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Summary

To transport lunar base elements to the Moon, large high-energy propulsion systems will be required. Advanced propulsion systems for lunar missions can significantly reduce launch mass and increase the delivered payload, resulting in significant launch cost savings.

In this report, the masses in low Earth orbit (LEO) are 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 addressed are (1) payload mass increases enabled with these systems; (2) system design issues involving the engine thrust levels, engine commonality between the transfer vehicle and the excursion vehicle; the number of launches to place the lunar mission vehicles into LEO; and (3) analyses of small lunar missions launched from a single Space Transportation System–Cargo (STS–C) flight.

Introduction

NASA is considering a vigorous new initiative to place a permanent base or settlement on the lunar surface (ref. 1) which will support a wide range of experiments in science and technology. The base may also support the first human missions to Mars. There may be a potentially significant infrastructure for producing propellants on the lunar surface that 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 vehicles are considerably larger than those for the past Apollo missions. 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 provide 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 the base into lunar orbit and onto the surface will require large spacecraft and large propellant loads. The lunar vehicles will require from 4 to 17 Space Transportation System–Cargo (STS–C) launches to be delivered into orbit. The advanced O_2/H_2 and metallized $O_2/H_2/Al$ systems require the lowest mass delivered to orbit (four launches) and potentially require 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 wide range of technologies that could provide lunar transportation are available or are in development. Many of these technologies can be combined to provide a significantly different vehicle from those used in the Apollo Program. Whereas Apollo used NTO/Aerozine–50 propellant for the service module 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 have many options available to them.

The chemical propulsion systems that are considered here include Earth-storable nitrogen tetroxide/monomethyl hydrazine (NTO/MMH), space-storable oxygen/methane (O_2/CH_4), cryogenic oxygen/hydrogen (O_2/H_2), and metallized $O_2/H_2/Al$ propellants. Metallized propellants have a high density or a high I_{sp}, or both (ref. 2). With these propellants, a metal (such as aluminum) is gelled with the fuel. The metal additive increases the propellant density and, potentially, the I_{sp} of the propellant combination. The effects of using metallized propellants will be discussed in more detail later in this report.

A mission designer and systems engineer will select the "best" propulsion and other technology options by looking at many factors, including the performance, size, reliability, life, and cost of the propulsion system; the number of systems involved in the transportation architecture; and the availability of new technologies such as metallized propellants. Other technologies that will greatly affect the lunar transportation system are aerobraking, lightweight cryogenic storage, and lightweight structures—these must be factored into the overall design process.

Determining the effects and potential benefits of advanced propulsion systems requires a series of systems analyses including lunar mission analyses and propulsion system design. These issues will be discussed in the following sections.

Lunar Mission Analyses

Each lunar mission scenario will require several individual missions including piloted missions, with and without cargo; unmanned cargo delivery missions; and test missions to prove system performance before committing expensive cargos to lunar flight. These missions will carry payloads ranging from

15 000 to 27 000 kg to the lunar surface (ref. 3) and may include both reusable and expendable vehicles. The current lunar mission scenarios include a lunar transfer vehicle (LTV), which travels between LEO and low lunar orbit (LLO), and a lunar excursion vehicle (LEV), which is delivered to lunar orbit by the LTV and which may be used in a fully automated mode or with an astronaut crew.

In this study, cargo missions delivering 27 000 kg payloads to the lunar surface were analyzed to establish the size of the transfer and excursion vehicles and to provide a relative comparison of each propulsion system's LEO masses and payload capabilities. The 27 000-kg payload mass is also representative of the largest lunar base elements: a pressurized module with an attached airlock (refs. 3 and 4).

Estimating Vehicle Masses

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

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

where

ΔV velocity change, m/s

g gravitational acceleration, 9.81 m/s

m_o initial mass, kg

m_f final mass, kg

The maneuver ΔV values, taken from reference 1, are listed in table I. The lunar missions are based in LEO—all of the mission elements are delivered to LEO by using the STS–C or other launch vehicles. The mission maneuvers begin with the preinjection preparation firing and the translunar injection (TLI).

Nine maneuvers are required for the transfer vehicle; four for the excursion vehicle (table I). 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 low lunar orbit (LLO) to be refueled and refitted with a payload for the next cargo landing. To return to Earth, the transfer vehicle delivers the trans-Earth injection (TEI) ΔV and a small ΔV during the trans-Earth coast. Aerobraking is typically used for the return to Earth orbit. (See table I to compare ΔV's for Earth departure and Earth return.) If an all-propulsive Earth orbit insertion (EOI) were conducted, the ΔV would be equal to that for translunar injection. The influence of aerobraking on the LEO initial mass will be discussed later in the report.

