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Propellant Trade Study for a Crew Space Vehicle

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A trade study is conducted to determine the best propellant combination for a notional crew space vehicle. The assumed 5000 ft/s spacecraft is divided into a command module and service module like Apollo and provides transportation of astronauts and supplies to low Earth orbit, the International Space Station, libration point one, and one-way transfer from lunar orbit to Earth. Twenty-five different propellant combinations are evaluated across nine important evaluation criteria that include mass, development, safety, complexity, reliability, flexibility, contamination, commonality, and Mars in-situ producibility. Nontoxic and Mars-producible are decided to be important requirements for an affordable Earth-moon-Mars exploration architecture. The assumptions when coupled with a mathematical model to estimate vehicle wet mass, lead to the recommendation of liquid oxygen and liquid methane for orbital maneuvering and gaseous oxygen with gaseous methane for reaction control. The new propellant combinations require up-front investment that includes new or modified engines, ground infrastructure, long term cryogenic storage technology, and for the later occupation of Mars in-situ production of methane and oxygen for propulsion.

Nomenclature

a	=	tank dome semi-major axis, ft
b	=	tank dome semi-minor axis, ft
e_W	=	weld efficiency
g_C	=	acceleration due to Earth gravity, ft/s^2
I_{SP}	=	vacuum specific impulse, s
$I_{SP,EFF}$	=	effective vacuum specific impulse, s
d_{TANK}	=	tank diameter, ft
K_{CYL}	=	cylinder wall thickness multiplier
K_{CR}	=	crown wall thickness multiplier
K_{KS}	=	knuckle stress factor
K_{KN}	=	knuckle wall thickness multiplier
L_{STAGE}	=	stage length, ft
L_{TANK}	=	tank length, ft
M_i	=	initial vehicle mass, lbm
M_f	=	final vehicle mass, lbm
M_{STR}	=	additional structural mass for tanks, lbm
MR	=	oxidizer-to-fuel mixture ratio
P_C	=	combustion chamber pressure, psia
P_T	=	maximum tank pressure, psia
S_W	=	maximum material working stress, psid
t_{CROWN}	=	dome wall thickness at crown, in
t_{CYL}	=	cylinder wall thickness, in
t_{KNUCK}	=	dome wall thickness at knuckle, in
ΔV	=	delta velocity, ft/s
ε	=	nozzle expansion ratio
η_{PACK}	=	tank packing efficiency
η_{ISP}	=	specific impulse efficiency

I. Introduction

SINCE the Gemini program forty years ago, human-rated space vehicles have used toxic hydrazine-based propellants for propulsion. Particularly, the current Space Shuttle's in-space propellants are nitrogen tetroxide (N2O4) and monomethylhydrazine (MMH). Both of these hypergolic, toxic fluids pose safety risks and require extensive human protective equipment during tank loading and post-flight processing that lead to high operations costs per flight. The renewed national interest in human space exploration to the moon and Mars will require a new fleet of human-rated spacecraft and pre-positioned resources to provide transportation and sustain human life. For such an Earth-moon-Mars exploration architecture the choice of spacecraft propellants is important because it directly affects the amount of mass that must be launched from Earth to establish the initial human presence as well as the mission costs once the system is operational. By use of several mathematical models and qualitative assessments various propellant combinations are evaluated for a notional multi-use crew space vehicle that travels to the International Space Station (ISS), performs rendezvous and docking with other spacecraft, and provides one-way travel between the moon and Earth.

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II. Approach

The propellant trade study for the crew space vehicle begins with definition of the spacecraft and its mission. Candidate propellants are selected and evaluated quantitatively in mass and qualitatively in several defined evaluation criteria. The results are tabulated and analyzed to determine a recommended propellant combination for the crew space vehicle.

A. Vehicle Description

The crew space vehicle is separated into two stages identical to that on Apollo. The forward stage or Command Module (CM) is a capsule that contains the human crew as well as subsystems needed for life support, communication, and control; it has a small three-degree-of-freedom (3DOF) propulsion system for re-entry on Earth. The aft stage or Service Module (SM) contains the main propulsion system including propellants and pressurants plus most of the power, avionics, and thermal control subsystems. The SM has a six-degree-of-freedom (6DOF) propulsion subsystem that provides propulsive power for major orbit changes, attitude control, spacecraft rendezvous, and docking. RCS propellants are common on the CM and SM and the in-space life of the spacecraft is up to one year. The NASA photograph in Fig. 1 depicts the Apollo 9 Command-Service Module (CSM) during integration at the Kennedy Space Center in 1968.

Figure 1. Apollo 9 CSM During Integration

In the present trade study the total dry mass of the spacecraft varies depending on the required propellant and pressurant tank sizes plus the propellant management hardware needed for each concept. Accordingly, only a base subsystem and vehicle structure mass that does not include propulsion is specified for the SM and CM. For the SM this non-propulsion system mass is assumed to be 7000 lbm and for the CM it is 12000 lbm. Also, structural mass is added to the SM by the mass model described below depending on the total length of the propellant tank arrangement. In addition to these masses the CM is assumed to carry 2000 lbm of cargo that may include astronauts.

B. Mission Description

The defining mission for the CSM is a notional space exploration trip that includes several in-space dockings and transportation from orbit around the moon to the Earth's surface. It is assumed that the CSM is placed in a circular low lunar orbit (LLO) by a stage from Earth and thus it only provides one-way, moon-Earth travel. The propulsion system in the SM provides all in-space maneuvering and moon-Earth transportation. At Earth entry, the expendable SM is jettisoned to burn up during descent. The CM continues down to the ground with the human crew where its propulsion system provides attitude control for the descent and safe return. A crew vehicle with this propulsive capability can also be used for orbital maneuvering in low Earth orbit (LEO) or LLO, rendezvous and docking with various spacecraft, transportation to the ISS, and transportation from either the moon or Earth to Lagrange libration point one (L1).

C. Propellant Selection

The ideal space exploration propellant has a high specific impulse and a high density. It is nontoxic to humans, nonhazardous to store, and noncontaminating to spacecraft. It is also compatible with most materials, sloshes little, is producible on Earth, moon, and Mars, and costs little. Finally, the ideal propellant uses existing, in-production engines and propulsion components.

In practice, unfortunately, only a few of these goals are typically met when selecting spacecraft propellants. For the CSM and its defined mission, 25 different combinations of OMS and RCS propellants are investigated. Table 1 lists the propellant cases studied which includes several state-of-the-art fluids in use today as well as cryogenic and solid-based concepts.

1. Storable OMS and RCS Propellants

Some of the propellants evaluated are storable in liquid phase at ambient Earth conditions and do not require active cooling unlike cryogenic propellants. Storable propellant propulsion systems have the advantage that they are simpler and potentially more reliable due to fewer devices that must operate correctly. Storable propellants can be practically used for both OMS and RCS functions.

N2O4 and MMH are listed first since they have been used so frequently for human-rated spacecraft due to their reliable hypergolic ignition, moderate specific impulse, and high storage densities. Apollo, Space Shuttle, and many satellites have used N2O4/MMH bipropellants, and thus there is a wide variety of qualified engines and components available for the CSM. However, special protective gear and training are required for working with these toxic liquids.[1] For this reason operations costs per mission are higher for toxic propellants than for nontoxic ones.[2,3] Toxic hydrazine (N2H4) also has an extensive spacecraft history. It is a monopropellant and a hypergolic bipropellant fuel that combines with N2O4 to allow a dual-mode propulsion option. Here, a bipropellant OMS is coupled with a monopropellant RCS using the same fuel. Such N2O4/N2H4 systems have been successfully used on the Chandra, A2100, and other spacecraft.[4,7] Two storable alcohols are also evaluated for OMS and RCS, specifically, toxic methanol (MeOH) and nontoxic ethanol (EtOH).

