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Feasibility Analysis of Cislunar Flight Using the Shuttle Orbiter

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Summary

A first order orbital mechanics analysis has been conducted to examine the possibility of utilizing the Space Shuttle orbiter to perform payload delivery missions to lunar orbit. In this mission scenario, a shuttle orbiter and external tank combination, refueled in earth orbit, would fly out to lunar distance, brake into lunar orbit, deploy payloads, and then fly back to earth orbit. The vehicle would utilize the orbiter's main engines and would have a fully-fueled external tank, which will have been carried into orbit by the Shuttle and then refueled at Space Station Freedom by Shuttle-C tankers.

In the analysis, the earth orbit of departure was constrained to be that of Space Station Freedom. Furthermore, no enhancements of the orbiter's thermal protection system were assumed. Therefore, earth orbit insertion maneuvers were constrained to be all propulsive. Only minimal constraints were placed on the lunar orbits and no consideration was given to possible landing sites for lunar surface payloads. Similarly, no constraints were placed upon the orbit of return. Only minimum energy, two-impulse transfers were examined for all of the mission phases.

The various phases and maneuvers of the mission are discussed for both a conventional (Apollo type) and an unconventional mission profile. The velocity impulses needed, and the propellant masses required are presented for all of the mission maneuvers. Maximum payload capabilities were determined for both of the mission profiles examined. In addition, other issues relating to the feasibility of such lunar shuttle missions are discussed.

The results of the analysis indicate that the Shuttle orbiter would be a poor vehicle for payload delivery missions to lunar orbit. The maximum payload to a circular 100 km lunar orbit is only about 3.2 mt. This performance is particularly poor when it is noted that the initial mass in earth orbit is in excess of 846 mt. While the analysis indicates that the use of unconventional mission profiles can greatly improve the payload performance, the orbiter is still shown not to be a viable vehicle for payload delivery missions to the lunar vicinity.

Acronyms and Symbols

ASRM	advance solid rocket motor
EOI	earth orbit insertion
ET	external tank [Shuttle]
ETO	Earth-to-orbit
HELO	highly-elliptic lunar orbit
HLLV	heavy-lift launch vehicle
IMLEO	initial mass in low earth orbit
L1	cis-lunar libration point
LEO	low earth orbit
LH2	liquid hydrogen
LLO	low lunar orbit
LOI	lunar orbit insertion
LOX	liquid oxygen
LTS	lunar transportation system
MLI	multi-layer insulation
M_{bo}	mass allowance for cryogen boil-off (kg)
M_f	propellant mass required for maneuver (kg)
M_m	mission mass--vehicle+payload+remaining propellant (kg)
M_p	payload mass (kg)
mt	metric tonne (1,000 kg)
OET	orbiter-external tank combination
OMS	orbital maneuvering system
RCS	reaction control system
SOI	sphere of influence [gravitational]
SSF	Space Station Freedom
SSME	Space Shuttle main engine
STS	Space Transportations System [Shuttle]
TEI	trans-earth injection
TEM	trans-earth midcourse correction maneuver(s)
TLI	trans-lunar injection
TLM	trans-lunar midcourse correction maneuver(s)
TPS	thermal protection system [reentry]
ΔV	velocity impulse (m/sec)
ΔV_{res}	reserve or contingency impulse (m/sec)

Introduction

In support of the Space Exploration Initiative, it has been proposed that the Nation's Space Transportation System (STS) could be used in an unconventional manner to advance the date of manned activities in the lunar vicinity[1]. Specifically, it has been proposed that perhaps the Shuttle orbiter could be utilized, along with a refueled external tank (ET), to itself deliver payloads to lunar orbit.

In this first order analysis, a modified Apollo-type mission approach was used wherein the orbiter-external tank (OET) vehicle initiates a trans-lunar injection (TLI) propulsive burn from low earth orbit (LEO) to enter a cis-lunar trajectory. After the lunar encounter begins, and just prior to perilune, the OET vehicle performs a lunar orbit insertion (LOI) burn to leave the free-return trajectory and brake the OET into low lunar orbit (LLO). At this point, the lunar cargo/payload is off-loaded/deployed. Following completion of lunar orbit operations, the OET initiates a trans-earth injection (TEI) burn to leave lunar orbit and enter an Earth return trajectory. However, unlike Apollo, the OET must perform an additional final propulsive burn to slow the vehicle into LEO instead of direct [atmospheric] entry. Direct entry is not possible due to the heating limitations of the orbiter's thermal protection system (TPS). This profile is illustrated in figure 1.

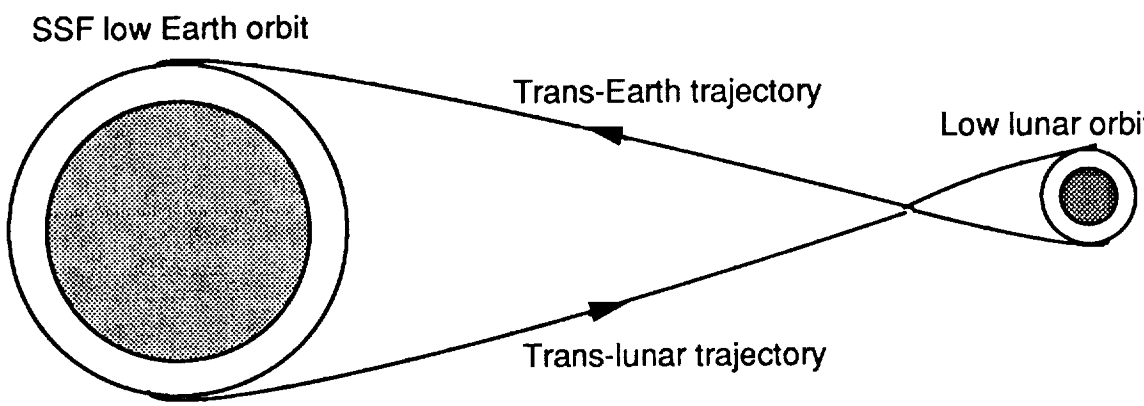

Figure 1. Lunar mission schematic. *The mission is similar to the Apollo approach except that upon return the spacecraft propulsively brakes into orbit instead of direct entry.*

Assumptions and Approach

To analyze the capabilities of the OET vehicle within this mission scenario, several assumptions were required to establish the necessary boundary conditions. First, the OET vehicle must be refueled in LEO. [This implies that the STS will carry the ET into orbit instead of jettisoning it prior to the orbital maneuvering system (OMS) circularization burn. It is assumed that sufficient OMS fuel could be carried for this purpose. The use of the advanced solid rocket motors (ASRM) should add enough additional capability for this purpose.] Refueling of the ET would most likely occur in the vicinity of Space Station Freedom (SSF) at a cryogen fuel depot. Therefore, the LEO of departure was assumed to be a circular orbit of 400 km altitude.

