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Lunar Missions Using Advanced Chemical Propulsion: System Design Issues

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To provide the transportation of lunar base elements to the Moon, large high-energy propulsion systems will be required. Advanced propulsion systems for lunar missions can provide significant launch mass reductions and payload increases. These mass reductions and added payload masses can be translated into significant launch cost savings for the lunar base missions. In this paper, the masses in low Earth orbit (LEO) were compared for several propulsion systems: nitrogen tetroxide/monomethyl hydrazine (NTO/MMH), oxygen/methane (O_2/CH_4), oxygen/hydrogen (O_2/H_2), and metallized $O_2/H_2/Al$ propellants. Also, the payload mass increases enabled with O_2/H_2 and $O_2/H_2/Al$ systems were addressed. In addition, many system design issues involving the engine thrust levels, engine commonality between the transfer vehicle and the excursion vehicle, and the number of launches to place the lunar mission vehicles into LEO will be discussed. Analyses of small lunar missions launched from a single STS-C flight are also presented.

Nomenclature

A, B, C	=	propulsion dry mass parameters
D	=	aerobrake mass fraction
E	=	mass parameter for leg structure
g	=	gravitational acceleration, 9.81 m/s^2
I_{sp}	=	specific impulse, s
ML	=	metal loading (fraction of oxidizer or fuel mass)
m_{entry}	=	total entry mass during aerobraking maneuver, kg
m_f	=	final mass, kg
m_{landed}	=	total landed mass on the surface, kg
m_o	=	initial mass, kg
T/W	=	thrust to weight
ΔV	=	velocity change, m/s
ϵ	=	expansion ratio
η	=	I_{sp} efficiency
ρ_m	=	density of metal in the oxidizer or fuel, kg/m^3
ρ_p	=	density of nonmetallized oxidizer or fuel, kg/m^3
$\rho_{p,m}$	=	density of metallized oxidizer or fuel, kg/m^3

Introduction

THE U.S. Space Program has considered a vigorous new initiative to place a permanent base or settlement on the lunar surface.[1] From this base, a wide range of technology experiments and science investigations will be conducted. Also, the base may support the first human missions to Mars. There may be a significant infrastructure for propellant production on the lunar surface. These propellants could be used for Mars flights launched from the vicinity of the Moon: either from lunar orbit or from a libration point.

The payloads being considered for the lunar missions are considerably larger than those of Apollo. Hence, the low Earth orbit (LEO) masses are very large. Applying advanced technologies, such as high specific impulse I_{sp}, chemical propulsion to these missions can enable large LEO mass reductions or significant payload increases. Several propulsion options for reducing the LEO mass will be analyzed and contrasted. This selection of the "best" technologies for the lunar mission can provide significant cost or schedule savings over the life of the lunar exploration program.

Placing the large elements needed for a base onto the lunar surface will require large spacecraft and large propellant loads. As will be discussed later in the paper, the LEO masses of the lunar vehicles range from 4 to 17 Space Transportation System-Cargo (STS-C) launches. The advanced oxygen/hydrogen (O_2/H_2) and metallized oxygen/hydrogen/aluminum ($O_2/H_2/Al$) systems require the lowest mass delivered to orbit (4 launches) and potentially provide the lowest cost for the overall transportation system. Advanced propulsion will lead to fewer launches for the missions and consequently a faster assembly rate and a reduced mission launch cost.

A number of propulsion technologies are available or are in development that could provide lunar transportation. Whereas Apollo used NTO/Aerozine-50 propellant for the service module service propulsion system and both lunar module ascent and descent propulsion systems, and the Saturn V O_2/H_2 third stage for the translunar injection, new lunar mission planners now have several options available to them.

The chemical propulsion systems that were considered range from Earth-storable nitrogen tetroxide/monomethyl hydrazine (NTO/MMH), space-storable propellants (O_2/CH_4, etc.), cryogenic O_2/H_2, and metallized propellants. Metallized propellants have a high density and/or a high I_{sp}.[2] A metal additive increases the propellant density and potentially the I_{sp} of the propellant combination. With these propellants, the metal (such as aluminum) is suspended in a gelled liquid fuel. The advantages and disadvantages of using metallized propellants have been documented elsewhere[19–22] and will be discussed later in this paper.

Selecting the "best" propulsion option is based on many factors, including the performance, size, reliability, and life of the propulsion system; the system cost; the number of systems in the transportation architecture; and the availability of the new technologies. Other technologies that can have significant effects on the lunar transportation system are aerobraking, cryogenic storage, and efficient structures. The effects of these technologies must be properly integrated into the overall design process.

To determine the effects and potential benefits of advanced propulsion systems, a series of systems analyses and trade

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studies including the lunar mission analyses and the propulsion system design are required. The mission, propulsion and systems analyses, and relevant design issues will be discussed in the following sections.

Lunar Missions

In each lunar mission scenario, several different types of missions are required. These missions range from piloted missions, with and without cargo delivery, unmanned missions carrying cargo, and checkout missions to validate system performance prior to committing expensive payloads to lunar flight. For these missions, both reusable and expendable vehicles have been considered. In these analyses, the payload to the lunar surface for each of these missions varies from 15,000 to 27,000 kg.[3]

The mission scenarios[3] include a lunar transfer vehicle (LTV) which transfers its payloads between LEO and low lunar orbit (LLO). Its payload is a lunar excursion vehicle (LEV) and its surface payload. The excursion vehicle may be used in a fully automated or a piloted mode. In this investigation, the cargo delivery missions with 27,000-kg payload to the lunar surface were analyzed. This mission is used to establish the size of the transfer and excursion vehicles. It also provides a relative comparison of each technologies' LEO masses and payload capabilities. The 27,000-kg payload mass is representative of the largest lunar base elements: a pressurized habitat module with an attached airlock.[3,4]

Mission Analysis: Lunar Mission ΔV

To estimate the vehicle masses, the maneuvers are described by a series of velocity changes (ΔV). The ΔV is computed using

$$\Delta V = I_{sp}\, g \ln\, (m_o/m_f)$$

The maneuver ΔV values are listed in Table 1 and were taken from Ref. 1. The lunar missions are based in LEO.

