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JAN.-FEB. 1965 J. SPACECRAFT VOL. 2, NO. 1

Spacecraft Propulsion Requirements for Lunar Missions

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This paper considers spacecraft propulsion requirements for lunar orbiting and landing missions, including requirements imposed by flight path, accelerometer errors, altitude determination errors, and impulse errors, with particular attention to minimizing the number of maneuvers, number of engines, and throttling requirements. It appears that a single fixed-thrust engine will satisfy the requirements for an orbiting mission. For landing from lunar orbit, a single regeneratively cooled engine with a throttling range as low as 3.5:1 appears acceptable; alternatively, if the system has a separate landing engine, a throttling range as low as 2:1 appears to be acceptable.

Nomenclature

g = gravitational acceleration at earth's surface
g_l = gravitational acceleration at lunar surface
H_h = hover altitude
H_0 = lunar orbit altitude
I_{sp} = specific impulse
M_0 = initial total mass of spacecraft
M_{pay} = payload mass
R_T = throttling ratio, T_{max}/T_{min}
T = thrust
T_{min} = minimum thrust
T_{max} = maximum thrust
V_∞ = hyperbolic excess velocity relative to the moon
W_{ll} = spacecraft lunar weight at landing
ΔV = characteristic or effective velocity increment
ΔV_0 = impulsive velocity increment

Introduction

All lunar missions are likely to involve midcourse corrections, and they may require various other maneuvers for entry into lunar orbit, landing, and return to earth. Questions of reliability (e.g., redundancy of engines and start-stop requirements) also enter, and for manned missions, abort (return to earth) capabilities must be included. Total impulse requirement and mass of the spacecraft propulsion system vary with all of these factors. This paper attempts to collate and summarize the various maneuver requirements, to assess their effect on the propulsion system design, and to determine the effect in terms of over-all system performance of minimizing propulsion-system requirements.

Presented as Preprint 63-76 at the IAS Annual Meeting, New York, January 21–23, 1963. This paper presents the results of one phase of research carried out under NASA Contract No. NAS 7-100. The author wishes to acknowledge that the data for Fig. 7 were developed by Rocketdyne under contract NAS 5-916 to NASA; the data for Figs. 10 and 12 are from the analyses of A. V. Zimmerman.[1] A. D. Harper and J. R. Wrobel of the Jet Propulsion Laboratory also performed analyses that have been used in the preparation of this paper.

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Maneuver Requirements

Midcourse Corrections

For either trans-earth or trans-lunar phases of flight, one to five midcourse corrections (totaling 100 to 500 fps) may be required. Since these maneuvers are relatively small, efficiency is usually secondary to reliability. Acceleration has little effect on efficiency (due to the small gravity gradient involved), but it may significantly affect maneuver accuracy. The minimum acceptable thrust is often determined by the null offset error of an integrating accelerometer used for shutoff control. Maximum thrust levels are often determined by the uncertainty in the impulse delivered after the shutdown command caused by uncertainties in propellant-valve response time and in the shutdown thrust transient. A fairly large range of accelerations can be used with relatively small impulse-magnitude errors. For example, if a 20-fps trajectory correction were required in a situation where the allowable error is $\pm 2\%$, the accelerometer null offset error is 0.0005 g, and the shutoff impulse uncertainty is equivalent to ± 0.05 sec of full-thrust operation, the range of usable accelerations would be 0.025 to 0.25 g. Subsequent discussion will show that the acceptable range of acceleration is important, because for most applications it is desirable to perform trajectory corrections with a propulsion system sized for additional spacecraft maneuvers.

Direct Landing from Trans-Lunar Trajectory

A direct landing maneuver can be considered in two parts: the main descent to a hovering position, and the subsequent soft landing on the lunar surface. The curves in Fig. 1 show the variations of hyperbolic excess velocity V_∞ with transit time for arrival at lunar apogee and arrival at lunar perigee. The velocity V_∞ would be the velocity of a vehicle relative to the moon at lunar impact if the moon were without gravitation.

Also shown on Fig. 1 is the ideal (minimum) velocity increment ΔV_0 required for a direct lunar landing. The parameter ΔV_0 is the velocity increment that would have to be applied by the propulsion system if the maneuver were made in an infinitesimally short time. The magnitude of ΔV_0 is independent of the angle between the spacecraft velocity vector and the lunar surface.

The variation of characteristic velocity increment ΔV with thrust-to-initial-mass ratio $T_{\max}/M_0$ for a vertical descent to a hover altitude is shown in Fig. 2. The parameter ΔV is the velocity increment, which could be provided by burning the same quantity of propellant during linear flight in a gravity-free, vacuum environment.

The throttling ratio $R_T = T_{\max}/T_{\min}$ required for a single engine with a minimum thrust equal to 75% of the spacecraft lunar weight at landing W_{ll} is shown in Fig. 2. This value of $T_{\min}/W_{ll}$ represents the largest value of minimum thrust that will allow a continuous landing maneuver to be performed efficiently. Figures 2a and 2b demonstrate that very high throttling ratios are required for a single engine if large penalties in ΔV are to be avoided; otherwise, vertical and near-vertical direct-landing trajectories require multiple engine arrangements.

The ignition altitude is shown as a function of thrust-to-initial-mass ratio in Fig. 2. If initial-altitude-measurement errors are assumed proportional to the altitude, then burnout altitude errors will be smaller when larger values of $T_{\max}/M_0$ are used. (This effect could, of course, be reduced by making corrections during descent.)