Second, minimum energy, 5-day Hohmann transfers were used to minimize the ΔV requirements. The TLI into the Hohmann trans-lunar trajectory was treated as a single burn, impulsive maneuver. In addition, a 50 m/sec midcourse capability was incorporated for the trans-lunar trajectory.

The third assumption was the lunar orbit selection, for which two approaches were examined. The first was a circular LLO at 100 km altitude. The second was a non-circular, highly-elliptic lunar orbit (HELO) with a perilune of 100 km and a period of 12-hours. The LOI into either the LLO or the HELO was treated as a single burn, impulsive maneuver at the point of closest approach during lunar encounter.

The earth-return transfer was treated as a single burn, impulsive maneuver from perilune. The TEI was calculated as a parabolic lunar escape. As with the trans-lunar trajectory, a 50 m/sec midcourse correction capability was incorporated for the trans-earth trajectory.

The final assumption necessary to establish the boundary conditions was the selection of the return LEO. An Apollo-style direct entry is not possible due to current orbiter TPS limitations. Since SSF's orbit regression and lunar orbital motion would make return to SSF impractical if not impossible (for a 14-day mission), the return LEO could be selected to optimize the ΔV requirements. Generally, propellant requirements to return to SSF or a nearby cryogen fuel depot would be excessive, and were thus not examined. For return to LEO, earth orbit insertion (EOI) was treated as a single burn, impulsive maneuver.

In addition to the assumptions necessary to establish the required boundary conditions, operations data was required regarding the Space Shuttle main engine (SSME) performance and the OET weight characteristics. Flight manifest weight data for OV-103, *Atlantis*, was used in all mass fraction calculations involving the OET vehicle.[2] This data is summarized in table 1. The specific impulse performance, Isp, of the SSME was taken to be the accepted value of 455 seconds.

Table 1. Mass data for OV-103, Atlantis. *These mass data were used as nominal values in all of the analyses.*

System/Component	Mass (kg)
Orbiter (inert)	76,070
External Tank (dry)	36,840
Personnel & Accommodations	1,660
Residual & Unusable Fluids	870
Reserve Fluids	1,070
In-flight Losses	1,670
Ascent Propulsion	2,240
Propellant--Usable RCS	3,170
Propellant--Usable OMS	10,340
Propellant--Usable ET	712,920
Totals	
Orbiter (wet)	97,090
Orbiter + ET (dry)	133,930
Orbiter + ET (wet)	846,850

Note: These data are for a particular shuttle mission--the masses of the systems change from time to time as systems are modified, and as new equipment is added. At this time, OV-103 is the lightest orbiter of the fleet; although the orbiter currently in construction is expected to be lighter.

Source: Rockwell International, Shuttle Orbiter Division; "Space Shuttle Orbiter Mass Properties Status Report," SD72-SH-0120-129, p. 7, Sept. 1989.

To address concerns of boil-off of the cryogenic propellants from the ET during a 14-day lunar mission, the following assumptions were incorporated. First, the foam insulation currently used on the exterior of the ET would be ineffective as an extended duration cryogen insulator. Therefore, it was assumed that part of the ET's foam insulation would be replaced with multi-layer insulation (MLI). However, some of the foam insulation would have to be retained as the MLI would have to be protected from aerodynamic loads during ascent to orbit. Weight changes due to this change is insulation were not accessed. Second, an estimate of 4 percent propellant boil-off per month was incorporated into the propellant mass calculations.

The method of patched conics was used to determine the ΔV's necessary for each of the velocity impulse maneuvers (TLI, LOI, TEI, and EOI). These patched conic calculations were based upon an Earth-Moon sphere of influence (SOI) transition point 318,100 km from the center of the Earth (66,300 km from the center of the Moon) as shown in figure 2. This corresponds to the average lunar distance (the distance varies slightly due to the eccentricity of the Moon's orbit about the Earth). Although this approach would be inappropriate for actual flight trajectory calculations, it is suitable for determining realistic estimates of ΔV requirements for Earth-Moon trajectories. Also, a 2 percent ΔV capability was added to all of the propulsive maneuvers to provide a contingency propellant reserve. This would provide enough extra propellant to perform an additional 2 percent of ΔV.

