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Mission Design for the Lunar Reconnaissance Orbiter

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MISSION DESIGN FOR THE LUNAR RECONNAISSANCE ORBITER

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The Lunar Reconnaissance Orbiter (LRO) will be the first mission under NASA's Vision for Space Exploration. LRO will fly in a low 50 km mean altitude lunar polar orbit. LRO will utilize a direct minimum energy lunar transfer and have a launch window of three days every two weeks. The launch window is defined by lunar orbit beta angle at times of extreme lighting conditions. This paper will define the LRO launch window and the science and engineering constraints that drive it. After lunar orbit insertion, LRO will be placed into a commissioning orbit for up to 60 days. This commissioning orbit will be a low altitude quasi-frozen orbit that minimizes stationkeeping costs during commissioning phase. LRO will use a repeating stationkeeping cycle with a pair of maneuvers every lunar sidereal period. The stationkeeping algorithm will bound LRO altitude, maintain ground station contact during maneuvers, and equally distribute periselene between northern and southern hemispheres. Orbit determination for LRO will be at the 50 m level with updated lunar gravity models. This paper will address the quasi-frozen orbit design, stationkeeping algorithms and low lunar orbit determination.

INTRODUCTION

The Lunar Reconnaissance Orbiter (LRO) is the first of the Lunar Precursor Robotic Program's (LPRP) missions to the moon. LRO is a one-year duration reconnaissance mission to be flown in a low (50 km) lunar polar orbit. It will be launched on an Atlas V launch vehicle in late 2008. The spacecraft is a three-axis stabilized nadir-pointing platform with a total mass, including fuel, of approximately 1850 kg. Figure 1 shows the current configuration of the LRO spacecraft. Seven instruments will fly on LRO to provide exploration/science measurements to characterize future robotic and human landing sites, identify potential lunar resources, and document lunar radiation relevant to human biological response. The seven LRO instruments are:

- Lunar Orbiter Laser Altimeter (LOLA) – global topography at high resolution
- Lunar Reconnaissance Orbiter Camera (LROC) – targeted images of the lunar surface
- Lunar Exploration Neutron Detector (LEND) – map the flux of neutrons from lunar surface
- Diviner Lunar Radiometer Experiment (DLRE)- map the temperature of lunar surface
- Lyman-Alpha Mapping Project (LAMP) – observe lunar surface in far ultraviolet
- Cosmic Ray Telescope for the Effects of Radiation (CRaTER) – background space radiation
- Mini-RF – S and X-band synthetic aperture radar (SAR)

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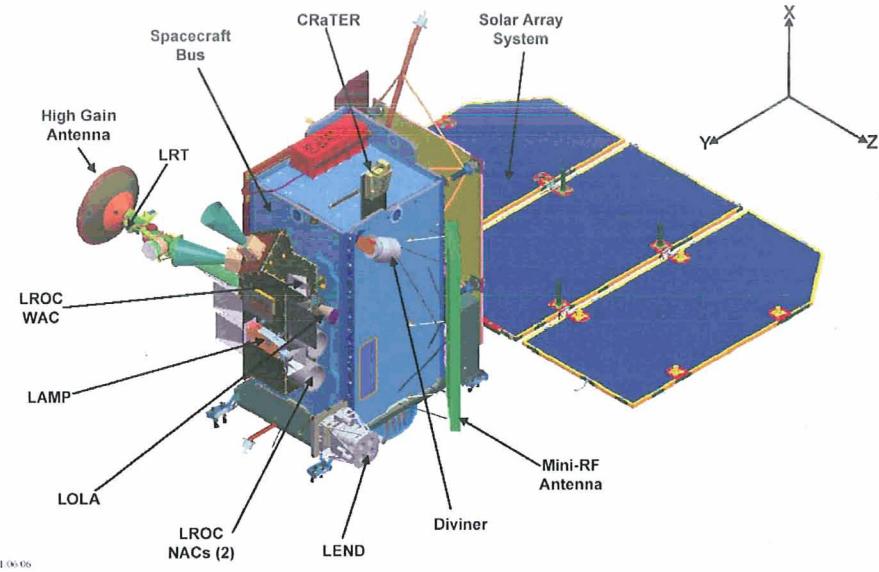


Figure 1: LRO Spacecraft

LRO will be launched into a minimum energy direct transfer to the Moon. After a four day transfer (other launch days may give a five day transfer), a series of Lunar Orbit Insertion (LOI) maneuvers will capture the spacecraft about the Moon and lower the altitude to a quasi-frozen orbit at 30×216 km altitude for commissioning. Commissioning phase will last up to sixty days. At the end of commissioning, the spacecraft will be maneuvered into the final mission orbit, a 50-km mean polar orbit. LRO will remain in the mission orbit for one year. Figure 2 shows the primary phases of the LRO mission.

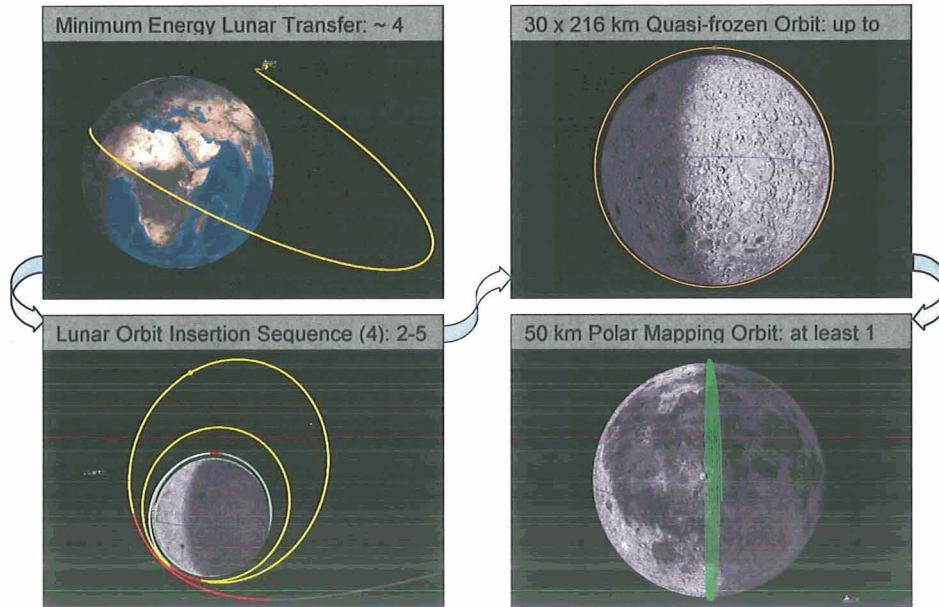


Figure 2: LRO Mission Phases

Once in mission orbit, the solar system can be viewed from a LRO perspective. Figure 3 depicts the motion of the Earth and Sun as seen from a fixed LRO orbit. By fixing the LRO orbital plane (vertical line through the Moon at the center), the Earth and Sun appear to circle the Moon. The Earth's period is the lunar sidereal period of 27.4 days. Over this period, the entire orbit plane is visible to Earth twice for about two days each. Part of the lunar orbit is occulted by the Moon during the rest of the time. The Sun's period about the orbit is one year. The spacecraft is in full Sun twice during the year for about two months each. During the rest of the year, LRO will experience eclipses of up to 48 minutes each orbit. Because of the single solar array on the spacecraft $-Y$ face, LRO will perform a yaw flip every six months to keep the Sun in the spacecraft $-Y$ hemisphere.