For some landings, hovering may be required for landing site selection or for final corrections. An idealized analysis was performed to show the net increase in ΔV_0 as a function of hover altitude (see Fig. 3). As the hover altitude is increased, the required ΔV_0 increases for the final landing maneuver and decreases for the initial descent maneuver. In general, a descent will require many minimum and maximum thrust periods to correct for errors in altitude and velocity; thus, many rapid starts and shutdowns would be required if on-off operation were used. When the system has continuously variable thrust, the descent from hover can be made by using one period of minimum thrust followed by a period of maximum thrust to bring the spacecraft to some small vertical velocity at which point the thrust could be adjusted for the final soft landing. Figure 4 shows the variation of ΔV with H_h (hover altitude), R_T, and $T_{\min}/W_{ll}$, and Fig. 5 shows the variation of ΔV with R_T (at optimum $T_{\min}/W_{ll}$) for various hover altitudes. For $H_h = 1000$ ft, the ΔV penalty for using $R_T = 2$ rather than 7 is approximately 100 fps.

For a manned mission, abort from any phase of a lunar mission is an extremely important consideration. For some lunar missions, the propulsion stage designed for the return phase of the mission can also provide abort capability during the initial phases of the mission. Aborts near the beginning of a vertical descent can require up to twice the capability required for a normal launch and return. For landings in which the trans-lunar trajectory makes an angle with the lunar surface of approximately 45° or less, the horizontal component of spacecraft velocity during descent is greater

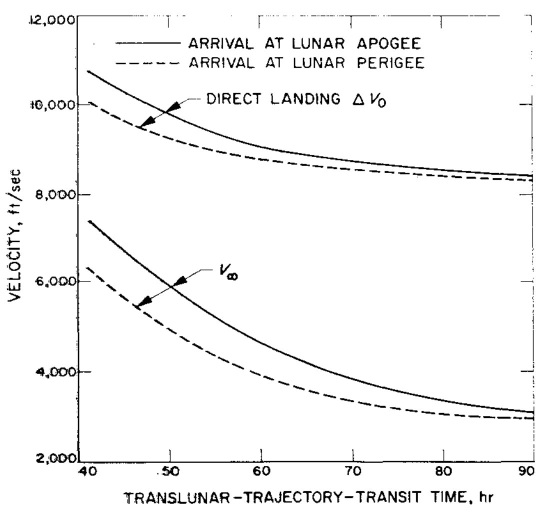

Fig. 1 Hyperbolic-excess velocity and landing velocity increment as a function of transit time.

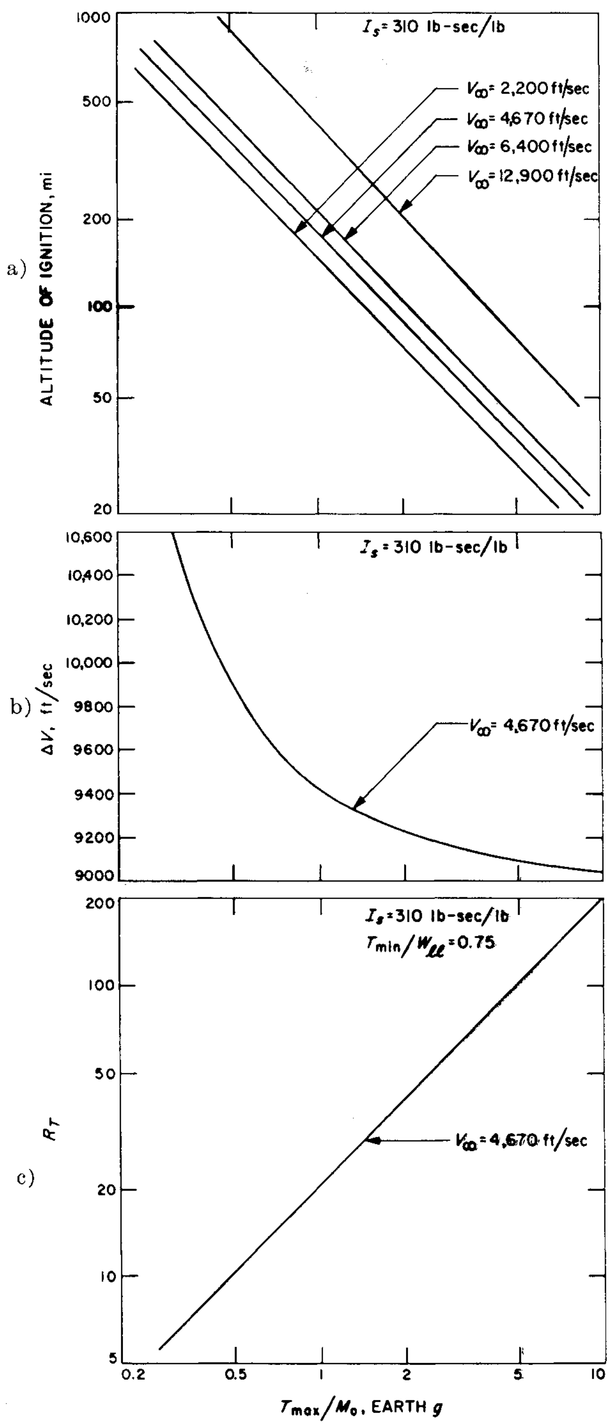

Fig. 2 Characteristic velocity increment, throttling ratio, and altitude of ignition for a vertical lunar landing.

