HIGH EARTH ORBIT DESIGN FOR LUNAR-ASSISTED MEDIUM CLASS EXPLORER MISSIONS

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ABSTRACT

This study investigates the application of high-Earth orbit (HEO) trajectories to missions requiring long on-target integration times, avoidance of the Earth's radiation belt, and minimal effects of Earth and Lunar shadow periods which could cause thermal/mechanical stresses on the science instruments. As used here, a HEO trajectory is a particular solution to the restricted 3-body problem in the Earth-Moon system with the orbit period being either ½ of, or ¼ of, the lunar sidereal period. A primary mission design goal is to find HEO trajectories where, for a five-year mission duration, the minimum perigee radius is greater than 7 Earth radii (R_E). This minimum perigee radius is chosen so that, for the duration of the mission, the perigee is always above the relatively heavily populated geosynchronous radius of 6.6 R_E. A secondary goal is to maintain as high an ecliptic inclination as possible for the duration of the mission to keep the apsis points well out of the Ecliptic plane. Mission design analysis was completed for launch dates in the month of June 2003, using both direct transfer and phasing loop transfer techniques, to a lunar swingby for final insertion into a HEO. Also provided are analysis results of eclipse patterns for the trajectories studied, as well as the effects of launch vehicle errors and launch delays.

1 – INTRODUCTION

A paper was presented at the 1994 Goddard Space Flight Center's Flight Mechanics Estimation Theory (FMET) Symposium titled *High Earth Orbit Design for Lunar Assisted Small Explorer Class Missions* (Reference 1). It described a lunar gravity-assisted method for achieving a class of stable HEO trajectories for which apogee is near the distance ($|R|$) of the Moon and perigee is in the range of 8 to 25 R_E. Lunar and solar perturbations usually cause dramatic changes in the elements of such highly elliptical orbits, but it was found possible to adjust the orbit period and phase with respect to the Moon so that the result is a quasi-periodic solution to the restricted three-body problem in the Earth-Moon-spacecraft system. This solution possesses great stability (in the sense that secular changes in key design parameters such as inclination and perigee height are eliminated, although these parameters may oscillate about some average value) and orbit lifetime without requiring spacecraft stationkeeping.

In 1994, it was noted that the advantages of the HEO orbit were freedom from passage through the Earth's radiation belts, small gravity gradient effects, and no atmospheric torques. Perigee altitudes remaining above geosynchronous altitude (~6.6 R_E), excellent coverage by a single ground station, and modest launch vehicle and spacecraft propulsion system requirements, as a result of the lunar gravity assist feature, seemed likely to make this an attractive orbit type for future missions. Additionally, a HEO has more launch opportunities per month than, for example, a lunar swingby Lagrange L2 point mission because there is greater flexibility in the choice of Moon location at lunar swingby. A drawback of the HEO, however, is that it is very difficult to analyze because of the complexity in modeling lunar and solar perturbations. One has to be careful that a chosen orbit meets all mission requirements. For example, one must verify that Transfer Trajectory Injection (TTI) dispersions and launch delays do not cause a nominal orbit to become unsuitable. Another possible disadvantage of the HEO orbit is the spacecraft disposal problem at the mission end of life (EOL). For example, this could be of concern if the long-term evolution of the orbit carried the spacecraft repeatedly through the relatively highly populated geosynchronous altitude.

Since 1994, however, the HEO orbit received little attention—perhaps because of the number of new missions utilizing libration point orbits about the Earth-Sun Lagrange L1 and L2 points. Yet HEO trajectories possess several advantages over the libration point orbits including the following: they are stable and much closer to the Earth; their station keeping requirements are negligible; and, it is much easier to achieve high data transmission rates than is the case for the libration point orbits. Consequently, several potential missions (Kronos, SNAP, Constellation-X, etc.) are now considering a HEO as their baseline trajectory design. It is beyond the scope of this paper to discuss the details and science objectives of these missions. Those interested in such matters are referred to the World Wide Web pages for those various missions (Reference 2). Most of the analysis presented here was performed for the Kronos mission but it is sufficiently general to be applicable to any proposed HEO mission. Various mission requirements might differ (e.g., lifetime, minimum perigee altitude, ecliptic inclination, etc.) but the behavior of the trajectories and methods of achieving them is similar in all cases.

Analysis Goals

The primary goal of this study is to find HEO trajectories where, for the entire five-year duration of the mission, the minimum perigee radius is always greater than 7 R_E, so that perigee is always well above the relatively heavily populated geosynchronous radius of 6.6 R_E. A secondary goal is to maintain the ecliptic inclination (ECL INC) as high as possible for the duration of the mission for shadow minimization.

This analysis will address the following properties of the proposed Kronos orbit:

- Dependence of delta velocity (delta-V or ΔV) at the post lunar encounter perigee as a function of orbit parameters and injection dispersions
- Characterization of launch window (days per month)
- Ecliptic Inclination Evolution
- Perigee Evolution
- Eclipse Overview
- Mission lifetime

A full parametric study of the proposed Kronos trajectory would require considerable effort because of the lack of analytic solutions to the 4-body (Sun, Earth, Moon, Kronos spacecraft (SC)) problem. To determine seasonal as well as monthly trends in ΔV, perigee radius, ecliptic inclination, etc., would require the development of three fully integrated and fully targeted 5-year trajectories for each launch day (nominal and ± 3 sigma (σ) launch vehicle injection energy error cases) as well as launch window analysis for each of the cases. This was beyond the scope of this preliminary analysis.