Table 1. Propellant Candidates

Concept	OMS Oxidizer	OMS Fuel	RCS Oxidizer	RCS Fuel
Case 1	N2O4	MMH	N2O4	MMH
Case 1a	N2O4	MMH	N2O4	MMH
Case 2	N2O4	N2H4	N2O4	N2H4
Case 3	N2O4	N2H4	-	N2H4
Case 4	-	N2H4	-	N2H4
Case 5	LO2	LH2	GO2	GH2
Case 6	LO2	LH2	N2O4	MMH
Case 7	LO2	LH2	GO2	EtOH
Case 8	LO2	LH2	-	N2O
Case 9	LO2	LCH4	GO2	GCH4
Case 10	LO2	LCH4	N2O4	MMH
Case 11	LO2	LCH4	GO2	MeOH
Case 12	LO2	LCH4	GO2	EtOH
Case 13	LO2	LCH4	-	N2O
Case 14	LO2	EtOH	GO2	EtOH
Case 15	LO2	MeOH	GO2	MeOH
Case 16	LO2	LC2H6	GO2	GC2H6
Case 17	LO2	LC2H6	N2O4	MMH
Case 18	LO2	LC2H6	GO2	EtOH
Case 19	LO2	LC3H8	GO2	GC3H8
Case 20	LO2	LC3H8	N2O4	MMH
Case 21	LO2	LC3H8	GO2	EtOH
Case 22	-	Solid	N2O4	MMH
Case 23	-	Solid	GO2	EtOH
Case 24	-	Hybrid	N2O4	MMH
Case 25	-	Hybrid	GO2	EtOH

2. Cryogenic OMS Propellants

Cryogenic OMS systems are studied that use normal boiling point (NBP) liquid oxygen (LO2) with a variety of fuels. Cryogenic fluids require active refrigeration for long duration missions to avoid large mass penalties from fluid loss due to boil-off and the required overpressure venting of the OMS tanks. It is estimated that a CSM with a one-year, in-space lifetime will require powered cryogenic coolers to thermally condition the OMS propellants in addition to passive thermodynamic vent systems (TVSs). The cryogenically stored fuels considered with LO2 include liquid hydrogen (LH2), liquid methane (LCH4), liquid ethane (LC2H6), and liquid propane (LC3H8). Of the four cryogenic fuels LH2 has the greatest design challenge because of its much colder temperature.

3. Difficulties with Cryogenic RCS Propellants

Cryogenic RCSs are not proposed in the present study due to difficulty with thermal conditioning in long RCS feed lines and potential erratic start profiles or delays. The primary issue with use of cryogenic liquids in an RCS is the ability to keep the propellants in liquid form at the thruster inlet. Here, stagnant cryogens in the feed lines over time will absorb heat, increase in temperature to saturation, and then boil within the lines. An initial engine start-up with two-phase propellants can cause large mixture ratio variations and cooling reductions that lead to catastrophic engine failure. Non-critical problems with unconditioned propellants at engine inlets include non-repeatable start profiles and ignition delays in the RCS. This effect is compounded by the engine thermal mass which ideally should be near propellant temperatures for smooth, repeatable, and reliable engine starts. The above problems can lead to more serious issues if the RCS is slow to start during a spacecraft rendezvous, docking, or collision avoidance maneuver. Another propellant feed system issue with a cryogenic RCS is the freezing of the propellant with the higher melting point in small thrusters. This can certainly occur if one propellant is storable and the other is cryogenic, however, propellant freezing can also happen with two cryogenic liquids.

In order to make a cryogenic RCS feasible both propellant conditioning and engine start issues must be addressed. One complex option for conditioning is to recirculate propellant within the vehicle with a separate set of lines, valves, and pumps to flow propellant from each thruster bank back to the propellant tanks. This could possibly be integrated with the long-term storage hardware needed for the propellant tanks where a recirculation loop is integrated with a thermodynamic vent system. However, a recirculation system is another set of hardware that increases both complexity and mass of the CSM and must function in order for the spacecraft to operate reliably and repeatably. Another concept to condition propellant is to regularly fire every thruster at scheduled intervals, but this approach wastes propellant, constrains orbital operations, increases contamination risk, and causes operational difficulties if the CSM is docked with other space assets. An overboard dump system could maintain conditioned propellants but at the expense of fluid loss, high contamination risk, and pre-planning similar to regularly firing the engines. One more option is to require that the delivered engines can start with multi-phase propellants. This would be difficult to do repeatably particularly for minimum impulse firings due to large mixture ratio variations.[5] It is for the above-mentioned reasons that a cryogenic RCS is not considered in the present investigation.

4. RCS Propellants

Although both fluids are toxic, N2O4 and MMH are investigated with multiple OMS propellants to determine the CSM mass with these well-developed, in-space propellants. Storable, nontoxic RCS alternatives include gaseous oxygen (GO2) with EtOH or a monopropellant N2O system. GO2/MeOH is included only for fuel commonality in the LO2/MeOH OMS case. In propellant types with cryogenic OMS propellants a gaseous RCS using the same propellants is always considered. Thus, GO2 is paired with gaseous hydrogen (GH2), gaseous methane (GCH4), gaseous ethane (GC2H6), and gaseous propane (GC3H8). Here, no liquid-gas conversion hardware is assumed, rather the needed gaseous propellants are stored in high-pressure bottles filled before launch.

5. Solid-based OMS Propellants

Solid rocket motors (SRMs) and hybrid solid motors are considered as OMEs in four of the cases studied. Each is combined with an RCS using either N2O4/MMH or GO2/EtOH.

6. Propellants Not Considered

The list of 25 propellant combinations for the CSM in Table 1 is not all-inclusive as many propellants are specifically not considered. Hydrogen peroxide (H2O2), for example, is not used because of storability and decomposition concerns.[6] H2O2 cannot use existing screen-based PMDs like other storable propellants for low-gravity liquid acquisition due to decomposition and screen material incompatibility. Any elastomeric bladder concepts will be unlikely to meet a one-year life requirement of wet storage in H2O2. H2O2 also becomes shock sensitive at moderate temperatures. Another propellant not considered is unsymmetrical dimethylhydrazine (UDMH) which performs similarly to the MMH fuel but is used mainly in Russian and European spacecraft and is no longer popular domestically. All hydrocarbon fuels heavier than propane (C3H8) are not considered due to their tendency for soot production and high spacecraft contamination potential. Ethylene or ethene (C2H4) is excluded for its high reactivity and low stability in storage. New propellants still in development such as HAN-based formulae are not considered nor are fluorine-based propellants that are difficult to handle, store safely, or are relatively untested.[7] Use of cold gas systems and other low-performance monopropellants are not specifically addressed; the N2O cases are considered to be representative of these propellant types.

D. Evaluation Criteria

Nine important criteria are defined to evaluate each propulsion system type including mass, development, safety, complexity, reliability, flexibility, contamination, commonality, and Mars in-situ producibility. Mass is computed *quantitatively* while the other eight parameters are determined *qualitatively*. Each criterion is described below.

1. Mass

The mass of the fully loaded CSM at launch from Earth is used for quantitative comparison of each propulsion concept. This mass, also termed the wet mass, is estimated mathematically as explained below and includes the dry CSM vehicle, all loaded propellants, pressurants, and cargo. It does not include the mass of a separate flight abort system. Propellant concepts with the lowest wet mass are considered the best. In the present study CSMs with wet masses less than 48,000 lbm are considered good, those with masses from 48,000 to 53,000 lbm are fair, and CSMs more massive than 53,000 lbm are relatively poor.

2. Development

The amount of technology and hardware development required is an important criterion since it defines the funding needed to build the first spacecraft and to establish the launch and processing facilities. New technology development frequently has unexpected technical obstacles that can lead to longer development cycles and higher costs than originally planned. New propellants require new ground infrastructure at launch and landing sites in order to load, drain, test, and analyze the CSM propellants. This also includes development of the procedures specific to the fluid and training of ground support personnel. For some propellants there are already existing OMS and RCS engines that can be used, however, many propellants considered require new engines to be developed. Also, new propellants need qualification programs for much of the valve and tank hardware before use on CSM. In the current analysis propulsion concepts using storable propellants and existing or modified OMS and RCS engines are rated as good in development. Systems with LH2 storage generally have a poor development rating while non-LH2 cryogenic propellants such as LO2, LCH4, LC2H6, and LC3H8 are considered fair in development because they have significantly higher liquid temperatures and less challenging design requirements than LH2. One exception for LH2 is Case 6 which has an LO2/LH2 OMS coupled with an N2O4/MMH RCS; this combination can use existing

OMEs, RCEs, small components, and ground infrastructure. Since engine and plumbing components are already developed and the only large development task is LH_2/LO_2 storage, Case 6 is rated as fair in development.

3. Safety

Safety of the propellant concepts is measured based on the toxicity and hazardous nature of the propellants. Toxic propellants that require significant protective gear and training have relatively poor safety when compared to low-toxicity propellants. Hypergolic propellants that react on contact such as N_2O_4/MMH or N_2O_4/N_2H_4 provide reliable ignition once in a rocket engine, but pose a large safety hazard to both the ground and flight crew if leaks occur. Solids are considered safe because human exposure to the propellant is highly unlikely. Hybrid OMEs in the study are assumed to have a nontoxic fluid propellant and thus are also safe. The propellant trade study safety criterion is considered poor for CSMs employing both a toxic OMS and RCS, poor if the OMS propellants are hypergolic, fair if the RCS is toxic and the OMS is nontoxic, and good if both the OMS and RCS propellants are nontoxic or safe .