TABLE I.—LUNAR MISSION MANEUVERS
[From ref. 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
Post-ascent orbital operations	5

The major mission maneuvers (translunar injection, lunar orbit insertion, trans-Earth injection, and Earth orbit insertion) from reference 1 are several hundred meters per second larger than those used in previous studies (refs. 5 and 6). A larger ΔV will require a larger propulsion system, which would allow a wider range of lunar departure opportunities, a longer launch window, and more flexibility to accommodate launch delays.

Staging

A series of transfer vehicles with differing numbers of stages have been considered. 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, which are expended after completing their respective maneuvers. This lightens the vehicle and reduces the mass that must be returned to Earth orbit. This, in turn, reduces the size and mass of the vehicle's aerobrake. A central vehicle "core" holds the propellant for the trans-Earth injection and the Earth orbit insertion maneuvers. This staging method allows the high-value engine module to be reused. No engines are expended with this staging method.

Only one set of drop tanks was considered in these analyses. The tanks had higher structural mass, but lower propellant mass fraction than those considered in reference 3 and in a personal communication from N. Brown, NASA Marshall Space Flight Center, Huntsville, Alabama, September 1989. Because of the lower mass fraction of the tank sets, there was no mass advantage to using more than one set of drop tanks (separate ones for translunar injection and lunar orbit insertion). In this work, drop tanks were used only to hold the translunar injection propellant.

Propulsion System Design

Engine Performance

The engine performance for each of the propellant combinations was estimated with a computer simulation code (Complex Equilibrium Compositions (CEC), ref. 7). The code-predicted I_{sp} was modified by using an engine I_{sp} efficiency η, which is the ratio of the delivered engine performance and the code-predicted I_{sp}. This reduction reflects losses due to the nozzle boundary layer, engine cycle inefficiencies, and other propulsion system losses. The engine efficiencies were derived from performance estimates (ref. 8 to 11) and comparisons with the vacuum I_{sp} predicted by the engine code.

Table II gives the design I_{sp} values selected for each propulsion system, and table III gives the engine mixture and nozzle expansion ratios. The engine chamber pressures were varied from 465 to 1500 psia, depending on 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. A nozzle expansion ratio of 400:1 to 1000:1 was selected for the transfer vehicle engine, again based on the designs of planned engines. The expansion ratio was reduced for the excursion vehicle because of packaging constraints that may limit the size of the nozzles. This reduction caused the I_{sp} to be reduced by 10 lb_f-s/lb_m; for example, the I_{sp} values for the space transfer engine (STE) were 485 lb_f-s/lb_m for the lunar transfer vehicle and 475 lb_f-s/lb_m for the lunar excursion vehicle.

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. Some of the issues that are important in determining the appropriate design for a metallized propulsion system are the propellant density, the performance, and the system dry mass.

TABLE II.—PROPULSION SYSTEM PERFORMANCE

Propellant	Specific impulse, I_{sp}, lb_f-s/lb_m		I_{sp} efficiency, η
	Lunar transfer vehicle	Lunar excursion vehicle	
NTO/MMH	340.0	330.0	0.940
O_2/CH_4[a]	360.8	350.8	.940
O_2/CH_4[b]	390.0	380.0	.940
O_2/H_2[c]	446.4	436.4	.962
O_2/H_2[d]	485.0	475.0	.984
$O_2/H_2/Al$	491.4	481.4	.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 III.—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-Percent aluminum loading in H_2.

Propellant Density

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

$$\rho_{p,m} = \frac{\left(\dfrac{\mathrm{ML}}{1 - \mathrm{ML}}\right) + 1}{\dfrac{\mathrm{ML}}{(1 - \mathrm{ML})\,\rho_m} + \dfrac{1}{\rho_p}}$$

where

$\rho_{p,m}$ density of metallized oxidizer or fuel, kg/m^3
ML metal loading (fraction of oxidizer or fuel mass)
ρ_m density of metal in oxidizer or fuel, kg/m^3
ρ_p density of nonmetallized oxidizer or fuel, kg/m^3

To deliver the maximal reduction in LEO mass or the maximal payload increase, trade studies must be conducted to determine the "best" 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 percent of the fuel mass. Since 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 60-percent loading performance level was selected from metal loading experience with solid rocket motors (Space Shuttle Transportation System, press information from Rockwell International, March 1982). The total metal loading of all of the propellant