Modeling Assumptions

All trajectories were computed in Earth-centered Mean-of-J2000 coordinates, with the Mission Analysis and Design Tool, commonly known by the name, 'Swingby' (Reference 3). The Earth centered trajectories were numerically integrated with an eight-order Runge-Kutta-Nystrom algorithm with adaptive step size control through sixth order, and had perturbation models consisting of an 8 x 8 Joint Gravity Model-2 (JGM-2) geopotential field, solar radiation pressure and point mass gravities assumed for the Sun and Moon (Reference 3). Planetary gravitation was not invoked for this preliminary work. All parameter targeting was performed using the differential correction trajectory-targeting scheme in Swingby. Swingby was used to calculate the trajectory from launch site to parking orbit insertion (POI). Manufacturer's data for the Delta II 7325 launch vehicle was used to give the insertion state into a 185-kilometer (km) parking orbit with a 28.7 degree (°, deg) inclination from a launch assumed to be from the Eastern Range. Swingby was also used to model the propagation (coast time (C)) from the POI to the TTI point. The mass of the Kronos spacecraft used in all propagations was 600 kilogram (kg), and all delta-V's used to simulate maneuvers by the launch vehicle (TTI) and spacecraft were modeled as impulsive.

2 – TRAJECTORY DESIGN APPROACH

The HEO can be established via either a direct transfer to the lunar gravity assist or via a series of phasing loops culminating in a lunar gravity assist. Detailed results for both methods will be discussed later. In either case, the spacecraft is launched into a highly elliptical orbit with apogee near the lunar distance. A close flyby of the Moon is used to provide an effective perigee raising delta-V (typically equivalent to 700 to 900 meters per second (m/s)) and also to target the desired perigee altitude, ecliptic inclination, and time of flight to the post-lunar encounter perigee (PLEP) passage. At the Moon, time of arrival and B-plane (B•T, B•R) parameters are used as targeting controls to establish a preliminary HEO trajectory (Reference 4). At the PLEP, a period adjust maneuver (PAM) is performed to adjust the mission orbit period to one-half the sidereal month. (A sidereal month is approximately 27.3 days.)

For both the direct and phasing loop techniques, an attempt was made to determine if the phase of the Moon on the launch day had any affect on the trajectory. Table 1 below lists the phases of the Moon for the month of interest, i.e., June 2003 (Reference 5).

Table 1: Start Times of Lunar Phases for June 2003

Lunar Phase	New Moon (NM)	First Quarter (FQ)	Full Moon (FM)	Last Quarter (LQ)
Start for June 2003	2003/05/31 04:20	2003/06/07 20:28	2003/06/14 11:16	2003/06/21 14:46

As an example of the influence of lunar phase at launch, we note that if the Earth-spacecraft-Moon (E-SC-M) angle is near 180° when the PAM is performed at PLEP, then the geometry shown in Figure 1 results. Figure 1 depicts a fixed-size phasing loop transfer to a HEO, displayed in an Earth-Moon Rotating coordinate frame. Subsequent apogee passages (where the Moon perturbs perigee height most strongly) occur when the Moon-Earth-Spacecraft angle is approximately 90° and the apogee passages are alternately on the leading, then trailing, side of the Moon's orbit. In this orbit, the lunar secular perturbations average, roughly, to zero resulting in significant long-term "stability." Solar perturbations do impact the trajectory as well, resulting in cyclic variations in the ecliptic inclination and perigee altitude. As will be shown, careful choice of initial conditions can yield a trajectory that maintains perigee above geosynchronous altitude and an inclination high enough to keep apogee out of the ecliptic plane to minimize the impact of eclipses.

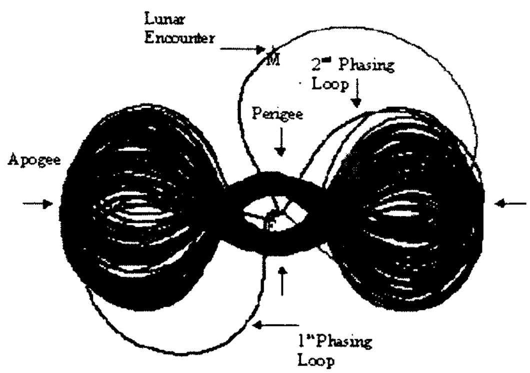

Figure 1: Establishing the HEO for Phasing Loop Transfer in Earth-Moon Rotating Coordinates

2.1 – Direct Transfer Technique

Direct transfer solutions with launch dates in June 2003 were sought. For these launch dates, the objective was to determine the number of nominal launch opportunities available while letting mission related goal values float. In this approach, we wished to obtain a PLEP inclination between 40° and 50° and a PLEP radius value between 7 R_E and 25 R_E. The aforementioned mission design goals are attainable by using the targeting variables, launch epoch, parking orbit coast time, and TTI delta-V magnitude (direction assumed along the instantaneous velocity vector). The TTI is assumed to correspond to the second cutoff of the 2nd stage, SECO2) of a Delta II launch vehicle for this class of mission. Before a trajectory for a given launch day can be calculated, an initial estimate of the launch vehicle launch epoch and low Earth parking orbit coast time, or duration, must be computed. A launch epoch and either a long coast (≈ 4500 sec) or a short coast (≈ 1500 sec) solution was calculated using the following:

- Launch location, launch vehicle outgoing asymptote angles, launch energy (C_3)
- Right ascension (RA) and declination (DEC) of the Moon approximately 3.5 days after launch (or at approximately 30 days for a phasing loop transfer to the lunar swingby point).

Table 2 gives an outline of the targeting approach used in designing a direct transfer to the Moon to achieve a nominal HEO mission trajectory. It lists the targeting variables that were used to achieve mission goals at specific points in the Kronos mission trajectory profile.