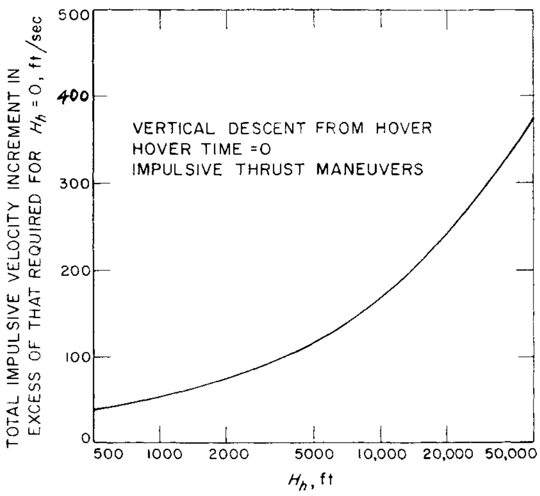

Fig. 3 Increase in characteristic velocity increment as a function of hover altitude.

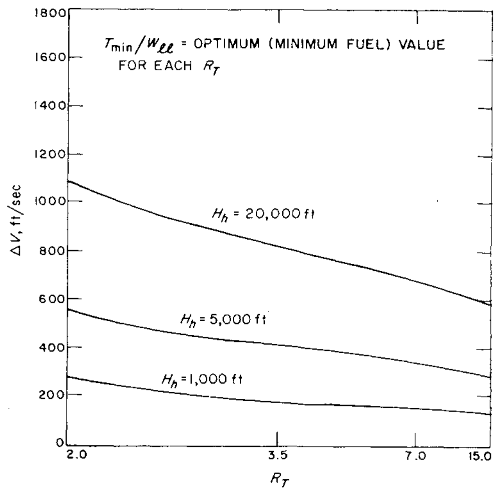

Fig. 5 Characteristic velocity increment for optimum descent from hover.

than the vertical component, so that the spacecraft possibly could be maneuvered to a lunar orbit, and, subsequently, be returned to earth using the propellant normally carried for return to earth from the lunar surface. Since the majority of landing sites on the visible lunar surface results in an angle between the trans-lunar trajectory and the lunar surface of more than 45°, it is concluded that, in general, direct lunar landing is not attractive for manned missions. However, the simplicity of this type of landing makes it very attractive for unmanned payloads, because only one descent maneuver is required and because the entire landing can be made with line-of-sight communications to earth.

Direct Return to Earth from the Lunar Surface

The propulsion requirements for direct return to earth are very similar to the requirements for direct landing, but they do not include requirements for variable thrust. The dependence of ΔV on the thrust-to-mass ratio for a vertical ascent is essentially the same as for the vertical descent shown in Fig. 2.

Entry into Lunar Orbit from Trans-Lunar Trajectory, or Return to Earth from Lunar Orbit

Figure 6 shows impulsive velocity increments ΔV_0 for entry into or exit from coplanar lunar orbits as a function of hyperbolic-excess velocity V_∞. Figure 7 shows ΔV as a function of thrust-to-initial-mass ratio T/M_0 for entry into circular orbits of various altitudes. (The curves for exit from lunar orbit are nearly identical to these.) Note that for a T/M_0 as low as 0.1 g, the ΔV penalty is only 85 fps. The accuracy of the maneuver for entry into lunar orbit will be affected by accelerometer errors much as are the trajectory correction maneuvers. If the accelerometer null error is ± 0.0005 earth g, and the average acceleration during the orbit entry maneuver is 0.12 earth g, the error for 3500 fps would be ± 1.5 fps, which would result in an eccentricity error of approximately 10 miles for a nominally circular orbit. The thrust-level requirement to enter or leave lunar orbit, therefore, appears to be compatible with the requirement for

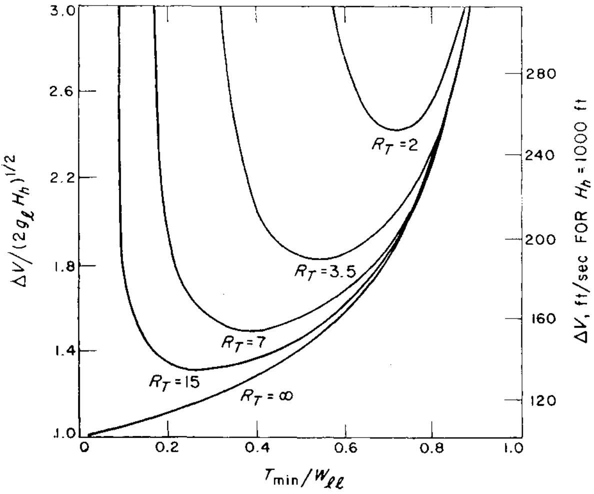

Fig. 4 Characteristic velocity increments for descent from hover to lunar surface.

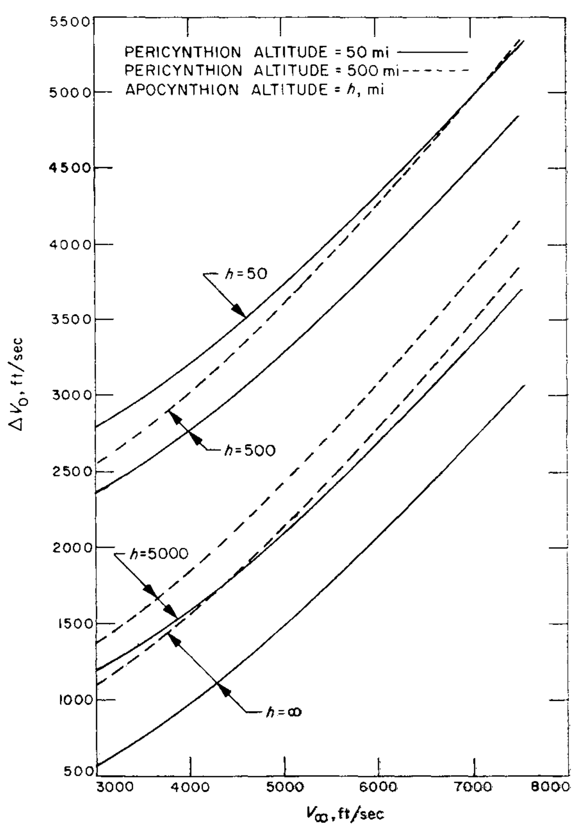

Fig. 6 Impulsive velocity increments required for entry into or exit from lunar orbits.