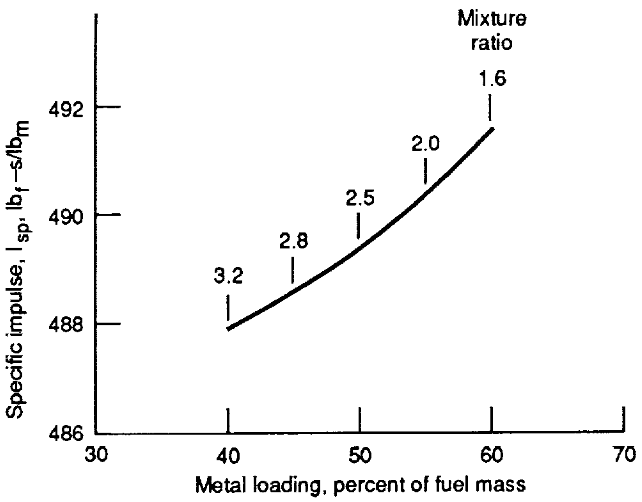

Figure 1.—Specific impulse versus metal loading (expansion ratio, 1000:1).

(oxidizer and fuel) of the propulsion system was 23 percent, which is comparable to that of solid propulsion systems. An I_{sp} of 491.4 lb_f-s/lb_m was delivered at a metal loading of 60 percent of Al in the H_2/Al, a nozzle expansion ratio 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 because, though 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 12 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 figure 1 (which has a metal loading of 60 percent) was selected.

Transfer Vehicle Mass-Scaling Equations

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

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

where

m_{dry}	dry mass
A,B,C	mass parameters
D	aerobrake mass fraction (0.1725)
m_{entry}	total entry mass during aerobraking maneuver, kg

Table IV provides the propulsion mass-scaling parameters for all the systems considered. These parameters include all the masses that are required to store and provide propellants to the main engines. Parameters provide a model for tanks, engines, feed system, thermal control, structure, residuals, and contingency. The parameter A of the scaling equations varied from 109 to 1364 for the lunar vehicles because of 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 tankset that has no engine components. Only the latter value of A, 1364, is shown in the table.

Propellant Tankage

The propellant tankage for all the systems uses a 50-psia maximal operating pressure. The propellant is stored at 30 psia. All of the tanks for O_2, H_2, and CH_4 are composed of aluminum alloy, whereas 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 percent of the total propellant mass. The percentage accommodates the added propellant mass for cryogenic propellant boiloff.

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 3722 psia. The storage pressure is 3444 psia. 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 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.

For thermal control, the cryogenic propellants (O_2, H_2, and CH_4) require a high-performance multilayer insulation and a thin-walled vacuum jacket sized for a 30-psia maximal operating pressure. After the vehicle reaches space, the space between the jacket and the tank is vented and evacuated. The storable propellants require only a lower-performance multilayer insulation.

Aerobraking

The aerobrake mass is 17.25 percent of the vehicle mass entering the atmosphere (refs. 6 and 13). The 17.25-percent mass factor represents 15 percent multiplied by 1.15, which

TABLE IV.—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	.1676	.0516
O_2/CH_4[b]	↓	.1669	.0463
O_2/H_2[c]		.1853	.0858
O_2/H_2[d]		.1811	.0806
O_2/H_2/Al		.1817	.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.

represents a contingency of 15 percent. This mass includes the payload, propulsion system dry mass, any propellant needed for the entry, and post-entry maneuvers and the aerobrake.

Excursion Vehicle Mass-Scaling and Design

The mass-scaling equation for the excursion vehicle stage is

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

where

D mass parameter for leg structure (0.02)

m_{landed} total landed mass on the surface, kg

The excursion vehicle is sized to give the ΔV values listed in table I. 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 low lunar orbit (LLO) empty. The excursion vehicle sizing parameters are similar to those for the transfer vehicle.

An important aspect of the excursion vehicle is its leg structure for support on the Moon. The leg is part of the descent stage and its mass is 2 percent of the total mass landed on the surface.

Results of Systems Analyses

In this section, analyses of the LEO mass, excursion vehicle masses, and the relative performance of the various chemical propulsion technologies will be discussed. The potential advantages of these technologies, in terms of increased payload and reduced mass in LEO, will also be discussed. Other system-level design considerations, such as thrust levels, engine firing times, and the potential for using small transfer and excursion vehicles for lunar exploration will be presented in the next section.