There are nine maneuvers required for the transfer vehicle. Four engine firings are delivered by the excursion vehicle. To depart Earth orbit, a 3300 m/s ΔV is required. A small maneuver is conducted during the translunar coast and the lunar orbit insertion (LOI) maneuver places the entire transfer vehicle and excursion vehicle into lunar orbit. The excursion vehicle descends to the surface, the payload is offloaded, and the vehicle ascends to orbit. The excursion vehicle remains in LLO to be refueled and refitted with a payload for the next landing.

To return to Earth, the transfer vehicle delivers the trans-Earth injection (TEI)/ΔV and a small midcourse correction ΔV during the trans-Earth coast. Aerobraking is typically used for the return to Earth orbit. Table 1 shows the ΔV for aerobraking into LEO. If an all-propulsive Earth orbit insertion (EOI) were conducted, the ΔV would be equal to that for the translunar injection (TLI). The influence of aerobraking on the LEO initial mass will be discussed later in the paper.

The major mission maneuvers (TLI, LOI, TEI, and EOI ΔV) from Ref. 1 are several hundred meters per second larger than those used in previous studies.[5,6] A larger ΔV will require a larger propellant load and propulsion system. The larger propulsion system would allow a wider range of lunar departure opportunities, provide a longer launch window, and give more flexibility to accommodate launch delays.

Table 1 Lunar mission maneuvers[1]

Maneuver	Change in velocity ΔV, m/s
Lunar transfer vehicle	
Preinjection preparation	10
Translunar injection (TLI)	3300
Translunar coast	10
Lunar orbit insertion (LOI)	1100
Lunar orbit operations	50
Trans-Earth injection (TEI)	1100
Trans-Earth coast	10
Earth orbit insertion (EOI)	40
Earth orbit operations	275
Lunar excursion vehicle	
Pre-deorbit preparation	5
Deorbit to landing	2000
Ascent to orbit	1900
Postascent orbital operations	5

Number of Stages

A series of transfer vehicles have been considered with differing numbers of stages. The current design being contemplated is known as the "stage and one-half." Here the propellant loads for the translunar injection ΔV (and, in some options, the lunar orbit insertion ΔV) are contained in separate drop tanks. These tanks are expended after completing their respective maneuvers. This reduces the mass that must be returned to Earth orbit. This, in turn, reduces the size and mass of the vehicles aerobrake. A central vehicle "core" holds the propellant for the trans-Earth injection and the Earth orbit insertion maneuvers. No engines are expended with this staging method and therefore the high-value engine module can be reused.

In the analyses presented here, only one set of drop tanks were considered. The tank sets considered here were of a lower propellant mass fraction than those considered in Refs. 3 and 7. The structural mass of the systems considered in the references was lower than that used in these analyses. Because of the lower mass fraction of the tank sets, there is no mass advantage to using more than one set of drop tanks (separate ones for TLI and LOI). In this work, drop tanks were only used to hold the translunar injection propellant.

Propulsion System Design

Engine Performance

Using a computer simulation code (Complex Equilibrium Compositions code, CEC),[8] the engine performance of the four propellant combinations was estimated. An engine I_{sp} efficiency was used to modify the code-predicted ideal I_{sp}. The I_{sp} efficiency η is the ratio of the delivered engine performance and the code-predicted I_{sp}. This reduction reflected the losses that may be incurred due to the nozzle boundary layer, engine cycle inefficiencies, and other propulsion system losses. The engine efficiencies were derived using the performance estimates from Refs. 9–12 and comparisons with the vacuum I_{sp} predicted by the engine code.

Table 2 provides the design I_{sp} values that were selected for the lunar mission cases. The engine mixture ratios are shown in Table 3. The engine chamber pressures were varied from 3.206 to 10.342 MPa. This variation was dependent upon the designs of the various engines under consideration for the Lunar-Mars initiative. The propellants were provided to the combustion chamber in the liquid state. An expansion ratio of 400:1 to 1000:1 was selected for the transfer vehicle engine, again based on the designs of planned engines. For the excursion

Table 2 Propulsion system performance

	Specific impulse I_{sp}, s		
Propellant	Lunar transfer vehicle	Lunar excursion vehicle	I_{sp} efficiency, η
NTO/MMH	340.0	330.0	0.940
O_2CH_4[a]	360.8	350.8	0.940
O_2CH_4[b]	390.0	380.0	0.940
O_2H_2[c]	446.4	436.4	0.962
O_2H_2[d]	485.0	475.0	0.984
$O_2/H_2/Al$	491.4	481.4	0.984

[a]Oxidizer-to-fuel ratio (O/F) = 3.4 for maximum I_{sp}.
[b]O/F = 3.9 for maximum I_{sp}. [c]O/F = 5.0. [d]O/F = 6.0.

Table 3 Propulsion system design parameters

Propellant	Mixture ratio	Expansion ratio, ϵ
NTO/MMH	2.0	400:1
O_2/CH_4	3.4	465:1
O_2/CH_4	3.9	1000:1
O_2/H_2	5.0	465:1
O_2/H_2	6.0	1000:1
$O_2/H_2/Al$	[a]1.6	1000:1

[a]60-% aluminum loading in H_2.

