

Contingency Trajectory Design for a Lunar Orbit Insertion Maneuver Failure by the LADEE Spacecraft

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This paper presents results from a contingency trajectory analysis performed for the Lunar Atmosphere & Dust Environment Explorer (LADEE) mission in the event of a missed lunar-orbit insertion (LOI) maneuver by the LADEE spacecraft. The effects of varying solar perturbations in the vicinity of the weak stability boundary (WSB) in the Sun-Earth system on the trajectory design are analyzed and discussed. It is shown that geocentric recovery trajectory options existed for the LADEE spacecraft, depending on the spacecraft's recovery time to perform an Earth escape-prevention maneuver after the hypothetical LOI maneuver failure and subsequent path traveled through the Sun-Earth WSB. If Earth-escape occurred, a heliocentric recovery option existed, but with reduced science capability for the spacecraft in an eccentric, not circular near-equatorial retrograde lunar orbit.

Nomenclature

<i>B-theta</i>	= Angle with respect to incoming hyperbolic asymptote at a body (deg)
<i>Bend Angle</i>	= Angle of trajectory change via gravity swingby (deg)
<i>C3</i>	= Orbit Energy (km^2/s^2)
<i>Geocentric</i>	= Earth-centered
<i>Heliocentric</i>	= Sun-centered
<i>LOI</i>	= Lunar Orbit Insertion (m/s)
<i>LV</i>	= launch vehicle
<i>DAA</i>	= Declination of the arrival asymptote (deg)
<i>TLI</i>	= trans-lunar injection (m/s)
ΔV	= delta-V, change in velocity (m/s)
<i>SE-L1</i>	= Sun-Earth Lagrange Point 1
<i>SE-L2</i>	= Sun-Earth Lagrange Point 2
<i>SOI</i>	= sphere-of-influence (km)
V_{inf}	= excess speed at a target body (m/s)
<i>WSB</i>	= weak-stability boundary, used to describe chaotic region near edge of Earth's SOI

I. Introduction

IN the event of a missed lunar-orbit insertion (LOI) maneuver by the Lunar Atmosphere & Dust Environment Explorer (LADEE) spacecraft, built, operated, and managed by NASA Ames Research Center (ARC) in Moffett Field, CA, it was the author's responsibility to design a rescue trajectory that would recover the spacecraft into its near-equatorial, circular retrograde lunar science orbit. Some in the press thought such a recovery to be impossible; *Universe Today* claimed the LOI to be a "do or die" [R] maneuver, in agreement with *Spaceflight101*'s assessment of missing LOI: "...the spacecraft would have passed the Moon with no hope of returning." [R] However, after receiving LADEE's planned LOI-state vector ≈ 2 weeks before the actual LOI, the author was able to design a viable rescue trajectory that was verified by the LADEE flight dynamics team and ready for use >10 days before LOI. Fortunately, this rescue trajectory was not flown as the LADEE spacecraft successfully performed its LOI maneuver on Oct. 6, 2013, notably by a skeleton crew during the U.S. Government Shutdown in 2013 (Oct. 1, to Nov. 17). Details of the selected contingency trajectory design and other considered designs are presented and discussed in the sections to follow.

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II. Assumptions & Constraints

The trajectory design and analysis was performed primarily with the Systems Tool Kit (STK) Astrogator module, which was used to plan maneuvers for the LADEE spacecraft during its nominal mission. For propagation of the spacecraft's trajectory, a 7th order Runge-Kutta-Fehlberg numerical integrator with 8th order error control was used with a high-fidelity N-body force model, the latter of which included solar radiation pressure (SRP), a Jacchia-Roberts Earth atmosphere model, and gravity field models for the Earth and Moon (initially 30X30 and 21X21, then increased to 70X70 and 100X100, respectively) as well as the Sun (4X0). Maneuvers were assumed to be instantaneous since no significant gravity losses were expected throughout the nominal or recovery mission.

The total available ΔV was constrained to < 920 m/s for nominal science operations. Per the science orbit, the spacecraft was required to enter a 250 km circular retrograde lunar orbit with inclination of 157 degrees to obtain the required science measurements at low-altitudes, passing from darkness into daylight over the lunar terminator [R]. DE421 is the ephemeris source used for both the Earth and Moon. LADEE's State Vector at the time of planned LOI (Julian Date 2456571.9531057) using the Earth J2000 Cartesian coordinate system: $[x, y, z, V_x, V_y, V_z] = [-324311 \text{ km}, -176241 \text{ km}, -81134.7 \text{ km}, 1.02087 \text{ km/s}, 0.78829 \text{ km/s}, 1.22223 \text{ km/s}]$

III. Trajectory Design Methodology & Selection

The following section provides details and analysis of the contingency trajectory design types considered, both geocentric and heliocentric, for use of the LADEE spacecraft in the case of a missed LOI maneuver.