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 by using a constant mass in LEO.

In figure 2, the LEO masses are contrasted for six systems: NTO/MMH (340 lb_f-s/lb_m I_{sp}), two O_2/CH_4 systems (360.8 and 390 lb_f-s/lb_m I_{sp}), two O_2/H_2 systems (446.4 and 485 lb_f-s/lb_m I_{sp}), and O_2/H_2/Al (491.4 lb_f-s/lb_m I_{sp}). Clearly, the propulsion options that provide the lowest LEO mass are

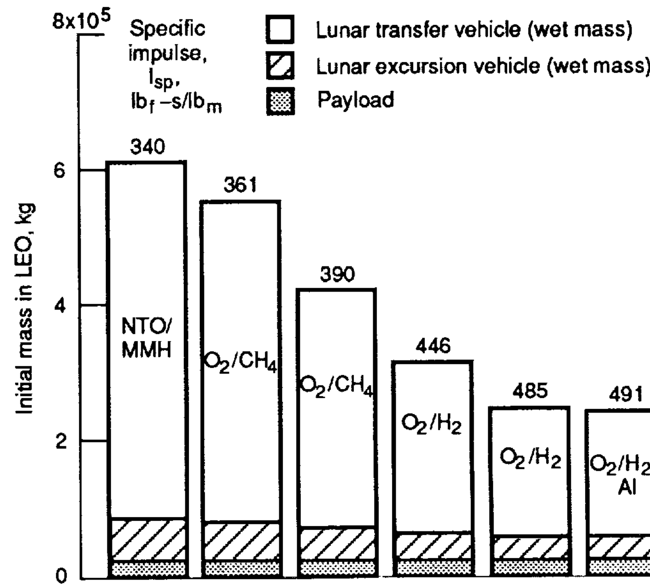

Figure 2.—LEO initial masses: chemical propulsion with aerobraking.

metallized O_2/H_2/Al and the space transfer engine (STE) O_2/H_2 system (485 lb_f-s/lb_m I_{sp}). Each O_2/H_2 vehicle (485 lb_f-s/lb_m I_{sp}) requires only 248 500 kg for the mission. Using the STE O_2/H_2 system provides 20-percent LEO mass reduction over the current-technology O_2/H_2 system (446.4 lb_f-s/lb_m I_{sp}). Metallized propellants provide a 23-percent mass reduction over the 446.4 lb_f-s/lb_m O_2/H_2 system.

The LEO mass performance of O_2/CH_4 propulsion is superior to that of 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-lb_f-s/lb_m I_{sp} system and 420 000 kg for the higher I_{sp} O_2/CH_4 vehicle. With storable NTO/MMH, the mass in LEO is considerably higher than that for any other case: 613 000 kg. For the large payloads that are being considered for the lunar base, neither the O_2/CH_4 nor the NTO/MMH options appear attractive for a lunar mission.

Aerobraking Versus 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 lb_f-s/lb_m I_{sp}) is very high: 17 launches for missions without aerobraking and 10 for missions with aerobraking. With O_2/H_2 propulsion (485 lb_f-s/lb_m I_{sp}), these numbers are reduced to five and four launches, respectively. Metallized O_2/H_2/Al propulsion (491.4 lb_f-s/lb_m I_{sp}) provides the same overall performance benefits: five and four launches for the all-propulsive and aerobraked cases, respectively.

In comparing the STE (485-lb_f-s/lb_m I_{sp}) all-propulsive case and the 446.4-lb_f-s/lb_m I_{sp} O_2/H_2 case with aerobraking, the LEO masses are comparable. Five launches are needed for the STE vehicle without aerobraking and the current-

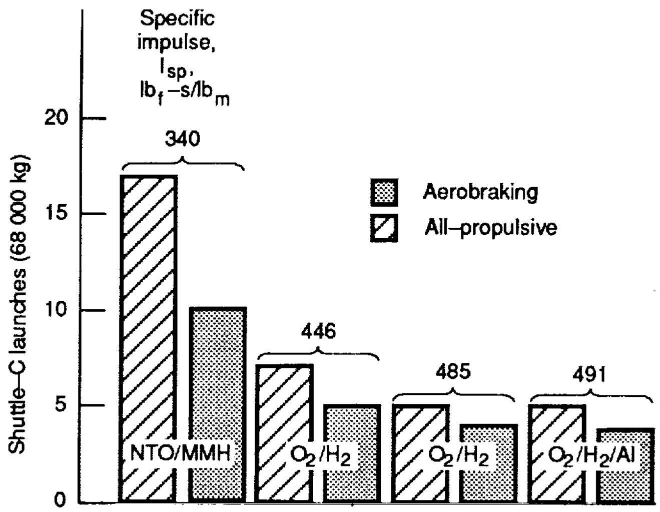

Figure 3.—STS-C launches: aerobraking and all-propulsive cases.

technology O_2/H_2 engine with aerobraking. This opens the possibility of using an all-propulsive vehicle for the initial lunar missions.