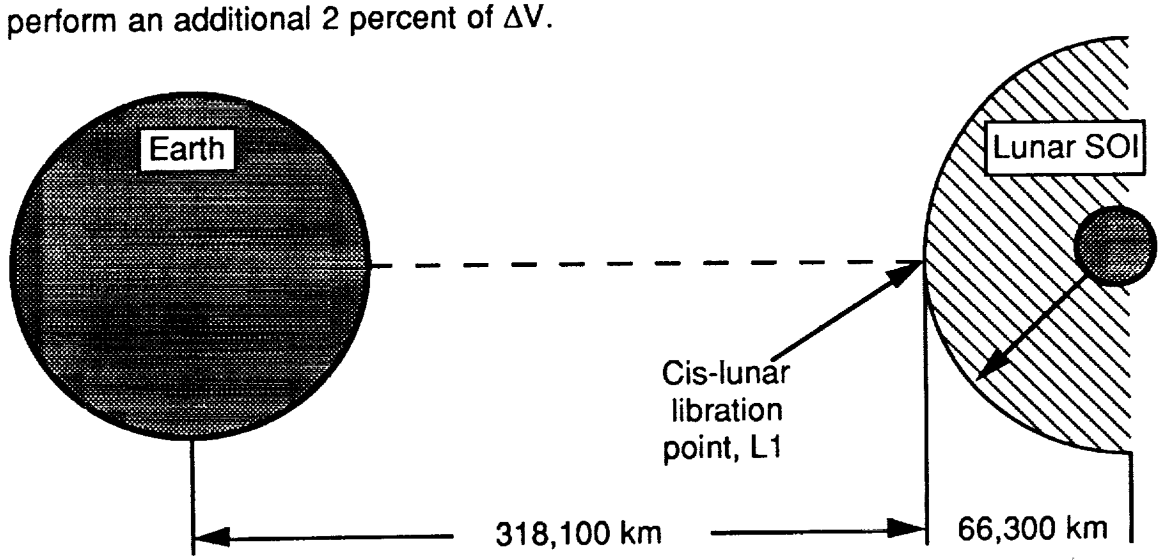

Figure 2. Earth-Lunar geometry. *The patched conics calculations are earth centered until the spacecraft encounters the lunar sphere of influence (LOI), at which time the calculations switch to moon centered.*

Results and Discussion

The analysis was conducted for two mission modes: the LLO mission mode and the HELO mission mode. The impulse (ΔV) calculations for the two modes are considered in parallel discussion, while the discussion of mission mass calculations is sequential.

Trans-Lunar Injection

Opportunities for a minimum-energy TLI occur whenever SSF's orbit plane intersects the future lunar position in the lunar orbit plane at the time of arrival (5 days for the Hohmann lunar transfer) as illustrated in figure 3. The Moon's orbital motion (~13°/day) and SSF's orbital regression (~7°/day) result in TLI windows which occur on the average every 9 days. However, half of these opportunities occur during lunar night; therefore, the usable opportunities occur about every 18 days. Each TLI window remains open for approximately 4 orbits of SSF.

The minimum energy Hohmann trajectory requires a velocity of 10,750 m/sec. Subtracting the vehicle's LEO velocity of 7,670 m/sec yields a required TLI burn impulse of 3,080 m/sec. Note that the use of this minimum energy trajectory generally precludes the use of a free-return abort trajectory. However, since the Orbiter, unlike Apollo, cannot make a direct atmospheric entry (due to TPS limitations), using a free-return abort due to propulsion system failure would be a moot point.

Lunar Orbit Insertion

While the use of minimum energy transfers generally precludes the use of lunar free-return, such trajectories permit unrestricted selection of lunar orbit inclination. Orientation for the lunar orbit occurs near the cis-lunar libration point, L1, via a small, midcourse correction maneuver . Since the relative velocity at this point in the trajectory is slow, the midcourse maneuver is small regardless of lunar orbit orientation. In addition, LOI requirements vary only slightly with the inclination of the lunar orbit. Thus, there is essentially no restriction on orbit inclination due to propulsive requirements. (Of course specific lunar landing sites greatly restrict the selection of orbit inclination.)

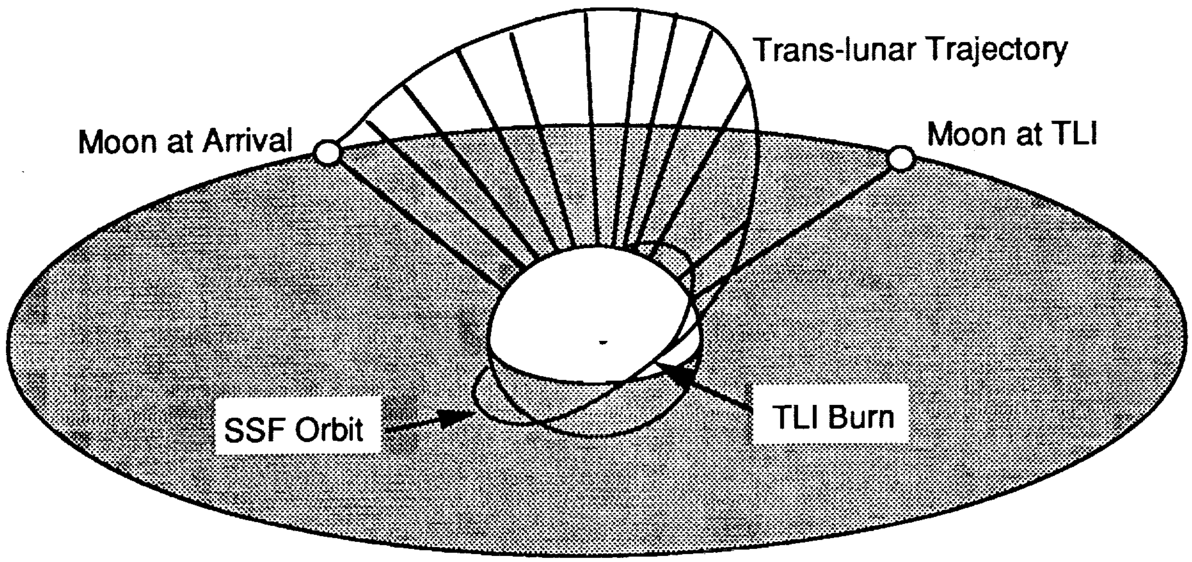

Figure 3. Trans-lunar Injection geometry. *The TLI geometry for a SSF departure requires that the orbital nodes of the SSF and Moon be properly aligned.*

Initially, a LLO approach similar to the Apollo lunar orbit rendezvous was examined to determine the LOI requirements. The orbital velocity for a circular LLO at an altitude of 100 km is about 1,630 m/sec. The hyperbolic approach for such a LLO would be targeted (via the midcourse maneuver) such that the closest approach would be 100 km altitude around the backside of the Moon at a velocity of approximately 2,490 m/sec. Therefore, the LOI burn would require an impulse of about 860 m/sec to brake into LLO.