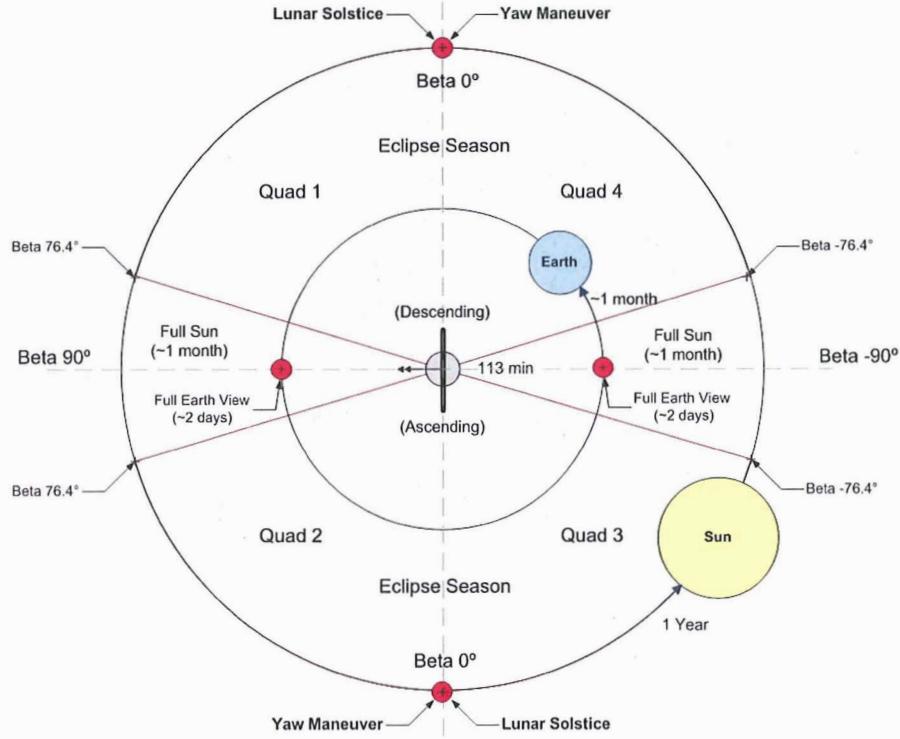


Figure 3: Solar System from LRO Perspective

The Flight Dynamics Analysis Branch (FDAB) at the Goddard Space Flight Center (GSFC) will provide all mission design, maneuver planning and orbit determination support for LRO. The FDAB has supported the last two NASA missions to the moon. The FDAB provided launch, maneuver planning, orbit determination and mission design support for Clementine¹ and Lunar Prospector². Clementine was a Naval Research Laboratory mission that flew in 1994 in a highly elliptical lunar orbit. Lunar Prospector (LP) was a Discovery class mission that launched in 1998. LP was in a low polar lunar mapping orbit at 100 km altitude for the nominal mission, then at 40 and 30 km for the six month extended mission.

LAUNCH WINDOW

LRO will be launched on an Atlas V 401 two stage launch vehicle (LV) on a direct lunar transfer. After Main Engine CutOff (MECO) -1, there will be a short coast phase to place the Transfer Trajectory Insertion (TTI) at the proper geocentric latitude to set the outgoing asymptote. At this point, the Centaur upper stage will re-ignite and place LRO onto the direct lunar transfer orbit.

The transfer time to the moon is a function of lunar phase and varies between about four and five days. The transfer trajectory is targeted for lunar insertion above the lunar south pole to a 90 deg lunar inclination and a post-insertion lunar periselene altitude of 216 km. The commissioning orbit has an argument of periapsis (ω) of 270 deg, so insertion over the lunar south pole minimizes the line-of-apsides rotation required for some contingency cases. The first lunar orbit insertion (LOI) maneuver will insert LRO into a 5-hr elliptical orbit with a ω of about 225 deg. After about four orbits, a series of additional LOI maneuvers are performed to reduce altitude, circularize the orbit and set the line-of-apsides for the commissioning orbit. After two months in commissioning orbit, a series of maneuvers are performed to achieve the mapping orbit at 50 km.

The lunar transfer trajectory is a minimum energy transfer to the moon. The launch C3 (energy), or equivalently the TTI ΔV , is a function of lunar phase. This varies primarily due to the eccentricity of the Moon's orbit. The transfer time of the minimum energy transfer varies too. There are two launch opportunities per day for a minimum energy direct transfer to the Moon: a short coast and a long coast. The two solutions have nearly the same TTI location but achieve lunar transfers in two different planes (shown in Figure 4). Each solution has a different launch time (the short coast solution has a later launch time) and a different coast time (the short coast solution has, obviously, the shorter coast time). LRO has no requirement to use exclusively either the long or short coast solution. However, due to LV restrictions, only one of the two opportunities per day will be used. The choice between the daily short or long coast solutions depends on many factors.

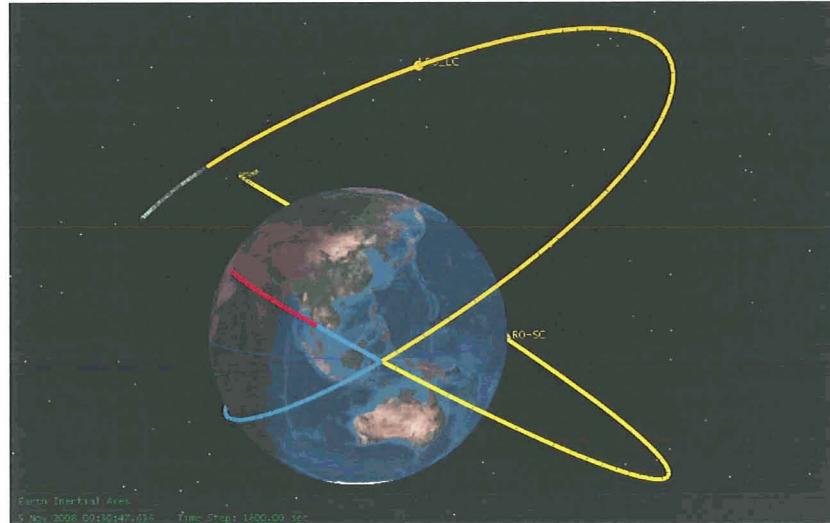


Figure 4: Short/Long Coast Solutions

The Moon's orbit is within 5 deg of the ecliptic plane. This places the sub-lunar point within about 28 deg of the Earth equator. As the Earth rotates, the sub-lunar point forms a line of constant latitude. This line of constant latitude varies over the lunar phase from about ± 28 deg. The TTI point is approximately at the anti-sub-lunar point because the lunar transfer trajectory is essentially a Holmann transfer to the moon. So if the sub-lunar point is at a positive latitude, the TTI must be at that negative latitude.

