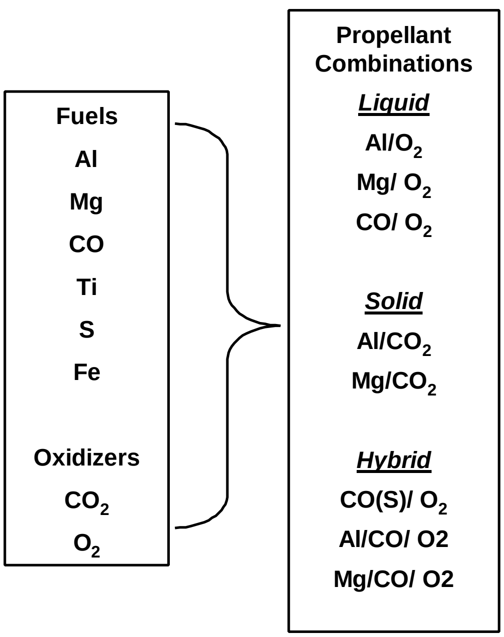

Figure 1. Raw materials for IMAV propulsion systems.

RESULTS

Theoretical Propellant Performance

To theoretically determine the performance and other design parameters of the candidate propulsion systems shown in Fig. 1, the NASA Chemical Equilibrium Computer Code (CEC) was used [2]. Different in-situ fuel and oxidizer combinations for each propellant type were modeled, simulated, and analyzed.

Figure 1 shows the different combinations that were investigated. Evaluation of rocket performance was based on the following criteria, specific impulse and the density of the oxidize/fuel mixture: I_{sp}, ρI_{sp}, and ρI_{sp}^2.

While I_{sp} is a suitable parameter to evaluate rocket performance, we also chose to consider ρI_{sp} and ρI_{sp}^2 because the ultimate criteria of the performance of a rocket propellant are flight parameters which reflect the effects of both specific impulse and propellant density [4]. The parameters ρI_{sp} and ρI_{sp}^2 can be derived from the rocket equation:

$$M_{pl} = \frac{\rho_p V_p}{e^{\left(\Delta v / g_o I_{sp}\right)} - 1} - M_{dry}$$

By expanding the exponential term and incorporating an infinite series expansion, the above equation reduces to the following linear relationship:

$$M_{pl} \propto \rho_p I_{sp}^n$$

where the variable n is related to the mission value of Δv. In a study by Zurawski and Green [7], an evaluation of several propellant combination performances demonstrated a linear relationship between delivered payload mass and ρI_{sp}^2 which can be seen in Figure 2. For the present study, since Δv is approximately the same as the mission outlined in the study by Zurawski and Green [7], n is also approximately 2. Therefore, the highest performing propellant combination will likely correspond with the propellant combination with the maximum value of ρI_{sp}^2 .

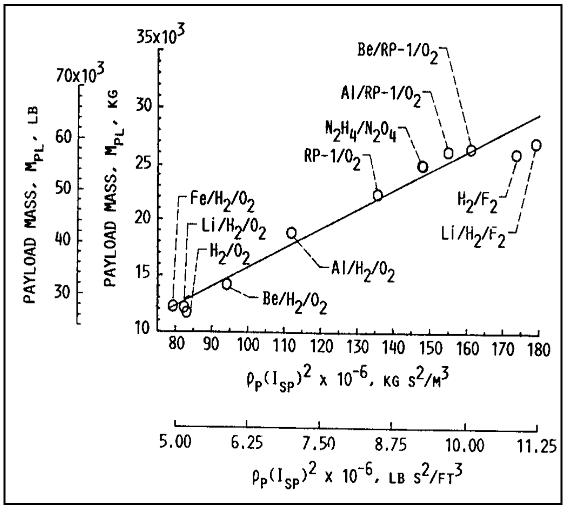

Figure 2. Delivered payload vs. ρI_{sp}^2 [5].

The above relationship makes it possible to use ρI_{sp}^2 as the preliminary criteria for the evaluation of the performance of rocket propulsion combinations. Accordingly, we calculated I_{sp}, ρI_{sp}, and ρI_{sp}^2 over a range of oxidizer/fuel ratios to simultaneously determine the optimum ratio.