A second approach was examined in effort to minimize the LOI and TEI propulsive requirements. In this approach, the spacecraft performs a similar hyperbolic approach maneuver such that the closest approach would again be 100 km altitude at a relative velocity of about 2,490 m/sec [around the backside of the Moon]. However, unlike the previous LLO scenario, a much smaller LOI burn impulse at the point of closest approach brakes the vehicle into a non-circular HELO with a 100 km altitude perilune.

In addition to greatly reducing the LOI propulsive requirement, the TEI impulse is likewise reduced due to the higher relative velocity at the orbit position where the TEI burn is initiated. Of course, both LOI and TEI are reduced for greater eccentricities. However, to be practical, the period of the HELO should be constrained to be no more that 12 hours and the stay time in lunar orbit must not exceed more than a few days to take maximum advantage of perilune positioning for TEI. While a 12-hour orbit might be considered at a disadvantage due to the communications blackout during lunar backside passage, this would be unfounded as the blackout time would actually be less for a HELO than a circular lunar orbit. This is because of the much higher relative velocity along the backside portion of the HELO. (Due to the very small lunar J2 gravitational potential [oblateness], orbital regression in not a concern for the short stay times envisioned here; and the lunar orbit will remain essentially inertial with respect to the Moon.)

A 12-hour HELO with a 100 km perilune has an orbital eccentricity of 0.70 and a apolune of 8,690 km. The required relative velocity at perilune is about 2,120 m/sec. Thus, the required LOI burn impulse is approximately 370 m/sec. This represents a 57 percent decrease in LOI propulsive requirements for the non-circular HELO approach versus the circular, LLO approach.

Trans-Earth Injection

As with TLI, opportunities for TEI occur whenever the lunar position in the lunar orbit plane intersects the future position of SSF's orbit plane at the time of arrival. This is the reverse of the situation illustrated in figure 3. The relative position of SSF in its orbit is not a factor as it completes each orbit in about 93 minutes. If minimum energy, 5-day trajectories are used each way, the proper alignment will occur on average after a 2-day stay in lunar orbit. The next such alignment will not occur (again on average) until 9 days later, or the 11th day in lunar orbit. In either case, the window will remain open for a TEI once around for 3 lunar orbits, but only once for a 12-hour HELO. Obviously, the orbital mechanics involved with returning to SSF greatly constrain mission scheduling and flexibility. For reasons discussed in the previous section, TEI is relatively unaffected by the inclination of the lunar orbit (or the desired inclination of the LEO).

The TEI requirements for both LLO's and HELO's were calculated as parabolic lunar escapes targeted on the cis-lunar libration point, L1. Parabolic escape from a 100 km LLO requires a relative velocity of 2,310 m/sec. Subtracting the relative orbital velocity of 1,630 m/sec yields the needed LLO TEI burn impulse of 680 m/sec. For injection from a 12-hour HELO, the TEI burn would be performed near perilune where the relative orbital velocity is about 2,120 m/sec. Therefore, the required HELO TEI burn impulse is about 190 m/sec. This represents a 72 percent decrease in TEI propulsive requirements for the non-circular HELO approach versus the circular, LLO approach.

Earth Orbit Insertion

As previously discussed, due to the TPS limitations the orbiter cannot make an Apollo-style direct entry upon Earth return. Therefore, an EOI maneuver is necessary to brake the spacecraft into the desired LEO. This is a serious handicap since the impulse needed is about the same as for TLI!

The minimum energy, 5-day return trajectory will result in the minimum EOI propulsive requirement. For such a return, the velocity at the targeted closest approach (400 km altitude for a return to a SSF orbit) would be about 10,790 m/sec. Thus, the required EOI burn impulse is 10,790 m/sec minus the 7,670 m/sec orbital velocity of a 400 km circular LEO, or approximately 3,120 m/sec.

Total Mission ΔV Requirements

Actual total mission ΔV requirements can vary substantially with the lunar geometry at time of launch. However, the data presented here represent a good estimate of the needed impulse capability for 14-day missions of both the LLO and HELO profiles. The LLO mission profile is, of course, the more expensive of the two due to the OET positioning much lower into the lunar gravity well. Such a 100 km LLO mission requires a total ΔV capability of 7,840 m/sec plus a 155 m/sec reserve. The HELO mission profile needs 6,860 m/sec plus a 136 m/sec reserve. Thus, the overall impulse savings for a HELO approach is about 12.5 percent. The ΔV's for each of the maneuvers and the totals are summarized in table 2.

Table 2. Summarized mission ΔV data. *Totals of the velocity impulses necessary for both the LLO and HELO missions and the reserve requirements for a 2 percent contingency per burn.*

Maneuver	ΔV (m/sec)	ΔV_{res} (m/sec)
TLI	3,080	62
TLM	50	---
LOI (LLO)	860	17
LOI (HELO)	370	8
TEI (LLO)	680	14
TEI (HELO)	190	4
TEM	50	---
EOI	3,120	62
Total LLO Mode	7,840	155
Total HELO mode	6,860	136

Propellant and Payload Mass Calculations

Mass calculations are obtained by use of the well known rocket equation which relates the vehicle propellant fraction and the specific impulse to the velocity impulse. For an unknown payload mass, the calculations are iterative in nature and are backed-out through each phase of the mission. Boil-off and reserve allowances are added before each of the maneuver calculations. The LLO and HELO mission modes are considered separately.

LLO Mission Results. For missions to a 100 km LLO, we see from table 2 that the final maneuver, EOI, requires a ΔV of 3,120 m/sec plus a 62 m/sec reserve. Thus, the total EOI requirement is about 3,182 m/sec. Assuming that no ET fuel is needed after EOI the final post-burn mass of the OET spacecraft is about 133,930 kg. If we make allowance for a 1,000 kg return contingency mass, then we have a final post-burn EOI mass of 134,930 kg. For this mass the rocket equation indicates that an EOI mass fraction of 0.4902 is needed, or 140,310 kg of LOX/LH2 propellant. Therefore, the pre-burn EOI mass is 275,240 kg.