LADEE Spacecraft's Nominal Trajectory

The nominal trajectory flown by the LADEE spacecraft, from launch on Sep. 5, 2013 to LOI on Oct. 6, 2013, is seen in Fig. 1 (top-left). This non-standard ≈ 1 -month lunar implemented eccentric Earth phasing orbits with apogee altitudes ranging from $\approx 275,000 \text{ km}$ (6.5-day period) to lunar distance (10-day period) and was flown by LADEE for multiple reasons including: 1) the launch vehicle (LV), a Minotaur-V launched from Wallops, VA, could not send the LADEE spacecraft (383kg initial mass) all the way to the Moon, 2) The LV's fifth stage injection accuracy was not expected to be as high as that of other larger, heavier, and more expensive LVs, thus the longer lunar transfer allowed ample time for the spacecraft to perform correction maneuvers (TCMs) to correct injection errors, 3) By varying the periods of the phasing orbits, the launch window could be lengthened.

Additionally, there were two types of nominal solution types considered for a given launch month, termed in-plane (IP) and out-of-plane (OP). IP trajectories are less inclined to the lunar orbit plane than OP trajectories (Fig. 1, bottom), and thus the former requires less LOI ΔV than the latter. However, OP solutions provided better shadow and lighting conditions for the spacecraft throughout the sub-lunar Earth phasing orbits and were thus preferred over IP solutions.

To understand effects of a missed LOI on IP and OP solutions, both types were propagated to the Moon's SOI after LOI throughout a July 2013 to June 2014 launch period. It was seen (Fig. 1, middle-left and top-right) that OP solutions (green) contain higher orbital energies than IP solutions (red). The primary reason for the energy difference is that IP solutions contain orbital planes less inclined to the lunar equator and thus receive higher energy-assists in the velocity direction of the Moon (Fig. 1, middle-right).

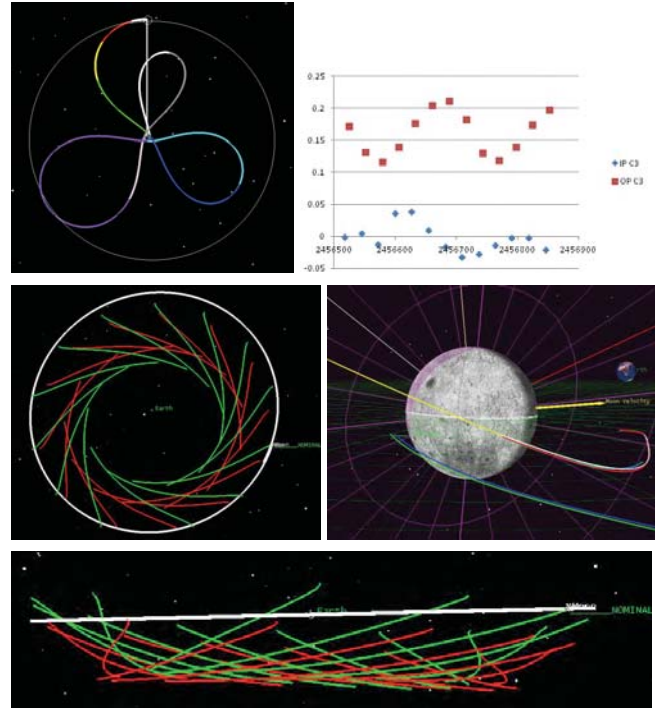


Figure 1. LADEE Nominal Trajectory & Effects of a Missed LOI Thereon. LADEE's Nominal Trajectory, Earth-centered, Earth-Moon rotating frame (top-left); Bplane plot and/or V_x plot, showing lunar swingby (middle-right); Post LOI-miss states, July 2013 to June 2014, IP (red), OP (green), (view from north of lunar orbit plane: middle-left; view edge-on lunar orbit plane, bottom); Orbital Energy vs Launch Date for IP & OP Solutions (middle-right).