The LV will launch east from the Eastern Test Range at Kennedy Space Center. The first orbit node is a descending node shortly after spacecraft separation. For times of the month with a southern latitude TTI (northern latitude moon), a short coast time of up to $\frac{1}{4}$ orbit (or about 23

minutes after the node crossing) is required. The long coast solution would have a coast time of about $\frac{1}{4}$ to $\frac{1}{2}$ orbit. For times of the month with a northern latitude TTI (southern latitude moon), a short coast time of up to $\frac{3}{4}$ orbit is required. The long coast solutions would have a coast time of about $\frac{3}{4}$ to one orbit. Each of these regions is identified as an Argument of Latitude (AoL) quadrant and is shown in Figure 5. The line through the TTI latitude crosses the parking orbit twice. These are the short and long coast solutions for that day³.

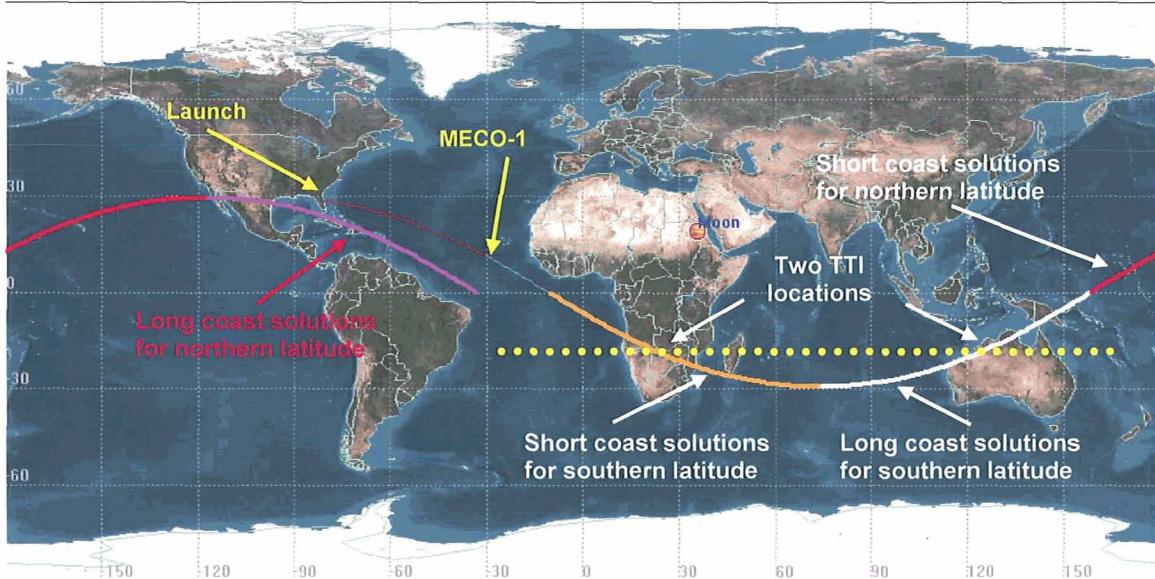


Figure 5: Short/Long Coast TTI Locations

LRO has a science constraint that further limits the launch window⁴. In order to identify regions of permanent shadow or lighting, these regions have to be viewed in extreme lighting conditions. The minimum energy transfer to the Moon gives a nearly constant beta-Earth angle. The beta-Earth angle is the angle between the lunar orbit plane and the Moon-Earth line. This beta-Earth angle is approximately 85 deg at lunar orbit insertion, regardless of lunar phase (see Figure 6). The beta-Sun angle at insertion is entirely dependent on lunar phase at the time of lunar arrival. The beta-Sun angle is the angle between the lunar orbit plane and the Moon-Sun line. The lunar orbit plane is essentially inertially fixed. Once in lunar orbit, the beta-Sun angle will process with the motion of the Earth about the Sun, or about one degree per day. The lunar pole is inclined about 1.4° to the ecliptic plane creating seasonal lighting conditions near the lunar poles. At the time of lunar solstice, the extreme lighting conditions are achieved (i.e. each of the lunar poles is in extreme light or extreme dark conditions). In order to fly directly over regions near the poles in the extreme lighting conditions, the beta-Sun angle must be near zero (i.e. the Sun must be near the lunar orbit plane). Thus, the requirement placed on LRO is that the beta-Sun angle at lunar solstice must be less than 20 degrees. Figure 7 shows the extreme lighting condition for summer solstice over the lunar south pole. The required lunar orbit plane would be a horizontal line in Figure 7. The beta-Sun angle at lunar solstice is set by the lunar phase at lunar arrival and the time between lunar arrival and lunar solstice, so it is entirely dependent on launch day. The beta angle at solstice constraint limits the LRO launch opportunities to one per day for a period of about three days every two weeks.