Table 2: Kronos Mission Targeting Event Sequence for a Direct Transfer Technique

Targeting Event	Initial Conditions	Variables	Goals
Establish transfer trajectory with right energy to reach the Moon.	Launch (L) Epoch Coast (C) Duration Coast to TTI	Impulsive TTI ΔV magnitude (along velocity)	$C_3 \approx -2.0\ km^2/s^2$
Achieve desired RA and DEC of Moon at L+3.5 days.	Propagate to Lunar Distance (370,000 km from Earth)	L, C and TTI ΔV magnitude	Lunar RA and DEC, Lunar Arrival Time
Perform Lunar gravity assist	Propagate to Periselene (closest approach to Moon)	Variables: L, C	B•T, B•R
Evaluate the E-SC-M angle, |R| and Ecliptic Inclination at the PLEP.	Propagate to the post-lunar encounter-descending node in the lunar orbital plane.	L, C and TTI ΔV magnitude	PLEP $\|R\| > 7\ R_E$, ECL INC > 40°, Delta RA = 0°, E-V-M ≈ 180°
Establish orbit period ≈ ½ of Lunar sidereal period	Propagate from PLEP to Post Lunar Encounter Apogee (PLEA)	PAM ΔV (along velocity)	Period ≈ 13.66 days
Examine orbit stability and ability to meet mission requirements.	Propagate for five years	-----	-----

Figure 2 is sample plot of an initial 1-month segment of a direct transfer trajectory viewed in solar rotating coordinates that was targeted for June 1, 2003 launch and then propagated for 5 years. The direct transfer to lunar swingby followed by the return to the PLEP (the PAM location) and a half-revolution of the resulting HEO is shown.

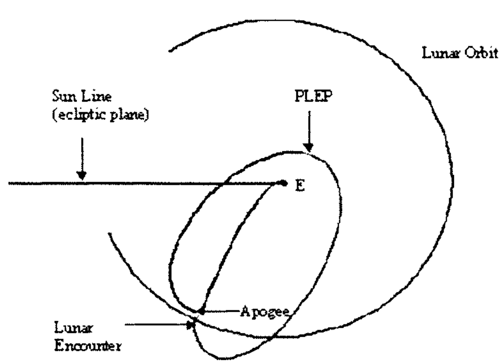

Figure 2: Direct Transfer to HEO

2.2 – Phasing Loop Transfer Technique

The phasing loop case involves completing 2.5 phasing orbits, or "loops" (so-called because consecutive revolutions do not lie on top of each other when plotted in a solar rotating reference frame as is typical in work with these orbits), before the lunar encounter occurs approximately 25 to 30 days after the time of launch. To properly phase the satellite trajectory for a Lunar Swingby without the use of maneuvers at perigee in the phasing loops, both phasing loop apogee radii must extend out to lunar orbit distance. Because the Moon's synodic period is approximately 29.5 days, the Sun-Earth-Moon geometry will be similar at encounter for both the direct transfer and phasing loop cases for launch on a given day of the month. That is, if a direct transfer launch took place at New Moon, the encounter would take place about 3 days after New Moon. A phasing loop launch at new Moon would encounter the Moon anywhere from a few days before the next New Moon (i.e., a synodic month later) to a day after the next New Moon. In other words, if encounter takes place with the Moon near the Sunline for a direct transfer, then a phasing loop transfer launched on the same day will also encounter the Moon when it is near the Sunline. This encounter geometry similarity is because the time spent in the phasing loops is nearly a complete month. The implication here is that a widened launch window could not be obtained by changing strategies by perhaps initially planning a phasing loop trajectory and then hoping to switch to a direct transfer scenario to buy some more time after the phasing loop window had passed. Other possible phasing loop designs such as the "stunted" 3.5 phasing loop scenario being used for the Microwave Anisotropy Probe (MAP) mission might be useful but have not been considered in this preliminary analysis.

The same analysis strategies (specific mission goal values) employed in the direct transfer case were also applied for the phasing loop cases in the month of June 2003. The targeting scheme for the phasing loop case involves one additional item; a maneuver at the first apogee (A1) may be needed to prevent both perigee radii in the phasing orbits from dropping below the Earth's surface. This apogee maneuver is unnecessary for a direct transfer scenario. This A1 maneuver maintains the perigee radius above an assumed minimum acceptable altitude of approximately 450 km.

Table 3 gives an outline of the approach that was taken in designing a phasing loop transfer to the Moon to achieve a nominal HEO mission trajectory. It lists the targeting controls used to achieve mission goals at specific points in the Kronos mission trajectory profile. In this outline, P1 is the first perigee after A1; P2 is the final perigee before the lunar encounter; and, the PLEP occurs at P3.

Table 3: Kronos Mission Targeting Event Sequence for a Phasing Loop Transfer Technique

Targeting Event	Initial Conditions	Variables	Goals
Establish transfer trajectory with right energy to reach the Moon.	Launch (L) Epoch Coast (C) Duration Coast to TTI	Impulsive TTI ΔV magnitude (along velocity)	$C_3 \approx$-2.0 km^2/s^2
Establish 1st phasing loop geometry	Propagate to A1	L, C, TTI ΔV magnitude	$\lvert R \rvert \approx$ 400,000 km (from Earth), RA, DEC of the Moon
Maintain Perigee Radius ($\lvert R \rvert$) > 7000 km	Propagate to P1 and P2	A1 ΔV	$\lvert R \rvert$ > 7000 km
Achieve desired RA and DEC of Moon at L+30 days.	Propagate to near Lunar Distance (370,000 km from Earth)	L, C and TTI ΔV magnitude	Lunar RA and DEC, Lunar Arrival Time
Perform Lunar gravity assist	Propagate to Periselene (closest approach to Moon)	Variables: L, C	B•T, B•R
Evaluate the E-SC-M angle, Radius and Ecliptic inclination at the PLEP.	Propagate to the Descending Node in the Lunar Orbital Plane.	L, C and TTI ΔV magnitude	PLEP $\lvert R \rvert > 7\ R_E$, ECL INC > 40°, Delta RA = 0°, E-V-M ≈ 180°
Establish orbit period ≈ ½ of Lunar	Propagate from PLEP to PLEA	PAM ΔV	Period

sidereal period		(along velocity)	≈ 13.66 days
Examine orbit stability and ability to meet mission requirements.	Propagate for five years	-----	-----

Figure 3 is a sample plot of a phasing loop trajectory that was targeted for a June 1, 2003 launch and propagated for 5 years as viewed in solar rotating coordinates. The figure depicts the initial portion of the mission from launch through the 2.5 phasing loop transfer to the Moon, on through the lunar swingby to the PLEP, and finally a half revolution in the HEO ending at the PLEA following the PLEP. The time of flight in this trajectory segment is approximately two months.

