Goddard Space Flight Center

LUNAR Reconnaissance Observer
ΔV Budget Estimate

Provided by

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Baseline Estimates Of ΔV Requirements For The Lunar 2008 Mission

An initial analysis was performed to assess the required impulsive DV budget, associated fuel allocation from finite maneuver modeling, and stationkeeping to meet both a 1year and 1 _ year mission life.

Assumptions and results: The nominal mission orbit was chosen as a 50 km circular orbit. A minimum energy direct transfer trajectory was use for this analysis and further analysis will address the use of weak stability transfers and gravity assisted transfers. Minimum energy transfers place the apoapsis at the lunar distance. Calibration of the propulsion system was not performed, so that errors from maneuver were not included in any propagation. No navigation errors were assumed and no margins are used in results. No fuel was held for end-of-mission demise. ΔV Results are reported for launch error corrections, lunar orbit insertion, stationkeeping, and placement into a frozen orbit at end of life. A spacecraft wet mass of 1000 kg was used.

Launch and Cislunar Transfer

A launch date of April 17, 2008 was chosen. Using an orbit that is Earth fixed with respect to the ETR launch site, a launch time and coast time that represents the time between the spacecraft separation and the injection onto the cis-lunar trajectory. The time between end of powered flight and insertion into the Earth parking orbit of 185 km circular was modeled as a curve fit to a launch profile. Figure 1 shows the cis-lunar transfer and figure 2 shows the ground track of this launch analysis.