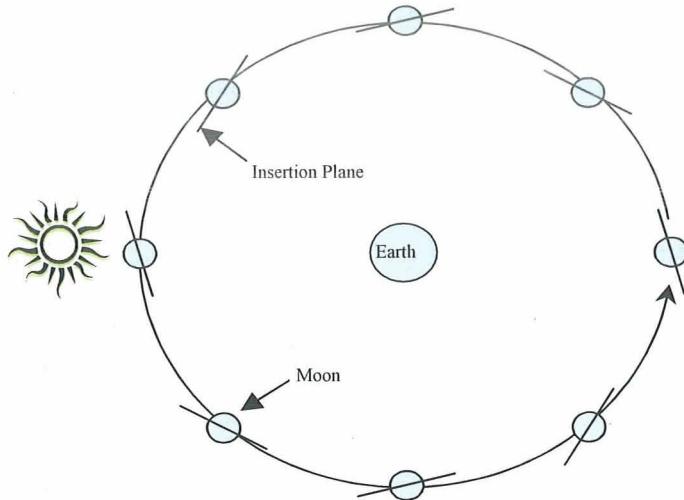


Figure 6: Beta-Earth Angle at Lunar Arrival

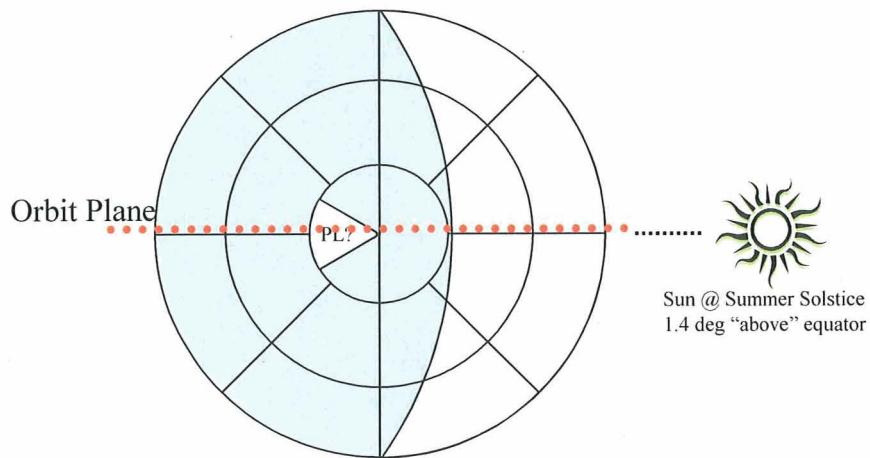


Figure 7: Extreme Lighting Condition at Lunar South Pole During Lunar Summer Solstice

An LRO requirement is to launch by the end of 2008. In the Fall of 2008, 14 launch opportunities, over five launch windows, have been identified. Each of the launch opportunities meets the beta angle at solstice constraint. The selection between long and short coast solutions each day is made based upon two criteria. First, the time to initial ground station acquisition after TTI is desired to be less than 10 minutes. Second, the total eclipse time during the transfer phase is desired to be less than 10 minutes. These two criteria generally lead to an obvious choice between the long and short coast solutions each day. These criteria preclude the selection of an AoL quadrant three solution since the time to first acquisition of signal (AOS) for these solutions is generally around 40 minutes. So for half the month, only one solution per day is available. Solutions that are well outside the above desired values are discarded and result in shorter launch windows. The launch opportunities are not all considered equal. The variation in ΔV cost for the first LOI maneuver is significant. The primary LRO launch day is October 28, 2008. This launch day is also near minimum LOI-1 ΔV cost. Launch opportunities two weeks later are significantly

more expensive in terms of LOI-1 ΔV . Table 1 shows the Fall 2008 LRO launch opportunities and key parameters.

Table 1: Fall 2008 LRO Launch Opportunities

	<u>Date</u>	<u>Local Launch Time</u>	<u>TTI Argument of Latitude (deg)</u>	<u>Transfer Trajectory Eclipse Time (min)</u>	<u>Time to 1st Ground Station AOS (min)</u>	<u>LOI ΔV (m/sec)</u>	<u>Beta-Sun Angle At Solstice (deg)</u>
1	10/28/2008	4:51	76.14	2.67	0.00	507.18	17.31
	10/29/2008	6:28	65.92	1.20	0.00	512.41	4.31
	10/30/2008	8:06	54.28	0.00	1.44	521.00	8.58
2	11/12/2008	4:44	240.48	0.00	3.31	552.81	7.83
	11/13/2008	6:37	228.57	0.00	4.61	557.81	8.47
3	11/25/2008	3:11	68.63	5.18	0.00	511.62	8.67
	11/26/2008	4:47	57.74	3.92	0.86	519.36	4.18
	11/27/2008	6:22	46.08	2.63	2.45	528.68	16.95
4	12/9/2008	22:12	295.17	0.00	10.80	550.35	18.03
	12/10/2008	22:33	304.37	0.00	9.36	547.90	3.02
	12/11/2008	5:44	221.66	0.00	5.47	569.82	16.47
5	12/22/2008	0:57	71.12	9.32	0.00	508.62	12.87
	12/23/2008	2:32	60.73	7.82	0.43	516.82	0.08
	12/24/2008	4:06	49.45	6.57	2.02	526.76	12.87

As a consequence of the beta angle at solstice constraint, the secular inclination drift during mission orbit is at a maximum. The lunar inclination of the mission orbit varies periodically due to lunar non-spherical gravitation effects. The amplitude of this variation is about ± 0.15 deg in osculating inclination. However, the mean inclination exhibits a secular drift due to third-body effects, primarily the Sun. This secular drift rate is highest, about 0.45 deg/year, for launch opportunities that meet the beta angle at solstice constraint. This drift rate is nearly zero for opportunities out-of-phase with the LRO launch window every two weeks. LRO will target the initial mission orbit inclination such that the mean inclination will cross 90 deg about six months into the one year nominal mission. This requires a biasing of the initial mean inclination of up to 0.25 deg. Figure 8 shows the mean inclination drift over the nominal mission for the primary launch date. LRO does not nominally plan to do any inclination maneuvers during the mission.

MID COURSE CORRECTIONS

LRO has only one planned mid-course correction (MCC) maneuver. This maneuver is planned for 24 hours after launch and is primarily to correct for 3- σ launch vehicle errors. LRO requires the velocity accuracy at launch vehicle separation to be within 3 m/sec. This uncertainty translates into a ΔV budget for MCC-1 of 28 m/sec.

No other MCC maneuvers are planned for LRO during the transfer phase. A placeholder has been inserted for an MCC-2 maneuver at launch plus 60 hours. MCC-2 would only be required with a nearly 3- σ launch vehicle error and a nearly 3- σ performance of the propulsion

system during execution of MCC-1. Even under these conditions, MCC-2 would only require 4 m/sec to retarget for lunar orbit insertion.