Generally speaking, the optimum oxidizer to fuel ratio (O/F) is near the stoichiometric ratio. Therefore, we determined the stoichiometric value for each propellant combination and then used the NASA CEC code to calculate propellant performance at O/F ratios below and above the stoichiometric ratio. For example, for the CO/O_2 liquid propellant combination:

$$CO + \frac{1}{2}O_2 \rightarrow CO_2$$

$$O/F_{Ratio} = \frac{\dot{m}_{O2}}{\dot{m}_{CO}} = \left(\frac{n_{O2}}{n_{CO}}\right)\left(\frac{MW_{O2}}{MW_{CO}}\right)$$

$$r = O/F = \left(\frac{0.5}{1}\right)\left(\frac{32}{28}\right) = 0.57$$

The overall propellant density of each propellant combination is calculated as follows:

$$r_{mix} = \frac{1+r}{\dfrac{r}{r_{ox}} + \dfrac{1-ML}{r_f} + \dfrac{ML}{r_m}}$$

where ML corresponds to the metal loading in cases where metal particles are used in conjunction with liquid or solid fuel.

With the main focus of this report to determine what type of propellant and propulsion system is optimal for the IMAV, we narrowed down the assessment to the best performing configuration by choosing one propellant combination for each class of rocket propulsion: liquid, solid, and hybrid. When analyzing the potential liquid propellant combinations, we considered cost, safety, and performance of propellant combinations.

We had three liquid propellant combinations which to choose from:

- Aluminum and LOX,
- Magnesium and LOX, and
- LCO and LOX.

These in-situ propellants were compared against methane and LOX, which is the propellant combination used in the baseline MAV. The first two liquid propellants mentioned are metallized liquid propellants where the metals are suspended in fine particulate form as a slurry or gel in the LOX oxidizer [8]. These two metallized propellants offer higher specific impulse and greater performance over the other liquid propellant combinations. Although the two metallized liquid propellants would provide a performance advantage, production of the fuel would be much more complex due to the process involving extraction of the metal particles from the surface of Mars. The two metallized propellants also have not been proven to be a safe or reliable fuel alternative since the fuel and oxidizer are mixed in a slurry potentially posing a explosion hazard. Therefore, although their performance is superior to LCO/LOX, we ruled out the metallized propellants as a possible liquid propellant for this mission.

The liquid propellants LCO/LOX and the baseline propellant methane/LOX have been compared thoroughly [5]. The methane/LOX combination was found to be a better performer with respect to LCO/LOX, but in order to produce the methane, 5.8 tons of liquid hydrogen would be needed to be transferred from Earth to Mars adding to payload cost [6]. Conversely, all the elements for the LCO/LOX propellant could easily be found on the surface of Mars therefore eliminating the cost of transporting 5.8 tons of hydrogen to Mars. Also the refining process to create methane is much more complex than the production of LCO and the energy required to produce methane would be 27% greater than producing LCO [6]. Based on cost, complexity, and safety, the LCO/LOX combination was the best choice for a liquid propellant.

Unlike the liquid rocket evaluation, the solid and hybrid choice for optimum propellant was based solely on performance. Thus, for each of the solid and hybrid propellants listed in Fig. 1, the top performer in terms of ρI_{sp}^2 was chosen as the candidate propellant combination for IMAV. In terms of this performance parameter, $Al/CO_{(S)}/O_2$ was chosen as a candidate hybrid propellant combination and $Al/CO_{2(S)}$ was chosen as a candidate solid propellant combination. Figure 3 is a plot of ρI_{sp}^2 vs. O/F for each of the identified propellant combinations.