4. Complexity

The complexity parameter relates to the number of propulsion components needed for the specific design. Thus, a solid or a monopropellant propulsion system has fewer valves and small components than a bipropellant system designed to the same mission requirements. Accordingly, in the current analysis the number of different propellants in each CSM are used to define the complexity ratings. Here, the fluid phase is also measured and so LO_2 and GO_2 count as two propellants. For the present CSM propellant research, concepts with two or less total propellants are considered good, those with three propellants are fair, and those with four propellants are poor.

5. Reliability

The probability that a given CSM concept can complete its mission is used for a reliability metric. Designs that have complex OMS propellant refrigeration and propellant management hardware are judged as less reliable than those that do not need such devices. LH_2 cryogenic systems in particular are considered less reliable in space than LO_2, LCH_4, LC_2H_6, and LC_3H_8 propellant storage systems due to the much greater cooling requirements and in turn the associated implications of a refrigeration system failure. The OMEs used for major orbit changes and the return to Earth are particularly critical to mission success and thus OMEs that use hypergolic or solid propellants with highly reliable ignition are best. In the present work propulsion concepts utilizing LH_2 are rated as relatively poor in reliability while other cryogenic propellants are considered fair. Hypergolic and solid OMEs are considered good, but OMEs with hybrid propellants are regarded as fair since they contain both a solid and liquid or gas subsystem that must function.

6. Flexibility

The flexibility of the CSM is defined as the ability of it to adapt to new missions that may include different orbit changes, thermal environments, or a mission life longer than the one-year design requirement. Thus, the most flexible vehicle will have an OMS propulsion system that is unaffected by unplanned trajectory changes, higher heating rates, and mission durations longer than one year. In the current CSM trade study, concepts utilizing solid rocket motors for OMEs are rated as poor since SRMs cannot be restarted multiple times without pre-installed propellant barriers that must be tailored to the specific mission. Conversely, storable liquid propellant OMEs and hybrid OMEs are good in flexibility since the engines can be started or stopped as needed. Here, the hybrid OMEs are assumed to use a storable liquid propellant. Systems that use liquid hydrogen are rated as poor since the cryocooler and propellant management designs are closely coupled to the expected thermal loads during flight. Other cryogenic propellants such as LO_2 and hydrocarbons are judged as fair in flexibility because of their higher-than-LH_2 operating temperatures and potentially greater design margins.

7. Contamination

The degree of contamination of nearby spacecraft and structures by the CSM's propulsion system is related to the chemical composition of the RCS propellants and their combustion products. OMS engines are typically fired a safe, far distance away from the other spacecraft, but the RCS is used for close-in, proximity operations where thruster plumes can impinge on critical surfaces. Here, the thruster exhaust can damage or cloud optical sensors and alter the radiative characteristics of spacecraft hardware. Similarly, propellant leaks may cause not only sensor damage but also structural damage to the CSM and nearby craft due to propellant chemical reactions with the spacecraft materials. N_2O_4/MMH thrusters have a poor contamination rating in the present study for their production of monomethylhydrazine nitrate. Combinations with N_2H_4 and the heavier hydrocarbon fuels studied are rated as fair while RCSs using N_2O monopropellant or GO_2 combined with GH_2, GCH_4, or alcohols are considered good with respect to contamination.

8. Commonality

The largest launch vehicle upper stages in use today have cryogenic LO_2 and LH_2 for propulsion. Unlike the CSM these stages do not have long, on-orbit life nor do they require significant micrometeoroid orbital debris

(MMOD) protection for the propellant tanks. Accordingly, a high-specific-impulse propulsion system that uses a high-density oxidizer and a low-density fuel is an acceptable design solution. By this reasoning and the potential for water ice on both the moon and Mars, oxygen-hydrogen propulsion is expected to be present in some of the vehicles used for space exploration. Potential O2/H2-propelled vehicles include trans-lunar injection (TLI) stages, trans-Earth injection (TEI) stages, and lunar/planetary landers. If O2 or H2 propellants are chosen for the CSM, there will be propellant commonality with these other spacecraft that could lead to operational cost savings and standardized common parts. Commonality in this case also relates to lunar in-situ producibility or the ability to manufacture oxygen and hydrogen from any available water ice on the moon. In the present study propellants using only O2 and H2 are considered good in commonality, those using only one propellant (always O2) are fair, and concepts with neither O2 nor H2 are poor.

9. Mars Producibility

The last evaluation criterion is Mars in-situ producibility or the potential for propellant production using only readily available Martian natural resources. Mars's atmosphere is mostly carbon dioxide (CO2) which also exists at the poles as dry ice along with water ice. Thus, carbon (C), hydrogen (H), and oxygen (O) elements are present at the Martian surface in relatively large quantities and available for propellant production. Spacecraft propellants made of only of carbon, hydrogen, and oxygen elements are, therefore, considered producible on Mars while those that include other elements such as nitrogen cannot be easily made on Mars. For the current analysis a concept is considered good in Mars producibility if both the OMS and RCS propellants are made from only C, H, and O. If both OMS and RCS use nitrogen or solid-based propellants then the rating is poor. Designs with a Mars-producible OMS but a nitrogen-based RCS are considered fair.

E. Mass Model

The CSM wet mass is computed iteratively based on the propellants selected and many important inputs for the spacecraft. The model is divided into the following five separate sections that are described hereafter: 1) propellant and pressurant, 2) engine characteristics, 3) delta-velocity and usable propellant, 4) tank design and loaded fluids, and 5) dry mass.

1. Propellant and Pressurant

The propellant and pressurant model receives fluid type, storage pressure, and storage temperature arguments and returns the thermal and physical properties of the fluid at the specified state point. The properties calculations are based on the well-established NIST Standard Reference Database for fluid thermodynamic and transport properties.[8] Note that solid and hybrid propellants do not use the NIST database and are treated differently as described below. The computed propellant and pressurant properties are used for sizing of the CSM fluid tanks. For all storable liquids and gases a temperature of 90 °F is used; this warm temperature is purposefully selected to provide conservative tank sizes from propellants with slightly lower densities than at standard conditions. All cryogenic OMS propellants are used at their NBP conditions and a 20 psia initial tank pressure. The operational pressure of each OMS tank is assumed to be 40 psia to feed the pump-fed OMEs. Storable liquid propellants used for RCS have tank pressures of 300 psia while gaseous propellant and pressurant bottles are filled to 6000 psia. Figure 2 illustrates the relative storage densities of each propellant investigated. As observed on the density plot N2O4 has the largest density of all propellants studied at 88 lbm/ft^3 while liquid oxygen is second densest at 71 lbm/ft^3. Hydrazine and alcohol-based fuels have moderate densities as do the heavier hydrocarbons. It is interesting to note that some of the nonliquid propellants at 6000 psia have densities that are larger than liquid methane's density. High-density propellants are desired to minimize vehicle size and mass, however, engine performance is also important.

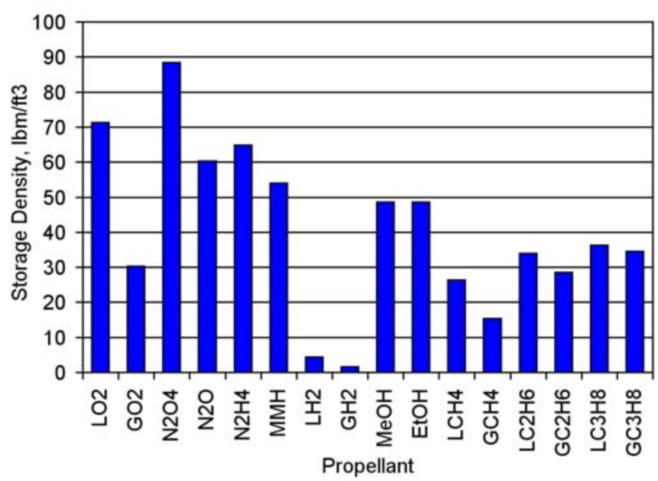

Figure 2. Propellant Storage Densities

2. Engine Characteristics

The NASA Gordon and Mcbride chemical equilibrium program is used for all engine specific impulse, I_{SP}, calculations.[9] The model takes reactant types, initial temperatures, mixture ratio, MR, chamber pressure, P_C, and area expansion, ε, as inputs and computes the theoretical maximum vacuum specific impulse assuming equilibrium nozzle chemistry for the particular engine.