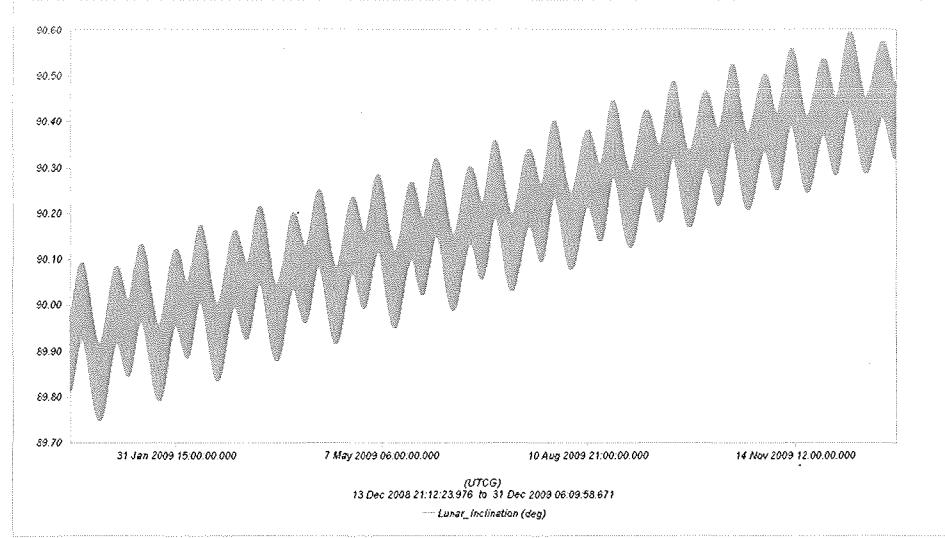


Figure 8: Mean Lunar Inclination over Nominal Mission for Primary Launch Date

LUNAR ORBIT INSERTION

The lunar orbit insertion sequence consists of a large LOI-1 maneuver followed by a series of smaller LOI maneuvers to place the spacecraft into the commissioning orbit. The LOI-1 maneuver is nominally 38 minutes long using all four 80-N insertion thrusters. This maneuver captures LRO into a 5-hr orbit about the Moon with a periselene altitude of 216 km. LOI-1 is planned to be such a long maneuver in order to minimize third-body effects on larger capture orbits and to allow multiple contingency options during LOI-1. LRO will remain in the capture orbit for about one day while the mission operations team assesses the spacecraft and propulsion system performance. Two additional LOI maneuvers are required to lower aposelene and eventually circularize the orbit at 216 km. Each of these LOI maneuvers is limited to 12 minutes in duration to minimize finite burn losses. A final LOI maneuver is required to set the proper argument of periapsis and periselene altitude of the commissioning orbit. Figure 9 shows a graphic of the LOI capture sequence for LRO.

The design of LOI-1 allows recovery from many contingencies that may occur during LOI-1. In general, only one half of the thrust planned for LOI-1 is required to capture about the Moon. In the event of a processor reset, and the complete loss of thrust, during the LOI-1 burn, there is ample time to reboot, reacquire and restart the burn. This results in a larger, but completely stable, capture orbit and has no ΔV penalty or other impact on the nominal mission. This capture scenario is depicted in Figure 10.

A LOI-1 restart is not always required. If the LOI-1 maneuver is interrupted after 20 minutes, a stable lunar capture orbit is already achieved and no restart will be attempted. The high aposelene altitude will be reduced with additional shorter LOI maneuvers at no ΔV penalty. If the LOI-1 maneuver is interrupted prior to 20 minutes, a restart will be attempted. If the restart is unsuccessful, and less than 10 minutes were achieved from LOI-1, then no lunar capture will occur. LRO will fly by the Moon and swing up out of the Ecliptic plane. With no further maneuvers, LRO will depart the Earth-Moon region and never return. However, recovery from

this scenario is possible with a large Deep Space Maneuver (DSM) within ten days of the lunar swingby. With a successful DSM, LRO could re-encounter the Moon and capture into a low polar orbit, though not as low as the nominal mission orbit. This would allow LRO to meet minimum mission requirements at at least the south pole.

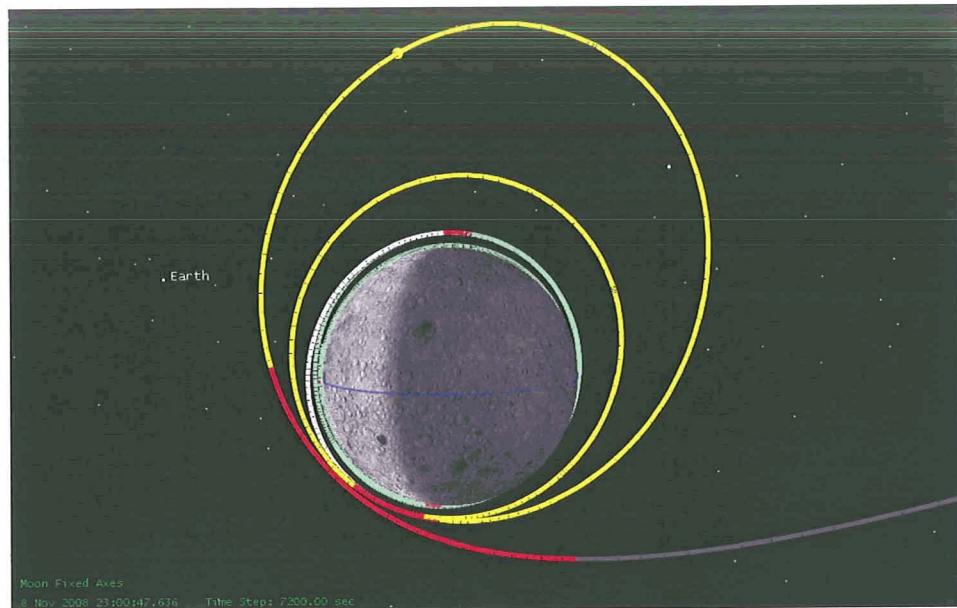


Figure 9: LRO Lunar Capture Sequence

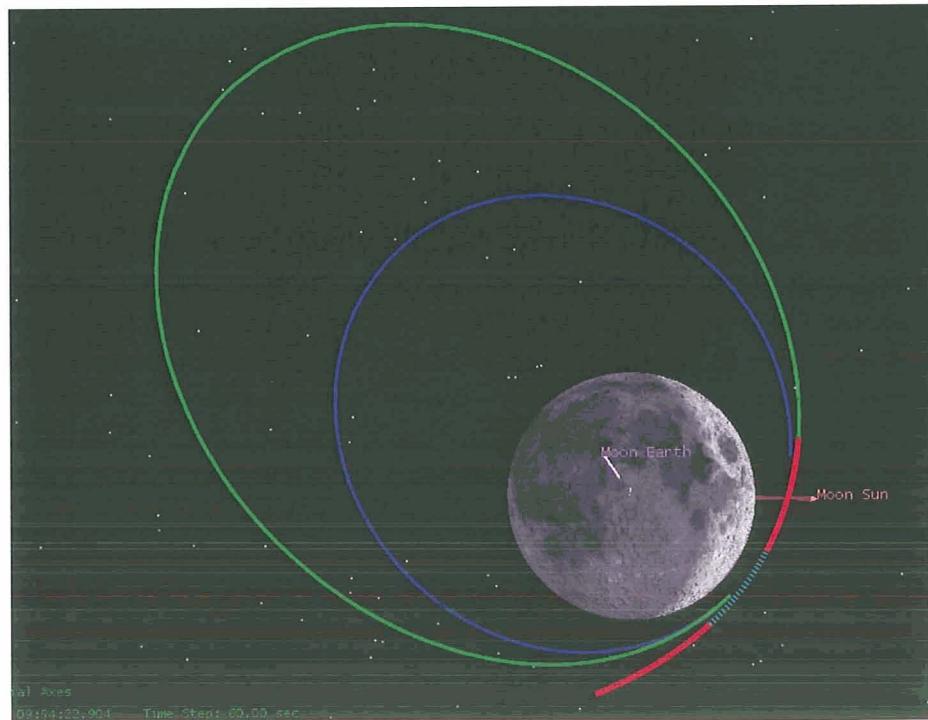


Figure 10: LOI-1 Restart Scenario

If a restart is unsuccessful and around 15 minutes was achieved from LOI-1, then LRO would be weakly captured about the Moon in a large 4-5 day orbit. In this large orbit, third-body effects would perturb the lunar inclination by about 30 degrees. LRO could still achieve a low lunar orbit but it would no longer be polar. Figure 11 shows the lunar capture orbits achieved for various duration LOI-1 maneuvers.