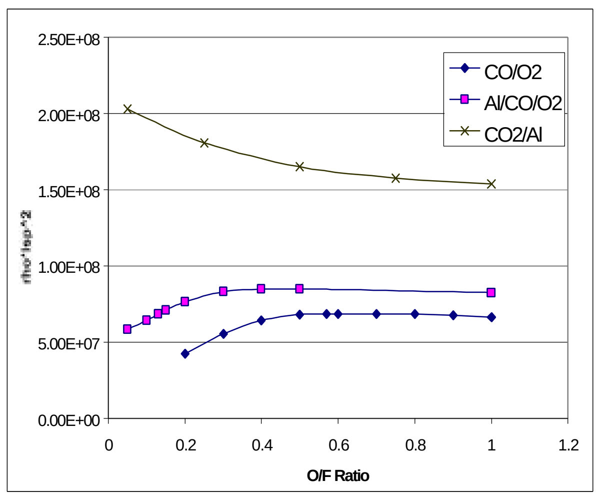

Figure 3. Calculated ρI_{sp}^2 vs. O/F ratio for liquid, hybrid and solid in-situ propellant candidates.

Having identified the top candidate propellant combination for each of the three rocket propulsion classes, we performed a more detailed analysis of each propulsion system and its corresponding candidate propellant.

Preliminary Design Configurations

Liquid Rocket Design

For the analysis of the liquid propellant operation, we used MS Excel to perform iterative design calculations taking into consideration all the factors affecting the performance of the CO/O_2 propellant. These calculations included vehicle payload, propellant characteristics, cargo bay size, and nozzle performance characteristics.

One assumption that we feel will help reduce costs and any additional need for testing is the use of a modified space shuttle main engine (SSME). By integrating a current main engine into the IMAV design, we set constraints that define the amount of thrust the IMAV can obtain. The current ratio of the area of the exhaust plane to the throat area of the nozzle on the SSME nozzle is 77.5.

$$\frac{A_e}{A_t} = 77.5 = \frac{\Gamma^2}{\left(\frac{P_e}{P_c}\right)^{1/g} C_F^o}$$

$$Thrust = C_F P_C A_T$$

We are able to calculate important parameters, C_f and the pressure of the chamber, which are the variable that define the thrust of the vehicle. From the calculation of thrust, we compare it to our overall weight of the IMAV. The ratio of the thrust over the weight must be greater than unity in oder for the vehicle to get off the surface of Mars. Once again, we used the NASA CEC code to calculate theoretical values for important parameters such as pressure ratios, temperature of the chamber, and molecular weight of the products. Using these parameters we extracted the remaining design variables including fuel, size of tanks, and the theoretical propellant performance from the spreadsheet.

The example Excel calculations are included in Appendix I for the chosen liquid propellant of CO/O_2. For comparison, the baseline spreadsheet of the reference mission is in Appendix II. The calculations examine how a methane/oxygen propellant might perform (CH_4/O2). The sheets illustrate that, while a large quantity of CO (compared to methane) is required to lift the IMAV into a low Mars orbit, there is a sufficient thrust to weight ratio and a reasonable propellant performance. The storage tank of the oxidizer was assumed to be spherical and the storage tank of CO needed to be cylindrical because of the large volume needed. Based on the theoretical size, will take up a majority of the available space in the cargo bay.

Solid Rocket Design

Next, we analyzed the solid rocket motor by calculating the ratio of the exposed solid burn area over the area of the throat using the following equation for equilibrium pressure in a solid rocket motor:

$$\frac{A_B}{A_t} = \frac{P_c^{1-n}}{r_p C^* a}$$

In this equation, P_c is the chamber pressure, n is the pressure exponent, r_p is the density of the propellant, C^* is the characteristic velocity (a measure of thermodynamic propellant performance) and a is the temperature coefficient. As in most rocket propulsion design studies, a nominal chamber pressure of 1000 PSIA was chosen. Once an approximate burn area ratio is calculated, the size of the fuel tanks can be determined.

To solve the equation we calculated the propellant density, and extracted C^* from the NASA CEC code. Since our optimum solid fuel is Al/CO_2(S), we assumed a pressure exponent according to similar solid fuels containing aluminum [9]. To determine a temperature exponent, we assumed a burning rate, $\dot{r}$ for the solid fuel and extracted a temperature exponent using:

$$a = \frac{\dot{r}}{P_c^n}$$

We calculated a burn area over throat area ratio of 156.4, which is comparable to other solid rocket engine designs [10]. Assuming a throat area similar to the SSME, the approximate initial burn area is 15 m^2. From this, we determined the approximate size of the rocket chamber assuming a cylindrical design. Figure 4 illustrates how the geometric shape of chamber depends upon the temperature exponent characteristic of the specific propellant.