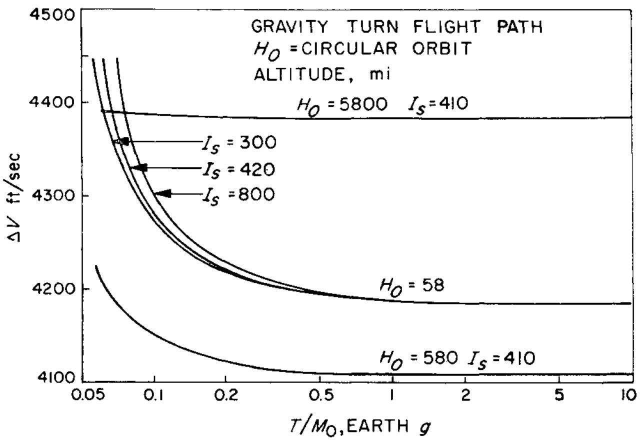

Fig. 7 Characteristic velocity increment requirements for entry into lunar orbit.

midcourse corrections and may allow use of the same propulsion system for both.

Landing from Lunar Orbit

In comparison with the direct descents, landing from lunar orbit is attractive, because propulsion systems with much lower thrust-to-mass ratios and throttling ratios can be used and because it allows abort during descent. For manned landings from lunar orbit, a stage sized for ascent from the lunar surface has sufficient capability to abort from any phase of the descent. Figure 8 shows the variation of total ΔV_0 for going from a trans-lunar trajectory into lunar orbit and descending to the surface as a function of hyperbolic-excess velocity for various lunar orbit altitudes H_0. The impulsive ΔV_0 requirements for a direct lunar landing are smaller than for landing from a lunar orbit with an altitude greater than zero. However, the advantages of the direct landing (H_0 = 0 in Fig. 8) in terms of impulse requirements are lost when the effects of finite thrust levels are considered. Characteristic velocity increments for landings employing descent from a 58-mile lunar orbit are lower than required for direct landing when the thrust-to-initial-mass ratio is below approximately 4 earth g's.

Figure 9 shows the four following trajectories for descent from lunar orbit.[1]

1) *Continuous constant-thrust descent.* Ignition of the propulsion system occurs in lunar orbit, and a descent to a hover altitude is effected utilizing continuous constant thrust and an optimum flight path (determined by a calculus of variations technique).

2) *Continuous variable-thrust descent.* Ignition occurs in lunar orbit. The descent (following an optimum flight path) is initially made at minimum thrust; at an optimum point during descent, the throttle is advanced to full thrust for the remainder of the descent to hover altitude.

3) *Hohmann-ellipse descent.* The propulsion system operates briefly to transfer the spacecraft from the initial lunar orbit to an elliptical orbit with a pericynthion of 50,000 ft. The initial maneuver occurs on the opposite side of the moon from the landing site. As the spacecraft nears the pericynthion of the ellipse, the propulsion system is restarted, and the descent is made to hover altitude following an optimum flight path.

4) *Synchronous-ellipse descent.* The propulsion system operates to transfer the spacecraft from the circular lunar orbit to an elliptical orbit of the same period. At the pericynthion altitude (~50,000 ft), the propulsion system is restarted for descent (optimum flight path) to hover altitude. The ignition point for the first maneuver can be approximately 90° or 270° from the landing site.

It is also possible to use descent trajectories[2] that reduce the range over the lunar surface between the initial maneuver (entry into the transfer ellipse) and the landing site from 180° to 25°–35° without significantly increasing the characteristic-velocity-increment requirements, as compared with

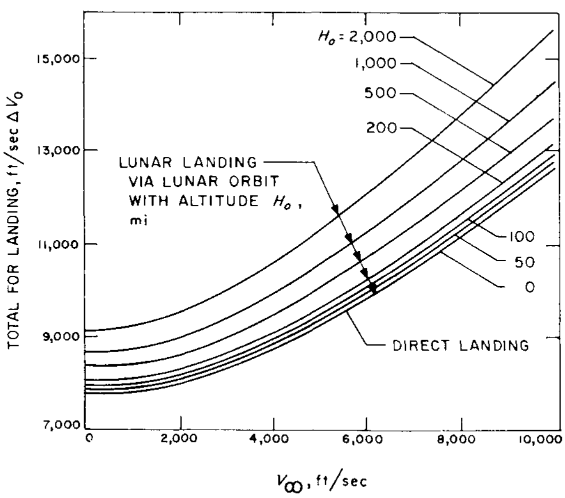

Fig. 8 Impulsive velocity increment requirements for landing.