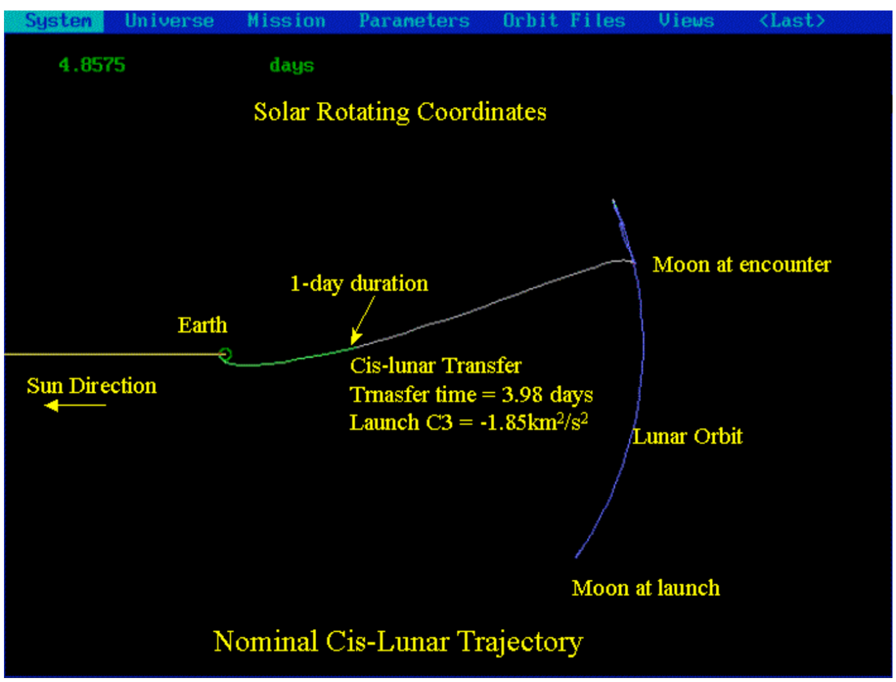

Figure 1. Cis-Lunar Transfer

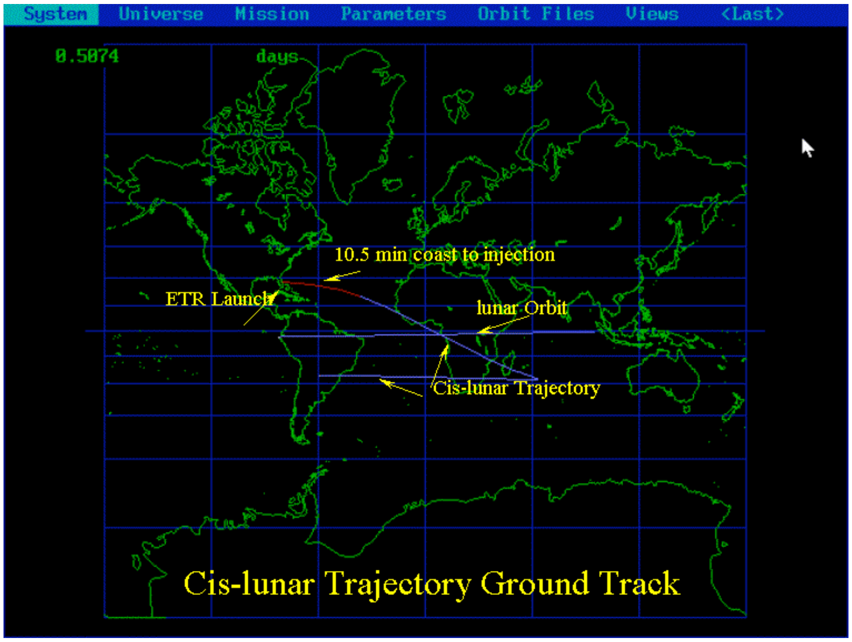

Figure 2. Ground Track.

The payload size for this launch is dependent upon the launch vehicle and the required energy to achieve the cis-lunar transfer trajectory. The energy is measured as launch C3 in units of km^2/sec^2. Using the information on the KSC web site for assessing payload mass per C3. Figure 3 lists mass vs. launch vehicle. Masses range from approximately 650kg to 1400kg.

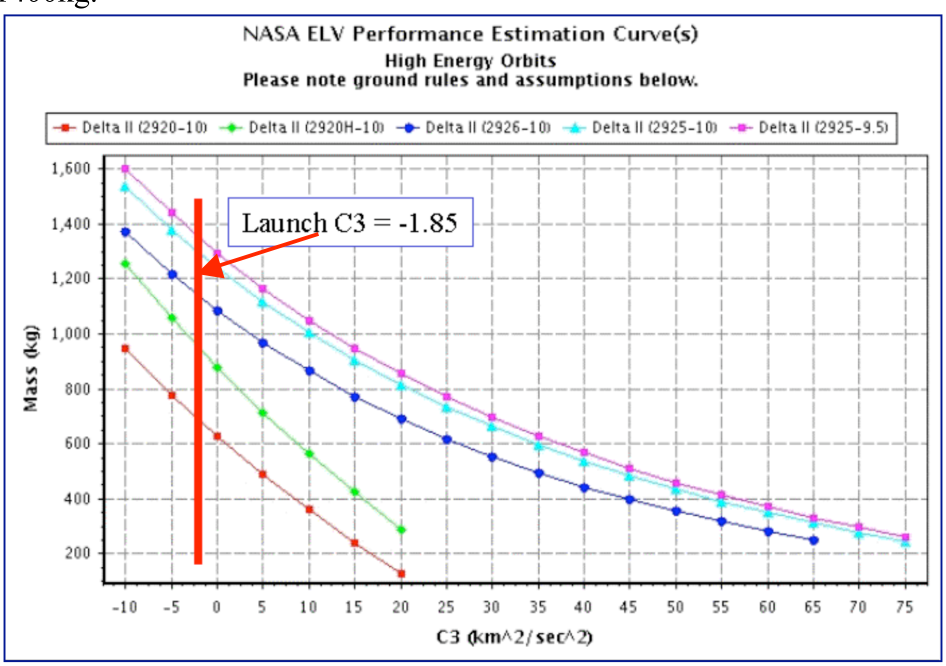

Figure 3. Launch Vehicle payload mass for C3 = $-1.85 km^2/s^2$

A simple launch contingency was approximated by assuming a launch under burn of -9m/s for a 3s value using a Star 37 or 48 solid upper stage. A maneuver correction was performed 1 day after the cis-lunar injection and resulted in a correction of 70m/s. Since the launch vehicle / upper stage is unknown at this time, it is recommended that launch ΔV budget of 75m/s be used to cover additional penalties and launch window delays. Figure 3 shows the effect of a –9m/s launch vehicle error on cis-lunar transfer.

