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Lunar Module Descent Mission Design

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Various lunar descent trajectories were analyzed that include the optimization of the Apollo constrained mission trajectory, a fully optimized minimum energy trajectory, and a optimal, constrained trajectory using current instrumentation technology. Trade studies were conducted to determine the impacts of mission assumptions, pilot in the loop/automated flight demands, and additional constraints for the present recurring missions to the same outpost landing site. For mission design at this conceptual phase of the program, the Apollo pre-mission planning was applied to account for known contingencies (hardware, instrumentation known uncertainties) and unknown unknowns. The mission Delta-V's are presented in a risk form of conservative, nominal, and optimistic range where 90 percent of Delta-V is derived by trajectory analysis and the other 10 percent was derived from a qualitative analysis from Apollo 11 pre-mission planning. The recommendations for the Delta Vs are the following: conservative (Apollo derived) (2262 m/s), nominal (2053 m/s), and optimistic (1799 m/s). Because of the qualitative nature of the results, the degree of autonomy assumed, additional safety considerations for a lunar outpost, and the impact of advanced instrumentation, more in-depth analyses are required to refine the present recommendations.

I. Nomenclature

ESAS	=	Exploration Systems Architecture Study
NASA	=	National Aeronautics and Space Administration
LEM	=	Lunar Exclusion Module
FPA	=	Flight Path Angle, degrees
g_L	=	lunar gravity, 1.622 m/s2
h	=	altitude, m
h_t	=	transition altitude from optimized to attitude constrained, m
POST	=	Program to Optimize Simulated Trajectories
r_L	=	lunar radius, 1738 km
t	=	time, s
T/W_L	=	thrust-to-lunar weight ratio
v	=	velocity, m/s
W	=	weight, N
γ	=	flight path angle, deg
ΔV	=	ideal velocity, m/s
$\Delta V_{characteristic}$	=	characteristic velocity increment (=vfinal – vinitial), m/s

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$\Delta V_{gravity\ losses}$ = velocity increment loss due to gravity, m/s
$\Delta V_{thrust\ vector\ losses}$ = velocity increment loss due to thrust misalignment, m/s
μ = gravitational parameter, 4902.801 km3/s2

II. Introduction

IN January 2004, President Bush addressed the nation and presented the NASA's Vision for Space Exploration. This vision included the completion of the International Space Station, the retirement of the Space Shuttle, the development of a crew exploration vehicle, and the return of humans to the moon by no later than 2020 and human Mars exploration in 2030. The Exploration System Architecture Study (ESAS) team was established to develop the baseline architecture that NASA would use to return humans to the moon. A general description of the architecture established during this study is Ref. 1. There are six main vehicle elements in this architecture: a Crew Launch Vehicle, a Cargo Launch Vehicle, a Earth Departure Stage, a Lunar Surface Access Module, a Crew Exploration Vehicle, and a Service Module. A more detailed description of each element can be found in Ref. 1. The architecture was designed using heritage space components where possible to help improve the overall cost and reliability.

The ESAS study was a 90-day conceptual study of the definition of a complete exploration architecture that laid the foundation for more in-depth analyses of each of the architecture elements. Currently, the detailed studies leading to preliminary design review are focused on the near-term elements of the Crew Launch Vehicle (ARES I), and the Crew Exploration Vehicle (Orion) for the space station mission. However, conceptual studies of the other ESAS elements are being conducted today to determine the mission and design requirements with the 7th manned lunar landing scheduled for 2020.

The lunar landing mission design has a rich history of trajectory analysis development that progressively matured over the course of the requirements refinement of the Apollo Lunar Excursion Module (LEM). The Apollo program was initiated on May 25, 1961, when President John F. Kennedy announced the goal of sending an American safely to the Moon before the end of the decade. This mandate resulted in the first lunar landing of the Eagle lunar lander on July 16, 1969.

Although there have been six manned lunar landings, the present architecture has a different concept of operations. There will be an outpost requiring multiple landings, will have unmanned cargo landings with full automation and forty years of technology advancement to improve performance, and advanced navigation sensors such as LIDAR. This paper will concentrate on defining the performance requirements for the lunar lander and will provide trade studies for propulsion and concept optimization in this early stage of development.

III. Lunar Landing History

A. Precursor Apollo Lunar Exclusion Module Studies

Most of the early references in the 1950s of lunar landing (as described in Ref. 2) were qualitative. Ref. 2 from 1959 was one of the first quantitative attempts of modeling soft lunar landing. The minimum energy case was presented to illustrate the absolute minimum performance. This case requires two burns - an impulse burn at high-lunar orbit (152 km) that directly transfers the LEM by an elliptical path to the surface (no mountains are assumed) followed by an impulsive stop. The resulting required performance, measured by ideal velocity increment - ΔV, is 1,742 m/s. A more practical approach was presented for the de-orbit and landing (using a non-rotating moon analysis) which started from the high-altitude circumlunar orbit (152 km) and then transferred by a minimum-energy elliptical path to the lowest acceptable altitude, the highest lunar mountain peak of approximately 10,000 meters. An impulsively stop is then made at this low-lunar attitude. The vehicle then goes into a vertical free fall descent, and then a final upward thrust is used to decelerate the vehicle for a soft landing. The ΔV, for the 152x15x0, was 1,956 m/s; this maneuver did require an extremely high thrust-to-weight ratio (T/W_L) of over 5.3 (weight based on lunar gravity) as compared to Apollo T/W_L of 1.8. As shown later, this de-orbit/landing technique has reasonable ΔV; however, the three engine starts reduce reliability, the requirement to start the engines during freefall is a safety concern, and finally, an extremely high system T/W_L is needed to stop the free fall.

Bennett became the lead for the Apollo lunar landing and wrote many of the papers during the design, development, and post-flight analyses of the Lunar Excursion Module. His paper is the first comprehensive "Apollo Working Paper" that conducted many of the initial performance trade studies.[3] The trajectory analyses included the calculus of variation technique established by Miele.[4] The concept of operations assumed an initial de-orbit transfer from a high-lunar circular orbit of 148.16 km (80 n.mi.) to an elliptical orbit with apoapsis of 15.24 km (50,000 ft) requiring a ΔV of 29 m/s, then a "Fuel Optimum Phase" using continuous powered descent with the thrust along the velocity vector providing minimum energy deceleration to a transition altitude that was varied from 1.524 km to 4.572 km (5,000 to 15,000 ft), followed by a "Landing Approach Transition" where the attitude of the LEM was varied from 90 degrees vertical for best pilot visibility to 140 degrees which is approaching fuel optimum descent to the surface, and ending with the "Final Translation and Touchdown" phase with initial conditions of 0.304 km (1,000 ft), velocity of 22.86 m/s (75 ft/s), a flight path angle of 0 degrees (vertical), and a vertical descent rate of 6.1 (20 fps). The final landing started at an altitude of 15.2 m (50 ft) with a descent rate of 1.02 m/s (3.33 ft/s). The initial T/W_L was 2.4 (0.4 Earth gravity) and was held at the maximum throttle setting of 1.0 and was throttled down to meet the constraints of the other phases. Meditch[5] in 1964 and Tawakley[6] in 1966 (Ref. 6 references Miele in 1958) concur that for optimum fuel consumption, an optimum fuel burn trajectory results with engines at maximum thrust throughout the trajectory. The performance results of Bennett are shown in Figure 1.[3] This figure presents several key results: 1) as stated above, the minimum fuel requirement using an elliptical transfer to the surface is a ΔV of 1,742 m/s, 2) T/W_L cannot be much lower than 1.8 and increases in T/W_L can significantly reduce ΔV up to a T/W_L of approximately 4.8 (however a trade exists between performance and the additional mass required for the additional thrust), 3) the initial de-orbit transfer from high, lunar circular orbit to the start of the fuel optimum phase should be as low an altitude as possible, thus the need to know the exact lunar terrain to mitigate any mountain impact), 4) pitching the LEM attitude from a fuel optimum of 140 degrees to a pilot visibility optimum of 90 degrees requires a significant addition of ΔV up to approximately 122 m/s.