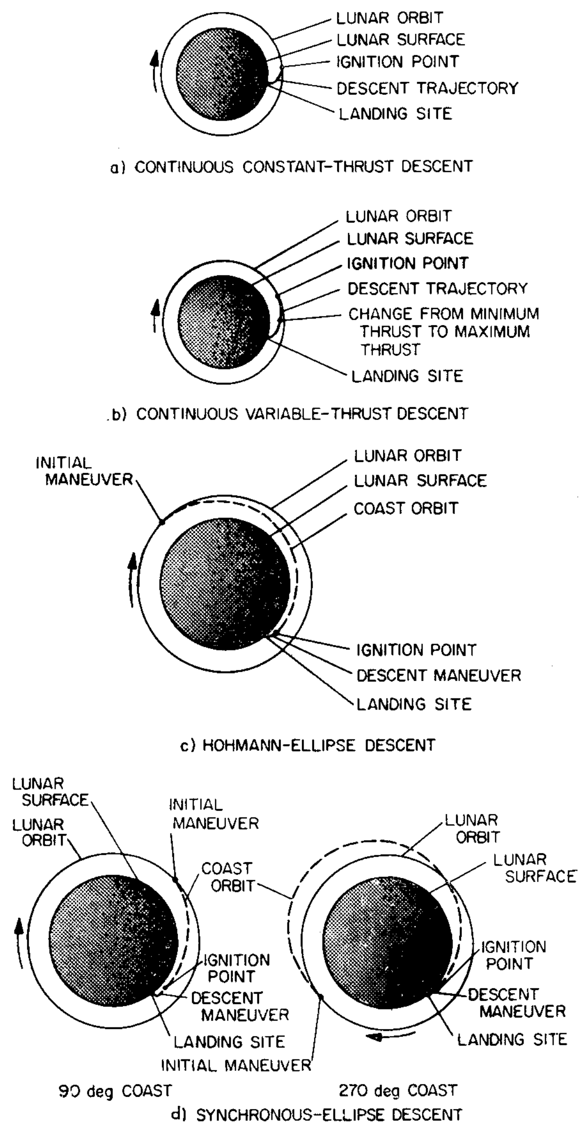

Fig. 9 Descent trajectories.

a Hohmann-ellipse descent. For a manned mission, however, these descents have a serious disadvantage in that a propulsion-system failure at the second ignition would leave the vehicle on an impact trajectory. Figure 10 (from data of Ref. 1) shows ΔV requirements as a function of throttling ratio with $T_{\min}/W_{ll} = 0.75$ for all four types of descent trajectories for descent from 58- and 116-mile circular-orbit altitudes. The differences in ΔV requirements among most of the trajectories are virtually independent of R_T. The Hohmann-ellipse descent is most efficient, and the synchronous-ellipse descent is least efficient. The continuous variable-thrust descent falls approximately midway between these two transfer-ellipse descents. The continuous constant-thrust descent becomes competitive only when maximum-thrust to initial-mass ratios on the order of 0.2 are used; it becomes very costly at the 116-mile alt (not shown).

Figure 11 illustrates the effect of R_T on total spacecraft mass per unit payload mass M_0/M_{pay} for the four descent trajectories; the ΔV requirements, engine mass (for a single regeneratively cooled engine with a chamber pressure of 150 psia and an expansion ratio of 40:1), and landing-gear mass are considered as variables. The minimum M_0/M_{pay} for the transfer-ellipse descents is reached at $R_T \sim 6$. For the continuous variable-thrust descent, M_0/M_{pay} is near optimum R_T range from 5 to 8. For the three most efficient descent trajectories, the difference between the mass ratio at $R_T = 3.5$ and at the optimum R_T for the same type of maneuver is approximately 2.7%. The efficiency of the continuous constant-thrust descent is within 1% of the continuous variable-thrust descent for $R_T < 3.5$.

In summary, the Hohmann-ellipse descent is most efficient; the continuous variable thrust and the synchronous-ellipse descents are approximately 1.7 and 2% less efficient, respectively. Using a single engine with $R_T = 3.5$ results in a penalty of approximately 2.7% in M_0/M_p for any of the descents except the continuous constant-thrust descent. At $R_T = 3.5$, the continuous constant-thrust descent penalizes M_0/M_{pay} by approximately 3.6%, compared with the optimum payload for the continuous variable-thrust descent. With the exception of the continuous constant-thrust descent, only small differences exist among the various types of descent from 58- to 116-mile lunar orbit altitudes. The transfer-ellipse descents (Hohmann or synchronous) require two ignitions under weightless conditions, whereas the continuous-thrust descents require only one ignition; reliability data show that approximately 50% of propulsion-system failures occur on or shortly after ignition; therefore, the continuous-thrust descents should be more reliable. The use of a throttling ratio of 3.5:1 rather than the near-optimum value of 7:1 results in a payload penalty of approximately 2.7%; an effective throttling ratio of 7:1 can be provided either by a single engine or by a combination of throttlable and fixed-thrust engines.

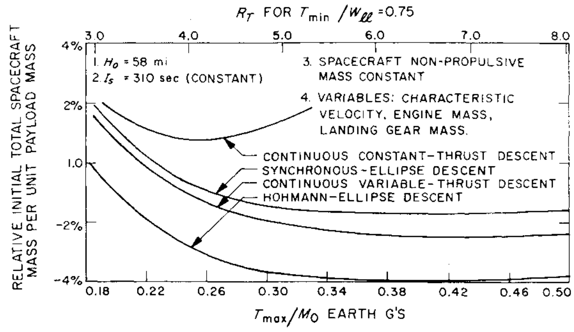

Fig. 11 Effect of descent trajectories and throttling ratio on spacecraft mass.