The maximum theoretical performance is never achieved in practice and thus an I_{SP} efficiency, η_{ISP}, is applied to each engine model dependant upon propellant type and operational mode. This yields the effective specific impulse, $I_{SP,EFF}$, that is used for vehicle design. OMEs with cryogenic fuels are assumed to have regeneratively-cooled nozzles and thus an I_{SP} efficiency of 96% is used; other bipropellant OMEs have an efficiency of 93%. RCS thrusters frequently operate in a pulsing mode where the engines are turned on and off rapidly. This type of use prevents the engine from reaching its steady-state performance level and thus all bi-propellant RCS thrusters use an efficiency of 86%.[10] In monopropellant systems such as N2H4 and N2O an efficiency of 99% is used. For the current propellant trade study three OMEs are assumed on the SM each with 5000 lbf of vacuum thrust for 15000 lbf total OME thrust. SM RCEs provide 25 lbf each for orbital maneuvering and docking, and the CM RCEs create 100 lbf thrust each for 3DOF control during Earth entry. Solid and hybrid-based OMEs have four gas generators feeding a common manifold that generates 15000 lbf of thrust.

Pump-fed orbital maneuvering engines are assumed for the present study to minimize CSM mass by avoiding heavy, high-pressure propellant tanks needed for a pressure-fed OMS. This is a departure from the designs of previous human-rated vehicles such as Apollo and Space Shuttle that have pressure-fed engines. Pressure-fed systems are generally more reliable than pump-fed ones, but as shown in the calculations below, a pump-fed OMS is lighter than pressure-fed for the mission considered. More importantly a pump-fed OMS design with low pressure tanks is easily scalable to bigger sizes without large mass penalties for more demanding exploration missions.

For the pump-fed OMEs a chamber pressure of 1000 psia is assumed while for the pressure-fed RCEs a chamber pressure of 150 psia is used. The only pressure-fed OMS evaluated (Case 1a) also has a P_C of 150 psia. The mixture ratio for each bipropellant OME is selected between 75-80% of the stoichiometric mixture ratio, MR_{STOICH}, which is consistent with existing engines in use today. RCEs typically have less cooling potential than larger OMEs and operate near 70% of MR_{STOICH} except for all gaseous thrusters which assume an MR equal to 50% of MR_{STOICH}. For area expansion the OMEs assume a 100:1 expansion nozzle while the RCEs use 25:1. In the solid and hybrid OME cases a detailed specific impulse model is not used, rather, both engines are assumed to have an $I_{SP,EFF}$ of 310 seconds. Thus, no performance differences are assumed for solid and hybrid propellants. The distinction between solid and hybrid-based OMEs is evident in the better flexibility rating of hybrids contrasted by the higher reliability and development maturity of solids.

Figures 3 and 4 depict the effective specific impulses of all OMS and RCS engines used in the crew space vehicle propellant trade study. The best OME specific impulse is produced by LO2 and LH2 at almost 450 seconds. The poor 240-second performance of monopropellant hydrazine is easily noticed. The remaining OMEs vary in effective specific impulse from 310 to 364 seconds. For the RCS systems in Fig. 4 the GO2/GH2 combination yields the highest pulse-mode specific impulse at 400 seconds while the monopropellants N2H4 and N2O have the worst at 221 and 195 seconds, respectively. Each other RCS engine has an $I_{SP,EFF}$ between 275 and 289 seconds.

Specific impulse, however, is not the only metric for propulsion system design; propellant density is also important. For example oxygen-hydrogen has an excellent engine specific impulse but requires a large vehicle to hold the low-density hydrogen. The best combination of engine performance and packaging varies depending upon the specific vehicle design and the delta velocity that the propulsion system must provide.

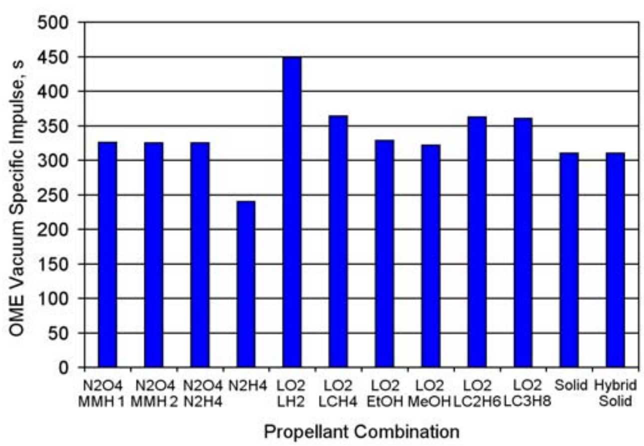

Figure 3. OME Effective Specific Impulse vs. Propellant Combination

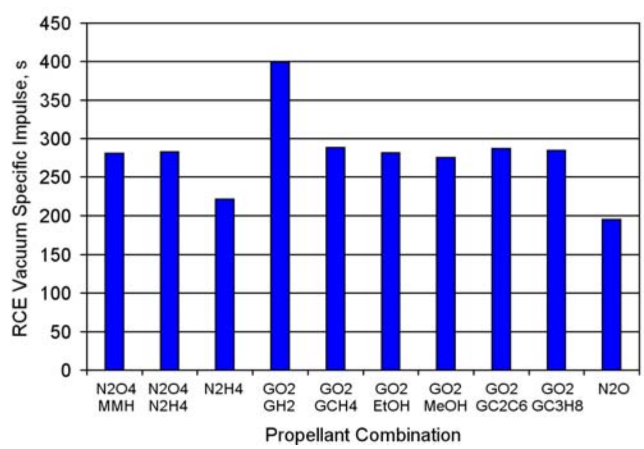

Figure 4. RCE Effective Specific Impulse vs. Propellant Combination

3. Delta Velocity and Usable Propellant

For the mission studied a service module delta velocity of 4900 ft/s is assumed while the command module capability is 100 ft/s. The SM mission includes propellant for the one-way trip from LLO to Earth plus enough ΔV for four rendezvous and docking attempts with other spacecraft. Here the CSM is assumed to perform the active role in the docking maneuvers with a passive, cooperative target. Allocations for a mid-course correction, one year of loiter in LLO or LEO, separation, and evasive maneuvers after undocking are also included in the 4900 ft/s capability. The 3DOF RCS in the CM uses its 100 ft/s for fine adjustment of the entry interface and attitude control during descent to the Earth's surface.

The amount of usable propellant needed by the OMS and RCS propulsion systems to complete the desired mission is calculated with the standard rocket equation shown below in Eqn. 1. For each maneuver the ΔV requirement, effective engine specific impulse, $I_{SP,EFF}$, and the initial mass, M_i, are input to determine the final vehicle mass, M_f, after each engine firing. This result of these calculations is the total usable propellant required for the mission including margin.

$$\Delta V = I_{SP,EFF} \cdot g_C \cdot \ln \frac{M_i}{M_f} \tag{1}$$

4. Tank Design and Loaded Fluid

The tank design model is used to estimate the dimensions and mass of the propellant tanks and gas bottles. Each pressure vessel must hold the required usable fluid plus any initial ullage gas, unusable or undrainable fluid, and internal PMD hardware. The tank volume also has an allocation for fluid mass gauge uncertainty. In the propellant trade study all liquid propellant tanks have an initial ullage volume of 5% the tank shell volume. Undrainable or unusable propellant is 2% by volume for all liquid tanks. Gas bottles are initially charged to 6000 psia and used down to 300 psia leaving a 5% residual. Internal PMD hardware in liquid propellant tanks are assumed to occupy 1% of the total shell volume. The specified mass gauge errors for liquid and gas tanks are 2% and 0.5%, respectively. It is noted that the usable propellant and loaded propellant differ by the gauge error plus the planned residual. By using the known fluid densities, the above allocations, and the usable propellant masses, the required tank shell volumes are readily computed.

To determine the dimensions for each tank a standard method of packaging is applied. The SM structural shell is assumed to have an inside diameter of 13 ft. Four OMS tanks are used in each design that are positioned at the rear of the SM shell. The largest diameters are used such that the four tanks fill as much of the shell diameter as possible with some margin for attachments and insulation. The diameters are also adjusted so that all fuel and oxidizer tanks have similar vertical heights. The RCS and pressurant tanks, which are much smaller than the OMS tanks, are packaged directly above the OMS tanks or in the spaces in between the OMS tanks; a six inch separation distance is included between the OMS and RCS cylindrical sections for plumbing hardware. One exception to this rule is the pressure-fed N2O4/MMH option in Case 1a which has integrated OMS and RCS tanks. In this concept most of the RCS and pressurant tanks fit in the spaces between the OMS tanks and thus a separate cylindrical section is not needed for these tanks. Similarly, the solid and hybrid OME concepts also have the RCS and pressurant tanks arranged around the OMEs in the SM shell.