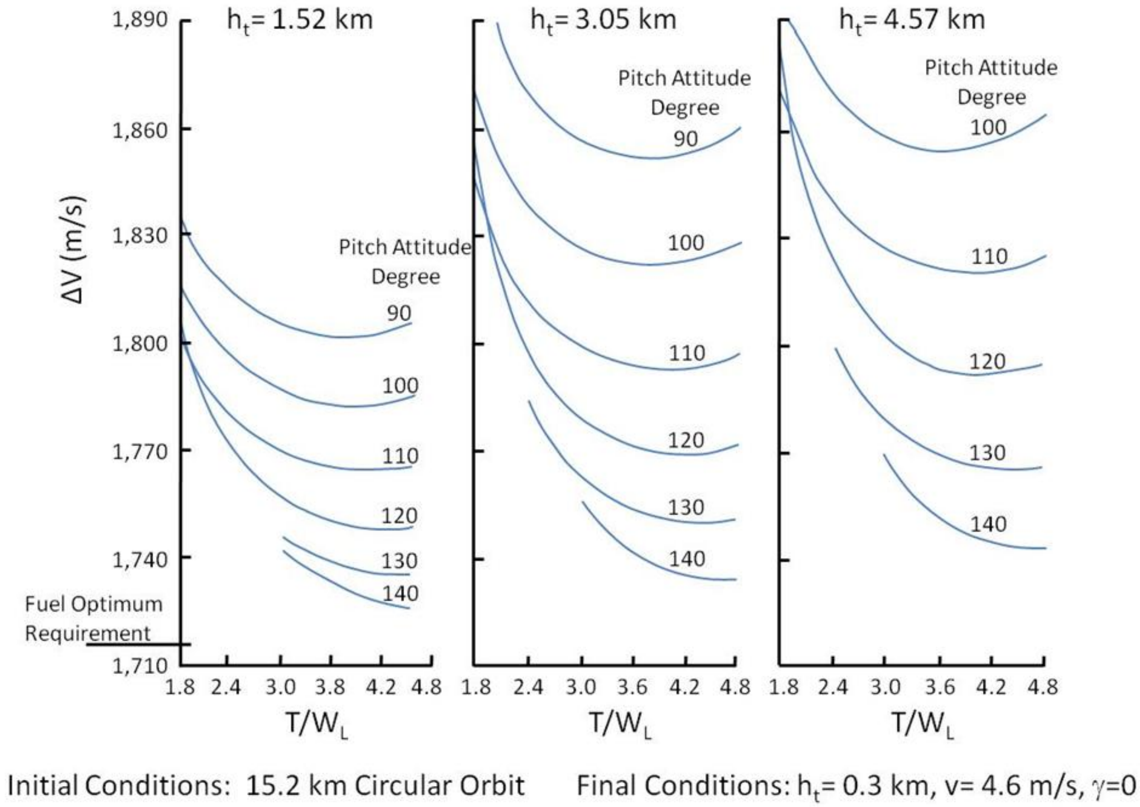


Figure 1. LEM Performance (does not include 29 m/s de-orbit DV (adopted from Ref. 3 pg 25)

From the previous references, it was assumed that thrust was aligned with the velocity vector. Thompson shows that ascent trajectories can be performed with a gravity turn where an initial impulse angle of attack (or gimbal angle) is used right after the vertical liftoff and then gravity automatically turns the vehicle to horizontal flight (zero flight path angle) at orbital conditions by using zero angle of attack throughout the trajectory.[7] Noting that

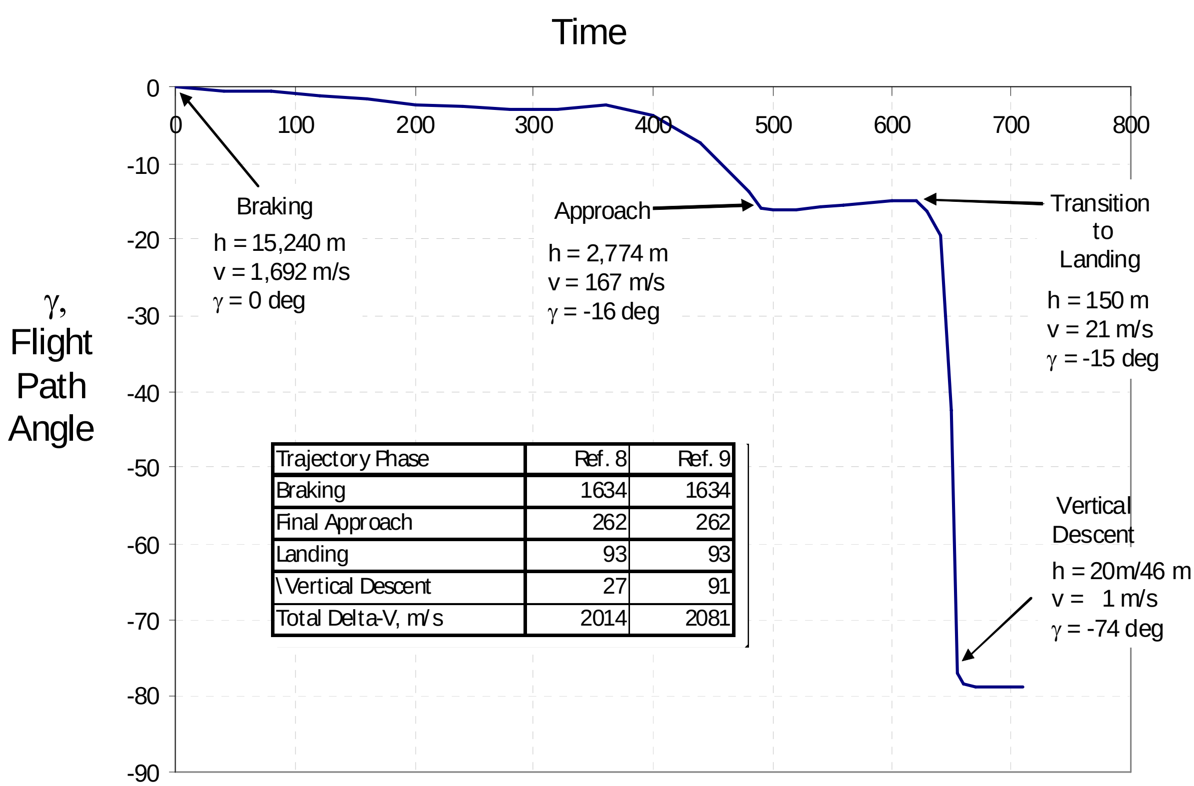
$$\Delta V = \Delta V_{characteristic} + \Delta V_{gravity\ losses} + \Delta V_{thrust\ vector\ losses}$$

(drag losses are zero in a vacuum) thrust vector losses are zero for zero angle of attack for acceleration (or 180 degree angle of attack for deceleration). However in Ref. 7, there is a theoretical analysis that shows that the "optimal" trajectory of minimum fuel burn is accomplished with varying angle of attack throughout the trajectory. This approach of using angle of attack trades lower gravity losses with thrust vector losses; however this approach uses extremely high angles of attack that limits this approach due to stability and control concerns. Using angle of attack to lower require ΔV performance is shown in the 1965 Ref. 8 where "to assume a zero angle of attack for all lunar descents is by no means optimum." Data in this reference showed that the improvement of fuel burn with thrust vectoring was a function of the LEM thrust-to-weight ratio.