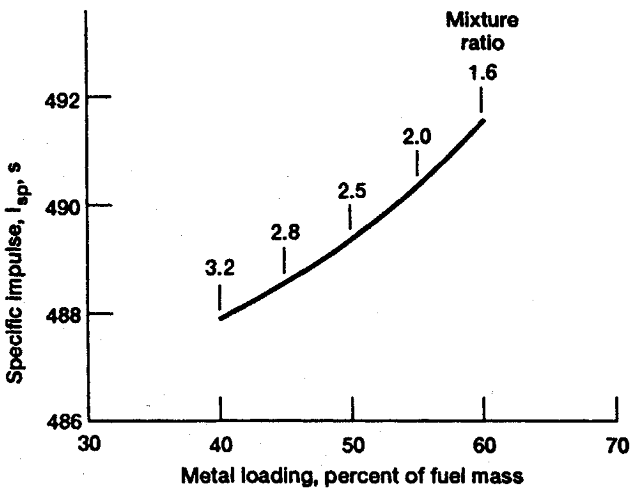

Fig. 1 Specific impulse vs metal loading (expansion ratio 1000:1).

vehicle, due to packaging constraints that may limit the size of the large expansion ratio nozzles, the expansion ratio was reduced, causing the I_{sp} to be reduced by 10 s. As an example, the I_{sp} values for the space transfer engine (STE) were 485 s (for the LTV) and 475 s (for the LEV), respectively.

Metallized Propellants

In selecting the "best" metallized system design, the propellant metal loading, its effects on the engine I_{sp}, and the propulsion system dry mass must be analyzed. The issues that are important in determining the appropriate design for a metallized propulsion system are the propellant density, the performance, the system dry mass, and the technology readiness.

Propellant Density

Using the aluminum loadings considered in the engine performance calculations, the propellant density for the H_2 fuel can increase from 70 kg/m^3 to 169 kg/m^3 (H_2 with a 60-% aluminum loading). The density increase is computed using[2]

$$\rho_{p,m} = 1/([1 - ML]/\rho_p] + ML/\rho_m)$$

Selection of the Best Density-I_{sp} Design Points

To deliver the maximal reduction in LEO mass or the maximal payload increase, trade studies must be conducted to determine the "best" combination of I_{sp} and density for each propulsion system. Figure 1 shows the results of one of these trade studies on I_{sp} for $O_2/H_2/Al$. The maximal metal loading considered was 60% (weight percent) of the fuel mass. The selection of the 60-% loading performance level was based on analyses of the metallized I_{sp} increases and the mass of the associated propellant tankage. A higher I_{sp} is produced at higher metal loadings. The mixture ratio was selected to deliver the highest I_{sp} for that metal loading. The total metal loading of all of the propellant (oxidizer and fuel) of the propulsion system was 23%, which is comparable to that of solid propulsion systems.[13] An I_{sp} of 491.4 s was delivered at a metal loading of 60% of aluminum in the hydrogen/aluminum (H_2/Al), a nozzle ϵ of 1000:1, and an engine mixture ratio of 1.55.

This I_{sp} design point, however, may require a heavier propulsion system than the nonmetallized design case. This is because the hydrogen tank is larger in the metallized case. Although the H_2/Al propellant is denser than H_2, the lower mixture ratio of the $O_2/H_2/Al$ system requires a larger fuel tank. Reference 14 compares the propulsion mass-scaling equations for several metal loadings. There is a small variation in the total mass of the propulsion system with the differing metal loadings. Based on the trade studies, the highest I_{sp} system of the range in Fig. 1 was selected, having a metal loading of 60%, which was the maximum considered in this analysis.

LTV Mass-Scaling Equations

In determining the dry mass of the transfer vehicles, the following general mass-scaling equation was used:

$$m_{dry} = A + Bm_p + Cm_p^{2/3} + Dm_{entry}$$

Table 4 provides the propulsion mass-scaling parameters for all of the systems considered. These parameters include all of the masses that are required to store and provide propellants to the main engines. They include tankage, engines, feed system, thermal control, structure, residuals, and contingency. The parameter A of the scaling equations varied over the range of 109–1364 for the lunar vehicles. The variation is due to the differing configurations and number of engines for each stage. For example, the 109 value of the parameter A is used for the feed system of a tank set that has no engine components. Only the latter value of A is shown in the table. The specific mixture ratios and the metal loadings are listed in Table 3.

Propellant Tanks

The propellant tankage for all of the systems uses a 0.345-MPa maximal operating pressure. The propellant is stored at 0.207 MPa psia. All of the tankage for O_2, H_2, and CH_4 is composed of aluminum alloy. The tanks for NTO and MMH are made of titanium. The flange factor and safety factor for the propellant tanks are 1.4 and 2.0, respectively. The safety factor is based on the tank material ultimate stress. The propellant residuals and holdup mass is 2.7% of the total propellant mass. The percentage accommodates the added propellant mass for cryogenic propellant boiloff.

Pressurization

Each cryogenic propulsion system uses autogenous pressurization. Only the NTO/MMH and the space-storable systems use regulated pressurization. The pressurant is helium. In the pressurant tank, the maximal operating pressure is 25.662 MPa. The storage pressure is 23.746 MPa. The flange factor and safety factor for the pressurant tanks are 1.1 and 2.0, respectively. For the autogenous systems, a small helium pressurization system is provided. This system provides a small amount of pressurant for the initial pressurization before the engine is ignited. It can pressurize one-tenth of the total propellant tank volume.