Sun-Earth WSB Effects and Contingency Trajectory Design Methods #1 & #2

For the analyzed 1-yr of possible LADEE LOI-states between July 2013 and June 2014 discussed in the previous section, a maneuver was performed 3 days after the LOI-miss to decrease the spacecraft's energy generally from hyperbolic to sub-parabolic. This escape-prevention (recovery) maneuver was performed in the anti-velocity direction in the spacecraft's orbit plane to allow a chance to re-encounter the Moon without entering heliocentric space. The cost of this recovery maneuver varies significantly depending on where the spacecraft's high-apogee is located in the Sun-Earth WSB region. For apogee-locations in quadrants II and IV (Fig. 2, top-left), solar gravity increases the spacecraft's Earth orbit energy while this energy is decreased by the Sun for apogee locations in quadrants I and III [R1, R2, R3]

Since LADEE's post LOI-miss trajectory is of higher energy than lunar orbit, WSB quadrants I and III are preferred high-apogee locations for the spacecraft, as the Sun will help decrease orbital energy and thus decrease the required recovery ΔV cost. This is observed in Fig 2 (top-right), where the lowest- ΔV lunar-return solutions (recovery ΔV as low as 37 m/s) are seen in the large outer "lobes", with their high-apogee locations in quadrants I and III. The highest- ΔV solutions require up to 236 m/s of recovery ΔV and contain quadrant II and IV apogee locations, as expected. The lowest- ΔV solutions generally take longer (≈ 7 months) to return to lunar-distance compared to higher-energy solutions (as low as ≈ 3 months). Although the LADEE spacecraft would have been able to perform the recovery maneuver for all analyzed cases, science operations would be reduced for these highest- ΔV solutions (> 200 m/s), which are seen to be IP solutions (Fig. 2, bottom-left). The noticeable recovery ΔV amplitude difference among IP and OP solutions is explained by the energy difference discussed in the previous section (Fig. 1, middle-right), i.e., IP solutions reach higher-energy Earth-orbits via the lunar swingby at missed-LOI and thus need more ΔV than OP solutions for lunar-return.

Furthermore, the post-LOI trajectories computed generally yield a spacecraft arrival at lunar-distance when the Moon is not there, since such low-energy returns are not naturally lunar-synchronous. This is seen in Fig. 3 (left) for the baseline LOI-miss state (Oct. 6, 2013), as the Moon is on the opposite side of the Earth when the spacecraft re-encounters lunar-distance (contingency design method #1). Thus the recovery ΔV values in Fig. 2 (top-right) represent minimum recovery ΔV requirements, since more ΔV may be required to solve this lunar phasing problem. For example, the baseline solution can solve this phasing problem by performing a relatively large maneuver at high-apogee (158 m/s of ΔV) (method #2, Fig. 3, right). Since changing the period of the single-loop trajectories is one-directional (i.e., increasing the period will generally result in Earth-escape), the use of single-loop trajectories are limited for the baseline case. Thus multiple-loop solution types are explored in the following sections.

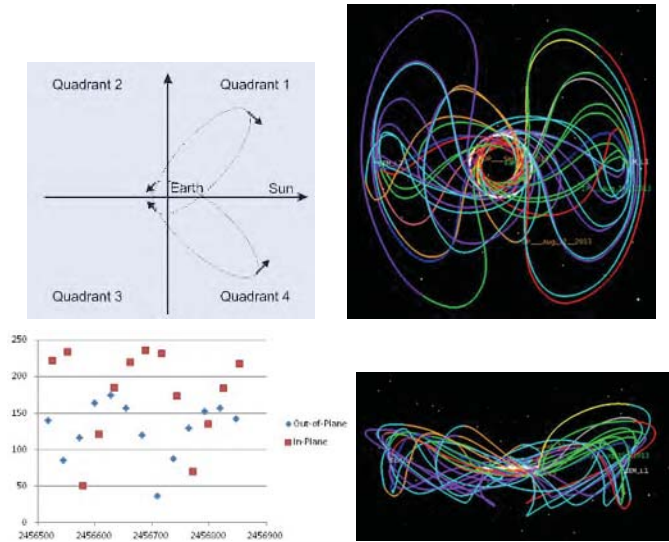


Figure 2. Post-LOI Miss States Recovered to Lunar Distance Sep 2013 to July 2014. shown in Earth-centered, Sun-Earth Rotating Frame (viewed from North of Lunar Orbit Plane, top-left; edge-on lunar orbit plane, bottom-left). With the spacecraft recovery time fixed at 3 days post LOI-miss, varying ΔV costs to recovery to lunar distance shown for all cases (bottom-left), depending on Sun-Earth WSB quadrant entered post LOI-miss (top-left).