Looking at this issue from another perspective, this all-propulsive STE vehicle has a relatively small launch mass penalty of one STS-C launch (five launches instead of four) over the case with advanced STE O_2/H_2 and aerobraking. This option may allow the lunar program, especially the initial lunar missions, to proceed if the aerobrake development program is slowed by technical difficulties. 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 figure 2, 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-percent LEO mass savings is possible by using the STE. A 23-percent LEO mass reduction is enabled over the 446.4-lb_f-s/lb_m I_{sp} engine for this mission by using metallized propellants.

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

Based on these analyses, metallized propellants will provide a modest benefit for lunar missions; however, they may not be deemed necessary given the relatively small advantage (3-percent added payload or 3-percent reduction in LEO mass) over the STE. If, at a later date, the NASA payload manifest requires the added payload benefit, metallized propulsion should be considered.

TABLE V.—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	$O_2/H_2/Al$
Lunar excursion vehicle		
Descent payload	27 000	27 871
Ascent payload	0	0
Adapter (payload to LEV)	1 421	1 467
Propellant tankage	498	503
Pressurization	107	119
Engines and feed system	1 240	1 240
Thermal control	1 153	1 160
Structure	1 773	1 784
Residuals and holdup	703	707
Contingency (10 percent)	547	551
Leg structure	788	807
Usable propellant	25 334	25 485
Total	60 564	[a]61 694
Lunar transfer vehicle		
Payload to LLO	60 564	61 694
Margin	436	[b]436
Capability to LLO	61 000	62 130
Payload returned to LEO	0	0
Adapter (payload to LEV)	3 211	3 270
Stage 2		
Propellant tankage	472	473
Pressurization	101	112
Engines and feed system	1 240	1 240
Thermal control	1 091	1 091
Structure	1 678	1 677
Residuals and holdup	665	665
Contingency (10 percent)	525	526
Aerobrake	2 030	2 044
Usable propellant	23 976	23 961
Adapter (interstage)	5 052	5 115
Stage 1		
Propellant tankage	2 450	2 437
Pressurization	524	576
Feed system	99	99
Thermal control	5 539	5 489
Structure	8 718	8 639
Residuals and holdup	3 456	3 425
Contingency (10 percent)	2 079	2 067
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.

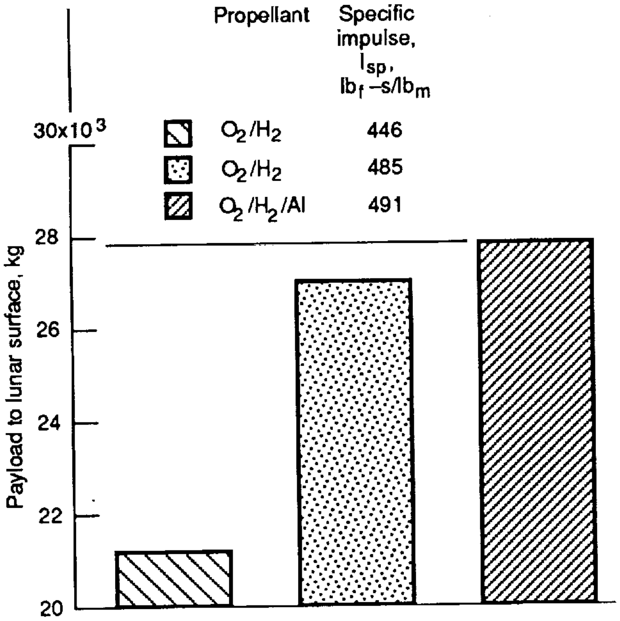

Figure 4.—Metallized $O_2/H_2/Al$ payload capability.