B. Apollo Mission Planning

The body of knowledge for LEM mission planning and post flight results from 1966 to the two LEM landings in 1969 is summarized in the "Apollo Experience Report" by Bennett.[9] Major differences between the initial performance analyses and the final mission plans included real-world impacts such as lunar surface hazard and avoidance maneuvers, pilot-in-the loop visibility and control, propulsion engine thrust constraints, known navigation errors, and contingency for unknowns. The LEM powered descent depended on the primary guidance, navigation, and control system; the descent propulsion system; the reaction control system; the landing radar; and the landing point designator. The Apollo descent strategy was to optimally descend with continuous thrust to a position where the pilot would have adequate time to observe the landing site and to provide adequate altitude, position, and velocity for the pilot to take the controls and land safely. The trajectory strategy is shown in Figure 2 and discussed in Ref. 9. "The lunar module powered-descent trajectory is initiated at pericynthion of 15.24 by 148.2 km (50,000 ft by 80 n.mi.) descent transfer orbit. The powered descent consists of three operation phases – braking, final approach, and landing. The "Braking" phase, initiated at pericynthion, is designed for efficient reduction of the orbital velocity and terminates at a position which is approximately ~ 2.7 km (9000 ft) altitude. The "Approach" phase is designed to allow for the pilot to visually (out-the-window) assess the landing area and for abort safety. This phase terminates at the "Transition to Landing" phase which is at approximately 150 m (500ft altitude). The "Landing" phase, beginning is designed to provide the crew with detailed visual assessment of the landing area and to provide compatibility for the pilot takeover from automatic control. This phase includes a slow vertical descent (~ -1 m/s) from approximately 20 m (65 ft) and terminates at the touchdown on the surface."[9] The total trajectory performance ΔV (Fig. 2) shows the initial baseline.[9] The final baseline trajectory for Apollo 11 mission planning extended the final vertical descent from an altitude of 20 m to 46 m in order to provide additional landing/control time for the pilot.[9] This additional 26 m changed the trajectory performance ΔV from 2014 m/s to 2081 m/s.

In order to determine the Apollo descent mission design-to requirement for ΔV, uncertainties, contingencies, margin, and pilot performance considerations were added to the trajectory ΔV as show in Table 1.[8]



Trajectory Phase	Ref. 8	Ref. 9
Braking	1634	1634
Final Approach	262	262
Landing	93	93
\Vertical Descent	27	91
Total Delta-V, m/s	2014	2081

Figure 2. Apollo Baseline Trajectory [Refs. 8 and 9]

	Propellant Required, kg	Propellant Remaining, kg	Delta V, m/s	Additional Delta-V, m/s
System Capacity		8,282.8		
Offloaded	34.2	8,248.6		
Useable	113.6	8,135.0		
Available for Delta-V		8,135.0		
Nominal required for Delta V (6827 fps)	7,693.3	441.7	2081	
Dispersions (~3σ)	132.4	309.2	2134	53
Contingencies		0.0		
Engine-valve malfunction (change in O/F)	29.3	279.9	2145	12
Redline low-level propellant sensor	31.2	248.7	2158	13
Redesignation (8 m/s; 610 m diameter)	46.7	202.0	2177	19
Manual hover (27 m/s)	65.3	136.7	2204	27
Margin		136.7	2261	57
		Additional ΔV, m/s		180
		Additional ΔV, percent		8.7%

Table 1. Apollo Pre-Mission Planning Performance [Ref.3]

Because of the known uncertainties (engine thrust, landing radar, and inertial measurement unit sensors), a Monte Carlo analysis was performed using the uncertainties of propulsion thrust, landing radar errors , terrain, and navigation gyros and accelerometer errors to determine the 3-sigma ΔV impact of 53 m/s on the baseline trajectory similar to Ref. 11. Contingencies of 25 m/s were added for known valve and sensor uncertainties. To account for potential hazards with the landing site, an extra 8 m/s was added for redesignation that provided an additional 610 m diameter landing site footprint. Also, an extra 3 seconds of vertical descent time was added to provide the pilot with a full 2 minutes of control time adding an additional ΔV of 27 m/s. Finally a margin of 2.5 percent (57 m/s) was added for unknowns. Thus, an additional 180 m/s or 8.7 percent was added to the trajectory ΔV to define the Apollo 11 pre-mission design-to a ΔV requirement of 2261 m/s.

The need for the contingencies and margin can be illustrated in the actual mission performance of the Apollo descent as show in Table 2.[12] As shown in the bottom of the table, all the missions used more ΔV than the ΔV computed from the trajectory analysis (called percent of AP11 pre-nominal ΔV – Table 2). With the 8.7 percent contingency and margin added to the trajectory Delta-V, Neil Armstrong, on the first landing of the Eagle, came fairly close to using all the LEM propellant with his hazard avoidance maneuver. With knowledge from each successive mission, the landings became more routine, and the propellant actually used was closer to the predicted mission trajectory ΔV.

	Apollo 11	Apollo 12	Apollo 14	Apollo 15	Apollo 16	Apollo 17
LM Gross, kg	15,095	15,224	16,187	16,438	16,437	16,448
LM Propellant Burned, kg	7,899	7,826	7,994	8,334	8,313	8,313
ΔV Used, m/s	2,216	2,159	2,037	2,115	2,108	2,106
LM Propellant Useable @ Cutoff	306	489	285	479	512	556
LM Mass at engine cutoff	7,196	7,397	8,193	8,104	8,124	8,135
ΔV Unused, m/s	130	205	106	182	195	212
Percent of AP11 pre-Mission ΔV	98%	95%	90%	94%	93%	93%
Percent of AP11 pre-Nominal ΔV	106%	104%	98%	102%	101%	101%

Table 2. Apollo Mission Performance [Ref. 12]

C. Literature Observations

Based on the literature for lunar powered descent and soft landing leading up to and including the Apollo planning and post-flight analyses, the following observations were made concerning the required performance ΔV:

1. The minimum energy ΔV is attained with an elliptical transfer from the lunar insertion altitude directly to the surface with an impulse burn to the surface of 33.3 m/s and an impulse stop at the surface of 1,714 m/s, for a total of 1,747 m/s. This is a theoretical minimum because of possible lunar mountain collisions and astronaut heart attacks caused by the frightening surface impulse maneuver.[2,3]
2. Theoretical analyses showed that using the maximum throttle provides the minimum fuel burn[5,6] and that angle of attack (or engine gimbal) may provide additional fuel economy.[7]
3. Several concepts of operations considered in the literature were constrained by the lunar topography, astronaut visibility of the landing site and pilot-in-the-loop considerations. Primary considerations that impact the performance ΔV are system T/W_L, initiation altitude of the continuous burn for the powered descent, pilot visibility considerations on approach such as time (or altitude or time of constant flight path hold) from the landing site/vehicle pitch attitude (vertical 90 degree attitude is best), redesignation for hazard avoidance, altitude of hover initiation, and rate of descent (time) for piloted landing, and other considerations such as known subsystem uncertainties, and overall contingency for unknown unknowns.
4. Using an optimal fuel burn trajectory, the performance ΔV ranged from 1755 km/s at a T/W_L of 4.8 and minimum observation altitude and a 140-degree attitude (which is near optimal) to 1,935 km/s for maximum observation altitude at a T/W_L of 1.8 and a100-degree pitch attitude (Fig. 1).