Ascent to Lunar Orbit from the Lunar Surface

For ascent, the spacecraft may go into lunar orbit either to extend the launch window for return to earth or to rendezvous with a mother ship. For this maneuver, the transfer-ellipse ascent trajectories appear to offer the greatest potential; they are similar to their counterpart descent trajectories, as shown in Fig. 9. The continuous variable-thrust ascent (similar to Fig. 9b) has initial attractiveness because it requires only one start and may reduce characteristic velocity requirements, but since the throttling is not otherwise required for ascent, a variable-thrust ascent propulsion system would seem to add an unwarranted complication. As with the descents, transfer ellipses that intersect the lunar surface could be used, but they appear to offer no advantages and were not considered because a propulsion-system failure at the second ignition would cause the vehicle to crash. If the spacecraft is manned, the ascent engine may be required to provide abort capability during descent, and an engine sized to these requirements may limit the choice of ascent trajectories. The abort requirement (not considered herein) is expected to limit the minimum thrust that can be used.

The ascent trajectories considered assume a vertical rise for 500 ft, followed by an optimum ascent to lunar orbit for the continuous constant-thrust ascent, or optimum ascent to the 50,000-ft pericynthion of a transfer ellipse. For transfer ellipse ascents with thrust-to-mass ratios greater than 0.7 earth g, the characteristic velocity increment begins increasing with increased thrust because of the 50,000-ft pericynthion altitude constraint (see Fig. 12). If a lower pericynthion altitude were allowed, the ΔV required would con-

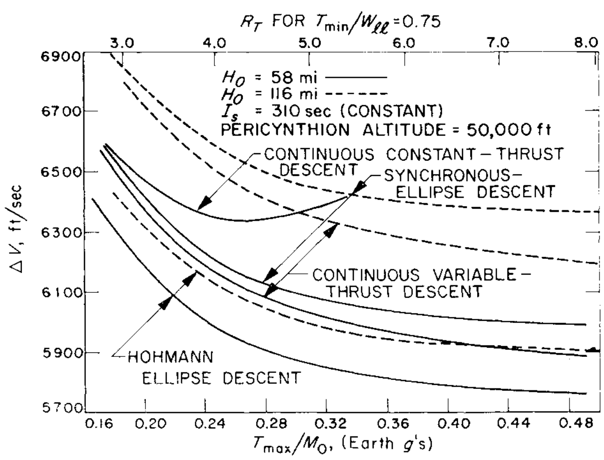

Fig. 10 Characteristic velocity increments for descent from a circular orbit to zero velocity at 500-ft lunar altitude.

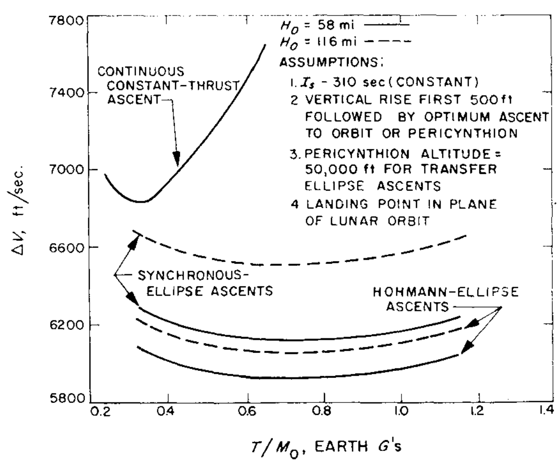

Fig. 12 Characteristic velocity increments for ascent to lunar orbit.

tinue to decrease slightly as thrust was increased. However, considering injection angle errors, nominal pericynthion altitudes below 50,000 ft could become dangerous if the spacecraft remained in the transfer ellipse for one complete orbit or more.

Unlike the descent maneuvers, the continuous constant-thrust ascent requires a ΔV approximately 700 fps greater than the synchronous-ellipse ascent in order to reach a 50-mile altitude lunar orbit. These greater ΔV requirements for the same T/M_0 are to be expected, since, for the same initial T/M_0, the thrust-to-mass ratio is lower during the vertical part of the ascent than during the near-vertical part of the descent. This represents a 7% penalty in total spacecraft mass for a thrust-to-takeoff-mass ratio of 0.33 earth g. For acceleration of 0.5 to 1.0 earth g, which may be required to satisfy descent-abort requirements, the penalty is even larger. Although a continuous-thrust ascent may be preferred for reliability, it appears that the associated penalties may be too great. If the lunar orbit rendezvous technique is used for return to earth, the required rendezvous propulsion system could also be used for the second maneuver from the transfer ellipse to circular orbit.

Rendezvous in Lunar Orbit

Rendezvous maneuvers have essentially the same requirements as shown for trajectory corrections; their magnitudes depend upon the accuracy of the guidance system. Epoch corrections may be required if an abort occurs during the descent or if an emergency condition requires that the spacecraft be launched when the orbiting vehicle is not in position for a direct rendezvous. Table 1 summarizes the maximum time and ΔV requirements for abort from the various types of descent trajectories. The descent using 270-°coast synchronous ellipse requires the longest times from abort to rendezvous. The Hohmann and 90°-coast synchronous-ellipse descents require the least time, 2.6 and 1.2 hr for 58- and 116-mile lunar orbits, respectively.

For an abort from any descent, the total impulsive ΔV_0 required for abort and rendezvous is always less than or equal to that required for normal launch and rendezvous. Abort trajectories can be selected which allow immediate return to the orbiting vehicle without utilizing an epoch correction orbit, but preliminary studies indicate that these trajectories require greater maneuver capability than is required for a normal launch. Less weight may be needed to provide redundancy in critical subsystems than to provide the additional spacecraft maneuver capability required for immediate return to the orbiting vehicle. Figure 13 shows maximum (worst position) required excess ΔV, compared with that for a normal launch and transfer to orbit, and for abort from the lunar surface as a function of maximum time to rendezvous.