The next previous phase of the mission is the trans-earth midcourse (TEM) correction maneuver. However, since this maneuver is relatively small, it can be conservatively considered by simply incorporating it into the TEI calculations. For the TEI maneuver the post-burn mass is the pre-burn EOI mass plus the allowance for propellant boil-off during the trans-earth trajectory. Using our estimate of 4 percent boil-off per month, this yields an allowance of 950 kg/day. Thus, the 5-day return requires an allowance of 4,750 kg. Therefore, the post-burn TEI mass is about 279,990 kg. The required TEI impulse is 680 m/sec plus a 14 m/sec reserve plus the 50 m/sec midcourse correction, or 744 m/sec. This yields a required TEI mass fraction of 0.8465, or 50,790 kg of LOX/LH2 propellant. The pre-burn TEI mass then becomes about 330,780 kg.

At this point the calculations are approached from the beginning, or TLI, and they proceed to LLO. The total impulse for this calculation is about 4,069 m/sec, including reserves. Starting with the initial mass of the fully-fueled OET plus payload, the TLI maneuver (including reserves) requires a ΔV of 3,142 m/sec and the resulting mass fraction is 0.4946. The process of determining the maximum payload mass now becomes iterative so we assume an initial payload mass and proceed with the calculations. (Subsequent iteration indicates a payload capability of 3,200 kg so the following calculations will use that value.) Thus, using a payload mass of 3,200 kg gives a resulting pre-burn TLI mass of 850,050 kg. From these data, the needed TLI propellant mass is about 429,610 kg of LOX/LH2 and the post-burn TLI mass is about 420,440 kg.

The next maneuver is the trans-lunar midcourse (TLM) correction, but again, this may be combined with the LOI maneuver for convenience. The LOI maneuver [including reserves and midcourse allowance] requires a ΔV of about 927 m/sec. The pre-burn LOI mass is adjusted from the post-burn TLI mass to account for the Δmass due to propellant boil-ff (4,750 kg) during the 5-day trans-lunar trajectory. Therefore, the pre-burn LOI mass is about 415,690 kg. The LOI maneuver requires a mass fraction of 0.8125 which works out to 77,940 kg of LOX/LH2 propellant and yields a post-burn LOI mass of about 337,750 kg.

Additional consideration for propellant boil-off must be given for the LLO stay-time. For a nominal 14-day mission, or 4 days in LLO, this boil-off allowance is 3,800 kg. Therefore, after subtracting the LLO boil-off allowance, the mass in LLO including the payload is roughly 333,950 kg. Since the *required* pre-burn TEI mass is about 330,780 kg, the result is approximately the pre-burn TEI mass plus the payload mass. A breakdown of the mission, propellant, and payload masses by maneuver is summarized in table 3.

HELO Mission Results. The payload and propellant masses for a HELO mission are determined just as in the case of the LLO mission. The only difference being a significant reduction in the ΔV's of the LLO and TEI maneuvers. Since the EOI maneuver is the same as that for a LLO mission, the calculations need not be repeated. Therefore, the post-burn TEI mass is, again, 279,990 kg.

The required TEI impulse is now 190 m/sec plus a 4 m/sec reserve. But, we must also add the 50 m/sec midcourse correction allowance for a total ΔV of 244 m/sec. This yields a required TEI mass fraction of 0.9468, or 15,730 kg of LOX/LH2 propellant. Therefore, the pre-burn TEI mass is about 295,720 kg.

Approaching the calculations now from the beginning, or TLI, this maneuver needs, again, 3,080 m/sec plus a 62 m/sec reserve, or 3,142 m/sec. (As before, subsequent iteration yields a payload capability of 141,670 kg so this number will be used in the following calculations.) Starting with the initial mass of the fully-fueled OET plus a payload of 141,670 kg, we have a total initial mission mass of 988,520 kg. The TLI mass fraction is, again, 0.4946. However, with the increased mission mass of the TLI maneuver requires about 499,600 kg of LOX/LH2 propellant which results in a post-burn TLI mass of around 488,920 kg.

The LOI maneuver for the HELO mission requires a ΔV of 370 m/sec plus a 8 m/sec reserve. Adding the midcourse correction allowance of 50 m/sec yields a total impulse of 428 m/sec. Adjusting the post-burn TLI mass to account for propellant boil-off (4,750 kg) during the trans-lunar trajectory results in a pre-burn LOI mass of 484,170 kg. The LOI maneuver needs a mass fraction of 0.9086, or about 44,250 kg of LOX/LH2 propellant. The resulting post-burn LOI mass is then approximately 439,920 kg.

Considering a propellant boil-off allowance of 3,800 kg during the 4-day stay in HELO leaves a mission mass of 436,120 kg. Since the *required* pre-burn TEI mass is 295,720 kg, this is approximately the pre-burn TEI mass plus the payload. Again, the masses for a HELO mission are summarized in table 3.

Table 3. Mission, propellant, and payload masses. *Summary of the approximate post-burn mission and propellant masses after each maneuver of the LLO and HELO type missions.*

LLO Mission	M_m (kg)	M_f (kg)	M_{bo} (kg)	M_p (kg)
TLI	420,440	429,610	4,750	3,200
LOI	337,750	77,940	3,800	3,200
TEI	279,990	50,790	4,750	1,000
EOI	134,930	140,310	N/A	1,000
HELO Mission	**M_m (kg)**	**M_f (kg)**	**M_{bo} (kg)**	**M_p (kg)**
TLI	488,920	499,600	4,750	141,670
LOI	439,920	44,250	3,800	141,670
TEI	279,990	15,730	4,750	1,000
EOI	134,930	190,310	N/A	1,000

Note: The mission mass, M_m, includes all of the post-burn mass including payload and propellant for upcoming burns.

Comparison of the LLO and HELO Profiles

While these calculations for the HELO mission mode indicate a maximum payload capability of about 141,670 kg, This number cannot be directly compared with the LLO mission payload capability of 3,200 kg. The reason is that the HELO mission mode examined thus far only delivers the payload to a 12-hour HELO. For a meaningful comparison we need to allow for the transfer of the payload from the HELO to a 100 km LLO.