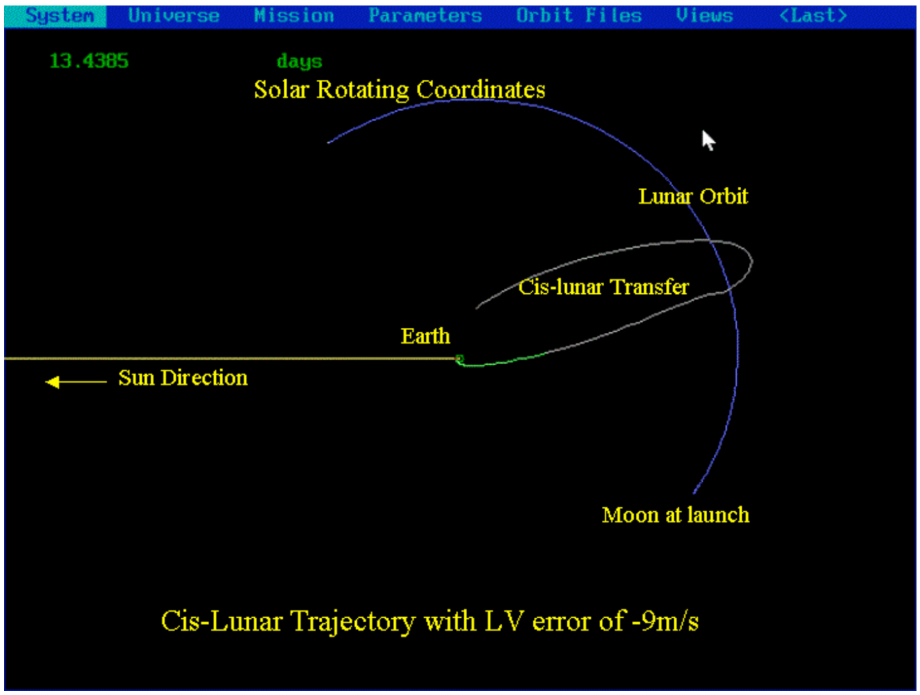

Figure 3 . Launch error impact on cis-lunar transfer

Insertion to Mission Orbit

The cis-lunar transfer achieves a periselene distance of 100km and takes a flight time of 3.9872 days. A lower target capture altitude can be used, but since the effect of finite burns results in a reduction in the periapsis altitude during the finite burn, a higher insertion orbit altitude was chosen for the initial capture. This also allows contingency for cis-lunar maneuver errors and navigation errors.

The launch mission orbit was achieved by using three maneuvers to capture and place the spacecraft into the 50km circular orbit. The first maneuver captured into an orbit with a 12 hour period. The second maneuver reduced the period to 6 hours. The third maneuver circularized at 100 km altitude. Two trim maneuvers are performed to attain the mission orbit of 50 km. Table 1 list the impulsive ΔVs and post burn orbit parameters. A total ΔV of 853.3 m/s is required for orbit capture and attainment of the 50 km mission orbit.

Using the impulsive ΔV as the first guess, a finite maneuver profile was developed using one 440N (100lb) thruster. A smaller system using 88N (four 5lb thrusters) was also assessed. Note that use of either 88N or 44N thrusters for the first capture maneuver resulted in performance that either did not capture into lunar orbit or resulted in impact with the lunar surface during the finite maneuver. Table 1 also list the finite maneuver information of ∆V, fuel used, maneuver durations based on event centered (usually periapsis) and post maneuver orbit. Figure 4 shows the capture into the mission orbit.

Table 1. Impulsive ∆V

	Impulsive DV (m/s)	Post DV Orbit (km)	440N Finite ∆V (m/s)	440N Finite Fuel (kg)	440N Finite Duration (min.)	Post Maneuver Orbit (km)
Maneuver 1	333	1838x10600	329	142.8	11.7	1830x10721
Maneuver 2	112	1838x5930	112	43.2	3.5	1830x5989
Maneuver 3	385	1838 circ	393	136.9	11.2	1787x1838
Maneuver 4	11	1788 x 1838	11	3.45	0.28	1787x1789
Maneuver 5	12.3	1788 circ	n/a	n/a	n/a	n/a