Thermal Control

For thermal control, the cryogenic propellants (O_2, H_2, and CH_4) use a high-performance multilayer insulation and a thin-

Table 4 Lunar vehicle mass-scaling parameters

Propellant	Parameter A	B	C
NTO/MMH	1348.55	0.1497	0.0000
O_2/CH_4[a]	1363.51	0.1676	0.0516
O_2/CH_4[b]	1363.51	0.1669	0.0463
O_2/H_2[c]	1363.51	0.1853	0.0858
O_2/H_2[d]	1363.51	0.1811	0.0806
$O_2/H_2/Al$	1363.51	0.1817	0.0798

[a]Oxidizer-to-fuel ratio (O/F) = 3.4. [b]O/F = 3.9. [c]O/F = 5.0. [d]O/F = 6.0.

walled vacuum jacket. The jacket is sized for a 0.207-MPa maximal operating pressure. After the vehicle reaches space, it is vented and evacuated. The storable propellants only require a lower-performance multilayer insulation.

Aerobrake

The aerobrake mass is 17.25% of the vehicle mass entering the atmosphere.[3,7,18] The 17.25% mass factor represents 15% with a 15% contingency. This mass includes the payload, propulsion system dry mass, any propellant needed for the entry and post entry maneuvers, and the aerobrake.

LEV Design and Sizing

The mass-scaling equation for the excursion vehicle stage is

$$m_{dry} = A + Bm_p + Cm_p^{2/3} + Em_{landed}$$

The excursion vehicle is sized to deliver the ΔV values listed in Table 1. In the baseline unmanned cargo mission scenarios, the payloads delivered to the surface have a total mass of 27,000 kg per flight and the vehicle returns to LLO empty. The LEV sizing parameters are similar to those for the LTV.

Leg Structure

An important aspect of the excursion vehicle is its leg structure to support it on the Moon. It is part of the descent stage and the leg mass is 2% of the total mass that is landed on the surface.

Technology Readiness

With metallized propellants, there are several factors that will influence the selection process: two-phase flow losses and their effect on I_{sp}, particulate erosion of the throat and nozzle, turbomachinery compatibility with thixotropic-gelled propellants, and the cryogenic boiloff difference between H_2 and gelled H_2/Al. These issues have been addressed in other works.[2,15,19,22] The readiness of $O_2/H_2/Al$ propellants for space missions is not on the same level as that with O_2/H_2. Metallized storable propellants have been tested for over 30 years[20] and their readiness level is much higher than cryogenic-gelled fuels. Many of the rocket combustion and nozzle issues that would face $O_2/H_2/Al$ have already been addressed and solved in the gelled storable propellant work. Issues with cryogenic storage that are specific to H_2/Al are also very encouraging.[22] The boiloff rate for gelled cryogens can be two to three times lower than for ungelled cryogens.[22] Although there are development issues with metallized $O_2/H_2/Al$, it is likely that none are insurmountable.

Systems Analysis Results

In this section, the relative performance of the various chemical propulsion technologies will be discussed, emphasizing the potential advantages of these technologies, in terms of payload advantage and reduced mass in LEO. Other system-level design considerations, such as propellant selection, thrust levels, and engine firing times will be evaluated. The potential of using small transfer and excursion vehicles for lunar exploration to reduce the number of Earth-to-orbit flights will be presented.

LEO Mass

The primary figures of merit used in these analyses are LEO initial mass, payload delivered to the surface, and number of STS-C launches. These figures of merit are the major comparative measures for understanding the specific and relative masses of the vehicles for lunar exploration. Many of the trade studies presented in the next section used the 27,000-kg payload delivery mission to the lunar surface (described previously) as the comparative basis. Other analyses estimate the payload delivery capability using a constant mass in LEO.

Cryogenic and Storable Propellants

In Fig. 2, the LEO masses are contrasted for six systems: NTO/MMH (340 s), two O_2/CH_4 systems (360.8 and 390 s), two O_2/H_2 systems (446.4 and 485 s), and $O_2/H_2/Al$ (491.4 s). Clearly, the propulsion options that provide the lowest LEO mass are metallized $O_2/H_2/Al$ and the STE-technology O_2/H_2 system. Each STE O_2/H_2 vehicle (485-s I_{sp}) requires only 248,500 kg for the mission, providing a 20-% LEO mass reduction over the current technology O_2/H_2 system (446.4 s). A 23-% mass reduction over the 446.4-s O_2/H_2 system is enabled with metallized propellants.

The LEO mass performance of O_2/CH_4 propulsion is superior to NTO/MMH but poor when compared with any of the O_2/H_2 systems. Over 549,000 kg are required for the 360.8-s I_{sp} system and 420,000 kg for the higher-I_{sp} O_2/CH_4 vehicle. Using storable NTO/MMH, the mass in LEO is considerably higher than that for any other case: 613,000 kg. For the large payloads being considered for the lunar base scenario, neither the O_2/CH_4 nor NTO/MMH options appear attractive for a lunar mission.

Aerobraking vs All-Propulsive

Both an aerobraking and an all-propulsive mission option were analyzed. Figure 3 shows that the number of STS-C launches for the storable propellant option (340-s I_{sp}) is very high: 17 launches for missions without aerobraking and ten for missions with aerobraking. Using STE O_2/H_2 propulsion (485-s I_{sp}), these numbers are reduced to 5 and 4 launches, respectively. Metallized $O_2/H_2/Al$ propulsion (491.4-s I_{sp}) provide the same

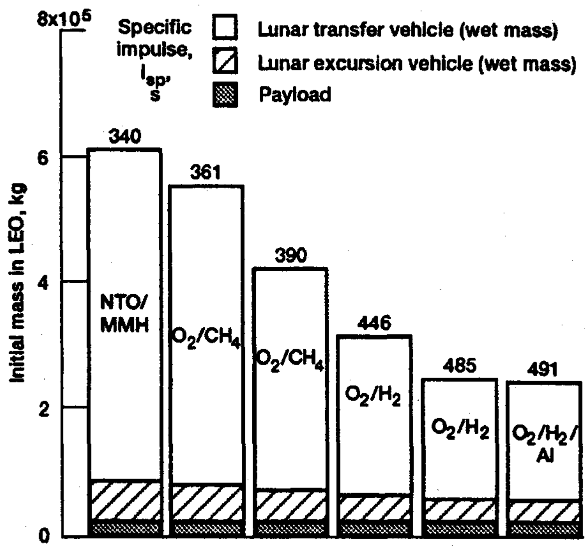

Fig. 2 LEO initial masses: chemical propulsion with aerobraking.