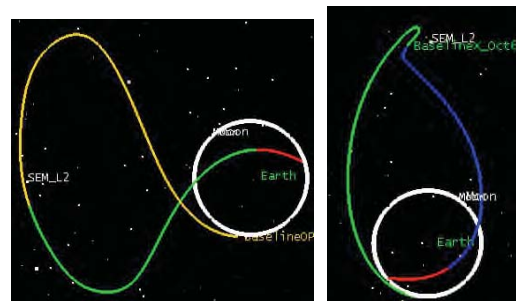


Figure 3. Single-Loop Recovery Solution shown for both Favorable (left) and Unfavorable (right) Sun-Earth WSB geometry with respect to the spacecraft's position at apogee.

Show Single-Loop WSB solution with 158 m/s at apogee

Contingency Trajectory Design Methods #3 & #4

By implementing multiple Earth-phasing orbits, the lunar-encounter phasing problem is solved. And if the apogee altitude is designed low enough to avoid undesirable WSB effects, it will remain essentially fixed in Earth-inertial space and thus rotate in the Sun-Earth rotating frame, the same frame used to define the discussed Sun-Earth WSB quadrants. The rate of this natural apogee rotation is dependent on the Earth's changing geometry with respect to the Sun throughout one full heliocentric revolution, thus it takes ≈ 1 year to rotate apogee a full 360 degrees; due to Earth's heliocentric motion, apogee is restricted to clockwise (CW) motion as viewed from north of the ecliptic plane. This rotation rate is observed in Fig. 4 (top), as the baseline LOI case establishes an (inner-WSB) apogee altitude of ≈ 1.2 million km; after 1 year of phasing orbits, the apogee has rotated ≈ 1 full revolution in ≈ 13 months. Since the apogee started in quadrant III, quadrant II is next visited after ≈ 3 -months of apogee rotation. Such a quadrant is favorable for lunar-return, which is shown for method #3 in Fig. 4 (bottom-left). However, the arrival V_{inf} upon lunar re-encounter yields a relatively large LOI ΔV requirement of 850 m/s. If apogee is rotated to quadrant IV instead (i.e., the next favorable quadrant) there is more help from the Sun to raise perigee and return to the Moon, but > 9 months of phasing loops are required, not including the final transfer to the Moon, which yields a total recovery duration of nearly one year (Fig. 4, seen in yellow, top and bottom-right).

Contingency Trajectory Design Method #5

By implementing the single Earth phasing orbit seen in method #3, but now with a 140 m/s recovery ΔV (compared to 129 m/s), the subsequent apogees are now in different locations. This location-difference is enough to yield a lower-energy return to the Moon in 168 days, 95 days earlier than method #3's lunar re-encounter date.

After the LOI-miss, the spacecraft is shown performing the recovery maneuver 3 days after missing the LOI maneuver (Fig. 5, A & B), followed by its first high-apogee located in quadrant 3 and subsequent perigee at $\approx 2,600$ km altitude (Fig. 5, C). The spacecraft performs lunar re-encounter maneuver (33 m/s of ΔV) at its second high apogee, at an altitude of ≈ 1.4 million km and located in quadrant II. However, the spacecraft reaches the Moon with a steep arrival declination > 80 deg (Fig. 5, D & Fig. 6, top), translating into lunar orbit options highly inclined from the lunar equator. Specifically, the orbit inclination is constrained between 79.6 and 98 deg (Fig. 6, bottom) and thus does not allow the spacecraft to achieve an acceptable science orbit (Fig. 6). Therefore, this lunar re-encounter is not used for the LOI-retry, but rather for a