III. Analysis and Trade Studies

A. Analysis

The Program to Optimize Simulated Trajectories (POST) was used for the trajectory performance calculations.[13] The POST is a generalized point mass, discrete parameter targeting and optimization program and provides the capability to target and optimize point mass trajectories for a powered or unpowered vehicle near an arbitrary rotating, oblate planet. For the present lunar study, a spherical, non-rotating model was used with the gravitational parameter, μ, equal to 4902.801 km^3/s^2 and radius, r_L equal to 1738 km. All trajectories were initiated at a circular lunar orbit altitude of 148.16 km (80 n.mi.).

B. Trade Studies

Optimal fuel burn with no constraints. The studies in Ref. 2 were extended to determine the optimal fuel burn as a function of T/W_L. As shown in Fig. 3, the theoretical minimum for a (Hohmann) direct elliptical transfer to the surface (ΔV=33 m/s) and an impulse burn on the surface (ΔV=1,714 m/s) is 1,747 km/s. The red line is the total ΔV from the initial circular lunar injection orbit altitude (1783 km) and the blue line is the ΔV from the transfer orbit to the surface. Note that the transfer orbit altitude changes (green line) with T/W_L, and that for optimal fuel burn cases, the start altitude is below the safe altitude of 6 to 15 km for the highest lunar mountain peaks. Trajectory adjustments have to be made for these cases. The difference between the red and blue lines is the de-orbit ΔV from the initial circular orbit of 1,783 km to the start of the continuous powered descent. As T/W_L increases from the Apollo T/W_L of 1.8 to 4.8, the ΔV decreases from 1,874 km/s to 1767 km/s. Thus, there is a system trade of ΔV versus the addition of addition engine mass to obtain the additional thrust. The ΔV difference between the minimum energy Hohmann and the optimal fuel burn are gravity losses as there are no thrust vector losses since the angle of attack is kept at a constant 180 degrees (directly opposite the flight path angle).

Optimal Apollo Trajectory. The POST program was used to optimize the Apollo trajectory using the same specific impulse (299 s), the same T/W_L (1.8), the same flight phases and constraints.

For the initiation of the continuous powered braking phase, the altitude and flight path angle were selected by the optimizer as well as the thrust angle. The throttle was set at 100%. The initiation of the approach phase (used for out-the-window pilot visibility to detect any hazards for redesignation.) was selected by the optimizer where the flight path angle is equal to the Apollo -16 degrees flight path angle hold. The hold time was 120 seconds. An optimal combination of thrust angle and throttle is determined to maintain this -16 degree flight path angle until an altitude of 150 m and velocity of 21 m/s is reached. At this state of altitude and velocity, the landing phase begins

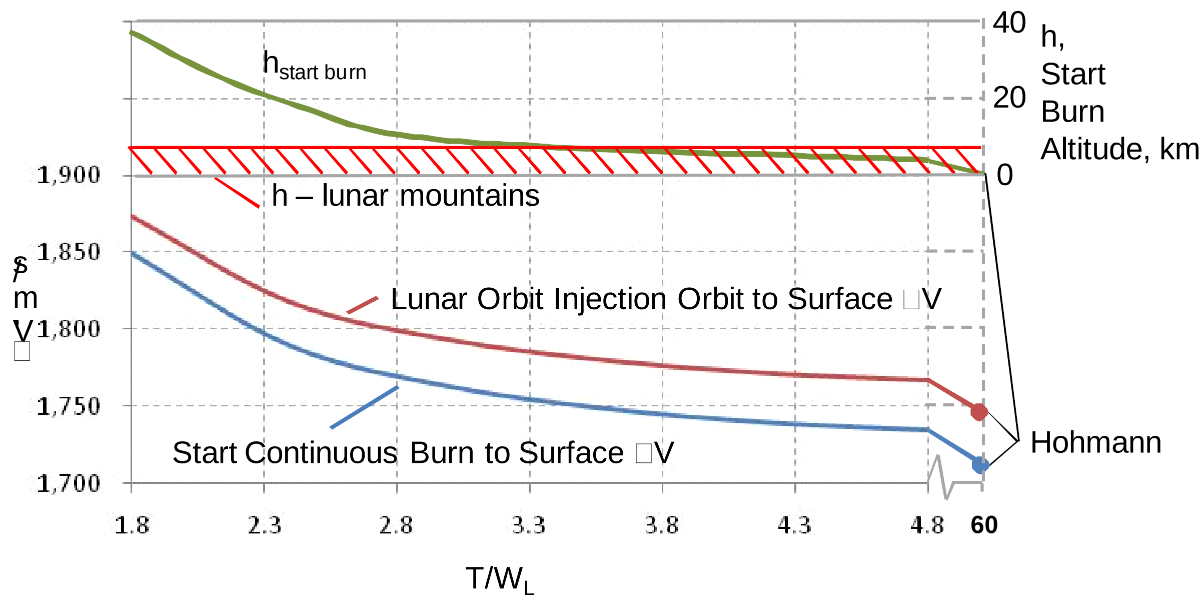


Figure 3. Optimal fuel burn performance.

where the optimal throttle and thrust angle is determined to reach 46 m altitude and -90 degree flight path angle for vertical descent. Vertical descent rate was held to 1 m/s descent velocity by varying thrust. It should be noted that the Apollo engine had constraints on throttling (no throttling between 100 and 50 percent), but none were administered with this optimal simulation.

Figure 4 compares the Apollo trajectory to the "optimal Apollo" trajectory. As shown in the figure, the braking phase of the optimal trajectory is somewhat lofted allowing a 100% throttle that slows the lander faster than Apollo, thus saving 42 m/s in ΔV. The total trajectory time was reduced by approximately 100 seconds. Also the