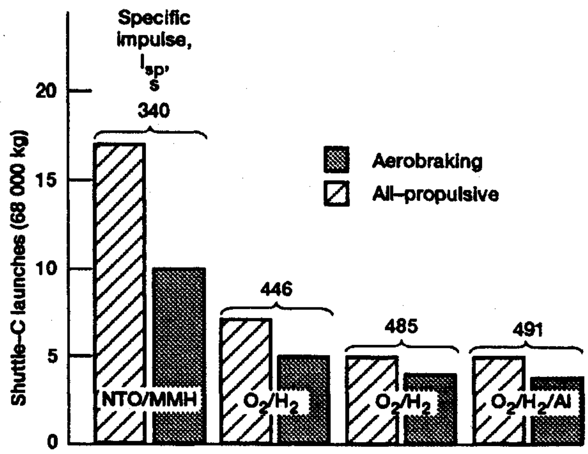

Fig. 3 STS-C launches: aerobraking and all-propulsive cases.

overall performance benefits, as measured in numbers of launches.

Five launches are needed for the STE vehicle without aerobraking and the current technology O_2/H_2 engine with aerobraking. This opens the possibility of using an all-propulsive vehicle for the initial lunar missions. This option may allow the lunar program to proceed with an option if the aerobrake development program is slowed by technical difficulties. Providing an option, such as relaxing or delaying the schedule of the aerobrake development and using an all-propulsive vehicle for the initial lunar missions may be attractive. It can reduce the program risk of tying the lunar missions to the aerobrake development schedule. Also, the mission risk of using an all-propulsive system may be perceived to be lower than that of an aerobraked vehicle.

Metallized Propellants

In Fig. 1, the mass of an advanced metallized propulsion system using $O_2/H_2/Al$ propellants is compared to an existing O_2/H_2 system and the STE. For the 27,000-kg payload mission, a 20-% LEO mass savings is possible using the STE. A 23-% LEO mass reduction is enabled over the 446.4-s I_{sp} engine by using metallized propellants.

Metallized propellants can also be used to increase the payload delivered to the lunar surface. Table 5 provides a mass summary for the LEV and LTV. An initial mass in LEO for the two cases was fixed at 248,500 kg. Figure 4 compares the payload capability of metallized cases and the other O_2/H_2 cases. Using metallized $O_2/H_2/Al$, a 870-kg (or a 3.2%) increase in payload is possible over the STE system.

Based on these analyses, metallized propellants will provide a modest benefit for the lunar missions considered. Thus metallized propellants, while they provide a payload benefit, may not be deemed necessary given the relatively small advantage (3% added payload or 3% reduction in LEO mass) over the STE. If, at a later date, the payload manifest required the added payload benefit, metallized propulsion could be considered.

A lunar transfer vehicle testbed for metallized propellants could be considered as an option. This propulsion technology can provide benefits on a future Mars mission. Metallized propellants do enable from 20 to 33% added payload for Mars missions.[14] The lunar environment can be used to test the vehicle engine performance and the operational differences with metallized propellant feed systems, such as gelled cryogen boiloff in zero gravity and long-term exposure of gelled fuels to the space and planetary surface environments. These would be important data to acquire for designing potential Mars injection, transfer, and excursion vehicles.

Excursion Vehicle LLO Mass

The mass in LLO was determined for a wide range of LEV I_{sp}. In Fig. 5, the total mass of the O_2/H_2 excursion vehicle in LLO is shown vs engine I_{sp}. An engine mixture ratio of 6:1 was used for all cases. It is a single-stage automated vehicle with a 27,000-kg payload to the surface and no payload returned to LLO. The excursion vehicle mass varies by 143 kg (one-seventh of a metric ton) per second of I_{sp} in the 445–465 s range whereas the mass in LLO varies by 111 kg (one-ninth of a metric ton) per second of I_{sp} for the range of 475–485 s. Overall, the sensitivity of the LLO mass to I_{sp} is low. Thus, the mass of the LEV will not be significantly affected by reductions in engine I_{sp}.

Table 5 Metallized $O_2/H_2/Al$ and O_2/H_2 mass summary for lunar excursion and transfer vehicles (unmanned cargo flight)

Element	Mass, kg O_2/H_2	Mass, kg $O_2/H_2/Al$
Lunar excursion vehicle		
Descent payload	27,000	27,871
Ascent payload	0	0
Adapter (payload to LEV)	1421	1467
Propellant tankage	498	503
Pressurization	107	119
Engines and feed system	1240	1240
Thermal control	1153	1160
Structure	1773	1784
Residuals and holdup	703	707
Contingency (10%)	547	551
Leg structure	788	807
Usable propellant	25,334	25,485
Total	60,564	61,694[a]
Lunar transfer vehicle		
Payload to LLO	60,564	61,694
Margin	436	436[b]
Capability to LLO	61,000	62,130
Payload returned to LEO	0	0
Adapter (payload to LEV)	3211	3270
Stage 2		
Propellant tankage	472	473
Pressurization	101	112
Engines and feed system	1240	1240
Thermal control	1091	1091
Structure	1678	1677
Residuals and holdup	665	665
Contingency (10%)	525	526
Aerobrake	2030	2044
Usable propellant	23,976	23,961
Adapter (interstage)	5052	5115
Stage 1		
Propellant tankage	2450	2437
Pressurization	524	576
Feed system	99	99
Thermal control	5539	5489
Structure	8718	8639
Residuals and holdup	3456	3425
Contingency (10%)	2079	2067
Usable propellant	124,538	123,411
Total	248,444	248,447