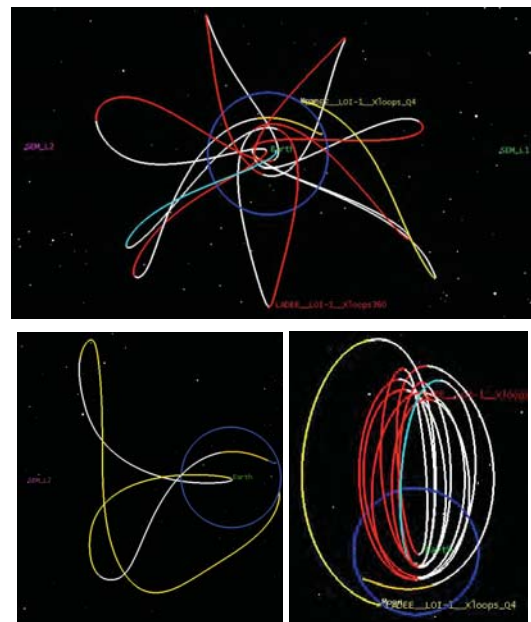


Figure 4. Apogee-Rotation in Sun-Earth Rotating Frame over 1 yr (top) & Multiple Phasing Orbit Solutions (bottom). Quadrant II (final apogee) Solution requires ≈ 80 m/s targeting ΔV for lunar re-encounter (bottom-left); Quadrant IV (final apogee) Solution shown in Earth-inertial frame, requiring ≈ 35 m/s targeting to re-encounter the Moon ≈ 1 yr post LOI-miss (bottom-right)

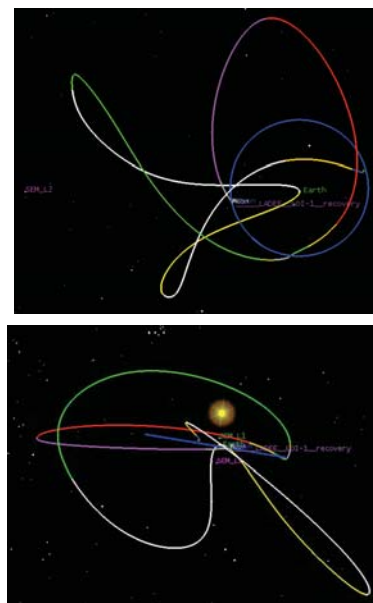


Figure 5. Selected Contingency Trajectory Design (Method #5). Trajectory is shown in Earth-centered, Sun-Earth Rotating frame, from north of (top) and edge-on the lunar orbit plane (bottom).

lunar swingby to change the orbital plane without use of propellant. This swingby is performed at a perilune altitude of $\approx 3,500$ km (shown in gray, Fig. 6, top-right) and changes the spacecraft's lunar orbit plane so as to achieve the 157 deg inclination required for the science orbit. The primary cost of this lunar swingby is ≈ 2 months, yielding a total recovery duration of ≈ 7 months, from the hypothetical LOI miss on Oct. 6, 2013 to LOI-retry on May XE, 2014 (Fig. 5, E). The total ΔV associated with this recovery option was **870 m/s**, about the same as that required for nominal LOI maneuver. Despite the necessity of a significant recovery maneuver for the recovery option (140 m/s), the LOI-retry ΔV (661 m/s) is $> 20\%$ lower than the nominal LOI ΔV (> 850 m/s) since a WSB lunar transfer trajectory allows the spacecraft to approach the Moon at a lower arrival V_{inf} as compared to the nominal lunar transfer [B]. This resulted in similar total ΔV requirements between the nominal and recovery cases. As such this contingency trajectory design was selected by the LADEE team to be flown in case of actual LOI maneuver failure.

Heliocentric Recovery Option

Two alternate mission modes (AMMs) result when reducing total mission ΔV in two ways: Reduce the science operations duration (AMM1) or change the orbit from circular to elliptical. AMM1's total ΔV requirement is reduced (by ≈ 100 m/s) via less station-keeping maneuvers while AMM2's LOI maneuver ΔV is less (> 400 m/s) in its eccentric orbit.

The transition points between these AMMs are defined with respect to spacecraft recovery time. To locate these transition points, the spacecraft recovery time was varied and the minimum ΔV cost of the recovery maneuver was calculated. As expected, for a fixed solution-type the recovery ΔV cost increases with time needed for spacecraft recovery (Fig. 7, left). After 6 days, the recovery ΔV cost is ≈ 200 m/s for the standard recovery solution. This yields a total mission ΔV requirement of ≈ 920 m/s, which is the total recovery ΔV allocation for the standard solution, thus the mission-mode changes to AMM1 beginning at ≈ 6 days. As AMM2 allocates **1,010 m/s** for total ΔV , the spacecraft recovery time transitions from AMM1 to AMM2 ≈ 10 days after missing LOI. AMM2 is notably a heliocentric solution that performs a reverse-WSB transfer upon Earth re-encounter a1 year after missing LOI. However, AMM2's elliptical lunar orbit drastically reduces the time spent by the spacecraft over the lunar terminator at low altitudes and thus AMM2 was considered only as a last-effort salvage opportunity, where other non-science measurements could be performed (e.g., laser-communication technology demonstration).