A lunar transfer vehicle testbed for metallized propellants should also be considered as an option. This propulsion technology can provide benefits on a future Mars mission. Metallized propellants do enable 20- to 33-percent added payload for Mars missions (ref. 12). The lunar environment can be used to test the vehicle engine performance and the operational differences with metallized propellant feed systems. 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 excursion vehicle I_{sp}'s (fig. 5). An engine mixture ratio of 6:1 was used for all cases. The excursion vehicle mass varied by 143 kg (one-seventh of a metric ton) per second of I_{sp} in the 445- to 465-lb_f-s/lb_m range, whereas the mass in LLO varied by 111 kg (one-ninth of a metric ton) per second of I_{sp} for the range of 475 to 485 lb_f-s/lb_m. Overall, the sensitivity of the LLO mass to I_{sp} is low. Thus, the mass of the excursion vehicle will not be significantly affected by reductions in engine I_{sp}.

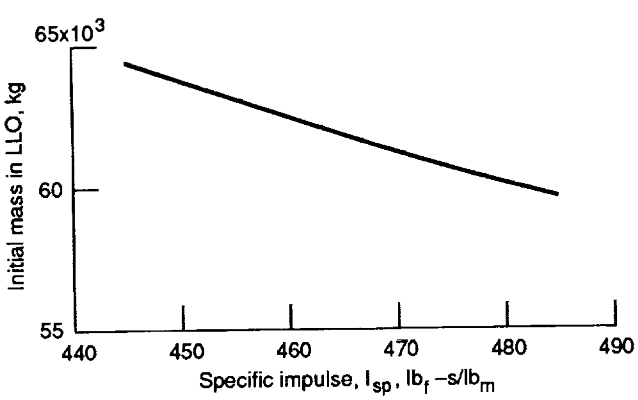

Figure 5.—Excursion vehicle mass in LLO versus specific impulse.

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 Technology

For all 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.

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. If metallized propellants are used, the propellant feed system must be designed to provide the non-Newtonian, thixotropic metallized propellant with the same reliability as the nonmetallized H_2. Currently, metallized propellants are fed to smaller propulsion systems with positive-displacement propellant expulsion devices such as diaphragms (ref. 14). A positive expulsion system and a pressure-fed system, however, are too impractical for large propellant tanks. For the extremely large propellant loads needed on the lunar missions, a different approach will be required. The propellant flow properties are being studied both experimentally and analytically to help determine the best propellant acquisition and feed system for these large propulsion systems.

Thrust Levels

The thrust-to-weight ratio (T/W) and the options for differing translunar trajectories should also be considered. In this report, thrust levels were selected to provide a common propulsion module for the transfer and excursion vehicles. A common module can potentially reduce the development cost for the lunar vehicle systems. Table VI provides the engine firing times for STE O_2/H_2 propulsion (485 lb_f-s/lb_m I_{sp}) with a 27 000-kg payload cargo delivery mission. Both 50 000- and 80 000-lb_f thrust levels were considered. These firing times for the translunar injection would force the selection of multiple firings or a higher thrust level for the transfer vehicle. A higher thrust level was not selected because that would require a higher thrust than that needed for the excursion vehicle. This would defeat the intent of providing a common engine module for both lunar vehicles.

TABLE VI.—LUNAR VEHICLE ENGINE FIRING TIMES FOR O_2/H_2 PROPULSION

[Specific impulse I_{sp}, 485 lb_f-s/lb_m; 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.]

Maneuver	Firing time, s	
Thrust level, lb_f	50 000	80 000
Lunar transfer vehicle		
Translunar injection (TLI)	2 670	1 670
Lunar orbit insertion (LOI)	430	270
Trans-Earth injection (TEI)	70	44
Earth orbit insertion (EOI)	16	10
Lunar excursion vehicle		
Deorbit to landing	450	280
Ascent to orbit	87	55

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 figure 6. The payload mass in the figure is the payload delivered to the surface by the excursion vehicle and the payload delivered to LLO by the transfer vehicle. The excursion vehicle T/W is 0.6 and that for the transfer vehicle is 0.1 in this figure. The excursion vehicle T/W is estimated on the basis of 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.