Transfer from a 12-hour HELO to a 100 km LLO can be accomplished with a single circularization burn at perilune. The required impulse is the difference between the 2,126 m/sec velocity at perilune and the 1,630 m/sec velocity of a

100 km circular LLO, or 496 m/sec. Now if we consider that part of the payload is a LLO kick-stage, we can determine an equivalent payload to LLO. Assuming that the kick-stage is a storable solid propellant module having a specific impulse of 350 seconds yields a required propellant mass of about 20,000 kg. Assuming a structural fraction of 20 percent for the kick-stage and payload assembly yields a structural weight of about 29,000 kg. Therefore, the mass remaining, 92,000 kg, is the maximum payload to LLO using a HELO mission mode with payload transfer to LLO.

The leverage of a HELO profile over a circular LLO profile is illustrated in figure 4. Obviously a HELO mission approach is advantageous to the LLO mission approach from the standpoint of maximizing delivered payload (92 mt versus 3.2 mt). However, a note of caution is in order since the effect is magnified here due to the extremely large mass of the OET which must be "moved" in and out of lunar orbit. In other words, the HELO mode is much more efficient in this case because it alleviates the need to lower the OET vehicle deeper into the lunar gravity well. The effect would not be as great for a more "conventional" LTS system.

The eccentricity of the elliptic orbit has a great magnifying effect upon the maximum payload capability. For the same perilune altitude (100 km), going from a circular orbit (zero eccentricity) to a mildly elliptic orbit with an eccentricity of 0.1 (100 x 508 km orbit) increases the payload capability by a factor of 6 to nearly 20,300 kg, well in excess of the maximum STS payload capability (about 17 mt). The relationship between payload capability and orbital eccentricity is nearly linear, as illustrated in figure 5. Note that the relationship of payload to orbital period is logarithmic in nature. Also note that a "full" STS payload could be carried to a 100 km perilune HELO with an eccentricity of 0.07 (100 x 214 km orbit).

The maximum payload to a circular LLO can also be greatly increased by allowing higher altitude orbits (which has the same effect of increasing the orbital period). For example, just increasing the orbit altitude to 200 km would more than double the payload capability, increasing it to 7,660 kg. The relationship between circular lunar orbit altitude and payload capability is shown in figure 6. Note that a "full" STS payload (about 17 mt) could be carried to a circular LLO of about 480 km altitude.

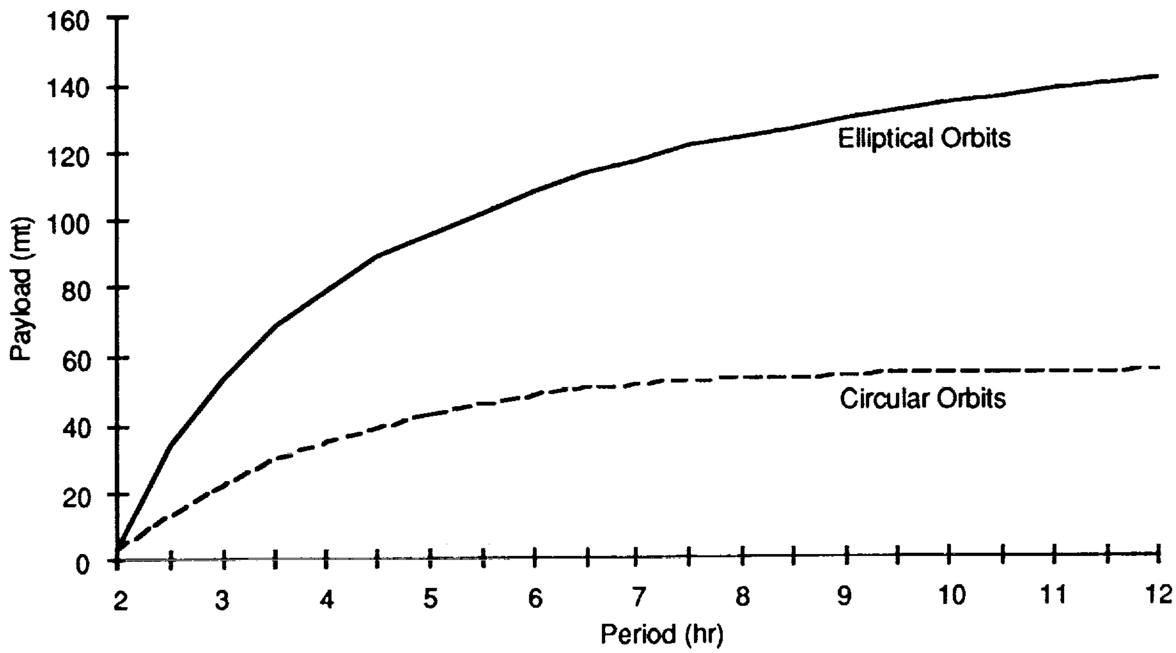

Figure 4. HELO and circular lunar orbit payload capability. The effect of orbital period on the payload capability is logarithmic for both the elliptic and circular profiles. The elliptic orbits all have the same 100 km perilune.

Obviously, a HELO mission profile would adversely effect the propulsive requirements for a lunar surface lander. When deployed from a HELO, such a lander would need additional impulse capability (in addition to what would be required from LLO) to slow to the velocity of a circular LLO. And, if the landing mission were manned or recoverable, this additional capability would have to be essentially doubled.

While the impact of a HELO mission profile on lander requirements is negative, the overall mission impact--even for a landing mission--is positive and the payload--orbital or lunar surface--is maximized. The explanation for this apparent contradiction is that for a HELO mission, only a transfer/lander vehicle and its payload need be "carried" through the extra ΔV requirements between the HELO and the LLO. While for a LLO mission profile, the entire OET vehicle with its TEI and EOI propellant, plus transfer/lander vehicle and payload must be "carried" through this extra ΔV necessary for a circular LLO. Therefore, it is much more efficient to deploy just the lander and payload "deeper" into the lunar gravity well.