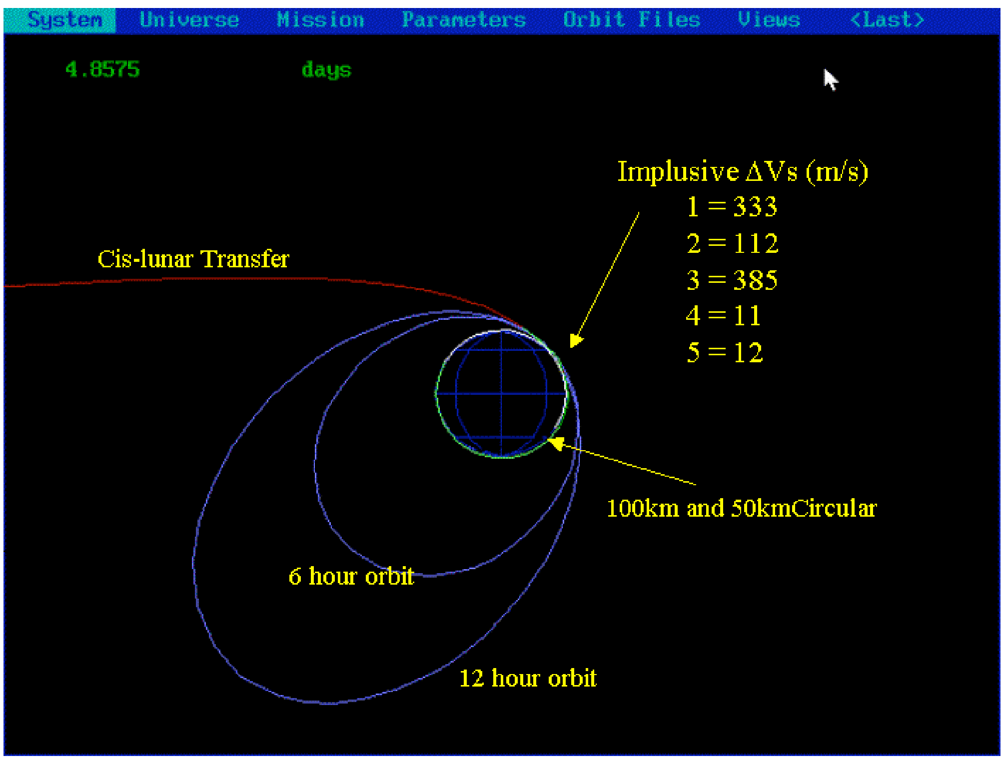

Figure 4. Capture Into The Mission Orbit

Stationkeeping

A stationkeeping / maintenance assessment was also made. The full 100 degree and order lunar potential model from Lunar Prospector was used for stationkeeping. The initial orbit was propagated for a 28-day duration and a maneuver was then performed to re-initialize the initial orbit condition of eccentricity and argument of periapsis. The method for quasi-frozen orbit maintenance was used. A seen in Figure 2, the polar plot shows the motion of the eccentricity and argument of periapsis. It was assumed that an altitude requirement of 50 km +/- 15 km would be used. Additional analysis will be performed to assess tighter altitude control, such as +/- 5 km. The used of the +/-15 km altitude control allows the mission to reduce the maneuvers to one per month. Figure 4 and 5 shows the altitude variation and the prediction of the eccentricity and argument of periapsis growth over one month.