Spacecraft Propulsion System Design

This section summarizes the spacecraft system penalties if maneuvers are chosen that minimize the propulsion requirements, particularly throttling requirements, the number of engines, and the number of operations. Since these particular requirements are not affected by the specific type of mechanization, they are considered in general with the assumption that minimizing all three is desirable. The effects for four representative missions are discussed. The small acceleration requirements for trajectory corrections and for

Table 1 Summary of maximum abort requirements for abort during descent

Type of descent trajectory	Phase of descent at which abort occurs	Lunar orbit alt, mile	Max[a] maneuver capability required for abort: Launch system,[b] fps	Rendezvous system,[c] fps	Max time, abort to rendezvous, hr
Continuous constant-thrust or continuous variable-thrust	Initial[d]	58	0	ΔV_R	<1
		116	0	ΔV_R	<1
	Final[e]	58	$\Delta V_H - 58$	$\Delta V_R + 58$	3.2
		116	$\Delta V_H - 130$	$\Delta V_R + 130$	1.7
Hohmann-ellipse	Initial maneuver	58	0	ΔV_R	<1
		116	0	ΔV_R	<1
	Transfer-ellipse coast	58	0	$\Delta V_R + 122$	4
		116	0	$\Delta V_R + 290$	4
	Main maneuver[f]	58	$\Delta V_H - 58$	$\Delta V_R + 58$	2.6
		116	$\Delta V_H - 130$	$\Delta V_R + 130$	1.2
Synchronous-ellipse, descent with 90° coast	Initial maneuver	58	0	$\Delta V_R + 142$	<1
		116	0	$\Delta V_R + 320$	<1
	Transfer-ellipse coast	58	0	$\Delta V_R + 142$	2
		116	0	$\Delta V_R + 320$	2
	Main maneuver	58	$\Delta V_H - 58$	$\Delta V_R + 58$	2.6
		116	$\Delta V_H - 130$	$\Delta V_R + 130$	1.2
Synchronous-ellipse, with 270° coast	Initial maneuver	58	0	$\Delta V_R + 142$	<1
		116	0	$\Delta V_R + 320$	<1
	Transfer-ellipse coast	58	0	$\Delta V_R + 142$	2
		116	0	$\Delta V_R + 320$	2
	Main maneuver	58	$\Delta V_H - 58$	$\Delta V_R + 58$	3.5
		116	$\Delta V_H - 130$	$\Delta V_R + 130$	4.3

[a] All aborts are assumed to use a three-maneuver abort trajectory, employ a 150-sec hover and landing maneuver, and use a 1.0-earth g ascent initial thrust-to-mass ratio.
[b] ΔV_H is the capability required for launch from the lunar surface into a Hohmann-ellipse ascent.
[c] ΔV_R is the capability required for a normal launch employing a maneuver from a Hohmann launch ascent ellipse to the lunar orbit and subsequent rendezvous and docking maneuvers.
[d] The initial phase of the descent is defined as orbit pericynthion = 50,000 ft or greater.
[e] The final phase of the descent is defined as orbit pericynthion less than 50,000 ft.
[f] The worst condition for abort is at the instant of landing.

entry into lunar orbit suggest that the same propulsion system might be used for all maneuvers, but, as noted previously, accelerometer-null errors lower the accuracy of the maneuver for entry into lunar orbit when very low-thrust levels are used. If the latter inaccuracy is small in comparison with errors from other sources, the thrust level of the engine used can be selected to minimize the spacecraft M_0/M_{pay}, considering the effects on performance of ΔV, engine mass, and associated spacecraft structure mass. If this thrust level would not result in unacceptable errors, use of a single engine for both maneuvers would appear best.

Direct Lunar Landing Mission

A spacecraft that lands directly from the trans-lunar trajectory requires trajectory corrections, descent to hover, and final landing maneuvers. The final landing requires a propulsion system with a minimum thrust less than the lunar weight of the spacecraft at landing, unless on-off propulsion system operation is used. Figure 2 shows the variation of ΔV with T_{max}/M_0 for a vertical descent, and throttling ratio vs T_{max}/M_0; these figures indicate that separate engines would be required for descent and final landing maneuvers unless wide-range throttling techniques were developed.

If one engine were designed for the main descent to the hover point, the optimum T_{max}/M_0 would probably be near 1 earth g for a liquid propellant engine[3] and near 2 to 3 earth g's for a solid propellant motor, again considering the effects of ΔV, engine mass, and associated structure mass. A separate engine could be designed for the descent from hover by varying the latter parameters to give the optimum T_{min}/W_{ll}, which would be slightly lower than the minimum fuel points on the curves of Fig. 4. If this engine were used for trans-lunar trajectory corrections, the T/M_0 for the correction maneuvers would be 0.03 earth g at minimum thrust, or 0.11 earth g at maximum thrust (based on $T_{min}/W_{ll} = 0.53$, $R_T = 3.5$, $I_{sp} = 310$, and $\Delta V = 10{,}000$ fps for the main descent maneuver). Since the acceptable range of T/M_0 for the landing engine falls within the acceptable range for performing trajectory corrections (0.025 to 0.25 earth g's), the same engine could be used for both maneuvers. If the landing engine is also used to provide thrust-vector control during operation of the main descent engine, it may be designed to a higher thrust level than the minimum-fuel point in order to provide more control capability. Throttling ratios as low as 2 appear to be acceptable for landings from hover altitudes up to about 5000 ft.