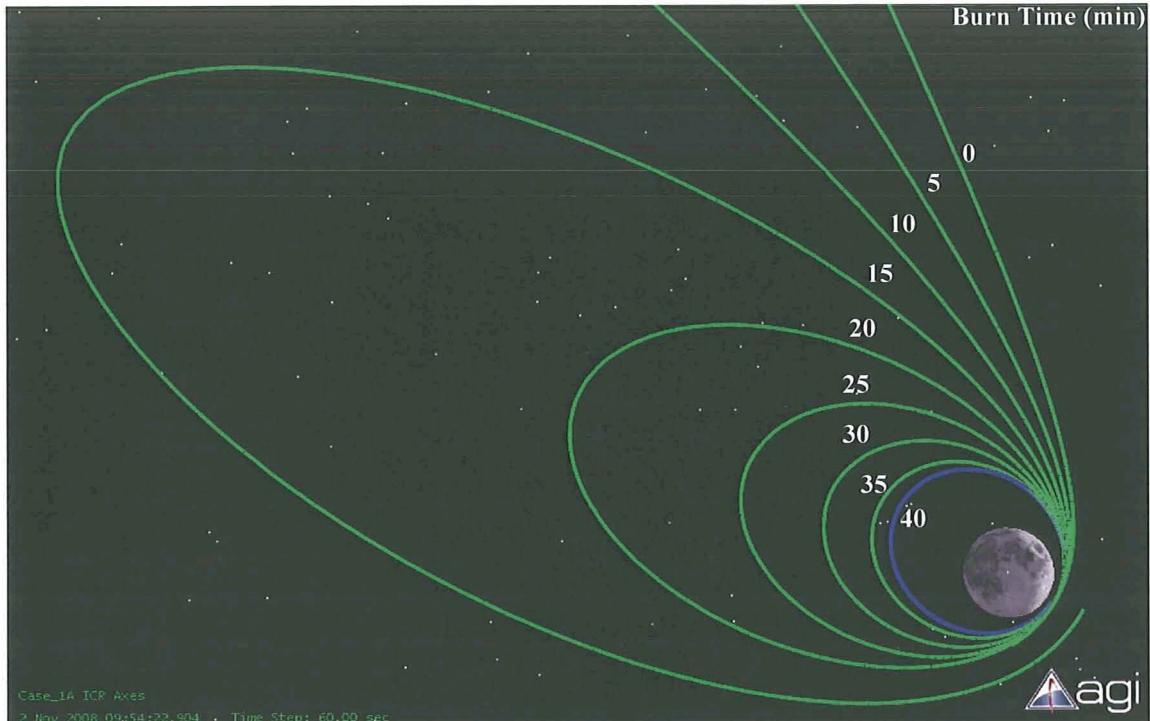


Figure 11: Lunar Capture Orbits by LOI-1 Burn Duration

FROZEN ORBIT

For many years, the term ‘frozen orbit’ has been most widely applied to spacecraft in Low Earth Orbit (LEO), but the term applies to any orbit where the parameters have been selected to fix one or more orbital elements in the presence of perturbations regardless of the central body. Frozen orbits at the Moon fall into two general classes. The first are high lunar orbits that are primarily perturbed by third-body effects^{5,6}. The second are low lunar orbits that are primarily perturbed by lunar non-spherical gravity effects. The altitude of LRO’s commissioning and nominal mission orbits clearly falls within the latter class.

Lunar non-spherical gravity effects cause a drift in eccentricity (ϵ) and argument of periapsis (ω)². This drift is similar from one lunar sidereal period to the next. At a specific initial condition, the end point of this drift can be made to coincide with the start point so that the drift pattern is exactly repeatable every month. The condition is considered quasi-frozen since the ϵ and ω are not fixed but merely bounded.

The quasi-frozen orbit is at an eccentricity of 0.043, or a 30 by 216 km altitude orbit. The ω is 270 deg, or a periselene directly over the lunar south pole³.

The frozen orbit can be maintained for many months, even years, without any stationkeeping (SK) ΔV . The variation in altitude is from 30×216 km to 45×195 km. ω varies from about 267 to 273 deg. Figure 12 shows the ϵ versus ω polar plot for the frozen orbit over four months. Each monthly ϵ versus ω pattern lies almost directly on top of the previous one.

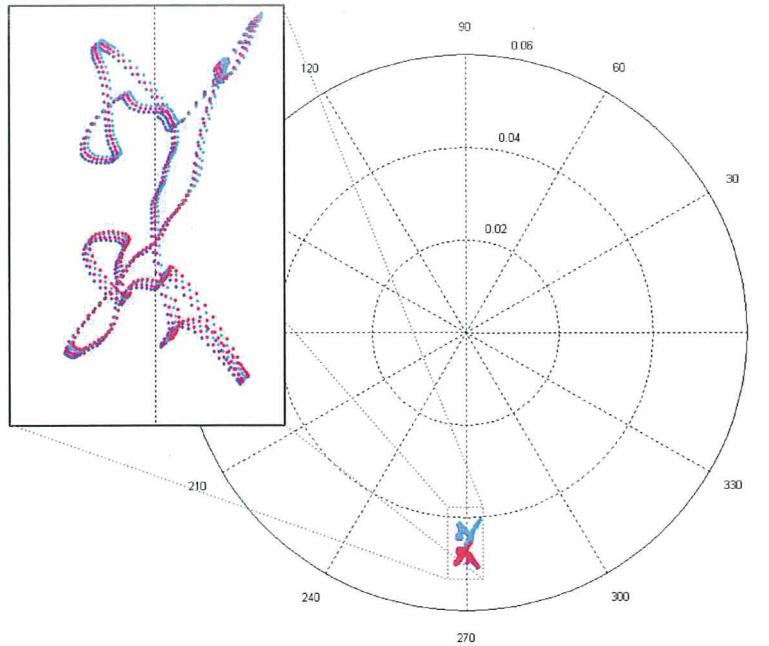


Figure 12: Frozen Orbit Evolution Over Four Months

The frozen orbit will be used for the initial two month instrument commissioning orbit during which time, no SK ΔV will occur. After the nominal one year mission, it is possible that LRO will be placed back into the frozen orbit. In this frozen orbit, LRO will have sufficient ΔV to maintain its orbit for many more years.