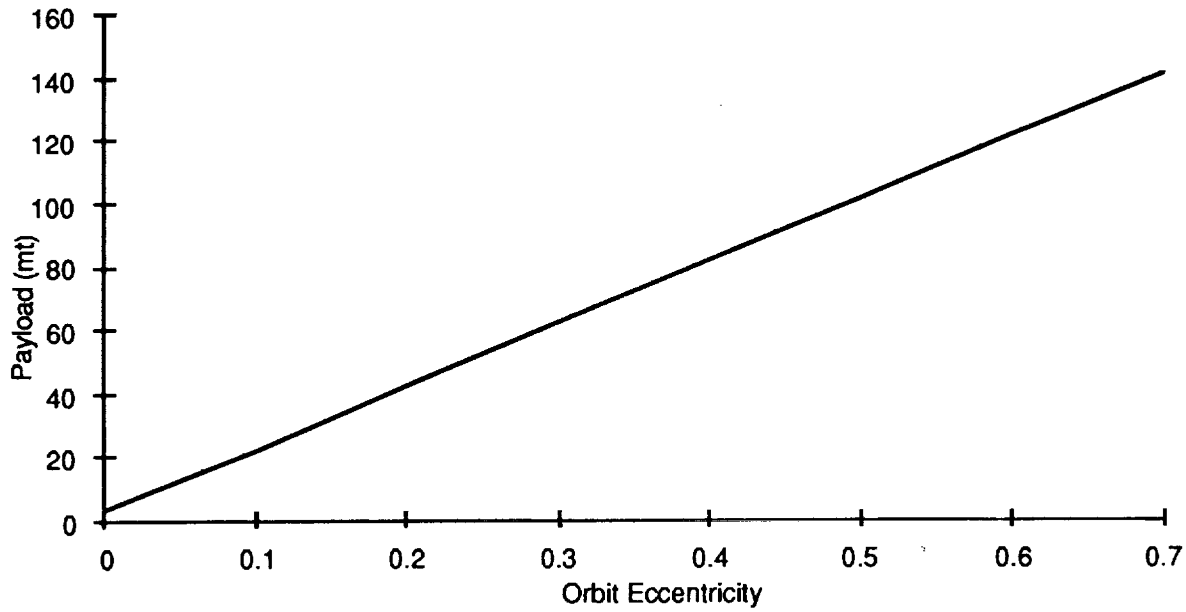

Figure 5. Effect of HELO eccentricity on payload capability. *The effect of increasing the eccentricity of the lunar orbit is to linearly increase payload capability for the same 100 km perilune altitude.*

Refueling the ET in LEO

While the STS can launch the orbiter, its payload, and probably its ET (if we assume ASRM's) into LEO, the ET must then be refueled with over 712 mt of LOX and LH2 propellant. If we assume that a Shuttle-C or similar type heavy-lift launch vehicle (HLLV) is available to perform refueling tanker missions we can assess the earth-to-orbit (ETO) requirements to support an OET lunar mission.

Studies indicate that a side-mount Shuttle-C vehicle in a tanker configuration could launch about 71 mt to a 400 km LEO[3]. Based on this capability it would take 10 launches of a Shuttle-C tanker to completely refuel the ET. However, if we assume that these launches could be conducted at a rate of 2 per month, an additional launch will be necessary to replenish propellant which would boil off during the interim. This does not address the hardware which would be needed (nor the technology which would have to be developed) to transfer the cryogen propellants from the tanker to the ET in a zero-G environment. Also, the infrastructure necessary to support a LEO cryogen fuel depot was not addressed.

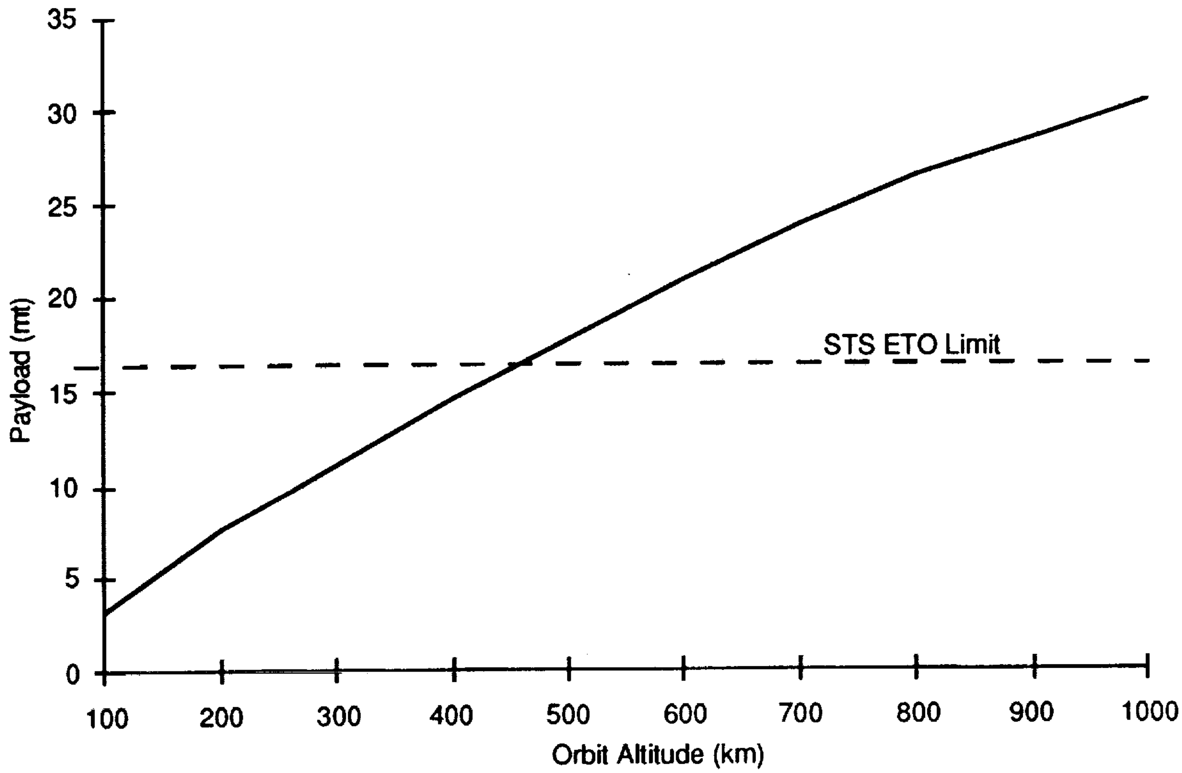

Figure 6. Effect of circular lunar orbit altitude on payload capability. *Increasing the altitude of the lunar orbit increases the payload in a nearly linear fashion.*

We can attempt to reduce the refueling requirement by reducing the payload mass. However, even for a 12-hour HELO mission profile, reducing the payload to zero would eliminate the need for only 1 Shuttle-C tanker flight.