Furthermore, all OMS and liquid RCS tanks use dome aspect ratios of 1.75:1 to reduce tank height for the required volume. This is important as explained in the next section due to the vehicle structural mass differences for different sized tanks. All high-pressure gas bottles have hemispherical domes no larger than 2.3 ft in diameter and tank aspect ratios, L_{TANK}/d_{TANK}, of less than 3:1. The above constraints are applied consistently to the analyzed propulsion concepts and several iterations are performed to maximize packaging efficiency, η_{PACK}. Here, η_{PACK} is defined as the volume of all propellant and pressurant tanks divided by the volume of the 13-ft cylindrical shells occupied. In the propellant trade study η_{PACK} varies from 41-62% for all cases.

Once tank dimensions are computed that meet the volume requirements and fit within the SM diameter, the mass of each pressure vessel is calculated with the structural design approach described in Ref. 11. First a tank material is selected where three different pressure vessel wall materials are employed in the present study. For storable liquid propellant tanks 6Al-4V titanium is used while cryogenic tanks are made from 2099 aluminum-lithium. Gaseous propellant and pressurant bottles are composite over wrapped pressure vessels (COPVs). The same characteristics are assumed for all types of COPVs independent of liner material.

The selected tank operating pressures vary based upon fluid type and function. OMS tanks are initially pressurized to 20 psia at lift-off and then ramped up once on-orbit to 40 psia in the pump-fed concepts and 300 psia in the Case 1a pressure-fed design. The initial pressure in liquid tanks is important because of the high hydrostatic pressures during launch which reaches over 7 g in acceleration. Maximum tank pressure is determined based on the maximum of the operating pressure in space and the pressure during ascent to orbit. RCS tanks are also initially pressurized to 20 psia but increased to 300 psia before activation in space. Gaseous propellant and pressurant bottles operate in blow-down mode, therefore, they are initially pressurized before launch to their beginning operating levels of 6000 psia.

Once the tank size, material properties, and maximum pressure are known, the membrane stress tank wall equations from Ref. 11 are used to determine the minimum wall thicknesses in each tank. Equation 2 determines the required wall thickness, t_{CYL}, for a cylindrical shell of radius, a, subjected to maximum differential pressure, P_T, material welding efficiency, e_W, and maximum working stress, S_W. If the predicted wall thickness from Eqn. 2 is less than minimum gauge for the material, then the minimum gauge thickness is used instead. A cylinder wall thickness multiplier K_{CYL} is included to increase tank wall thickness to meet buckling requirements.

$$t_{CYL} = \frac{K_{CYL} P_T a}{S_W e_W} \tag{2}$$

Equation 3 computes the wall thickness, $t_{KNUCKLE}$, at the knuckle of an ellipsoidal dome with a semi-major axis, a, semi-minor axis, b, joined to a cylindrical shell of radius, a. K_{KS} is a knuckle stress factor multiplier that is a function of the tank dome aspect ratio, a/b, and K_{KN} is a multiplier used to increase material thickness in buckling problems.

$$t_{KNUCKLE} = \frac{K_{KN} K_{KS} P_T a}{S_W e_W} \tag{3}$$

Equation 4 calculates the wall thickness, t_{CROWN}, needed at the crown or apex of an ellipsoidal tank dome. A stress multiplier, K_{CR}, is also included to directly vary the material thickness here if needed.

$$t_{CROWN} = \frac{K_{CR} P_T a^2}{2 S_W e_W b} \tag{4}$$

Before finalizing the tank wall thicknesses the buckling tests described in Ref. 11 are performed to determine if additional material is needed in the dome or cylinder to prevent buckling due to negative tank pressures or axial compressive loading. If the cylinder or ellipsoid buckling test fails for a particular tank design, the wall thicknesses predicted by Eqns. 1-3 are increased until the tank passes the test.

Some additional general assumptions for the tank sizing calculations include a yield safety factor of 1.5, burst safety factor of 2.0, buckling coefficient of 0.10, no compressive loading, and a -1 psid buckling requirement. The welding efficiency varies depending upon material used. Aluminum-lithium that uses friction-stir welding has an efficiency of 0.75 while titanium's achievable welding efficiency is 0.90. The OMS and RCS pressurization systems on the SM are always integrated into to one, however, separate bottles are used for oxidizer and fuel pressurization. If propellant management devices (PMDs) are present, the mass of each PMD is estimated by an empirical formula proportional to tank surface area. All cryogenic PMDs are assumed to be an extra 50% heavier than the results of this relationship due to integrated thermodynamic vent systems. Total mass for each propellant tank is the shell mass plus the PMD mass.

5. *Dry Mass*

The CSM dry mass is defined as the sum of the following masses: 1) base subsystems and cargo, 2) tanks and bottles, 3) propellant and pressurization plumbing, 4) OMS and RCS engines, 5) cryogenic refrigeration hardware, 6) cryogenic tank insulation, 7) additional SM structural mass for propellant tanks and 8) propulsion system structural attachments.

The base subsystem and cargo masses are assumed inputs into the analysis. The SM base mass is 7000 lbm, the CM base mass is 12000 lbm, and the CM cargo mass is 2000 lbm. Tank and bottle masses for the propellants and pressurants are computed in the previously-described tank design and loaded fluid model. The dry mass model merely reads the pressure vessel masses from this part of the code and includes them in the total mass sum. Propellant and pressurant plumbing includes small propulsion system components such as valves, lines, pressure regulators, filters, disconnects, pressurization diffusers, orifices, burst disks, and instrumentation not including wiring. Mass estimates for all plumbing items are treated with simple models based on the number of different propellants used in the OMS and RCS. For the SM plumbing mass estimate, concepts that use monopropellants in both the OMS and RCS are 400 lbm, those with bipropellants for OMS and RCS are 800 lbm, and designs with a mix of monopropellants and bipropellants are 600 lbm. Thus, the more complex the feed and pressurization subsystems are, the greater the plumbing mass.

For the 5000-lbf OMEs in the SM all liquid-based engines are assumed to have a mass of 250 lbm. The masses of solid-based designs are determined from a 0.8 propellant mass fraction assumption and 5% residual propellant. The RCEs in the SM are each assumed to have a mass of 9 lbm. The RCEs in the CM are slightly heavier at 10 lbm, since unlike the SM RCEs, they are insulated and buried in the vehicle structure with scarfed nozzles flush with the CM outer mold line.

The cryogenic refrigeration mass includes the mass of the cryocooler plus any additional vehicle mass needed for integrated radiators, vapor-cooled shields, heat exchangers, and ancillary hardware. The mass for the additional power needed to run the cryocooler is also included in this value. Conversely, TVS mass is already accounted for in the tank PMD estimate, and thus it is not included in the cryogenic refrigeration mass. One cryocooler is used per cryogenic tank and thus a fully cryogenic OMS has four cryocoolers which can provide some fault tolerance to the CSM cold storage. A simple tiered model is used to specify the cryogenic refrigeration hardware mass per tank where 75 lbm is used for LO2 and LCH4, 225 lbm is used for LH2, and 50 lbm is used for LC2H6 and LC3H8.

Cryogenic tank insulation consists of one inch of foam with a density of 2.2 lbm/ft^3 installed on each cryogenic tank with an additional 50% of that mass for insulation attachments. The foam insulation mass is proportional to the surface area of the tank and thus it varies with propellant selection. One-hundred layers of double-alumized Mylar multi-layer insulation (MLI) is placed over the foam insulation to minimize radiative heat transfer in space. An effective MLI mass density of 17 g/m^2 is used and an additional 50% of the MLI mass is assumed for a purge blanket and attachments. MLI mass is also proportional to tank surface area.

The base subsystem and structural mass assumes that the entire SM propellant and pressure tanks only occupy a 5.2 ft length in the 13-ft SM shell. Thus, for propulsion system designs that are longer than 5.2 ft additional structural mass must be added to stretch the SM primary structure and MMOD protection to accommodate the larger tanks. Here, the structural mass change per unit stage length change, dM_{STR}/dL_{STAGE}, is assumed to be 500 lbm/ft. To indicate the magnitude of this parameter, 500 lbm/ft is equivalent to a 13-ft diameter SM shell with 0.5-inch thick titanium or 0.8-inch thick aluminum walls. The value of dM_{STR}/dL_{STAGE} is important because it determines the relative benefit of specific impulse versus propellant density for the spacecraft and mission analyzed.

Finally, the dry mass computation includes a 5% allocation for attachment of propulsion system hardware to the vehicle structure. This 5% factor is applied last in the calculations and thus it includes all of the previously described dry mass items.