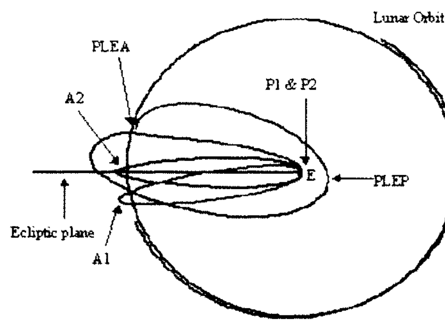

Figure 3: Phasing Loop Transfer to HEO

2.3 – Eclipse Analysis Approach

One important goal of the Kronos mission is to survive eclipses. Considering the geometry of the HEO, the trajectory crosses the ecliptic plane twice per orbit and it is around these nodal crossings that an eclipse is most likely to occur. Though eclipses cannot be designed out of the mission entirely, attempts can be made to minimize the eclipse frequency and duration. The easiest way to reduce eclipses is to orient the line-of-apsides out of the ecliptic plane. The current method uses the Moon's gravity to torque the orbit plane out of the ecliptic. This is an essential part of the trajectory design. Another option is to perform a pair of line-of-apsides maneuvers, which can rotate apogee and perigee out of the ecliptic plane. This method was not investigated for this analysis, as it is a plainly costly option in terms of delta-V.

2.4 – Launch Vehicle Errors and Launch Window Approach

Three sigma launch vehicle energy errors were examined for three direct transfer cases and three phasing loop transfer cases with launch dates in June 2003. In this analysis, a ±10.5 m/s (corresponding to $\pm 3\sigma$ errors) magnitude impulsive ΔV error is applied after TTI for each of the nominal cases examined. These numbers are based on analysis for the MAP project for a similar launch vehicle. The following variables used in the baseline trajectories were frozen for the launch error cases:

- Launch Epoch
- Coast Duration
- Impulsive nominal TTI ΔV magnitude and direction

For the direct transfer cases, a mid-course correction ΔV introduced only 7 hours or more after TTI will be available as a control to target to the B-Plane incoming asymptote. One needs to wait this long after TTI to perform the maneuver in order for an accurate Orbit Determination (OD) solution to become available. The direct transfer cases that were analyzed used an energy correction impulsive ΔV 7 hours after TTI. The phasing loop cases use impulsive maneuvers at each perigee (P1 and P2) in the phasing loops to control the timing and encounter geometry of the lunar swingby. The perigee delta-Vs are applied tangentially to the orbit and affect the apogee distance and period of the subsequent phasing loop.

A launch window analysis was performed for each of three selected nominal direct transfer cases and each of three selected nominal phasing loop cases for June 2003. For a given trajectory, a number of cases are analyzed as part of the launch window analysis. One has to analyze the nominal (no TTI dispersions) trajectory as well as the $\pm 3\sigma$ TTI dispersion cases. Additionally, for each of these three (nominal TTI, $\pm 3\sigma$ TTI) cases for a given trajectory, one has to introduce launch delays of varying magnitude in order to determine the affect of both TTI dispersions and launch delay on the final mission orbit.

2.5 – Solar and Lunar Perturbation Analysis

High apogee, highly eccentric orbits are extremely sensitive to lunisolar perturbations. In many cases, they cause perigee to decay to the point of impact with the Earth's surface. Hence, there is frequently the need to do a perigee-raising maneuver at A1 for the pre-swingby phasing loops. To test the conjecture that the chosen HEO geometry eliminates such mission terminating secular perturbations, however, each HEO trajectory was propagated to the end of the 5-year mission and plots of perigee radius, ecliptic inclination, declination of apogee, etc. vs. time were generated in each case. In addition, to determine the relative impact of the Sun and Moon individually on the trajectory, we numerically integrated representative HEO trajectories with full force modeling of Sun, Earth, and Moon and then integrated the trajectories again with the Moon perturbing-body effects eliminated from the force modeling. The time variations of the orbital elements in both cases were compared graphically. Finally, anticipating that end of life disposal questions will inevitably arise, we looked at the long term variation of the HEO orbital elements by propagating several of the trajectories for 100 years to see just how stable the HEO orbits are.

3 – TRAJECTORY DESIGN RESULTS

3.1 – Direct Transfer and Phasing Loop Technique

As previously discussed, the goal for the direct transfer and phasing loop transfer launch dates is to find an initial PLEP ecliptic inclination and PLEP radius that yield perigee radii above 7 R_E and maximum ecliptic inclination over the mission lifetime. After calculating numerous direct and phasing loop transfer trajectories, it was determined that it is not possible to maintain an ecliptic inclination between 40° to 70° for an entire 5-year mission. Perturbations due to the Sun and Moon force the ecliptic inclination down to a value as low as 20°. Only with the use of plane change maneuvers of prohibitive size can the trajectory inclination be maintained between 40° and 70°. The strategy used to achieve the PLEP ecliptic inclination and radii goals was to target to orbits with initial PLEP ecliptic inclinations greater than or equal to 40° and PLEP radii between 10 and 15 R_E. These desired goals were achieved for 21 launch dates in June 2003 (Reference 6). The combination of the direct and phasing loop transfer techniques provided launch opportunities from the beginning of New Moon phase, to the beginning of Full Moon phase and covering the duration of Last Quarter. No direct transfer designed orbits meeting the aforementioned criteria were found for Full Moon phase and no phasing loop designed orbits meeting the aforementioned criteria were found for Last Quarter phase, as shown in Figure 4.