ORBIT DETERMINATION

Orbit determination (OD) accuracy in low lunar orbit is almost entirely dependent on the non-spherical lunar gravity model. The recent Lunar Prospector (LP) mission provided valuable tracking data at 100, 40 and 30 km altitude⁷. As a result of LP, the current lunar gravity models are an order of magnitude more accurate than those of a decade ago. However, there is still no tracking data available on the far side of the moon, which severely limits the accuracy of all lunar potential models. Significant improvement in gravity modeling is possible if backside tracking is obtained from either a subsatellite or relay spacecraft. The best model currently available for lunar orbit navigation is the LP165Q model. LP165Q was developed by Alex Konopliv at JPL⁸ and is a spherical harmonic expansion of the gravitational potential that includes up to degree and order 165.

LRO will be tracked via a network of Earth ground stations called the Space Communication Network (SCN). The SCN consists of the Deep Space Network (DSN), a new dedicated 18-m S-band antenna at White Sands, a network of Universal Space Network (USN) S-band tracking stations and the laser tracking station at Greenbelt, Maryland.

LRO will be tracked via the S-band tracking network for 30 minutes of every lunar orbit. The lunar orbit period is approximately two hours. Tracking measurements will consist of two-way coherent range and Doppler measurements. Doppler measurements will be accurate to 1 mm/sec ($1-\sigma$) from White Sands and 4 mm/sec ($1-\sigma$) from the USN⁴. In addition, LRO will be tracked by the laser ranging station at Greenbelt whenever weather allows. The laser range measurement will consist of a one-way forward range measurement time-tagged onboard using an Ultra-Stable oscillator (USO). The laser range measurement will be downlinked with science data. Figure 13 shows the ground system for the LRO mission.

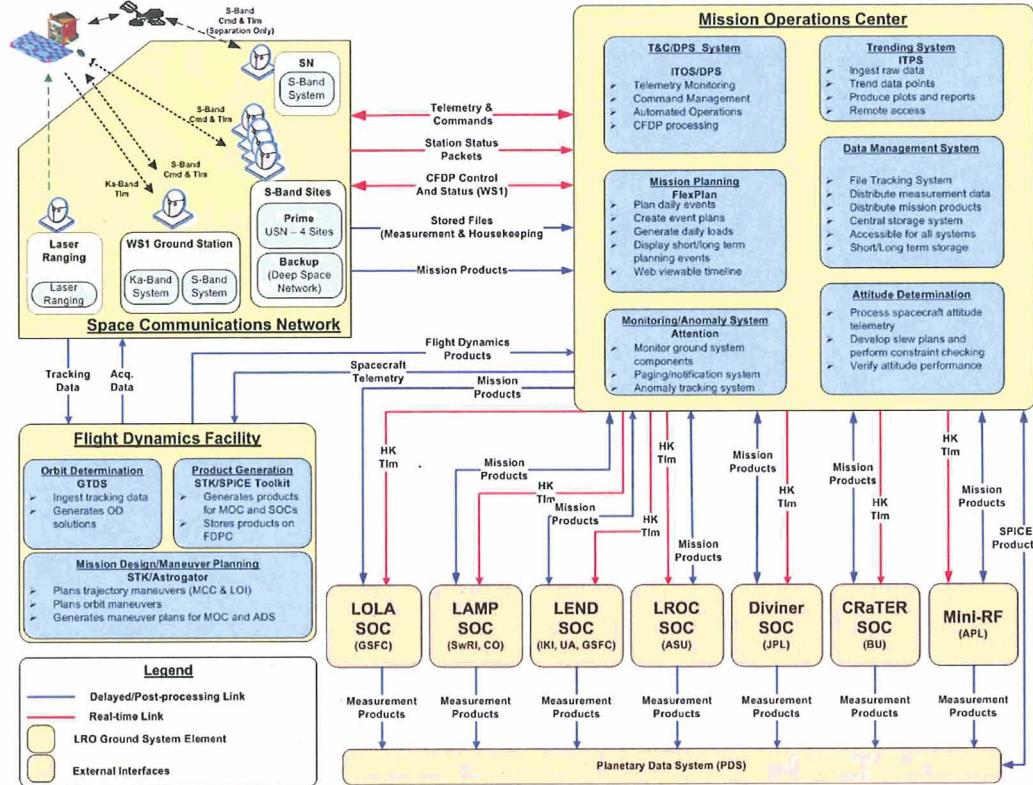


Figure 13: LRO Ground System

Orbit determination (OD) will be performed daily for LRO using all S-band tracking measurements. The accuracy requirement for these solutions is 500 m (RSS, $1-\sigma$) and 18 m radial ($1-\sigma$). The expected accuracy of these solutions is around 100 m (RSS, $1-\sigma$). These daily navigation solutions will be used for maneuver and event planning. The predictive requirement of these solutions is 800 m (along-track, $1-\sigma$) after 84 hours of prediction. This translates to about one half second error in event timing even at three days out.

LOLA will obtain laser altimeter data to the lunar surface. As the spacecraft orbits the moon, the ground tracks will overlap. This altimeter data of the same point on the surface from different orbits creates what are called “crossovers”. The enormous amount of crossover data made available from a one year mission can then be used as an additional measurement set. This measurement data, in addition to the S-band tracking data, can be used to update the LRO definitive orbit and even the lunar gravity model. The LOLA team will update the lunar gravity model periodically throughout the mission.

The Flight Dynamics Facility (FDF) at Goddard will re-process all LRO tracking data and generate definitive ephemerides using the latest available lunar gravity model. This reprocessing will occur twice. The first will be around four months into the nominal mission. The second will be after the completion of the nominal mission. These updated definitive solutions will be accurate to 50 m (RSS, 1- σ) and 1 m radial (1- σ). These solutions will be posted at the Planetary Data System (PDS).

STATIONKEEPING

Stationkeeping (SK) is required because the non-spherical lunar gravity effects cause the lunar eccentricity to drift. As eccentricity drifts, lunar altitude varies. LRO will be flown at a mean altitude of 50 km. Altitude is required to be maintained within ± 20 km.

LRO has several requirements and constraints on the mission design. First, the SK maneuvers are required to be in view of ground tracking stations. Since the SK maneuvers are performed in pairs approximately $\frac{1}{2}$ orbit apart, this constraint restricts SK maneuvers to orbit orientations when the entire LRO orbit is visible to Earth, or lunar longitudes of approximately 90 and 270 deg. Second, LRO has a mission requirement to maintain altitude within ± 20 km but a goal to maintain the altitude as tightly as possible within the limits of the ΔV budget. This altitude variation can be controlled by limiting the eccentricity box during the nominal mission. Third, LRO's instruments obtain measurement data that is sensitive to altitude. The LRO science team has placed a constraint that the ω , which places the location of periselene, or closest approach to the moon, not be biased to either the northern or southern hemispheres. Last, SK maneuvers require a significant amount of fuel. The SK algorithm must minimize SK ΔV while meeting all other constraints.