III. Results and Discussion

A summary of the propellant trade study results for all cases investigated is illustrated in Fig. 5. Each propellant combination and case number is shown at the left of the chart while the ratings in each evaluation criterion are on the right side. The results are color coded where green indicates *good*, yellow indicates *fair*, and red indicates *poor*. The legend at the bottom of Fig. 5 lists the evaluation rules for each category. As can be seen in the chart there is no obvious choice for the CSM since each propellant concept has various benefits and detriments, however, some trends can be identified and relative comparisons made.

A. Propellant Trends

The bipropellant combinations using N2O4, MMH, and N2H4 have low masses as do most concepts utilizing hydrocarbon fuels. The lowest wet CSM mass is achieved by Case 20 at 46411 lbm; it has an LO2/LC3H8 OMS with an N2O4/MMH RCS. However, it is mentioned that the uncertainty in wet mass predictions are most likely larger than the differences between some of the low mass winners where some cases differ by only about 20 lbm. The all-oxygen-hydrogen concept, Case 5, has the largest mass of the designs studied at 64848 lbm; this is due to the large tanks that hold the low density hydrogen and must be packaged in the SM shell.

The monopropellant hydrazine solution in Case 4 also has a high mass at 62254 lbm due to its low-performance OME and the corresponding large propellant mass required. The solid and hybrid-based solutions also performed poorly at 53800 to 54867 lbm due their 0.8 mass fraction and relatively low OME specific impulse of 310 s when compared to liquid propellant OMEs.

Concept	OMS Oxidizer	OMS Fuel	RCS Oxidizer	RCS Fuel	Mass	Development	Safety	Complexity	Reliability	Flexibility	Contamination	Commonality	Mars
Case 1	N2O4	MMH	N2O4	MMH	46742	Existing HW	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2O4/MMH	No O2 or H2	Nitrogen
Case 1a	N2O4	MMH	N2O4	MMH	47274	Existing HW	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2O4/MMH	No O2 or H2	Nitrogen
Case 2	N2O4	N2H4	N2O4	N2H4	46640	OME	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2H4	No O2 or H2	Nitrogen
Case 3	N2O4	N2H4	-	N2H4	47411	OME	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2H4	No O2 or H2	Nitrogen
Case 4	-	N2H4	-	N2H4	62254	OME	Toxic	1 Prop	Monoprop	Storable Prop	N2H4	No O2 or H2	Nitrogen
Case 5	LO2	LH2	GO2	GH2	64848	LH2 Cryo	Nontoxic	4 Props	LH2 Cryo	LH2 Cryo	O2/H2	O2/H2 OMS	H & O
Case 6	LO2	LH2	N2O4	MMH	49709	LH2 & Exist HW	Some Toxic	4 Props	LH2 Cryo	LH2 Cryo	N2O4/MMH	O2/H2 OMS	Mix
Case 7	LO2	LH2	GO2	EtOH	50834	LH2 Cryo	Nontoxic	4 Props	LH2 Cryo	LH2 Cryo	O2/Alcohol	O2/H2 OMS	H, C, & O
Case 8	LO2	LH2	-	N2O	51899	LH2 Cryo	Nontoxic	3 Props	LH2 Cryo	LH2 Cryo	N2O	O2/H2 OMS	Mix
Case 9	LO2	LCH4	GO2	GCH4	48641	Cryo & OME	Nontoxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/CH4	O2-Based OMS	H, C, & O
Case 10	LO2	LCH4	N2O4	MMH	46906	Cryo & OME	Some Toxic	4 Props	LO2 & LCH4	LO2 & LCH4	N2O4/MMH	O2-Based OMS	Mix
Case 11	LO2	LCH4	GO2	MeOH	48028	Cryo & OME	Some Toxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/Alcohol	O2-Based OMS	H, C, & O
Case 12	LO2	LCH4	GO2	EtOH	47889	Cryo & OME	Nontoxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/Alcohol	O2-Based OMS	H, C, & O
Case 13	LO2	LCH4	-	N2O	48832	Cryo & OME	Nontoxic	3 Props	LO2 & LCH4	LO2 & LCH4	N2O	O2-Based OMS	Mix
Case 14	LO2	EtOH	GO2	EtOH	49374	Cryo & OME	Nontoxic	3 Props	LO2 Only	LO2 Only	O2/Alcohol	O2-Based OMS	H, C, & O
Case 15	LO2	MeOH	GO2	MeOH	50490	Cryo & OME	Toxic	3 Props	LO2 Only	LO2 Only	O2/Alcohol	O2-Based OMS	H, C, & O
Case 16	LO2	LC2H6	GO2	GC2H6	47439	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	O2/HC	O2-Based OMS	H, C, & O
Case 17	LO2	LC2H6	N2O4	MMH	46437	Cryo & OME	Some Toxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	N2O4/MMH	O2-Based OMS	Mix
Case 18	LO2	LC2H6	GO2	EtOH	47207	Cryo & OME	Nontoxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	O2/Alcohol	O2-Based OMS	H, C, & O
Case 19	LO2	LC3H8	GO2	GC3H8	47372	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC3H8	LO2 & LC3H8	O2/HC	O2-Based OMS	H, C, & O
Case 20	LO2	LC3H8	N2O4	MMH	46411	Cryo & OME	Some Toxic	4 Props	LO2 & LC3H8	LO2 & LC3H8	N2O4/MMH	O2-Based OMS	Mix
Case 21	LO2	LC3H8	GO2	EtOH	47188	Cryo & OME	Nontoxic	4 Props	LO2 & LC2H8	LO2 & LC2H8	O2/Alcohol	O2-Based OMS	H, C, & O
Case 22	-	Solid	N2O4	MMH	53800	Existing HW	Some Toxic	3 Props	Solid	Solid	N2O4/MMH	No O2 or H2	Solid & Nitrogen
Case 23	-	Solid	GO2	EtOH	54867	Modified RCE	Nontoxic, Solid	3 Props	Solid	Solid	O2/Alcohol	No O2 or H2	Mix
Case 24	-	Hybrid	N2O4	MMH	53800	OME	Some Toxic	4 Props	Hybrid	Storable Prop	N2O4/MMH	No O2 or H2	Solid & Nitrogen
Case 25	-	Hybrid	GO2	EtOH	54867	OME	Nontoxic, Hybrid	4 Props	Hybrid	Storable Prop	O2/Alcohol	No O2 or H2	Mix

Legend:	< 48K lbm	Existing or Modified HW	Nontoxic	1 - 2 Propellants	Hypergolic, Solid, or Mono OMS	Storable Props	O2 with H2, CH4, Alcohols, & N2O	O2/H2 OMS	H,C,O Only
	48 - 53K lbm	Cryo Storage or 1 Engine Dev	Some Toxic	3 Propellants	Non-LH2 Cryos or Hybrid OMS	Non-LH2 Cryos	N2H4 Fuel or O2/Heavy HCs	O2-Based OMS	Mix
	> 53K lbm	LH2 Cryo Storage or 2 Engine Devs	Toxic	4 Propellants	LH2 Cryo OMS	LH2 or Solids	N2O4/MMH	No O2 or H2	Nitrogen or Solid

Figure 5. Results Summary

Another trend noticed is that N2O4/MMH and N2O4/N2H4 provide the lightest RCS solutions for the cases studied. For example consider Cases 19-21 which all have an oxygen-propane OMS. Case 20 with an N2O4/MMH OMS is 778 lbm lighter than Case 21 with a GO2/EtOH RCS which is 183 lbm lighter than Case 19 with a GO2/GC3H8 RCS.

A comparison of Case 1 and Case 1a illustrates the benefit of using a pump-fed OMS instead of a pressure-fed one for the vehicle and 5000 ft/s mission studied. Here, the pump-fed concept is 532 lbm lighter than the pressure-fed version due mainly to the heavier 300 psia integrated OMS/RCS propellant tanks. Thus, as mission delta velocities for CSM-like spacecraft increase above 5000 ft/s, the mass benefits of a pump-fed OMS become even greater. For this reason a pump-fed approach is more scalable to other space exploration tasks than a pressure-fed OMS.

The best development solutions occur with the existing Space Shuttle and satellite propellants N2O4 and MMH. Here, in Cases 1 and 1a, many already qualified existing engines and components are available from suppliers. Solid OME designs with either an N2O4/MMH RCS (Case 22) or a GO2/EtOH RCS (Case 23) also have a good development outlook since existing solids and oxygen-ethanol thrusters may be used or modified. Some propellant combinations are new to both OMS and RCS and thus both new OMEs and RCEs must be developed along with any needed cryogenic storage technology. In this category are Case 16 with an LO2/LC2H6 OMS and a GO2/GC2H6 RCS and Case 19 with an LO2/LC3H8 OMS and a GO2/GC3H8 RCS. Systems with the greatest development challenges have long-term liquid hydrogen storage.