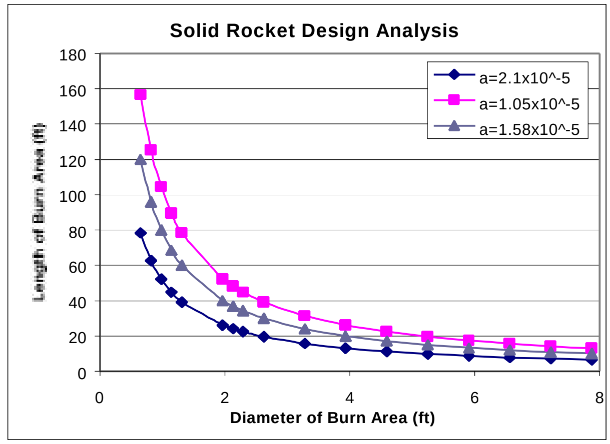

Figure 4. Length vs. diameter for Al/CO_2 solid rocket propellant motor.

The appropriate length and diameter of the burn area were determined to be 20 ft by 2 ft respectively assuming a temperature exponent equal to 2.1 x 10^{-5}. After including the overall diameter of the fuel, insulation, and the wall thickness, the tank diameter is approximately 7 ft and the length of the rocket is approximately 30 ft, which satisfies size constraints.

Hybrid Rocket Design Analysis

Analysis of the hybrid rocket design is somewhat difficult due to the lack of information and testing available on hybrid propellants. Orbitec is conducting a series of tests with various hybrid fuels that are applicable for Mars-based vehicle [12]. A shown in Figure 5, the group published the following graph comparing some of the fuels they tested. The focus of the paper was analyzing data for solid CO and oxygen. While there is no aluminum doping in the fuel, we assumed it be an accurate starting point based on the data for Al/$CH_{4(S)}$ fuel, which is also included in the figure. The graph contains a plot of the average fuel regression rate, $\dot{r}$ as a function of the average oxidizer mass flux, G_o. Through a logarithmic derivation of the standard empirical fuel burning rate formula, we were able to that if we estimate a data line representing $Al/CO/O_2$ on Figure 5 and extract a slope, we can determine an approximate temperature and pressure exponent from the graph.

$$\dot{r} = aG_o^n$$
$$\log \dot{r} = \log a + n \log G_o$$
$$y = b + mx$$

Calculating the burn area allows us to approximately determine rocket chamber size according to the following equation:

$$A_b = \frac{\dot{m}}{aG_o^n r_f}$$

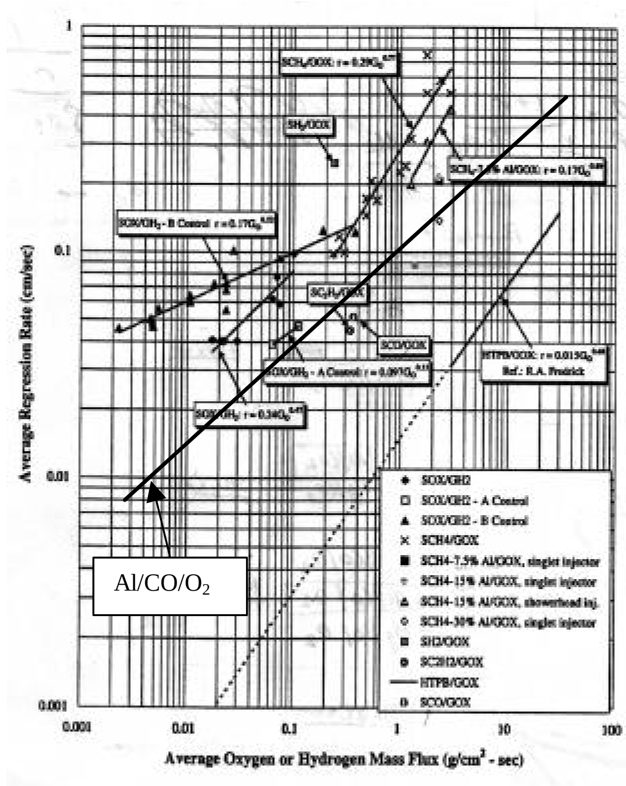

Figure 5. Fuel regression rate for hybrid rocket propellant combinations [12].