[a]Total masses of the excursion vehicles differ because, for a constant mass in LEO for the combined excursion and transfer vehicles, the metallized propulsion option will allow a larger excursion vehicle mass to be delivered to lunar orbit and thus more payload delivered to the surface.
[b]The margin is used to accommodate any LEV mass contingencies.

System Design Issues

After examining the global issues of the LEO mass and the payload capabilities of the propulsion options, several issues regarding the overall system design should be addressed: engine technology availability, thrust levels, and the use of small lunar vehicles on a single STS-C flight.

Engine Efficiency

For all of the new engine designs that are postulated, engine efficiency will be a critical issue. Assuring the highest possible performance will require component and system technology programs and engine development programs for the O_2/H_2 and metallized $O_2/H_2/Al$. Investing in these propulsion technologies will be important not only for the lunar missions but also for the future Mars exploration program.

Pump-Fed Metallized Propulsion

With the very high performance O_2/H_2 systems being considered for lunar exploration, a pump-fed engine is required. Pressure-fed propulsion systems typically require larger masses for propellant tankage and pressurization systems. Using metallized propellants, the propellant feed system must be designed to provide the non-Newtonian, thixotropic metallized propellant with the same reliability as the non-metallized H_2.[19–21] Currently, metallized propellants are fed to smaller propulsion systems with positive-displacement propellant expulsion devices (diaphragms, etc.).[15] A positive expulsion system and a pressure-fed system, however, are deemed too heavy or impractical for large propellant tanks. For the extremely large propellant loads

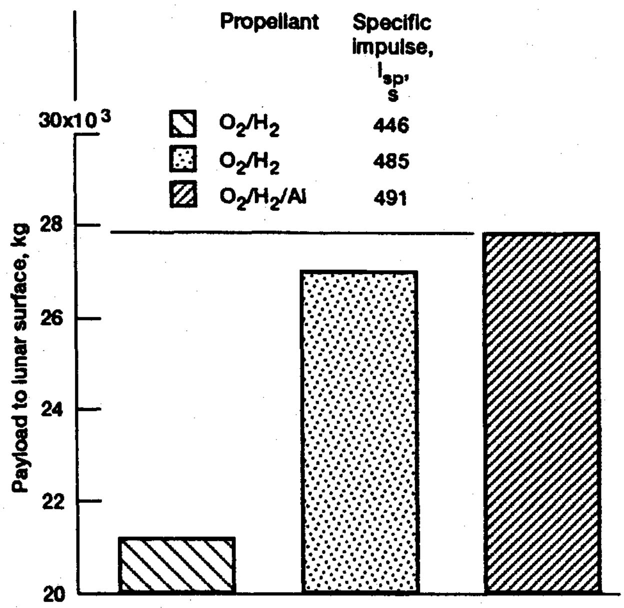

Fig. 4 Metallized $O_2/H_2/Al$ payload capability.

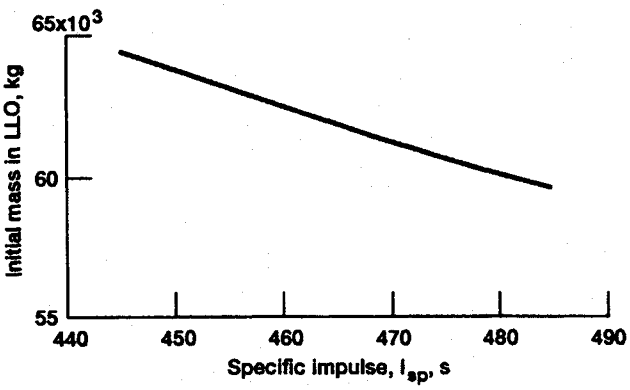

Fig. 5 Excursion vehicle mass in LLO vs specific impulse.

needed on the lunar missions, a different approach will be required. The propellant flow properties have been studied both experimentally and analytically.[19-21] These studies will help determine the best propellant acquisition and feed system for these large propulsion systems.

Thrust Level: Common Propulsion Module

In the selection of the thrust level, the vehicle thrust to weight T/W and the options for differing translunar trajectories that would promote engine commonality should be considered. Using a common engine module can reduce the development cost for any lunar vehicle program. Table 6 provides the engine firing times for STE O_2/H_2 propulsion (485-s I_{sp}) with a 27,000-kg payload cargo delivery mission. Both 222,400- and 355,840-N thrust levels were considered. These firing times for the translunar injection would force the selection of multiple firings or a higher thrust level for TLI. A higher thrust level was not selected because that would require a higher thrust than that needed for the LEV. This would defeat the intent of providing a common engine module for both lunar vehicles.