This illustrates the real problem with using the shuttle orbiter for lunar flight: the overall mass of the vehicle that must be returned propulsively to LEO simply overburdens the system. The mass of the dry orbiter alone is in excess of 130 mt and the ET, which cannot be jettisoned prior to the completion of all propulsive maneuvers, has a mass of more than 36 mt. Thus, the vehicle would be a massive, single-stage LTS which, at LEO departure, would be in excess of 846 mt.

Other Concerns

Other issues which could impact the feasibility of using the shuttle orbiter for lunar flight were identified although they were not considered in the analysis. For example, it was assumed that all payloads would be integrated into the orbiter's payload bay on the ground a priori. This assumption effectively restricts the lunar payload mass to the maximum earth to orbit (ETO) payload capability of the STS--about 17 mt. In addition, this does not include payload integration provisions which would further reduce payload mass by at least 1 mt.

Another issue which will adversely effect payload capability is weight growth due to orbiter modifications necessary for lunar flight. For example, the following modifications would have to be addressed: i) Upgrade of the SSME's for essentially double their current operating time between inspection and refurbishment; ii) Modification of the SSME's to enable vacuum start with multiple restart capability; iii) Modification of the ET propellant handling plumbing to enable zero-G SSME startup; iv) Installation of MLI on the ET to reduce propellant boiloff; v) Increase in OMS propellant capacity needed to allow the ET to be carried into orbit; and vi) Installation of crew radiation protection and additional consumables for extended duration missions. Certainly there are other items not mentioned here which could also push the vehicle's weight and subsequently reduce the payload capability (not only the lunar payload, but also the ETO capability).

In addition to the above issues, such a lunar shuttle system would require an additional infrastructure to support the LEO cryogen fuel depot orbital node. This infrastructure would have to support 11 refueling flights per lunar mission which would entail autonomous rendezvous and docking, propellant transfer, and orbital debris protection.

Conclusions

The largest factor leading to the conclusion that the shuttle orbiter would be grossly inefficient as a LTS is the initial mass in LEO (IMLEO) requirements of such a system. While the STS can place the orbiter, its payload, and probably its ET into orbit, the then empty ET must be refueled with over 712 mt of LOX and LH2 propellants. This yields a total IMLEO of 846 mt. Therefore, the ratio of IMLEO to payload delivered to LLO is about 50. (Assuming a "full" STS payload of 17 mt and a 100 x 214 km LLO.) Obviously, from an economic standpoint such a large IMLEO to payload ratio makes the orbiter very unattractive as a LTS.

From a technical standpoint, delivering and transfering 712 mt of cryogenic propellants to the orbiting ET would require a greater ETO capability than could be obtained with a Shuttle-C system. As shown, use of a Shuttle-C type system would require a flight rate of about 2 per month just to refuel the ET. Such a flight rate is obviously not practical. Even the largest of the HLLV concepts being considered for support of future Mars missions do not exceed 250 mt lift capability--requiring a flight rate of 1 per 45 days just to support OET lunar missions.

HELO's can be used to greatly increase the payload capability. However, the payload capability for moderately- to highly-elliptic lunar orbits quickly exceeds the ETO payload capability of the STS. While a Shuttle-C or a HLLV could be used to bring these large payloads to the orbiter in LEO, the payload bay volume envelope would be exhausted. Also, the orbital operations required to transfer payloads to the orbiter would be difficult and time consuming. It is probably unrealistic to expect the orbiter to be able to accommodate payloads much over 20 mt. Nevertheless, the results here illustrate the performance pay-off of HELO's for a bimodal LTS.

Because the transfer system (the orbiter plus ET) is such a massive vehicle, the system favors extremely large payload masses to be anywhere near efficient. To approach the efficiency of a bimodal LTS, the LLO payload capability would have to be near 150 mt. Therefore, such efficiencies cannot be realistically obtained since a 150 mt payload could only be delivered to a HELO with a period in excess of 12-hours. Use of such an exaggerated HELO would require a sizable transfer stage to transfer the payload to a more realistic LLO.

Other issues, unrelated to performance [directly], could seriously constrain a LTS based on the OET. Obviously, significant advancement in zero-G cryogen handling technology would be required for on-orbit refueling of the ET. In addition, modifications to an OET to support lunar flight would be extensive. Probably the most extensive of these would be modification and certification of the SSME's for double their current operating time between inspection, multiple restart capability, and zero-G start ability.

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15. Supplementary Notes

16. Abstract

A first order orbital mechanics analysis has been conducted to examine the possibility of utilizing the Space Shuttle Orbiter to perform payload delivery missions to lunar orbit. In the analysis, the earth orbit of departure was constrained to be that of Space Station Freedom. Furthermore, no enhancements of the Orbiter's thermal protection system were assumed. Therefore, earth orbit insertion maneuvers were constrained to be all propulsive. Only minimal constraints were placed on the lunar orbits and no consideration was given to possible landing sites for lunar surface payloads. The various phases and maneuvers of the mission are discussed for both a conventional (Apollo type) and an unconventional mission profile. The velocity impulses needed, and the propellant masses required are presented for all of the mission maneuvers. Maximum payload capabilities were determined for both of the mission profiles examined. In addition, other issues relating to the feasibility of such lunar shuttle missions are discussed. The results of the analysis indicate that the Shuttle Orbiter would be a poor vehicle for payload delivery missions to lunar orbit.

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