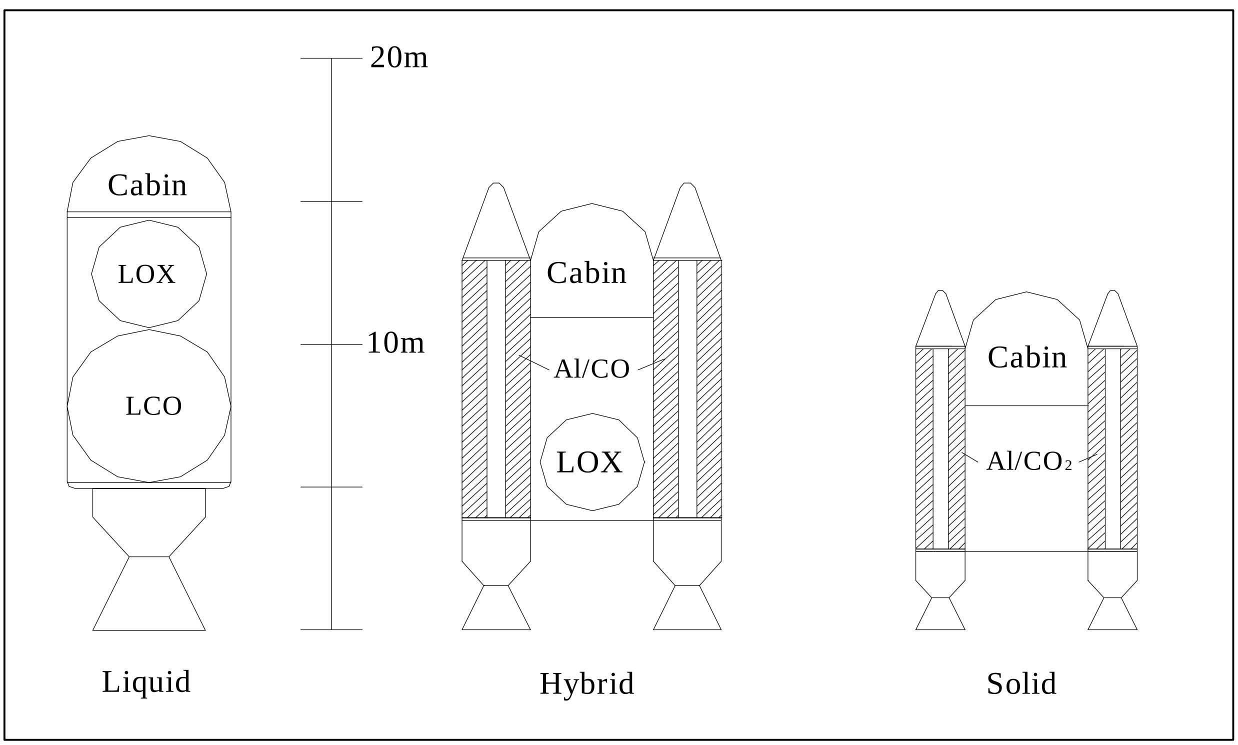

Figure 6. Approximate sizes of the MAV for each system according to the design calculations.

The approximate burn area is a function of average regression rate, $\dot{r}$, which we already know is a function of the mass flux rate, the temperature, pressure exponents, fuel density, and the fuel mass flow rate.

The calculations yield a burn area of around 25 m^2. Based on this calculation, it is possible to determine the size of the remaining solid rocket motor components. Following the same formula of the solid rocket design, we determined an overall tank diameter of 10 ft. In addition to the rocket chamber there must also be an oxidizer tank upstream resulting in an overall length of the rocket is an estimated 50 ft. In Figure 6, the scaled illustration shows the size comparison of all three different systems. This contrast is related to certain rocket design specifications such as the density of the propellant, the engine layout for each system, and the overall performance of the propellant.