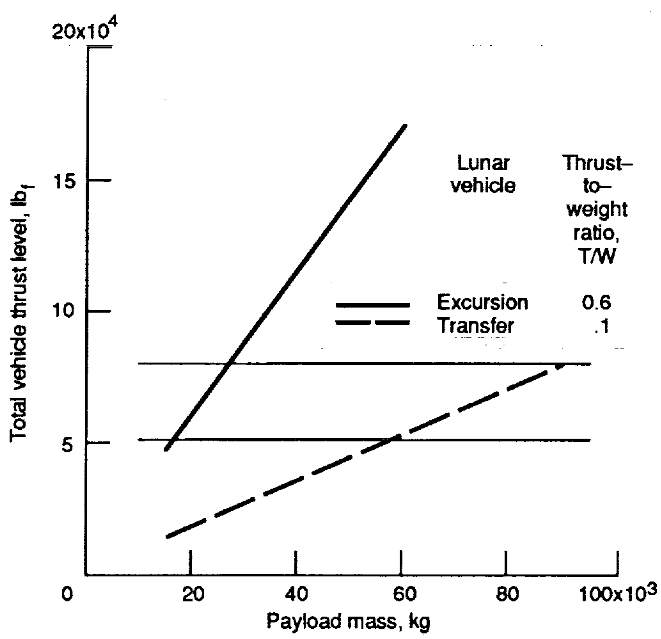

Figure 6.—Thrust versus payload mass.

A transfer vehicle T/W range of 0.1 to 0.225 has been suggested (ref. 14). 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 excursion and transfer vehicles. 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 transfer vehicle T/W's were traded with that of the excursion vehicle to determine the region where a common thrust level was possible. If the thrust levels for the vehicles were 80 000 lb_f, a common engine module can be used for both the lunar transfer and the lunar landing. This 80 000-lb_f thrust level can allow an excursion vehicle to place up to 27 000 kg on the lunar surface and allow a transfer vehicle 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 80 000-lb_f thrust level, the transfer vehicle has an initial T/W of 0.15 (80 000 lb_f/(248 500 kg × 2.2046 lb_m/kg)). With this T/W, however, multiple firings for the translunar injection will be required.

The number of engine firings and their effect on Earth departure (or translunar injection) is described in table VII. After each firing, the transfer vehicle is on a transfer ellipse. Successive firings of the engines are performed when the vehicle returns to the orbit perigee. Trip times were estimated by a method derived from reference 15. In each case, the total ΔV for the translunar injection maneuver is divided equally among the firings. For two firings, the total added time for the LEO departure is 3.78 hr. 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 an acceptable tradeoff for having a common engine for both the excursion vehicle and the transfer vehicle.

Small Missions on a Single STS-C Flight

During the Apollo Program, a series of studies were conducted to assess the payloads that might be delivered to

TABLE VII.—LUNAR TRANSFER VEHICLE: MULTIPLE FIRINGS FOR TRANSLUNAR INJECTION

Number of firings	Total added trip time, hr	Intermediate altitudes, km
2	3.78	11 400
3	9.02	6 004 21 780
4	15.32	4 178 11 400 31 607

the Moon by post-Apollo missions and the construction of a lunar base: the Apollo Extension System (AES), the Apollo Logistics Support System (ALSS), and the Lunar Exploration System for Apollo (LESA, ref. 16). Table VIII lists the potential mission payload masses for exploring the surface with rovers and slowly building a semipermanent lunar base. The mass per "shot", in some cases, is an average 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 on 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 proposed NASA lunar payloads. It is clear that the lunar program will be expensive. Perhaps one way to reduce this cost is to reduce the size of the payloads that are under consideration. By down-sizing the payloads, the overall transportation vehicles can be smaller and less costly.

A small-scale lunar mission and its ability to fit into a smaller launch mass was analyzed. Figure 7 compares several types of O_2/H_2 propulsion for the small LESA-class transportation system. The LESA-class system requires only one STS–C launch for a complete lunar mission. Table IX 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. Though these payloads are smaller than those proposed by NASA in the 90-Day Study (ref. 2), they are comparable to the payload masses considered for the LESA Saturn V lunar missions. These smaller missions could be particulary effective during the construction of a lunar base.

The first lunar missions could be flown from single STS–C flights to 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 to 10 months (with one or two launches per month (refs. 1 and 17)) to have all of the elements assembled. The small mission could depart from LEO soon after arriving in orbit.

TABLE VIII.—AVERAGE PAYLOAD TO THE LUNAR SURFACE PER SATURN V EXPENDED

[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.]