Thrust to Weight (T/W)

A series of analyses were conducted to find a common range of thrust level for the two vehicles. The needed O_2/H_2 thrust levels for a lunar transfer vehicle and the lunar excursion vehicles are shown in Fig. 6. The payload mass in the figure is that delivered to the surface by the excursion vehicle and the payload that is delivered to LLO by the transfer vehicle. The LEV T/W is 0.6 and that for the LTV is 0.1 in this figure. The excursion vehicle T/W is estimated based on the thrust level needed for lunar descent and the need to provide engine redundancy in case of failure. For a four-engine module, the total thrust delivered by two engines should still provide the required landing thrust level. This allows the module to suffer a single engine failure and still maintain the thrust axis through the vehicle's center of gravity. To maintain the alignment of the thrust axis (if one engine were to fail), the engine opposite the failed one would be shut down and the mission continued with two engines.

Table 6 Lunar vehicle engine firing times for O_2/H_2 propulsion[a]

Maneuver	Firing time, s	
Thrust level, N	222,400	355,840
Lunar transfer vehicle		
Translunar injection (TLI)	2670	1670
Lunar orbit insertion (LOI)	430	270
Trans-Earth insertion (TEI)	70	44
Earth orbit injection (EOI)	16	10
Lunar excursion vehicle		
Deorbit to landing	450	280
Ascent to orbit	87	55

[a]Specific impulse I_{sp}, 485 s; mission assumptions: for LEV, 27,000 kg to surface, 0 kg returned to LLO; for LTV, 61,000 kg to LLO, 0 kg returned to LEO.

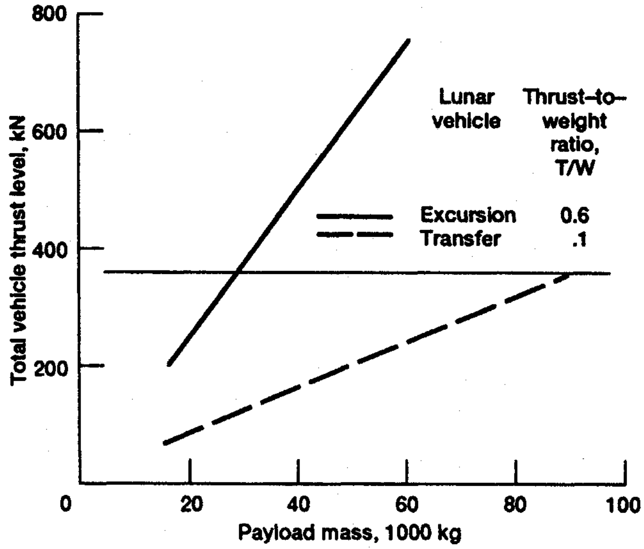

Fig. 6 Thrust vs payload mass.

A transfer vehicle T/W range of 0.1–0.225 has been suggested.[3] For the higher T/W (0.225), the gravity losses for the translunar injection are small. However, this T/W will not allow a common module to be used for both the LEV and the LTV. The lower T/W (0.1) will require a longer firing time for the propulsion module. To minimize the potential gravity losses from the longer firing time, multiple firings will be needed. The LTV T/Ws were traded with that of the LEV to determine the region where a common thrust level was possible. If the thrust levels for the vehicles were 355,840 N, a common engine module can be used for both the lunar transfer and the lunar landing. This 355,840-N thrust level can allow an excursion vehicle to place up to 27,000 kg on the lunar surface and allow a LTV to deliver up to 90,000 kg to LLO. The current design for the transfer vehicle requires only 61,000 kg be delivered to LLO. At the 355,840-N thrust level, the LTV has an initial T/W of (355,840 N/[248,500 kg $\times$ 9.81 m/s^2] =) 0.15. With this T/W, however, multiple firings for the translunar injection will be required.

TLI Perigee Firings

The number of engine firings and their effect upon the Earth departure (or translunar injection) is described in Table 7. After each firing, the LTV is on a transfer ellipse. Successive firings of the engines are performed when the vehicle returns to the orbit perigee. The method for estimating the trip times was derived from Ref. 16. In each case, the total ΔV for the TLI maneuver is divided equally amongst the firings. For two TLI firings, the total added time for the LEO departure is 3.78 h. Additional multiple firings of the Earth departure stage will add several hours to the total time required for the translunar injection. This added time, however, is traded with the need for a common engine for both the excursion vehicle and the transfer vehicle. Adding the short amount of time to the total trip can reduce the need for developing separate propulsion modules for the two lunar vehicles. The crew, however, would be exposed to the Van Allen Belts several times and this option may be rejected for this reason alone.

Small Missions on One STS-C Flight

During the Apollo Program, a series of studies were conducted, assessing the payloads that might be delivered for the construction of a lunar base. These studies were called Apollo Extension System (AES), Apollo Logistics Support System (ALSS), and Lunar Exploration System for Apollo (LESA).[17] Table 8 lists the mission payload masses that were considered. These payloads were used for exploring the surface with rovers and slowly building a semipermanent lunar base. The mass per "shot" is an average, in some cases, of several Saturn V launches. Some of the averaged launches are only to deliver crew with a minimal payload. Other missions are dedicated cargo missions. Each of their payloads and transportation systems were designed to be flown using Apollo-derived vehicles: Saturn V, the command and service modules, and the lunar module.

Many of the missions analyzed for the post-Apollo program were designed to deliver payloads that are relatively small compared to the recently proposed NASA lunar payloads. It is clear that the lunar program will be expensive. Perhaps one avenue to reducing this cost is to reduce the size of the payloads that are under consideration. By downsizing the payloads, the overall transportation vehicles can be smaller and less costly.

Table 7 Lunar transfer vehicle: multiple firings for translunar injection

Number of firings	Total added trip time, h	Intermediate altitudes, km
2	3.78	11,400
3	9.02	6004
		21,780
4	15.32	4178
		11,400
		31,607

Table 8 Average payload to the lunar surface per Saturn V expended[a]

Type of mission	Weight of equipment delivered per launch, kg
Apollo	113.4
AES	567.0
ALSS	1814.4
LESA 1	4128.5
LESA 3	2293.7–4989.6

[a]From Ref. 16; mission assumptions: for LEV, cargo delivered to surface, 0 kg returned to LLO; for LTV, LEV delivered to LLO, 0 kg returned to LEO.