CONCLUSIONS AND RECOMMENDATIONS

After thorough analysis of several in-situ propellants, we chose three candidate fuel/ oxidizer combinations, one from each rocket propulsion class. We then compared the performance, cost, and feasibility of the hybrid, solid, and liquid IMAV design configurations.

Liquid CO/O_2, the recommended choice for a liquid propulsion system is a capable fuel oxidizer combination for the IMAV. Although the theoretical performance is lower than the candidate hybrid fuel ($Al/CO/O_2$) and the solid fuel (CO_2/Al), the liquid CO/O_2 still offers many benefits. Derived from the abundant supply of CO_2 in the Martian atmosphere CO/O_2 is a safe and storable liquid in-situ propellant. CO/O_2 would also be easy to manufacture in a propellant production plant on Mars using presently available technology. The production has a minimal impact on the Martian environment and the testability of CO and O_2 make it possible to determine any uncertainties associated with this fuel choice. The design configuration demonstrates the large size of fuel tanks due to the quantity of required fuel to lift the IMAV into Martian orbit. While this requires that much of the space in the shuttle cargo bay would be occupied it does not eliminate CO/O_2 as an effectively performing fuel source. Finally, and perhaps most significantly, development of a reliable, safe CO/O_2 engine can be readily achieved using today's technology. Indeed, neither the SSME derivative engine (shown in Figure 7) described in this report nor the pair of RL-10 derivatives described in the Mars reference mission would require a quantum leap in technology.

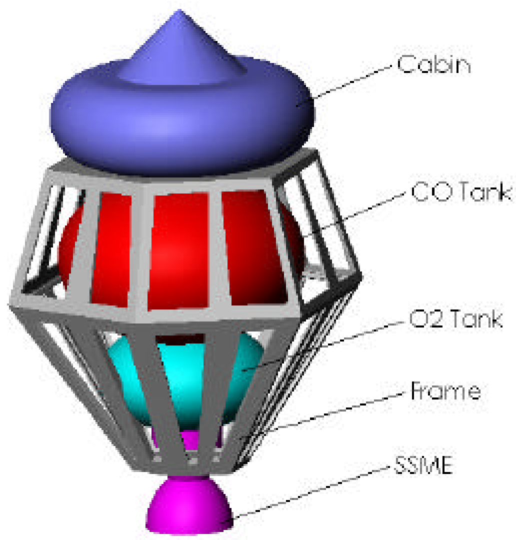

Figure 7. SolidWorks© illustration of CO/O_2 Mars Ascent Vehicle.

The recommended solid fuel choice, Al/CO_2, proved to be the best performing fuel of the three we examined. The tank size was very compact due to the density of the fuel, which in turn reduced the total weight of the IMAV design configuration. However, with solid propellant, there are severe political, economic, and safety drawbacks[11]. While proving to be extremely compact and simple, at the same time, solid fuel poses the greatest associated risk factor. With no known method, the need for technological advances in the mining of the aluminum metal from the Martian soil inflates the cost of the mission. The propellant production method must be extremely refined due to solid fuel's sensitivity to cracks and bubbles [9]. The use of solid Al/CO_2 also imposes more detrimental environmental effects to the Martian environment. Mining for the metal will create surface disturbances whose effects are undeterminable. Solid propellants' safety risk is widely known due to the uncontrollability of the rocket motor and the inability to throttle the engine or abort the launch once it has been fired. Finally, no data exists to verify that the Al/CO_2 combination would be ignitable and burn with a burning rate high enough for the development of a feasible rocket engine.

The recommended hybrid propellant $Al/CO/O_2$, performed slightly better than the recommended liquid propellant. Although it did not perform as well as the solid, the hybrid is a safer design because it is controllable. Unfortunately, comparatively little testing has been done with hybrid rocket motors in general, which delays mission time in order to conduct more thorough research and experimentation. Similar political and economic pressures of the solid are also associated with the hybrid design because of the need for the development of advance metal mining techniques to extract the aluminum metals from the soil. As stated before this will increase mission expenses, and possibly cause environmental damage to the Martian surface. The design for the hybrid shows the need for large oxidizer tanks and an extended rocket chamber for the solid CO/Al fuel mixture.