LRO will perform a repeat SK cycle every lunar orbit period, or 27.4 days³. A pair of SK maneuvers will be performed, in view of tracking stations, to reset the ϵ and ω to their initial values. These maneuvers are performed at the same ϵ , ω and lunar longitude every month. The drift pattern of ϵ versus ω is centered such that periselene is in the lunar southern hemisphere 50% of the time. The ΔV cost of these SK maneuvers is about 11 m/sec per month. Figure 14 shows the repeating ϵ vs ω pattern. Eccentricity is the radial component and ω is the angular component in the polar plot. A marker is shown at each ascending node, so greater spacing indicates a faster drift in the phase space. The numeric annotations next to some markers represent the lunar longitude of that ascending node. Note that the start and end of the pattern is at approximately 270 deg lunar longitude, as required in order to see the entire orbit from Earth. The entire pattern extends only slightly beyond an eccentricity of 0.008, which translates to an altitude variation of +/- 15.1 km. Additionally, the pattern is translated such that half of the markers are in the top (northern hemisphere periselene) and half in the bottom (southern hemisphere periselene) of the polar plot.

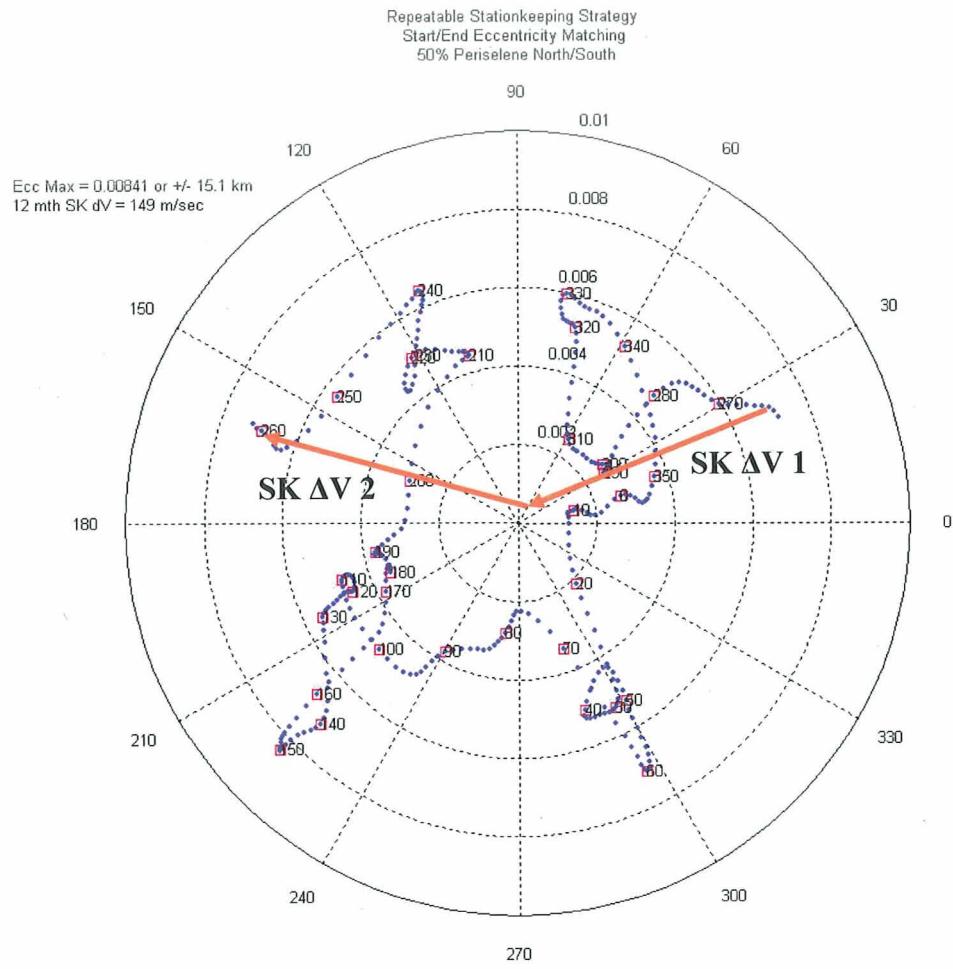


Figure 14: LRO Repeat Stationkeeping Cycle

ΔV BUDGET

LRO launch mass is required to be less than 2000 kg, of which about 900 kg is fuel mass. Most of the fuel is used to insert into the low lunar orbit. Table 2 shows the current baseline ΔV budget. The extended mission ΔV budget is a placeholder only. Since the actual extended mission will not be finalized until midway through the LRO mission, this ΔV placeholder gives a ΔV allocation for any likely extended mission scenario. The LOI-1 maneuver accounts for launch at any time during the month. If LRO launches on the primary launch day, an immediate 60 m/sec ΔV savings will be achieved.

Table 2: LRO Baseline ΔV Budget

Maneuver/Phase	ΔV (m/sec)	Fuel (kg)	Notes
Mid-Course Correction	28	26.4	3- σ LV errors
Lunar Capture (LOI-1)	567	444.9	Deterministic
LOI-2	185	121.8	Deterministic
LOI-3	133	81.3	Deterministic
LOI-4	41	24.1	Deterministic
Mission Orbit Insertion	50	28.8	Deterministic
Mission Orbit S/K	150	84.1	Deterministic
Extended Mission	65	33.5	3 yrs Frozen
Momentum Unloading	–	17.0	Worst Case
Other (Residuals)	–	15.4	Conservative
Unallocated Margin	39	20.9	Conservative
Total	1258	898	

CONCLUSION

The LRO mission is the first of many planned LPRP missions to the moon. LRO will launch on an Atlas V 401 LV on a minimum energy transfer to the moon. Science constraints give LRO a launch window of about three days every two weeks. Lunar orbit insertion consists of a series of insertion maneuvers. The first LOI maneuver is designed to be very robust and allows multiple recoveries from contingencies. After LOI, LRO will be placed into an instrument-commissioning orbit for two months. This commissioning orbit will be at the lunar frozen orbit condition. At this unique location, the mean eccentricity and AoP drifts are small enough to eliminate the need for SK maneuvers during the commissioning phase. After two months, LRO will be transitioned into the nominal circular polar mission orbit at 50 km. Because eccentricity and argument of periapsis drift over time, SK maneuvers are required to maintain altitude. A repeating SK algorithm has been designed to meet a number of mission constraints. Orbit determination will be performed with the most current lunar gravity model and near continuous S-band tracking coverage. At the end of the nominal mission, a ΔV allocation will allow a variety of opportunities for an extended mission.

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