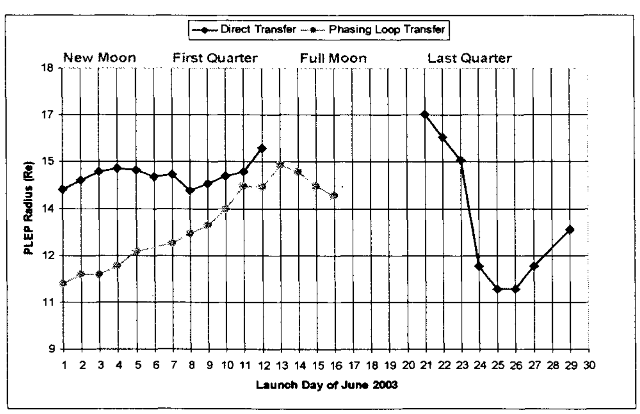

Figure 4: Nominal June 2003 Direct and Phasing Loop Transfer PLEP Radius Summary

For the direct transfer designed orbits launched during New Moon and First Quarter, a varying long parking orbit coast solution was used to achieve mission goals while a short coast solution was determined to be optimal for the launch dates occurring in Last Quarter. For the phasing loop transfer designed orbits launched during New Moon and First Quarter, a varying short coast solution was used to achieve mission goals. Long coast solutions were found for only three launch dates occurring around Full Moon. However, these trajectories barely met the ecliptic inclination constraint and violated the E-SC-M angle constraint of $150.0° < \text{E-SC-M} \leq 180°$. It was found that maintaining the E-SC-M goal keeps the lunar perturbation effects down to a minimum.

The launch dates attempted near or during New Moon tend to have the highest total mission delta-V. It was seen in the results that the higher PAM ΔVs were correlated with a higher PLEP radius. Additionally, the PLEP radius value is a determining factor in maintaining an altitude above 7 R_E for the duration of the mission. Figure 5 shows the total delta-v results for nominal direct and phasing loop transfer launch dates during New Moon through Last Quarter, respectively. Notice that the total delta-V for the phasing loop transfer trajectories is greater than the direct transfer cases. This difference is attributed to the A1 maneuver that is needed to maintain the perigee radii in the phasing loops above a safe altitude (greater than 450 km). Figure 6 shows that the Sun's perturbations effectively act as a negative delta-V when the spacecraft is near apogee and the HEO is oriented as at point A. In this case, the perturbation is opposite the direction of the spacecraft velocity and the P1 altitude will decrease. At the beginning of New Moon or just after point C from Figure 6, the A1 ΔV is at 12 m/s. It then reaches a peak of 27 m/s for the June 7 launch date and then decreases to a minimum of 5 m/s on the June 14 launch date during the beginning of Full Moon. The situation is just reversed at point B where the Sun's perturbation is in the direction of the velocity and acts as a positive apogee delta-V, thus increasing the perigee altitude. At points C and D, the Sun has a minimal effect on the P1 height.

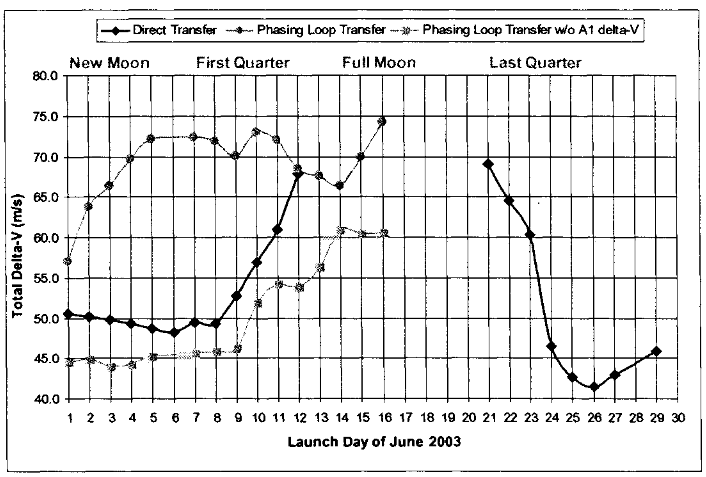

Figure 5: Nominal June 2003 Direct and Phasing Loop Transfer PLEP Radius Summary

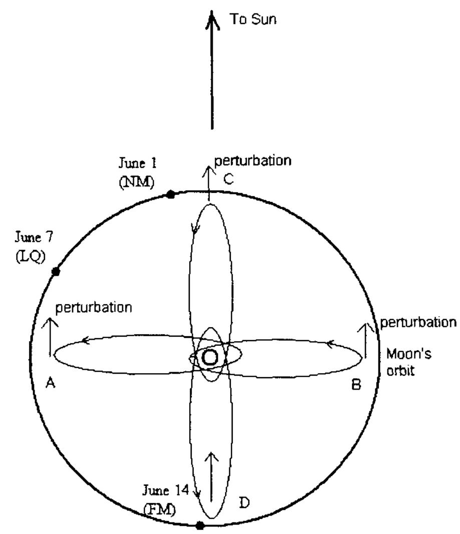

Figure 6: Effect of Solar Perturbations on 1st Phasing Loop Perigee Pass Height

The ecliptic inclination and the perigee and apogee radius oscillate over time but the semimajor axis remains constant. Figures 7 and 8 show the evolution of ecliptic inclination, perigee and apogee radius after lunar encounter for a mission lifetime of six years for the complete set of nominal trajectories calculated for the month of June. Figure 7 demonstrates that the ecliptic inclination values achieve a minimum in the 20 to 25° range, but start to increase after 2 years and eventually reach values between 50 and 65°. Once the ecliptic inclination and PLEP radius reach a minimum (apogee radius is at a maximum) as shown in Figure 8, both begin to increase and then eventually surpass the initial PLEP values. During this latter time, apogee radius decreases to a minimum. The apogee radii values during the first 2 years increase to a peak value of 70 R_E and then decrease to a range of 50 to 60

R_E toward the end. The perigee radii values initially decrease to 7 R_E after 2 years but ultimately peak at approximately 28 R_E after six years.