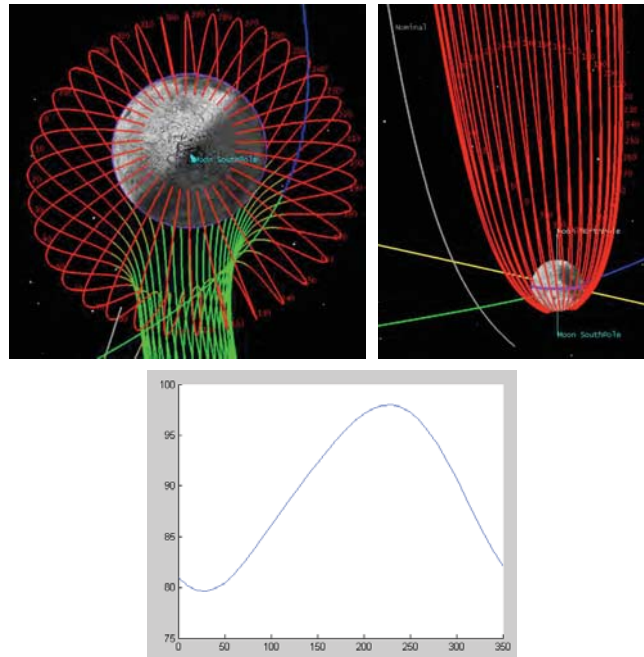


Figure 6. Effects of Arrival Declination on Lunar Orbit Inclination. *B*-plane centered at Moon with incoming lunar declination of > 80 deg (left); Lunar Orbit Inclination shown for a full 360-degree *B*-theta range (bottom).

3500 km Moon swingby altitude

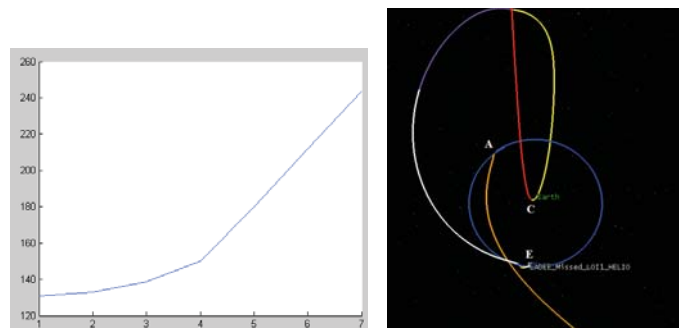


Figure 7. Effects of Spacecraft Recovery Time on Recovery ΔV (left) and Heliocentric Recovery Solution-Type (right). *Mission-mode is dependent on spacecraft recovery time for fixed solution-type* (left); *Example of a heliocentric return to the Moon via a reverse-WSB transfer* (right).

IV. Conclusion

Since the spacecraft was required to pass from darkness to light (over the terminator) for dust collection, the nominal trajectory design yielded a leading-edge lunar swingby as the spacecraft first passed in front of the Moon and beyond lunar orbit before performing its LOI maneuver on the Moon's trailing edge. This lunar approach was shown to decrease (Earth) orbital energy for all cases and thus decrease the recovery ΔV requirement. OP solution types required a lower worst-case recovery ΔV than IP solutions, but significant variations in this ΔV requirement were observed throughout a year-long launch period due to varying solar gravity perturbations in the Sun-Earth WSB region. It was shown the LADEE spacecraft could have returned to lunar orbit for all analyzed launch months and for both IP and OP nominal trajectory solution, depending on the amount of time needed for the spacecraft to recover (e.g. for reasons related to communications, propulsion, safe-mode, et al) after a missed LOI.

There are applications of the selected trajectory design beyond the scope of strictly missed LOI maneuvers. In particular, elements of this design can be flown for secondary spacecraft dropped off in significantly eccentric and/or inclined orbits (with respect to the lunar orbit plane), especially if the mission requires a near-equatorial lunar orbit. Such a secondary spacecraft can avoid restricting the primary payload's launch window (e.g., launch time of day) by utilizing similar Earth-centered phasing loops seen for this selected design to wait for its apogee to rotate to a favorable Sun-Earth WSB quadrant. Per the near-equatorial lunar orbit, such an orbit could be designed for a spacecraft needing to maximize its time spent in the radio-quiet zone on the lunar farside, where unique space-observing opportunities exist; the lunar swingby element of this design would be used to attain near-equatorial inclination in lunar orbit. The apogee-rotation design element can be extended to most other systems as well without the necessity of a significantly massive moon (e.g., Sun-Venus), while the latter cannot.

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