Type of mission	Weight of equipment delivered per launch	
	lb_m	kg
Apollo	250	113.4
AES	1 250	567.0
ALSS	4 000	1814.4
LESA 1	9 300	4128.5
LESA 3	6 600 to 11 000	2993.7 to 4989.6

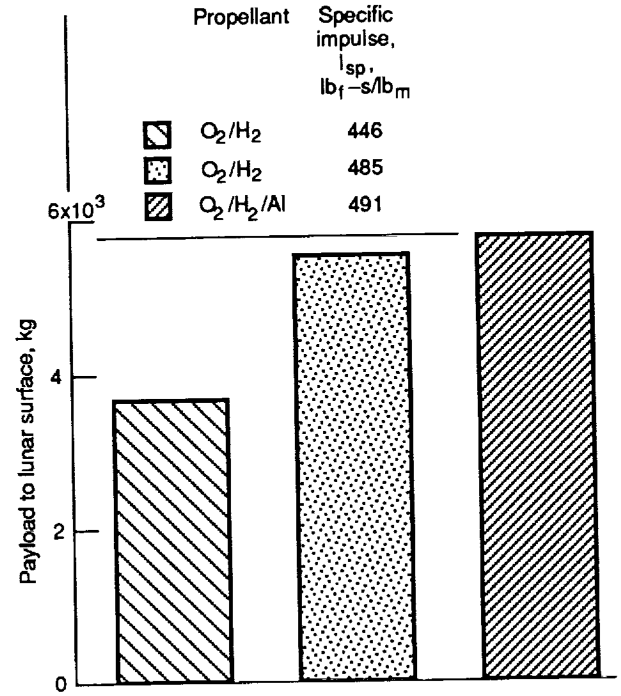

Figure 7.—Single STS–C payload capability (STS–C capacity, 68 000 kg).

TABLE IX.—LESA-CLASS FLIGHT PAYLOAD CAPABILITIES: SINGLE STS-C LAUNCH WITH METALLIZED $O_2/H_2/Al$ AND O_2/H_2 PROPULSION

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

Specific impulse, I_{sp}, lb_f-s/lb_m	STS-C capacity, kg	
	68 000	71 000
	Surface payload, kg	
446.4	3 400	3 670
485	5 087	5 455
491.4	5 330	5 710

If not used for the construction of a lunar base, these smaller missions might be used to explore areas away from the lunar base: the rugged cratered areas near Tycho and Copernicus, the lunar poles, and the lunar far side. Also, an engineering precursor for the Mars mission could be flown with metallized propellants to test the engine technology and long-term propellant storage properties in the lunar environment.

Conclusions

Advanced chemical propulsion is a powerful tool for reducing total system transportation cost. Neither NTO/MMH nor O_2/CH_4 propulsion 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 the existing O_2/H_2 system (446.4 lb_f-s/lb_m I_{sp}). The space transfer engine (STE) system provides a 20-percent LEO mass reduction over the 446.4-lb_f-s/lb_m I_{sp} system. Metallized $O_2/H_2/Al$ provides a 23-percent LEO mass savings over the current-technology O_2/H_2 system.

Using the STE in an all-propulsive mission option requires the same mass in LEO as an O_2/H_2 system with a 446.4-lb_f-s/lb_m I_{sp} using aerobraking. Each system requires five STS–C launches. The all-propulsive STE vehicle has only a small launch mass penalty of one STS–C launch over the STE vehicle with aerobraking. The all-propulsive STE option for the lunar transfer vehicle allows the aerobrake development to be delayed or have its schedule relaxed without delaying the lunar program.

The STE technology program is vigorously 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 and, more importantly, significant payload increases for missions to Mars. Using metallized propulsion in a lunar testbed vehicle to prove this technology for Mars flights is therefore recommended.

An 80 000-lb_f 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 thrust-to-weight ratio (T/W) for the transfer vehicle. The low T/W will require multiple firings to be performed for the translunar injection. Only 4 to 15 hr (required for two and four firings, respectively) 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. Though the payloads delivered by these missions are significantly smaller than those proposed by the current NASA scenarios, these missions may provide an option for scientific and engineering precursors early in the lunar base scenario: testing aerobraking, the advanced STE, and metallized propellants.

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Lewis Research Center
National Aeronautics and Space Administration
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16. Abstract

To transport lunar base elements to the Moon, large high-energy propulsion systems will be required. Advanced propulsion systems for lunar missions can significantly reduce launch mass and increase the delivered payload, resulting in significant launch cost savings. In this report, the masses in low Earth orbit (LEO) are 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 addressed are (1) payload mass increases enabled with these systems; (2) 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; and (3) analyses of small lunar missions launched from a single Space Transportation System-Cargo (STS-C) flight.

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