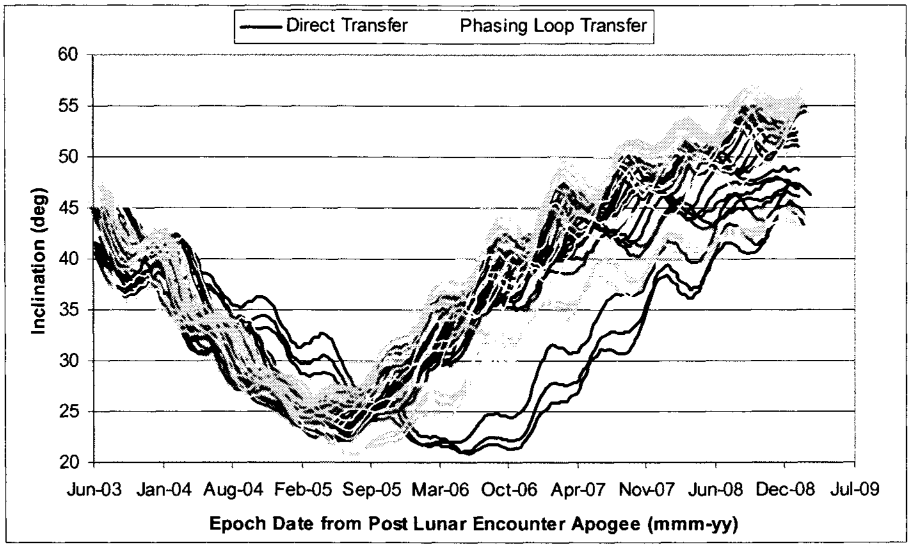

Figure 7: Evolution of Ecliptic Inclination for Nominal Trajectories

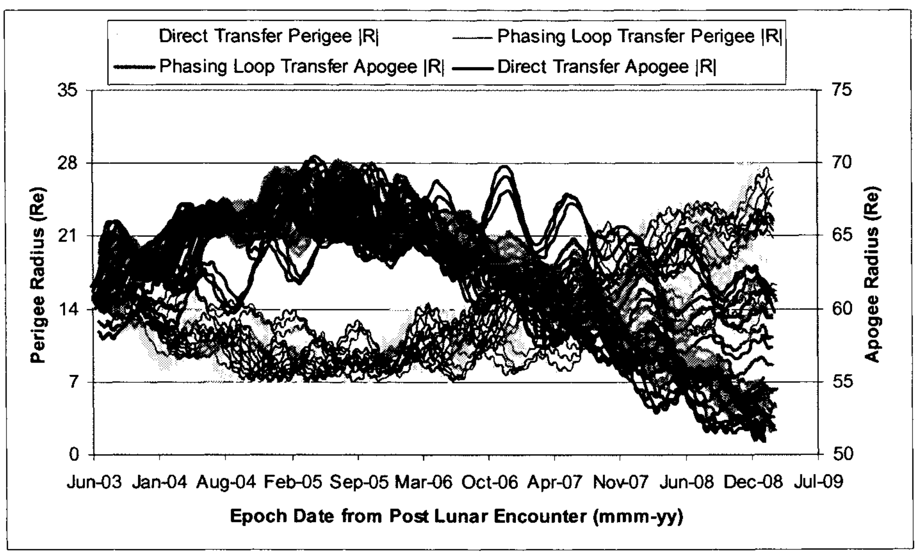

Figure 8: Evolution of Apogee and Perigee Radius for Nominal Trajectories

3.2 – Eclipse Event Results

Figure 9 shows the eclipse duration during a 6-year mission for all the computed launch dates in June. The eclipse events are mostly Earth shadow events with only a couple of lunar eclipse (penumbra) events appearing during each six-year mission. The eclipse cycle for each of the computed trajectories ranged from two to three per year during their six year mission propagations. Note that usually any eclipse event exceeding 60 minutes in duration will involve an umbra cone (except cases involving the Moon, which will only have penumbras). The majority of eclipse events have durations of 200 minutes or less and mostly occur at true anomalies approximately 90° off of apogee, or where the trajectory plane intersects the ecliptic plane. It was observed that when the line-of-apsides of the HEO is close to the Earth-Sun line and the ecliptic declination is below 25 °, the eclipse durations have exceeded 200 minutes and tended to occur less than five days away from apogee. Figure 10 shows the evolution of the ecliptic declination at apogee for all the computed trajectories in June. Also, if the ecliptic declination at apogee approaches zero, it increases the likelihood of long duration eclipses that could exceed a half-day. Figure 11 is a sample plot of a direct transfer trajectory that was targeted for a June 26, 2003 launch and propagated for one orbit starting after PLEA. The figure depicts the geometry of the line-of apsides with respect to the ecliptic plane and even though the ecliptic inclination of this orbit is ≈ 45°, the line-of-apsides is only ≈ 20° off the ecliptic plane.

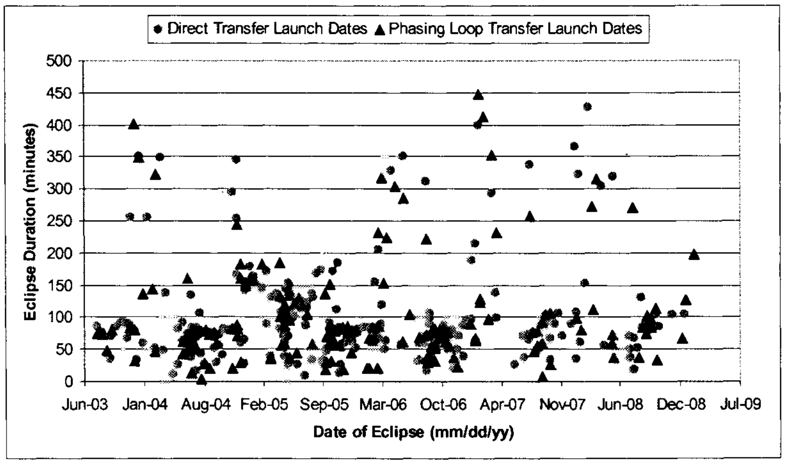

Figure 9: Eclipse Duration for Nominal Trajectories in June 2003

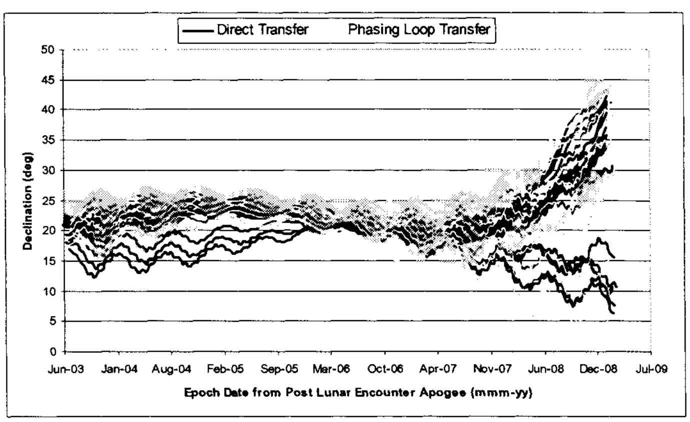

Figure 10: Evolution of Ecliptic Declination for Nominal Trajectories in June 2003