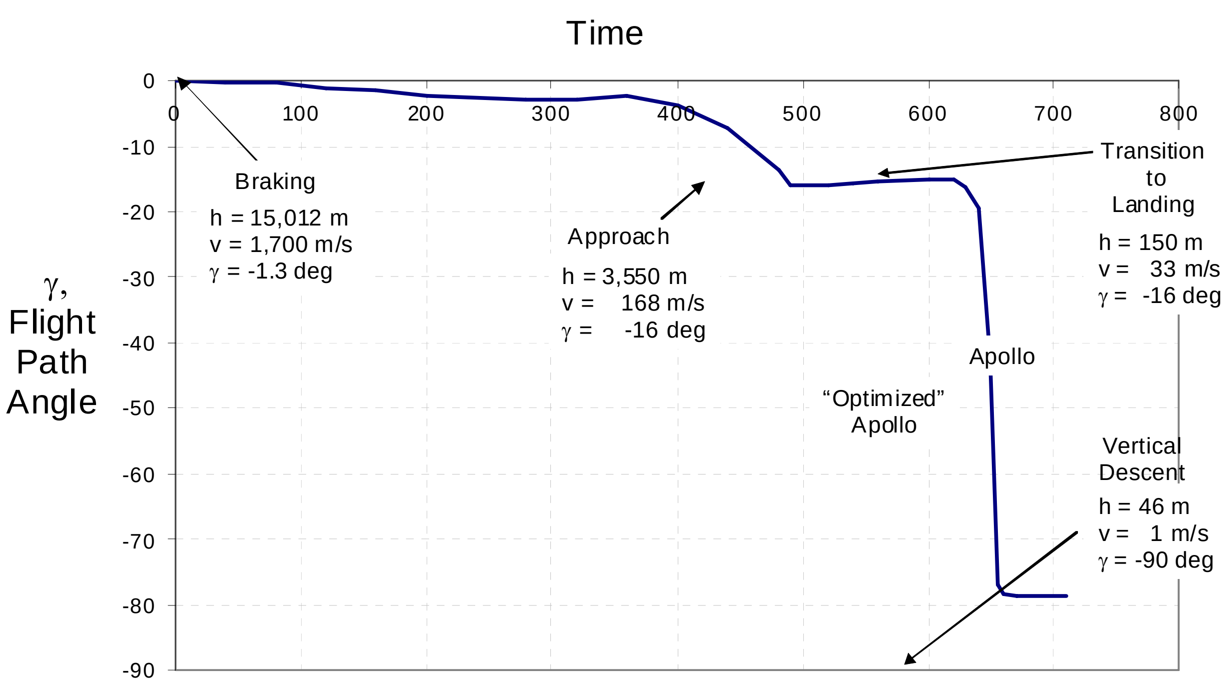


Figure 4. Comparison of Apollo and "Optimal" Apollo Trajectories.

optimization of the Landing Phase saves an additional 44 m/s. Comparing the phase ΔVs of Figure 4 to Figure 2, approximately 100 m/s were saved over the complete trajectory.

Eliminate the approach phase. The next trajectory simulation eliminates the approach phase of holding the minus 16-degree flight path angle (Fig. 5). This trajectory assumes a flight instrument landing; however, the final hover and the constant decent rate is retained for final pilot-in-the-loop landing. The direct path is more compatible with current LIDAR systems that would rather have a vertical -90 trajectory to reduce navigational errors. As shown in Figure 5, the trajectory takes a much more direct path and is shorter in total time. The ΔV was reduced to 1877 m/s, a 204 m/s (9.8 percent) savings over the Apollo baseline trajectory.

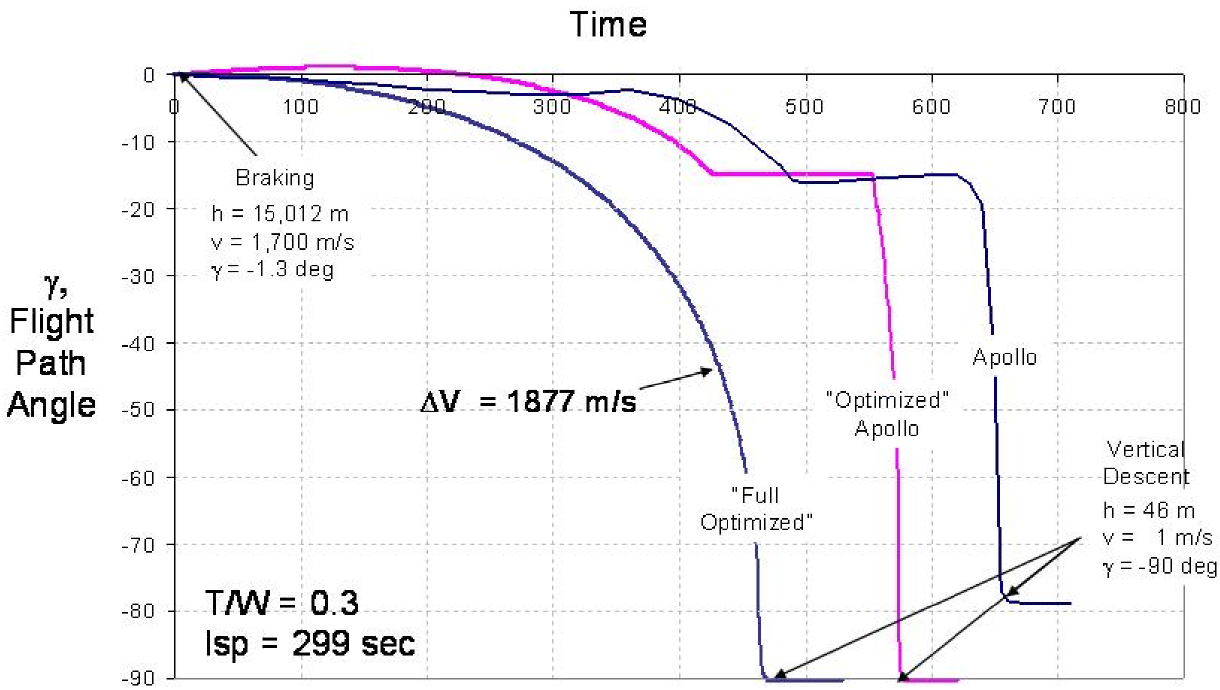


Figure 5. "Full Optimized" Descent

Compromise trajectory. The next trajectory (Fig. 6) considered was a compromise trajectory using a -45 degree flight path angle hold was considered. The flight path hold time was not changed in this trajectory. Resulting ΔV for this compromise trajectory is 2,015 m/s.

Thrust to Weight Trade. Figure 7 shows the results of changing the initial T/W_L for 1.8 (Apollo) to 4.6. Comparing Figure 7 to Figure 3, the flight-path hold trajectories with the required throttle in the last phases of flight, do not have the improvement with T/W_L that the unconstrained "optimal" cases do. However, it should be noted that for all cases, the improvement in trajectory ΔV must be traded with the additional mass of the engines for final thrust-to-weight ratio selection.

Approach Hold Time. The flight path hold time is determined by the time a pilot needs to ascertain that the landing site is clear for landing or there is a hazard requiring avoidance maneuvers; thus, in order to determine the observation and response times, pilot-in-the-loop and operations support simulations need to be conducted.