For safety only the hydrazine-based options in Cases 1-4 and the oxygen-methanol design in Case 15 are rated poorly due to protection needed for handling. It is interesting to note that many nontoxic options are competitive solutions with the toxic hypergols for the CSM design. In complexity the concepts with an OMS and RCS using dissimilar bipropellants are the most complex due to the number of different propellants and in turn plumbing components. For example Case 9 has LO2, LCH4, GO2, and GCH4 propellants which all require qualified components specific to the propellant type, pressure, temperature, operations, and maintenance. The least complex propellant system is Case 4 which uses monopropellant N2H4 for both OMS and RCS; unfortunately this simple approach is much heavier than other more complex options. Reliability is shown to be good for the hypergolic and monopropellant concepts in Cases 1-4 and for the solid-based designs in Cases 22-23. Systems with LH2 storage score the worst in reliability due to the larger, more complex, and technically challenging refrigeration subsystem required.

Since the reliability and flexibility ratings are both related to the functionality of the cryogenic cooling units, both parameters have similar trends for the cases studied with some exceptions. Solid propellants are deemed poor in flexibility because the OMS impulse per firing cannot be varied efficiently after the motor is manufactured. Although propellant barriers can be installed to provide discreet burns and impulse segregation, the size of the impulse must be known a priori. Hybrid motors on the other hand can be started and stopped multiple times with no preplanning required, therefore, they are rated good in flexibility similar to storable liquid propellant options. For the hybrids in this study, it is assumed that the nonsolid propellant is storable and nontoxic such as N2O.

The contamination rating considers only the selected RCS propellants and not OMS. It is noticed that many of the concepts are rated poorly in contamination due to their N2O4/MMH RCSs. The commonality results are straightforward and relate only to the OMS propellant choice. Cases 5-8 with LO2/LH2 OMS are best in commonality while those with no oxygen or hydrogen (Cases 1-4 and 22-25) worst. The largest number of propellant combinations are rated as fair in commonality due to their use of oxygen. Note that LO2 or GO2 propellant supplies could be integrated with life support breathing oxygen storage to provide subsystem synergy and a reduced mass. A similar integration and mass savings could occur if oxygen-based fuel-cells are used for power on the CSM.

Mars propellant in-situ producibility is straightforward where if all the propellants are made from elements available on Mars, the evaluation is good. Several of the propellant combinations considered are potentially manufacturable on Mars. Several others, which are rated as fair, have Mars-producible OMS propellants but N2O4/MMH RCSs. In these scenarios the RCS propellants needed for the return trip to Earth from Mars must be carried from Earth. Depending on the amount of RCS ΔV required by the mission the mass penalties vary. For the CSM analyzed here, which does not travel to Mars, approximately 600 ft/s of the total 5000 ft/s are allocated to the RCS. For example Case 20 with an LO2/C3H8 OMS and an N2O4/MMH RCS carries over 3000 lbm of RCS propellant just for the lunar mission. To carry this quantity extra quantity of propellant for the return trip would incur a large mass penalty on any chemical propulsion spacecraft using these propellants and traveling to Mars. With this reasoning and by assuming that Mars is the target destination for the space exploration initiative, an in-situ propellant production capability is an important criterion.

B. Relative Comparisons

To recommend a propellant combination for the human-rated CSM the relative merits of each candidate are evaluated in several criteria. If development ease is the most important metric for propellant selection, then the cases illustrated in Fig. 6 are best. As shown the familiar Space Shuttle and satellite propellants N2O4 and MMH in Case 1 (pump-fed) and Case 1a (pressure-fed) have a good development outlook because of the existing propulsion hardware available and the use of storable instead of cryogenic fluids. A solid OMS with an N2O4/MMH RCS, Case 22, requires little development; existing SRM designs can be modified and the RCS system built with in-production, qualified, heritage hardware. Similarly, Case 23 which uses a solid OMS with a GO2/EtOH RCS also has a low development risk. It is noted that Case 23 is the only concept that has good ratings in both development and safety. In contrast the hypergolic N2O4/MMH cases are light, simple, reliable, and flexible but poisonous and contaminating.

Concept	OMS Ox	OMS Fuel	RCS Ox	RCS Fuel	Mass	Dev	Safety	Complex	Reliability	Flex	Contam	Common	Mars
Case 1	N2O4	MMH	N2O4	MMH	46742	Existing HW	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2O4/MMH	No O2 or H2	Nitrogen
Case 1a	N2O4	MMH	N2O4	MMH	47274	Existing HW	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2O4/MMH	No O2 or H2	Nitrogen
Case 22	-	Solid	N2O4	MMH	53800	Existing HW	Some Toxic	3 Props	Solid	Solid	N2O4/MMH	No O2 or H2	Solid & Nitrogen
Case 23	-	Solid	GO2	EtOH	54867	Modified RCE	Nontoxic, Solid	3 Props	Solid	Solid	O2/Alcohol	No O2 or H2	Mix

Figure 6. Propellant Combinations with the Easiest Development

If, however, the wet mass of the vehicle is the most significant evaluation criterion, then the cases listed in Fig. 7 are best. Here, the five cases with the lowest mass are shown which include the pump-fed hypergolic solutions in Cases 1 and 2 and the cryogenic hydrocarbon OMS cases in Cases 10, 17, and 20. All of these low mass concepts have toxic propellants for reaction control. Case 1 looks attractive from development, complexity, reliability, and flexibility view point, but not when considering safety, contamination, commonality, and Mars producibility. The liquid-propane-based OMS is calculated as the lightest followed by ethane, then methane. Thus, in this study the higher density of propane proved to be more beneficial to a low vehicle mass than the higher engine specific impulses of methane or ethane systems.

Concept	OMS Ox	OMS Fuel	RCS Ox	RCS Fuel	Mass	Dev	Safety	Complex	Reliability	Flex	Contam	Common	Mars
Case 1	N2O4	MMH	N2O4	MMH	46742	Existing HW	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2O4/MMH	No O2 or H2	Nitrogen
Case 2	N2O4	N2H4	N2O4	N2H4	46640	OME	Toxic, Hypergol	2 Props	Hypergolic	Storable Prop	N2H4	No O2 or H2	Nitrogen
Case 10	LO2	LCH4	N2O4	MMH	46906	Cryo & OME	Some Toxic	4 Props	LO2 & LCH4	LO2 & LCH4	N2O4/MMH	O2-Based OMS	Mix
Case 17	LO2	LC2H6	N2O4	MMH	46437	Cryo & OME	Some Toxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	N2O4/MMH	O2-Based OMS	Mix
Case 20	LO2	LC3H8	N2O4	MMH	46411	Cryo & OME	Some Toxic	4 Props	LO2 & LC3H8	LO2 & LC3H8	N2O4/MMH	O2-Based OMS	Mix

Figure 7. Propellant Combinations with the Lowest Mass

Another approach is to require that only nontoxic propellants are used for the CSM which leads to safety and recurring cost benefits. Since the chosen propellants will be used for many years on many space flights, the issues of accidents and operations costs become important. Nontoxic propulsion systems have been estimated to significantly reduce operations costs below those for toxic, hypergolic propellants.[2,3] With a nontoxic requirement 13 propulsion concepts are eliminated including the heritage N2O4/MMH propellants. Thirteen propulsion concepts remain, but some are superior to others in important evaluation criteria.

Missions to Mars and the establishment of a human presence will be both technically and financially challenging when considering the amount of mass that must be lifted to orbit from Earth. Carrying return propellants to far away destinations such as Mars will have increasing mass and cost penalties as the destinations become more distant. Accordingly, in-situ propellant production is an enabling technology that will allow affordable access to the solar system with increased exploration capability per spacecraft. Based on these arguments Mars in-situ producibility is enforced as a requirement for the CSM propellants. This requirement leaves nine remaining solutions but eliminates the solid and hybrid OME designs plus RCS concepts that use an N2O RCS. At this point the last LO2/LH2 OMS concepts are eliminated due to their considerably higher masses, worse development, lower reliability, and less flexibility. Seven propellant combination concepts now remain as shown in Fig. 8 which are all nontoxic and producible on Mars.