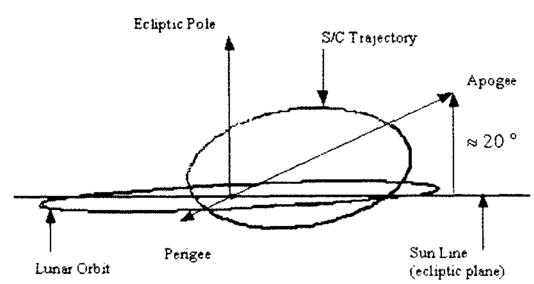

Figure 11: Geometry of the Line-of-Apsides of a Nominal June 26, 2003 Direct Transfer

3.3 – Launch Vehicle Errors and Launch Window Results

For this analysis, one starts with a nominal trajectory that meets the baseline mission constraints. One then first introduces TTI energy dispersion errors and then combinations of energy dispersion errors and launch delays. Launch delays of 5, 10, and 20 minutes were analyzed. If a given launch delay results in a mission orbit that does not meet the previous derived constraints, then larger launch delays are not analyzed. Note that the PLEP targets were allowed to float, but not to the extent of constraint violation, for the dispersion and launch delay cases, if needed, to maintain a suitable mission orbit.

A set of six nominal trajectories, three from the direct transfer design and three from the phasing loop design, were investigated for this analysis. It was assumed that if all 6-baseline cases can meet all mission conditions that were established earlier, then it should be possible to establish several additional launch opportunities near the epoch of each nominal case examined.

Table 4 shows a summary of mission total delta-V results for the direct and phasing loop-designed nominal, $\pm 3\sigma$ error, and launch window delay (+5, +10, +20 minute) cases together with the corresponding lunar phase at launch for the selected launch dates in June 2003. Both the direct transfer design and the phasing loop design methods established the ability to maintain a 20-minute launch window, as delta-V results were acceptable in all cases. Notice the phasing loop approach generally requires less ΔV to correct launch vehicle errors and allows more time to correct potential spacecraft problems. It does, however, take longer to get to the Moon and requires two or more passes through the Earth's radiation belts. Note that the launch dates, which occur when the Moon is near conjunction with the Sun (New Moon or Last Quarter), yield the most favorable launch windows.

Table 4: TTI Error Dispersions and Launch Delay Delta-V Summary

Error Case Event	Direct Transfer (m/s)	Lunar Phase	Phasing Loop Transfer (m/s)	Lunar Phase
Nominal Trajectory Maximum Total ΔV	69.1	LQ	74.2	FM
Nominal Trajectory Minimum Total ΔV	41.5	LQ	57.1	NM
+3-Sigma Trajectory Maximum Total ΔV	110.7	FQ	121.9	NM
+3-Sigma Trajectory Minimum Total ΔV	97.6	NM	112.7	FQ
-3-Sigma Trajectory Maximum Total ΔV	114.5	FQ	95.1	FQ
-3-Sigma Trajectory Minimum Total ΔV	87.8	LQ	91.8	NM
5-Minute Delay Trajectory Maximum Total ΔV	89.0	NM	76.0	FQ
5-Minute Delay Trajectory Minimum Total ΔV	57.2	LQ	67.4	NM
+3-Sigma/5-Minute Delay Trajectory Maximum Total ΔV	126.6	NM	120.3	NM
+3-Sigma/5-Minute Delay Trajectory Minimum Total ΔV	99.0	LQ	114.9	FQ
-3-Sigma/5-Minute Delay Trajectory Maximum Total ΔV	110.6	FQ	100.4	FQ
-3-Sigma/5-Minute Delay Trajectory Minimum Total ΔV	81.6	NM	92.5	NM
10-Minute Delay Trajectory Maximum Total ΔV	128.1	NM	82.3	FQ
+3-Sigma/10-Minute Delay Trajectory Minimum Total ΔV	163.1	NM	117.9	FQ
-3-Sigma/10-Minute Delay Trajectory Maximum Total ΔV	100.9	NM	105.8	FQ
20-Minute Delay Trajectory Maximum Total ΔV	85.1	LQ	71.6	NM
+3-Sigma/20-Minute Delay Trajectory Minimum Total ΔV	135.5	LQ	117.3	NM
-3-Sigma/20-Minute Delay Trajectory Maximum Total ΔV	94.6	LQ	95.9	NM

3.4 – Solar and Lunar Perturbation Analysis Results

It is well known that perturbations due to the Sun on the Earth-Moon system cause both secular and cyclic changes in the Moon's orbital elements. For example, the Moon's line-of-nodes regresses continuously completing a full cycle in about 18.6 years and the line-of-apsides rotates through a full 360° in approximately half this time. There are cyclic variations in the Moon's inclination and eccentricity. For the HEO, we expect the Sun to affect the trajectories in a similar manner but we expect no secular changes in perigee radius and inclination because of our choice of initial Sun-Earth-Moon-SC geometry and orbital period. Figures 12, 13, and 14 show the time variation of perigee radius, ecliptic inclination, and argument of perigee for a typical HEO propagated both with and without lunar gravity.