After significant consideration we have chosen the CO/O_2 liquid rocket engine as the optimum design configuration for the IMAV. The design offers a relatively high performance, the safety of a liquid engine, and the possibility of rapid development time due to the reliance on current and proven technology.

This paper is the culmination of a one-semester class on the fundamentals of rocket propulsion. The class helped us gain an extensive knowledge in the field of rocket propulsion, which we applied to the research and analysis conducted for this project.

Over the past year a separate group of students, as part of their Senior Engineering Clinic project, have built a hybrid rocket motor and test stand as shown in Figure 8. The work they have done provides a stepping stone for future effort into this project. The lack of test data on hybrid rocket engines was very limiting to this study, but by using the available test stand we can help develop the required data. A future project will be conducted to developing a hybrid rocket using $Al/CO/O_2$, our choice propellant, and verify our calculations with experimental data.

Figure 8. Test firing of hybrid rocket motor built by Rowan University students. The burning propellant is a Al/Epoxy/GOX mixture.

The research community has realized that in-situ propellants are the most viable option for a manned mission to Mars, but little research has gone into the harvesting of the elements to produce the propellants. It is evident that metallized propellants provide more energy but more research can be done on how to remotely obtain the metals from the Martian soil. A future study in the development of liquid carbon monoxide and oxygen from the Martian environment would allow us to more clearly illustrate the time and costs associated with our suggested fuel source.

OUTREACH

As engineers it is our duty not only to innovate but also to educate. With that in mind we looked to inform faculty members, our peers, the public, and the media regarding the scope of this project and the human exploration of Mars. Through presentations at local elementary schools we hope to inspire further interest in children to investigate the world around them and beyond.

Poster presentations on campus, such as the annual STEM (Science Technology Engineering Math) Symposium at Rowan University, provide outreach to students and faculty from a variety of studies. A distinguished campus-wide event, participation in the symposium exposed the project to hundreds of students, faculty, and guests.

As part of the Mechanical Engineering Department policy the design project was subject to a mid-semester design review. For the review a presentation was given to professors and students, who had the opportunity to question, critique, and evaluate the project team's progress. The development of a web page has allowed the project to be exposed to any interested party with access to the Internet. The web page provides important information on the project and the HEDS-UP Forum.

As part of a continuing cooperative relationship between Rowan University and our local newspaper, the Gloucester County Times, publicity for the project and the team's participation in the HEDS-UP Forum is already underway. Upon return from Houston final interviews with the team will take place and an article will be released.

APPENDICES

I. Baseline Spreadsheet for CO/O_2 Propellant

II. Baseline Spreadsheet for CH_4/O_2 Propellant

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Appendix I. Baseline Spreadsheet for CO/O_2 Propellant