Lunar Landing via Lunar Orbit

Figure 10 shows that the penalties in ΔV are relatively small for throttling ratios as low as 3, and Fig. 11 shows that the effect of using $R_T = 3.5$ rather than the more nearly optimum value of 7 results in a 2.7% penalty in M_0/M_{pay}. Thus, it appears that the entire landing maneuver from lunar orbit can be performed efficiently by a single engine with a relatively low-throttling range. Figure 7 shows that the ΔV for entry into lunar orbit is affected very little by T/M_0 for values greater than 0.1 earth g. If the descent from lunar orbit is accomplished with an engine with $R_T = 3.5$, then T_{max}/M_0 would be 0.153 earth g for entry into lunar orbit, assuming that $T_{min}/W_{ll} = 0.75$, $I_{sp} = 310$ sec, and that the spacecraft uses the same engine to perform a 3500-fps maneuver for entry into lunar orbit. The total penalty for the orbit-entry maneuver resulting from using this T_{max}/M_0 rather than infinity is approximately 35 fps, or about 0.3% increase in total spacecraft mass. The relatively low value of T_{max}/M_0 (0.153 earth g) suggests that this propulsion system could also be used for trans-lunar trajectory corrections. If the available R_T were 7, the T_{max}/M_0 for the entry into lunar orbit maneuver would be approximately 0.3 earth g, which would reduce the ΔV penalty to 20 fps.

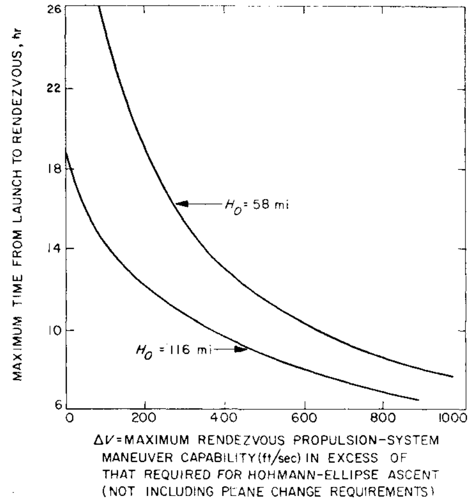

Fig. 13 Maximum abort requirements for abort launch from lunar surface.

In summary, lunar landing via lunar orbit could be accomplished using a single engine, whereas direct landing would require separate descent and landing engines. A direct landing requires one maneuver, whereas landing via lunar orbit requires a minimum of two maneuvers, or three maneuvers if a transfer ellipse is used for descent from lunar orbit. For many trans-lunar trajectories, the maneuver for entry into lunar orbit must be performed behind the moon where the spacecraft cannot communicate directly with earth; lunar landing via lunar orbit allows landing at any point on the lunar surface, whereas direct landing is restricted primarily to the visible side. Thus, the best type of landing for a particular mission depends upon the importance of system complexity, operational complexity, and restriction on landing site location.

Lunar Landing and Return via Lunar Orbit Rendezvous

This mission requires trans-lunar and trans-earth trajectory corrections, entry into and exit from lunar orbit, ascent to lunar orbit, and rendezvous. As discussed previously, the T_{max}/M_0 requirements for entry into or exit from lunar orbit are compatible with trajectory correction requirements. It appears possible to design the spacecraft so that the engine used for entry into and exit from lunar orbit can be used for trajectory corrections. The following is an example of such a design.

Assume that the total mass of the spacecraft that affects the lunar landing and return to lunar orbit is one-third of the total injected mass of the two spacecraft and that the entry into and exit from lunar orbit maneuvers each requires $\Delta V = 3500$ fps for $I_{sp} = 310$ sec. The penalties in ΔV would be approximately 160 fps for entry into orbit and approximately 5 fps for exit from orbit for a thrust level of 7% of the complete spacecraft earth weight at injection. An engine of this size would result in an acceleration of 0.07 earth g for trans-lunar trajectory corrections and 0.42 for trans-earth trajectory corrections. Selecting a lower-thrust level would reduce errors during trans-earth trajectory corrections but increase ΔV penalties for entry into and exit from lunar orbit. Selecting a higher thrust level would have the converse effects. A thrust of 7% of the total injected earth weight would also be well suited for emergency rendezvous maneuvers in lunar orbit.

If a propulsion system is designed specifically for descent from lunar orbit to the lunar surface, the same arguments apply as were developed in the discussion for the lunar landing mission via lunar orbit; it seems practical to use a single engine for the descent stage even if a modest throttling range must be used. Advanced propulsion system designs that allow wide-range throttling are not necessarily required. Of the descent trajectories compared in Figs. 9–11, the continuous variable-thrust descent is probably best, because it eliminates the restart requirement for the descent engine.

The ascent stage of the landing spacecraft, in general, must be designed for abort during descent, ascent from the lunar surface, and rendezvous. Abort during descent is expected to require a relatively high T/M_0 for the ascent stage in order to provide sufficient time for detecting an abort condition and initiating the abort maneuver to avoid impact with the lunar surface. Assuming that $T/M_0 = 0.5$ to 1.0 earth g is needed to fulfill abort requirements, the engine would probably be too large for rendezvous maneuvers. For example, if the ascent-stage engine provides a $T/M_0 = 1.0$ earth g, the nominal ΔV is 6000 fps, and the I_{sp} is 310 sec, then the ascent engine would provide a T/M_0 of 1.86 earth g for rendezvous maneuvers. Therefore, a separate propulsion system may be required in order to obtain sufficient accuracy for rendezvous.

In summary, it should be possible to design a single engine with $T_{max}/M_0 \simeq 0.07$ which could be used for trans-lunar and trans-earth trajectory corrections, as well as entry into and exit from lunar orbit maneuvers and emergency rendezvous maneuvers in lunar orbit. A single engine with a modest throttling range could be used for the descent of the landing spacecraft. Since relatively high T_{max}/M_0 may be required for the ascent engine to satisfy abort requirements, it may not be well suited to perform rendezvous maneuvers, and a separate rendezvous engine may be required.

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