A small-scale lunar mission and its ability to fit into a smaller launch vehicle was analyzed. Figure 7 compares several types of O_2/H_2 propulsion for the small LESA-class transportation system. The LESA-class system only requires one STS-C launch for a complete lunar mission. Table 9 compares the payload capabilities for several O_2/H_2 propulsion technologies for these small missions. Two STS-C payload capabilities were used: 68,000 and 71,000 kg. Aerobraking is used to return to LEO. The payloads for the two systems are significantly different: 5335 kg (for 68,000 kg STS-C) for the small vehicle and 27,000 kg for the currently planned system. Although these payloads are smaller than those proposed by NASA in Refs. 1 and 3, they are comparable to the payload masses considered for the LESA Saturn-V lunar missions. Comparable exploration to that proposed in the post-Apollo era could thus be accomplished using these smaller missions, perhaps prior to the establishment of a complete lunar base.

The first lunar missions could be flown from single STS-C flights and eliminate the complexity of orbital assembly. A single launch also reduces the time between Earth launch of the first piece of the lunar spacecraft (on the first of multiple STS-C launches) and the mission departure from LEO. The four-to-five STS-C launches required for the planned NASA lunar missions may require 8–10 months (with one or two launches per month)[1,18] to have all of the elements assembled. The small mission could depart from LEO soon after arriving in orbit.

Table 9 LESA-class flight payload capabilities: single STS-C launch with metallized $O_2/H_2/Al$ and O_2/H_2 propulsion[a]

Specific impulse I_{sp}, s	STS-C capacity, kg: 68,000	71,000
	Surface payload, kg	
446.4	3400	3670
485	5087	5455
491.4	5330	5710

[a]Mission assumptions: for LEV, cargo delivered to surface, 0 kg returned to LLO; for LTV, LEV delivered to LLO, 0 kg returned to LEO.

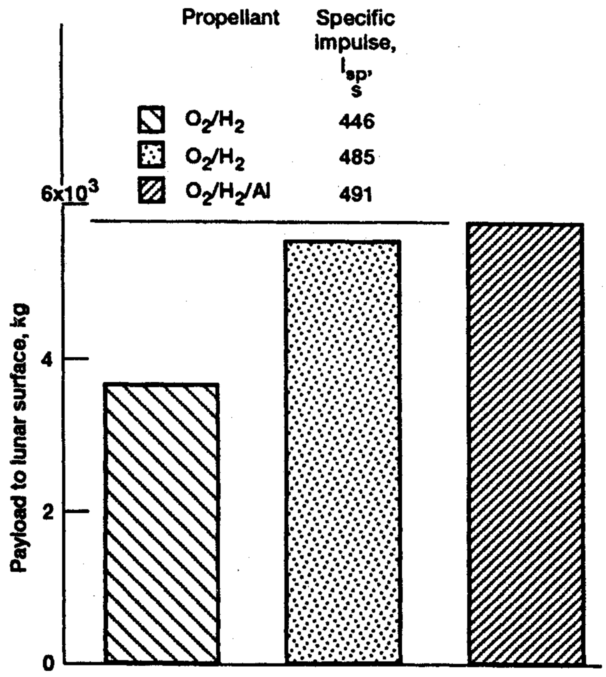

Fig. 7 Single STS-C payload capability (STS-C capacity, 68,000 kg).

If not used for the construction of a lunar base, these smaller missions might be used as science and engineering precursors to explore areas away from the lunar base: rugged cratered areas near Tycho and Copernicus, flights to the lunar poles, and exploration of the lunar farside. As was previously discussed, an engineering precursor for the Mars mission could be flown with metallized propellants. This vehicle would test the engine technology and long-term propellant storage properties in the lunar environment.

Conclusions

Advanced chemical propulsion is a powerful tool in the reduction of total system transportation mass and cost. Neither NTO/MMH nor O_2/CH_4 systems provide any LEO mass benefit over the O_2/H_2 systems. Advanced O_2/H_2 and $O_2/H_2/Al$ can both provide additional payload over existing O_2/H_2 systems (446.4 s). The STE system provides a 20% LEO mass reduction over the 446.4-s I_{sp} system. Metallized $O_2/H_2/Al$ enables a 3.2% LEO mass savings over the STE technology O_2/H_2 system.

Using the STE in an all-propulsive mission mode requires the same mass in LEO as the current O_2/H_2 system using an aerobrake. Each vehicle requires five STS-C launches. Using only one additional STS-C flight, this all-propulsive option for the lunar transfer vehicle allows the development of aerobrake technology to be delayed or eliminated.

The STE technology program is progressing toward a development program to support the lunar missions. Metallized propulsion is only in the formative stages. It promises modest benefits for lunar missions, although it shows potential for very significant payload increases for missions to Mars. Using metallized propulsion in a lunar testbed vehicle to prove this technology for Mars flights is therefore a possibility.

An 355,840-N thrust level for the LTV and LEV allows a common engine module to be used for both vehicles. This thrust level will produce a low T/W ratio for the transfer vehicle at Earth departure and will require multiple firings. Only 4–15 h are added to the total lunar trip time.

Small lunar missions, flown from a single STS-C vehicle, can deliver lunar payloads comparable to that proposed for the post-Apollo LESA exploration missions. Although their payloads are small, these missions may provide an option for scientific and engineering precursors early in a lunar base scenario: testing aerobraking, the advanced STE, and metallized propellants.

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