Mission Parameters				**Vehicle Envelope: Space Shuttle Cargo Bay**			
Delta V	4500	m/s		Diameter	4.572	m	
Ambient Pressure	0.101526	psia		Length	18.288	m	
Isp	303	s		Volume	300.23	m^3	
g o [m/s^2]	9.8	m/s^2		Ms, Max	22727	kg	
g mars [in/s^2]	3.72	m/s^2		Mass SSME	2900	kg	
tb	120	s					
ML	2000	lbs					
Ms	22727	kg					
				Propellant Charactericstics			
Mission Calculations				Fuel	CO		
Max g's	37.5			Fuel Density	806	kg/m^3	
R	5.28			Oxidizer	O_2		
Final Vehicle Mass	22727			Oxidizer Density	1140	kg/m^3	
Initial Vehicle Mass	120222.56			O/F opt	0.5		CEC
Mp	97495.56	kg		Tc	3439	K	CEC
Mdot	812.46	kg/s		C*	1379	m/s	CEC
Mdot,ox	270.82	kg/s		MW	37.529	g/mol	CEC
Mdot,f	541.64	kg/s		little gamma	1.1235		CEC
Weight	1224638.4	N		big gamma	0.63		
Thrust	2413542.37	N					
Thrust/Weight	1.97			**Nozzle and Performance Characteristics**			
				Pc	6894000	1000	psia
Vehicle Calculations				Pc/Pa	9849.69		
Total Vehicle Mass	124963.10	kg		Pe/Pc, adapted	0.000101526		
Vehicle Dry Mass	22727	kg		Ae/At, actual	76.28		
Mox	32498.52	kg		Ae/At, adapted	77.5		
Mf	64997.04	kg		Pe/Pc, actual	0.0012908		CEC
Vox	28.57	m^3		Cfo	1.96		CEC
Vf	80.64	m^3		CF,sea level	2.154		
Dia, ox sphere	3.79	m		At	0.162	m^2	
Dia, fu cylinder	4.13	m		Ae	12.397	m^2	
Length of fuel cylinder	6	m		dia t	0.513	m	
Engine Mass	2900	kg		dia e	4.48	m	
Tank Mat'l Density	2800	kg/m^3		Thrust	2413542.371	N	
Tank Wall Thickness	0.00635	m		Isp	303.1274372	s	
Ox tank mass	401.87	kg					
Fuel Tank mass	478.66	kg					
Passengers(4)	260	kg					
Passenger Equipment	100	kg					
Surface Samples	100	kg					
Experimental Data	200	kg					
Navigational Equipment	300	kg					

Appendix II. - Baseline Spreadsheet for Reference Mission

Mission Parameters				**Vehicle Envelope: Space Shuttle Cargo Bay**			
Delta V	4500	m/s		Diameter	4.57	m	
Ambient Pressure	0.101526	psia		Length	18.28	m	
Isp	406	s		Volume	300.23	m^3	
g o [m/s^2]	9.8	m/s^2		Ms, Max	22727	kg	
g mars [in/s^2]	3.72	m/s^2		Mass –RL-10 (2)	300	kg	
tb	120	s					
ML		lbs					
Ms	22727	kg					
				Propellant Charactericstics			
Mission Calculations				Fuel	CH4		
Max g's	37.5			Fuel Density	717	kg/m^3	
R	3.46			Oxidizer	O_2		
Final Vehicle Mass	22727			Oxidizer Density	1140	kg/m^3	
Initial Vehicle Mass	78786.56			O/F opt	3.4		CEC
Mp	56059.56	kg		Tc	3543	K	CEC
Mdot	467.16	kg/s		C*	1859.1	m/s	CEC
Mdot,ox	360.98	kg/s		MW	21.21	g/mol	CEC
Mdot,f	106.17	kg/s		little gamma	1.128		CEC
Weight	817580.30	N		big gamma	0.63		
Thrust	1859241.2	N					
Thrust/Weight	2.27			**Nozzle and Performance Characteristics**			
				Pc	6894000	1000	psia
				Pc/Pa	9849.69		
Vehicle Calculations				Pe/Pc, adapted	0.000101526		
Total Vehicle Mass	83426.56	kg		Ae/At, actual	69.63		
Vehicle Dry Mass	22727	kg		Ae/At, adapted	77.5		
Mox	43318.75	kg		Pe/Pc, actual	0.001392		CEC
Mf	12740.81	kg		Cfo	1.96		CEC
Vox	37.99	kg		CF,sea level	2.140		
Vf	17.76	kg		At	0.125	m^2	
Dia, ox sphere	4.17	kg		Ae	8.772	m^2	
Dia, fu sphere	3.23	kg		dia t	0.4519	m	
Engine Mass	300	kg		dia e	3.7710	m	
Tank Mat'l Density	2800	kg		Thrust	1859241.19	N	
Tank Wall Thickness	0.00635	kg		Isp	406.10	s	
Ox tank mass	486.66	kg					
Fuel Tank mass	293.32	kg					
Passengers(4)	260	kg					
Passenger Equipment	100	kg					
Surface Samples	100	kg					
Experimental Data	200	kg					
Navigational Equipment	300	kg					