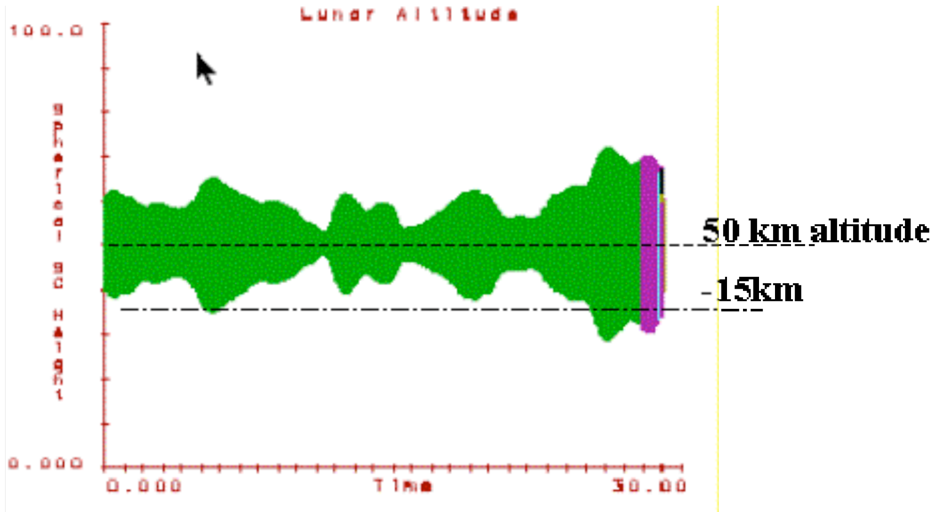

Figure 4. Altitude variation over one month

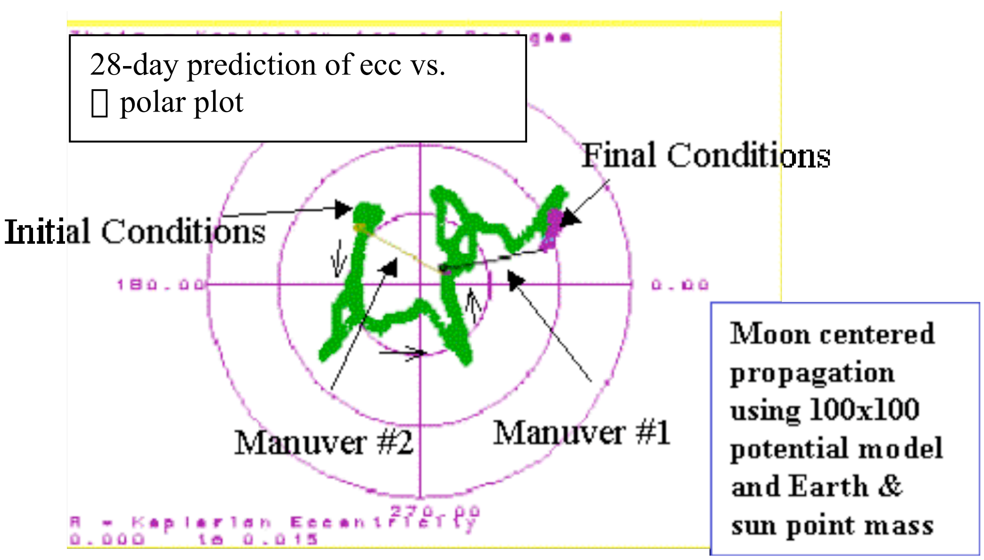

Figure 5. Precession of the eccentricity and argument of periapsis for a 100x100 potential model

Given the prediction of the orbit, the monthly stationkeeping maneuver is performed as two burns, one to 'circularize' and the second to target the eccentricity (0.006) and argument of periapsis (ω= 165 degrees). The ΔV cost is 11.64 m/s. Yearly stationkeeping cost are 11.64 m/s * 12 months = 139.7 m/s.

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Mission □V requirements

Given the assumptions and initial orbit, a total mission □V (rounded to the nearest whole number) is shown in Table 2. A total of 326.4 kg of fuel is used for the insertion assuming an initial 1000kg spacecraft wet mass. This initial mass was used since the launch vehicle error correction may not need to be performed. The fuel used for the stationkeeping based on the rocket equation is 3.63 kg per maneuver (of 11.64 m/s). For one year the stationkeeping fuel would be 43.44 kg and for 1 and _ year it would be 65.25 kg. The transfer maneuver would require approximately 14kg. The total fuel without launch vehicle error correction is then 383.8 kg for a 1 year mission and 405.6 kg for a 1 and _ year mission.

Table 2. Total Mission □V

	1-Year Mission □V Budget (m/s)	**1-Year Mission Fuel Budget (kg)**	**1 & 1/2 Year Mission □V Budget (m/s)**	**1 & 1/2-Year Mission Fuel Budget (kg)**
LV error correction	75	34	75	34
Insertion	854	326	854	326
Stationkeeping	140	43	210	65
Transfer to frozen orbit	50	14	50	14
Total DV Budget with EOL Frozen Orbit	**1119**	**417**	**1189**	**439**

For comparison, the Lunar Prospector mission required 58 m/s for launch vehicle error correction, 898.5 m/s for insertion into the 100km circular orbit. Stationkeeping maneuvers can be scheduled similar to those of lunar prospector by using the 28 days cycle so that the orbit plane is perpendicular to the ground line of sight for full coverage of all maneuvers.

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