Concept	OMS Ox	OMS Fuel	RCS Ox	RCS Fuel	Mass	Dev	Safety	Complex	Reliability	Flex	Contam	Common	Mars
Case 9	LO2	LCH4	GO2	GCH4	48641	Cryo & OME	Nontoxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/CH4	O2-Based OMS	H, C, & O
Case 12	LO2	LCH4	GO2	EtOH	47889	Cryo & OME	Nontoxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/Alcohol	O2-Based OMS	H, C, & O
Case 14	LO2	EtOH	GO2	EtOH	49374	Cryo & OME	Nontoxic	3 Props	LO2 Only	LO2 Only	O2/Alcohol	O2-Based OMS	H, C, & O
Case 16	LO2	LC2H6	GO2	GC2H6	47439	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	O2/HC	O2-Based OMS	H, C, & O
Case 18	LO2	LC2H6	GO2	EtOH	47207	Cryo & OME	Nontoxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	O2/Alcohol	O2-Based OMS	H, C, & O
Case 19	LO2	LC3H8	GO2	GC3H8	47372	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC3H8	LO2 & LC3H8	O2/HC	O2-Based OMS	H, C, & O
Case 21	LO2	LC3H8	GO2	EtOH	47188	Cryo & OME	Nontoxic	4 Props	LO2 & LC2H8	LO2 & LC2H8	O2/Alcohol	O2-Based OMS	H, C, & O

Figure 8. Propellant Combinations with Best Safety and Mars In-Situ Producibility

All of the propellants in Fig. 8 are made of carbon, oxygen, and hydrogen and could theoretically be manufactured from the carbon dioxide and water on Mars. Several studies and experiments have already been completed to investigate propellant production methods on Mars.[12,13] Here, Zirconia CO2 Electrolysis (ZCE), Sabatier/Water Electrolysis (SWE), and Reverse Water Gas Shift (RWGS) are leading process candidates.[14] However, much of the work relates to oxygen, hydrogen, methane, methanol, or water production and not ethanol.

On Earth ethanol is mainly made organically, but due to the inefficiency of this method producing ethanol on Mars using food waste for example is probably not practical. Synthetic EtOH on Earth can be made from ethylene and water vapor, where the ethylene is acquired either organically or by cracking longer hydrocarbon chain molecules. To make EtOH the same way on Mars would require new methods that fabricate C2H4 from CO2 and water. Accordingly, due to the immaturity of in-situ ethanol production for Mars, the ethanol-based RCS concepts are eliminated leaving only hydrocarbon fuels for the CSM RCS.

The remaining three propellant combinations that all use hydrocarbon fuels for both OMS and RCS are illustrated in Fig. 9; they are Cases 9, 16, and 19. Notice that more development is required for the ethane and propane solutions than for methane due to the new RCEs needed and the greater likelihood of coking problems in the engine. The packaging benefits of the denser ethane and propane systems lead to about a 1300 lbm mass savings compared to a methane-fueled system. However, GC2H6 and GC3H8 RCS engines are more contaminating than GCH4 which burns cleaner. Similar to the ethanol argument above, fabrication of the heavier hydrocarbons ethane and propane is more complex and has less mature in-situ production methods.

Concept	OMS Ox	OMS Fuel	RCS Ox	RCS Fuel	Mass	Dev	Safety	Complex	Reliability	Flex	Contam	Common	Mars
Case 9	LO2	LCH4	GO2	GCH4	48641	Cryo & OME	Nontoxic	4 Props	LO2 & LCH4	LO2 & LCH4	O2/CH4	O2-Based OMS	H, C, & O
Case 16	LO2	LC2H6	GO2	GC2H6	47439	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC2H6	LO2 & LC2H6	O2/HC	O2-Based OMS	H, C, & O
Case 19	LO2	LC3H8	GO2	GC3H8	47372	Cryo, OME, RCE	Nontoxic	4 Props	LO2 & LC3H8	LO2 & LC3H8	O2/HC	O2-Based OMS	H, C, & O

Figure 9. Final Three Propellant Combinations for the Crew Space Vehicle

It is judged that the production, development, and contamination advantages of CH4 in Case 9 are more beneficial to a sustained, long-term Mars exploration program than a 1300 lbm CSM mass savings. Therefore, due to the aforementioned reasons, the ethane and propane-based solutions are dropped from consideration in the study. This leaves Case 9 with the LO2/LCH4 OMS and GO2/GCH4 RCS as the recommended propellant combination for the crew space vehicle. If, however, an efficient in-situ production method is developed for ethanol, ethane, or propane, then the propellant selection should be revisited.

C. Recommended Work

For the selected LO2/LCH4 OMS and GO2/GCH4 RCS, six recommendations are made for hardware development in preparation for use for the space exploration initiative.

First, an expendable and reliable LO2/LCH4 pump-fed engine for orbital maneuvering systems should be developed. The engine should be based on existing in-production LO2/LH2 engines to minimize the development effort required. A suggested thrust level is 5000 lbf per engine which provides thrust for both the burn from LLO to Earth and for some high-altitude flight abort capability. Mixture ratio should be at least 3.0 with a target near 3.4. With a 100:1 nozzle the new engine should yield a vacuum specific impulse at steady-state flow of near 363 seconds.

Second, modify existing GO2/GCH4 engines into dual-mode pressure-fed engines for the SM and CM reaction control systems. Here, dual-mode means that the engine can create two different thrust levels where the higher one uses the full flow rate of the chamber while the lower thrust level is created by only the chamber igniter. A suggested full-flow thrust of 100 lbf is suggested with a 10 lbf igniter thrust; with this approach one thruster can serve as both a primary and vernier RCE on a CSM-like vehicle. Also, similar thruster designs can be used for both the RM and CM RCEs with some part commonality. An expected specific impulse in pulse-mode operation for the SM GO2/GCH4 RCE should be near 289 s for a 25:1 nozzle. The CM engine, however, must be buried in the CM structure and thus designed and insulated to minimize heat transfer to the vehicle. Both the engine work and the CM installation issue should be addressed.

Third, a cryogenic storage capability for large quantities of LO2 and LCH4 should be developed both for vehicles and remote storage depots. Here, it may be possible to use common hardware for both the oxidizer and fuel since the boiling temperatures are only about 40 °F apart. Integration of the cryogenic hardware with other vehicle subsystems should be studied to reduce vehicle mass. The vehicle storage systems should be designed for at least a one-year of mission life while the depots should last longer.

Forth, the Earth ground infrastructure required for the LCH4 and GCH4 propellants should be developed to allow filling and draining of propellant tanks, propellant analysis, and propulsion system testing. Since all fluids are nontoxic, these new facility capabilities should cost less than those developed to process toxic hypergols.

Fifth, work should continue towards in-situ methane and oxygen production on Mars. An investigation should also be conducted to evaluate the producibility of heavier hydrocarbons and ethanol on Mars due to their potential for vehicle mass reductions and less complex long-term storage designs. An in-situ propellant production robotic mission to Mars should be considered to demonstrate autonomous placement of in-situ resources plus on-Mars production and storage of LCH4 and LO2.

The sixth recommendation is to investigate the benefits of LCH4-GCH4 and LO2-GO2 liquid-gas conversion to determine the impacts to mass and reliability of space exploration vehicles. Use of liquid-gas conversion can replace gaseous propellant bottles with a lower mass and volume system using RCS accumulators.

IV. Conclusion

The defined multi-use crew space vehicle provides a flexible capability for space exploration. It provides access to low Earth orbit, the ISS, and L1, but also can perform one-way transportation between the Earth and moon. For a vehicle of this size and 5000 ft/s propulsive capability, pump-fed main engines with low-pressure tanks lead to a lighter design than a pressure-fed system similar to the Space Shuttle's. A pump-fed, crew space vehicle design approach can be scaled to larger sizes that have greater delta velocity but without the large mass penalties that would result from a pressure-fed system with heavy propellant tanks. For the fleet of spacecraft needed in the Earth-moon-Mars exploration initiative, affordability will define how much can be done and how soon. The amount of mass launched from Earth and in turn the launch costs can be decreased if return propellant is produced at the target destination. Thus, by requiring Mars producibility in the propellants used for exploration this cost savings can be realized. Safety and operational costs reductions per mission can both be achieved by using nontoxic propellants. For new exploration space vehicles any change in propellants away from the heritage, toxic N2O4 and MMH propellants will incur development costs and risk, and thus there is a certain attractiveness to using these existing toxic propellants, their qualified hardware, and their well-established ground infrastructure. However, over a multi-decade space exploration program a nontoxic propulsion approach can save considerable funds that can be focused elsewhere and provide a safer environment for the human astronauts and ground support team. Accordingly, it is on the basis of these nontoxic, Mars-extensible arguments, the mass estimates, and the other evaluation criteria that a liquid-oxygen/liquid methane OMS is recommended for the crew space vehicle with a gaseous-oxygen/gaseous-methane RCS.

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