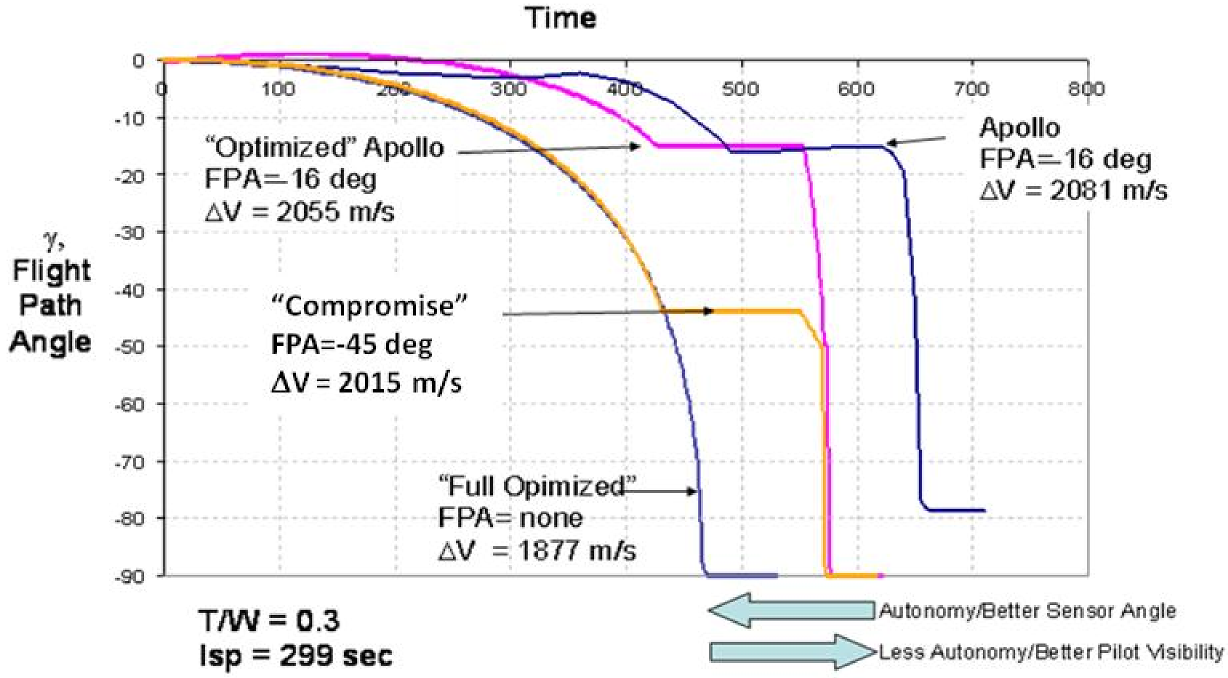


Figure 6. Compromise Trajectory comparison

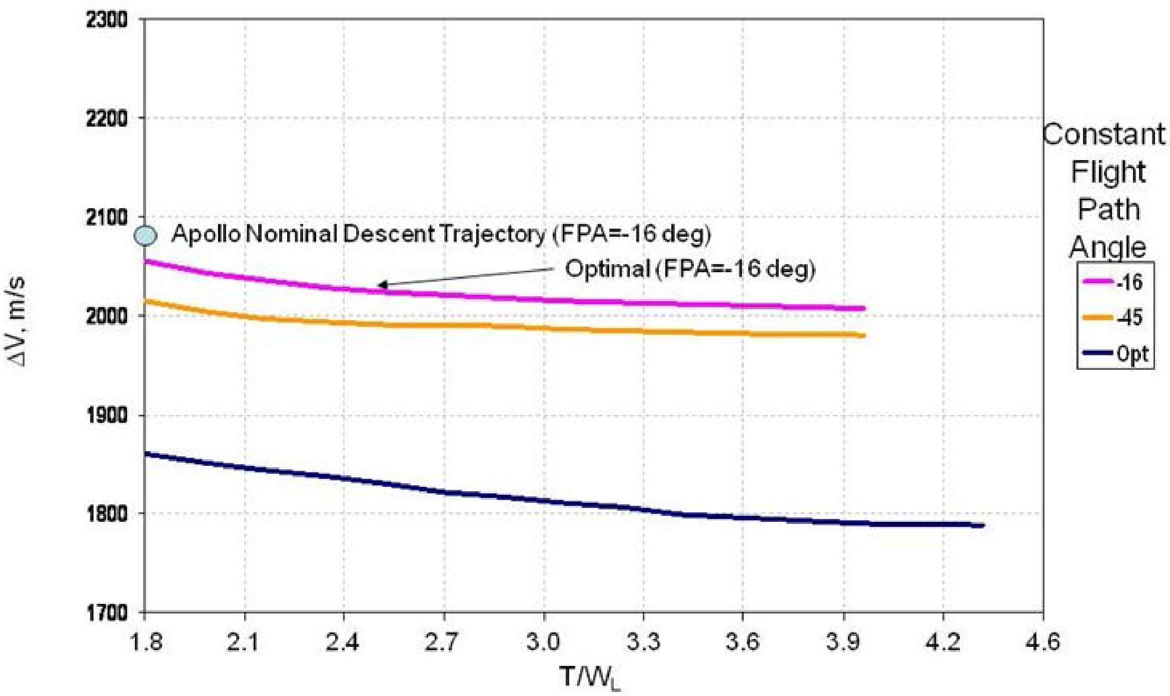


Figure 7. T/W Trade

Comparing the "Full Optimized" trajectory in Figure 6 with the -45 degree flight path hold trajectory, the trajectories follow the same path until the approach phase is reached. The impact of varying the length of flight path hold time is show in Fig. 8 where the Apollo hold time was 120 seconds. As shown in the chart, there is approximately a 50 m/s difference in the ΔV between the Apollo hold time for the pilot and the no hold time required for full autonomous flight assuming that full instrument scans can be made and hazard avoidance maneuvers can be initiated during the unaltered optimized trajectory.

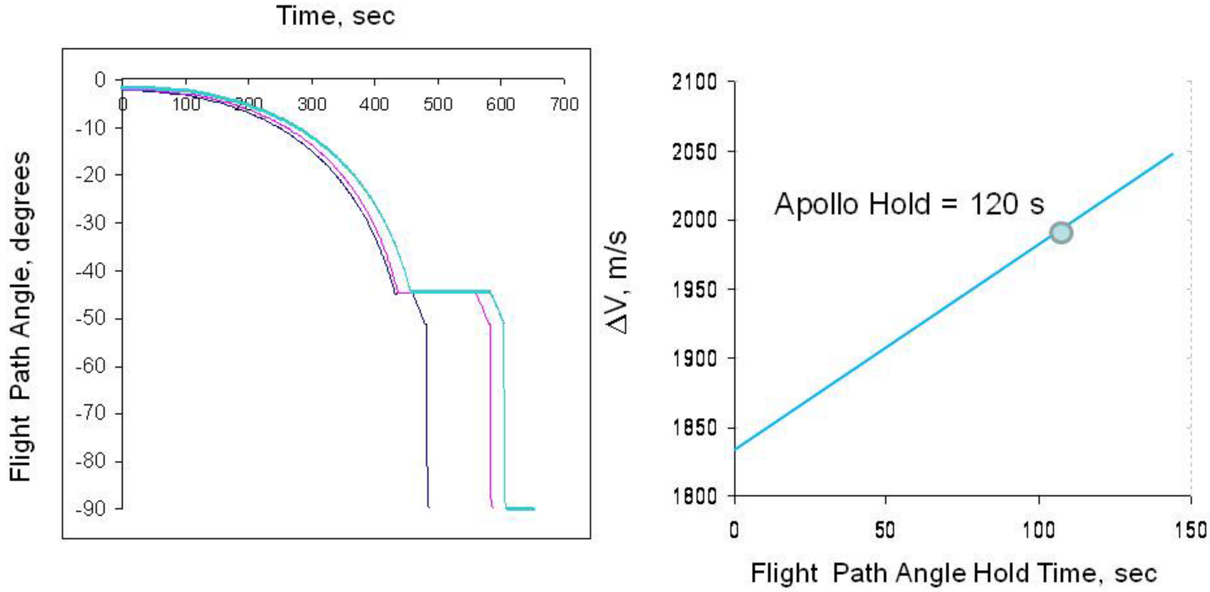


Figure 8. Flight Path Angle Hold Time.

Final Descent Vertical Distance and Time. The final phase of the trajectory is the hover and slow descent rate for landing. As mentioned earlier in the report,[9] the Apollo Vertical Descent distance was extended from 20 to 46 m (20 to 46 seconds on -1 m/s vertical decent velocity) in order to provide extra time and distance for the final landing due to pilot constraints. In addition, for the Apollo 11 pre-mission planning, an extra 17 m (17 seconds) was added for additional margin. Figure 9 shows the impact of the Vertical Descent distance (and descent time assuming a 1 m/s descent rate) has on trajectory ΔV. This ΔV performance requirement for vertical descent is all gravity losses determined by

$$\Delta V_{\text{gravity losses}} = g_L\, t = 1.622 \text{ m/s}^2 \cdot t$$

On the Apollo missions, the final descent maneuver was actually flown by several iterations of hover, pilot observation, pilot maneuver decision, and partial descent. On the first mission, astronaut Neil Armstrong observed boulders at the landing site and diverted until a clear site was found. As shown in Table 2 (Apollo 11), 98 percent of the total on-board propellant was burned and 6 percent more propellant was used than determined by the nominal trajectory. Commander Neil A. Armstrong's comments on his landing maneuver were the following - "I [was] just absolutely adamant about my God-given right to be wishy-washy about where I was going to land."[8]

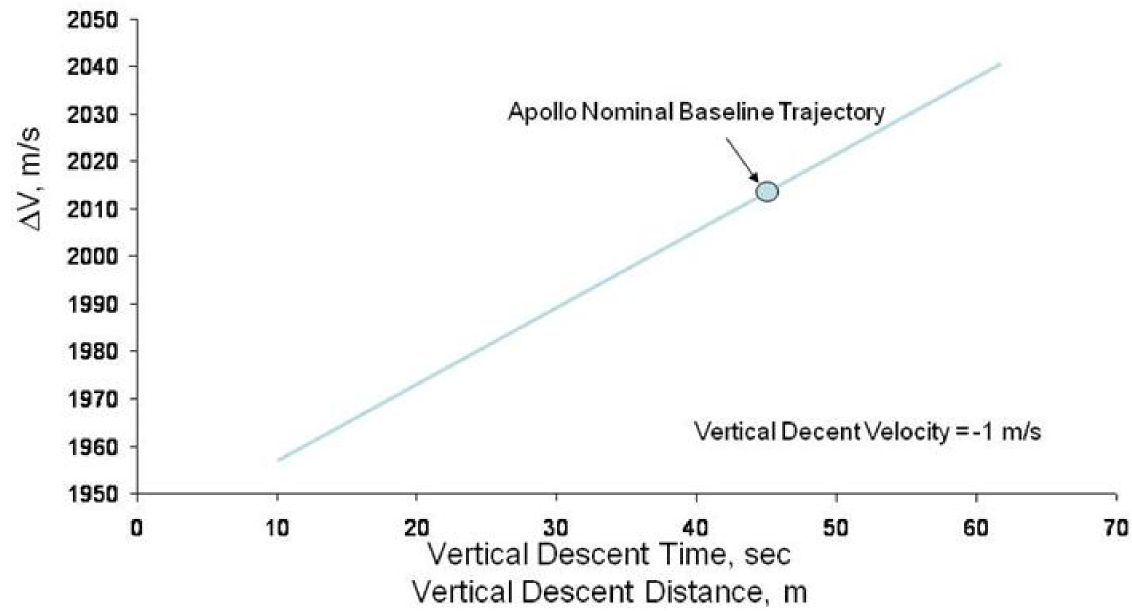


Figure 9. Performance Requirements for Final Vertical Descent.

Site Redesignation. During the nominal Apollo descent flight, the crew had 120 seconds of approach phase. During the approach phase the landing site is visible and the crew can determine if the target landing site is safe for landing. In the event that the landing site is deemed unsafe, a new landing site would be chosen and a redesignation trajectory would be flown.

Table 3 shows the redesignation landing footprint options of a 25 and 50 meter radius circle. Also, the redesignation impact ΔV is shown for making the landing change for 110, 50, and 0 seconds from the end of the approach phase. The impacts show that an early redesignation decision can actually save ΔV (negative ΔV in Table 3) because the 120 second approach phase (flight path angle hold) is terminated early to start the redesignation and landing maneuvers. For the uprange and cross range cases at 50 seconds, extra powered braking performance is required. At time to landing equal to 0, the assumption is that the required ΔV is all gravity loss and that the diversion velocity is 8 m/s, the same diversion velocity as Apollo. As shown in the previous section, the required ΔV is simply a function of time (or distance divided by diversion rate). From the table, it is shown that no extra ΔV is required for redesignation if the landing site hazard can be detected early enough. As shown in Table 2, the Apollo requirement for redesignation was 19 m/s where the assumption was a much larger redesignation footprint of 305 meter radius. In this present study, the landing site topography would be very accurately defined with the Lunar Reconnaissance Orbiter; thus a smaller redesignation footprint circle was assumed in the analysis.

Redesignation ΔV Impact m/s			
Time to Landing (sec)	**110**	**50**	**0**
Uprange			
25 meter	**-7.26**	**0.24**	**5.07**
50 meter	**-6.96**	**5.44**	**10.14**
Downrange			
25 meter	**-7.76**	**-2.76**	**5.07**
50 meter	**-8.06**	**-3.56**	**10.14**
Crossrange			
25 meter	**-7.76**	**6.24**	**5.07**
50 meter	**-7.56**	**7.34**	**10.14**

Table 3. Redesignation ΔV Requirements, m/s

Instantaneous Impact Point. In the present lunar exploration scenario, two types of missions are planned: short stay 7-day missions (Apollo had 3-day missions) at various landing sites on the moon and support of a continuous stay at an outpost using 180-day missions requiring both cargo only and human/cargo payloads. For these outpost

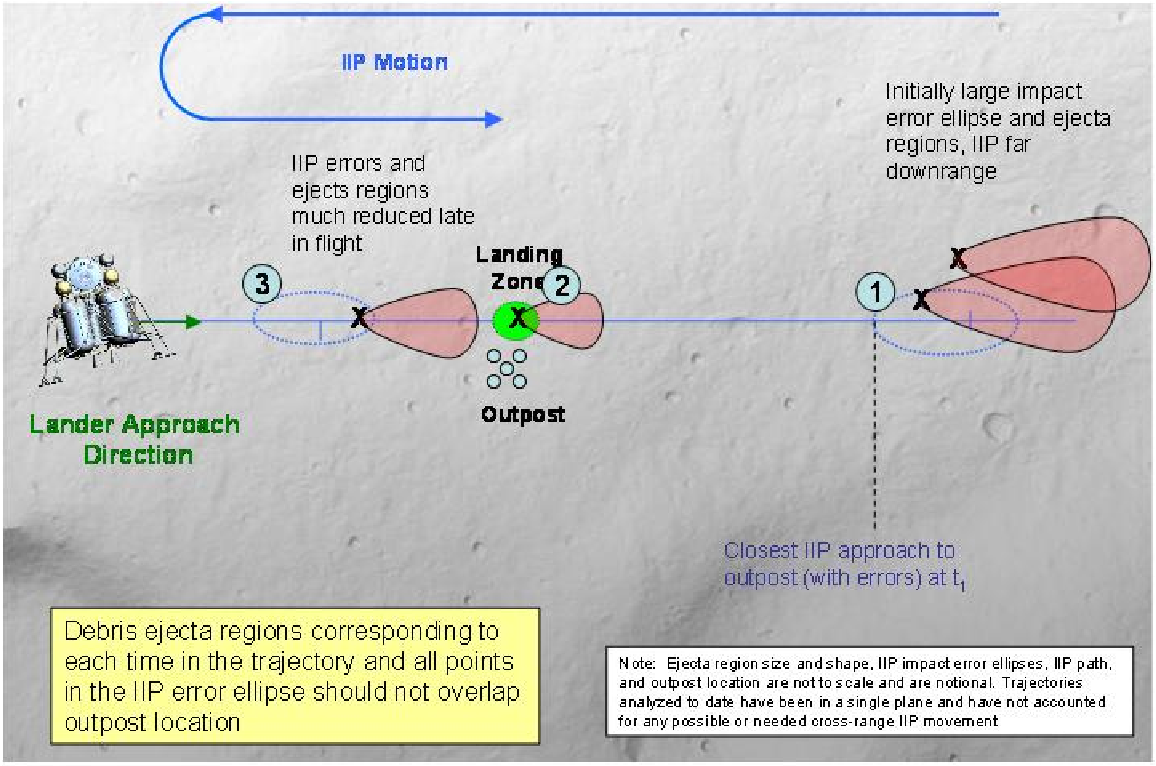


Figure 10. Definition of the Instantaneous Impact Point

missions, there is a concern about the safety of the outpost if a lunar lander loses power on the approach where the lander may either impact the outpost or contaminate the outpost with lunar regolith ejecta if the lander crashed in near the outpost.

The landing point for an all engine shutdown is called the instantaneous impact point (IIP). Figure 10 illustrates the IIPs relative to the planned landing zone. Initially, during the braking stage, the IIP would be downrange of the landing zone as illustrated by (1). As the lander approaches the landing zone, the IIP approaches and passes over the landing zone (2). The IIP is then uprange of the landing zone, moving further away on the uprange side of the zone, and then approaching the zone until landing.

Figure 11 shows two trajectories from Figure 6, the "Full Optimize" and the "Compromise" :where the approach flight path hold is -45 degrees. For both trajectories, the IIP starts on the right hand side of the figure. As the lander approaches the landing sight, the IIP gets closer to the landing zone on the downrange side until the IIP is at the outpost. The lander IIP then switches to the uprange, gets further away, and then again approaches the landing zone from the uprange side.

A strategy to mitigate the impact (no pun intended) is to have the lander dogleg into the landing zone by staying a

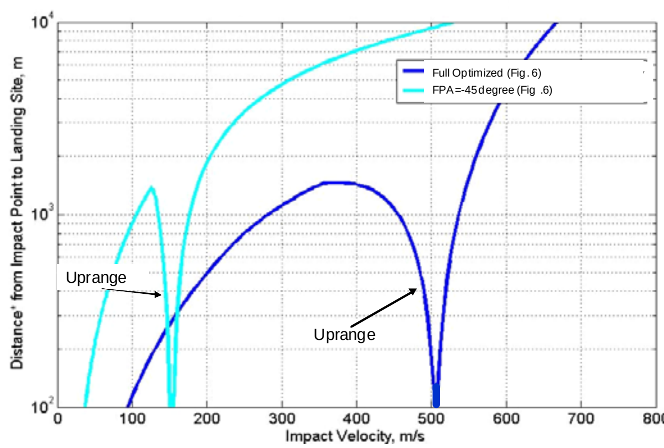


Figure 11. Instantaneous Impact Point.

constrained distance from the cross range side of the landing zone. Figure 12 shows four trajectories with this dogleg maneuver for various offset distances.

Figure 13 shows the performance requirement of distance offset and dogleg into the landing zone. The performance penalty for this maneuver is approximately 19 m/s ΔV for a 1 km cross range maneuver.

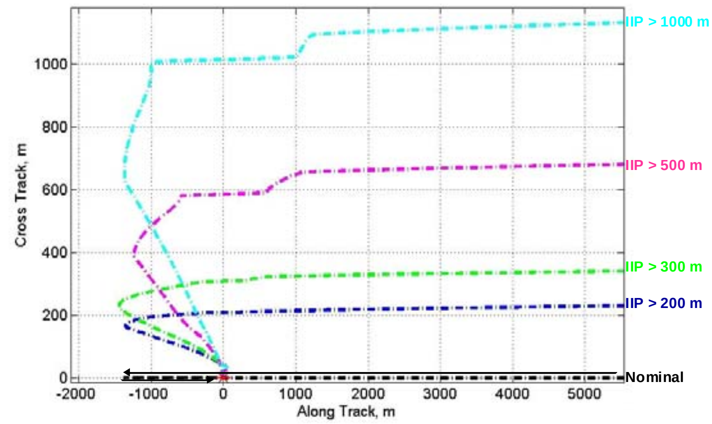


Figure 12. Instantaneous Impact Point Trajectories.

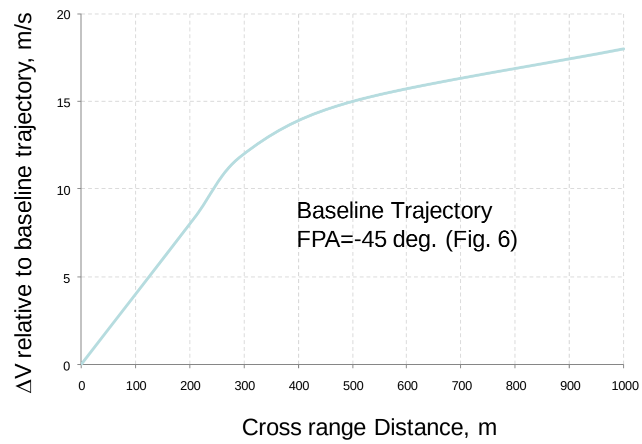


Figure 13. Impact of cross range on ΔV.

IV. Results and Discussion

Table 4 shows the conceptual performance ΔV recommendations based on the current conceptual state of the design and supporting analyses. The recommendations are given in terms of conservative, nominal, and optimistic. The conservative performance is Apollo based with the addition of 19 m/s for a 1 km cross range capability to dogleg into the outpost landing site. The (extremely) optimistic recommendation is not really a recommendation but provides an absolute minimum ΔV requirement for reference. As shown, the nominal is between the conservative

	Assumptions			ΔV, m/s		
	Conservative	Nominal	Optimistic	Conservative	Nominal	Optimistic
Nomial Trajectory	Pre-Mission Apollo	FPA=-45	Optimized	2081	2015	1877
T/W	1.8	2.4	3.0	0	-20	-30
Dispersions, ΔV m/s	53	27	0	53	27	0
Approach Time, sec	120	60	0	0	-10	-20
Vertical Descent Time, sec	46	23	12	0	-37	-55
Vertical Descent Margin, ΔV m/s	17	8	0	27	14	0
Redesignation footprint radius, m	305	50	25	0	15	8
Instantaneous Impact Cross Range, m	1000	500	200	19	10	5
Hardware Uncertainties, ΔV m/s	25	12	0	25	12	0
Additional Margin, ΔV m/s	57	28	14	57	28	14
			Total Mission ΔV, m/s	2262	2053	1799

Table 4. Conceptual Mission Planning Recommendations.

and optimistic recommendation and represents a starting performance ΔV assumption for the initial configuration development studies.

V. Conclusions

Various optimal descent trajectories were analyzed that include direct minimum energy, constrained case for current sensor technology, and the Apollo trajectory. Trade studies were conducted to determine impacts of mission assumptions and pilot in the loop and sensor flight demands. For mission design, the Apollo pre-mission methodology was applied to account for known contingencies (hardware, instrumentation known uncertainties) and unknown unknowns. The mission Delta-V's are presented in a risk form of conservative, nominal, and optimistic range where 90 percent of Delta-V was derived by detailed trajectory analysis, and the other 10 percent was derived from a qualitative analysis from Apollo 11 pre-mission planning. The recommendations for the Delta Vs are the following: conservative (Apollo derived) (2262 m/s), nominal (2053 m/s), and optimistic (1799 m/s); however the optimistic value represents an absolute minimum requirement for reference. Because of the qualitative nature of the some of the results, the degree of autonomy assumed, the additional safety considerations for a lunar outpost, and the impact of advanced instrumentation, additional in-depth analyses are required to refine the current recommendations.

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