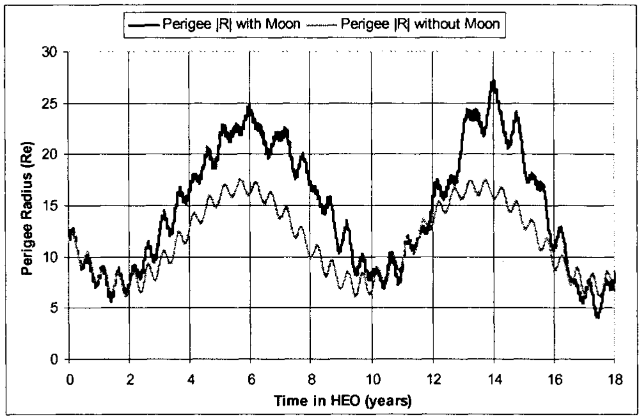

Figure 12: Evolution of Perigee Radius with and without Lunar Perturbations

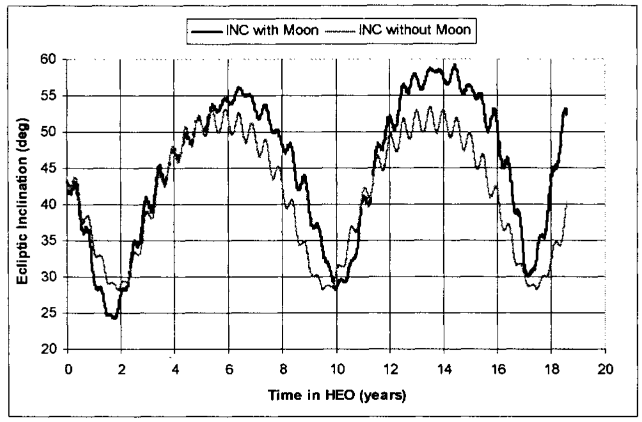

Figure 13: Evolution of Ecliptic Inclination with and without Lunar Perturbations

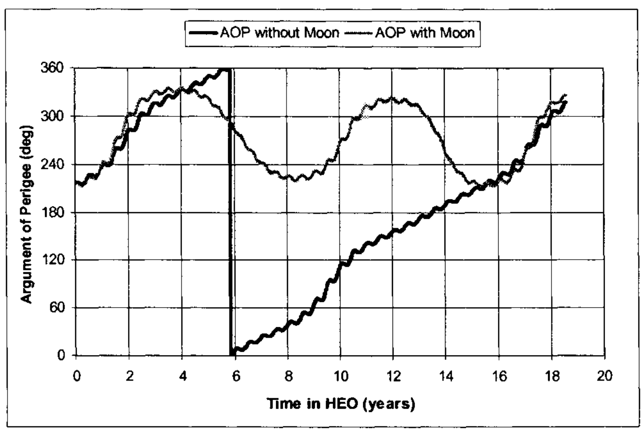

Figure 14: Evolution of Argument of Perigee with and without Lunar Perturbations

As was expected, the Sun dominates the behavior of the orbit but the Moon adds some interesting complications. The Moon affects the inclination very little but tends to increase the amplitude of the perigee radius oscillation. The minimum perigee radius is almost unaffected but the maximum increases.

Interestingly, the right ascension of the ascending node regresses in a manner unsurprisingly similar to that of the Moon, completing one revolution in about 19 years. The argument of perigee, however, oscillates about a mean value of approximately 275° with an amplitude of 50° to 75° (there is apparently a long-term variation in the amplitude) and a period of approximately 8 years. This would seem to indicate that, at least in this case, the line-of-apsides does not advance secularly. The implication of this for our purpose is that with judicious choice of launch day or with a line-of-apsides rotating maneuver, the apogee position can be controlled within limits. This could be useful, for example, if it were desirable to keep apogee in the Northern Hemisphere for communications or other purposes.

Unfortunately, nature is seldom simple and when we examined a case with a very low ecliptic inclination, this oscillatory behavior of the line-of-apsides was not observed. Instead, the line-of-apsides rotated secularly at approximately the same rate as the nodal regression. At this point, we are not certain whether there is some critical initial inclination at which the line-of-apsides begins to oscillate instead of rotating or whether some other factor is responsible for this behavior. In this preliminary investigation, we did not complete a parametric study for a range of initial inclinations.

Another fascinating phenomenon was observed when we attempted to examine the long-term stability of the HEO by propagating the trajectories for 100 years. Figures 15 and 16 show the behavior of the argument of perigee vs. time for both the initial ecliptic inclination cases of 6° and 44°. For the 6° case, the variation is quite regular for the entire 100 years except that there is a long-term increase in the average value of inclination and its amplitude. For the 44° case, however, the behavior is unremarkable until approximately 75 to 80 years into the propagation. At this point, the nodal regression rate slows by roughly one-half, the line-of-apsides stops oscillating and begins advancing secularly at about one rotation every 8 years, and the ecliptic inclination becomes nearly constant near its maximum value as does the perigee radius also. At this time, we can offer no hypotheses about this anomaly nor can we, with any degree of assurance, suggest it might be a software problem. It is a curious problem awaiting future investigation and resolution.

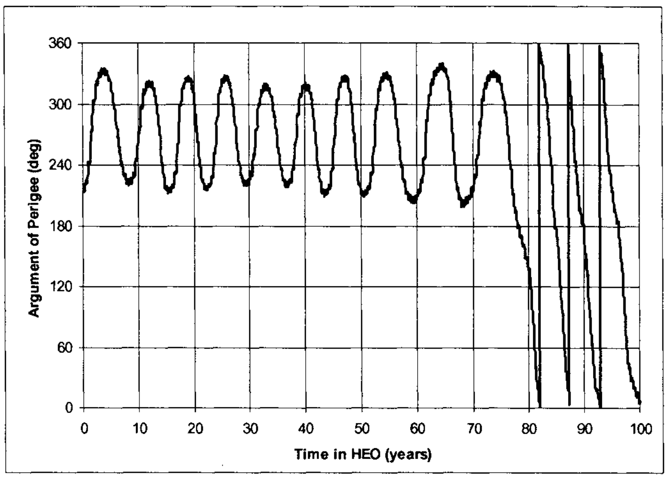

Figure 15: Evolution of Argument of Perigee with Lunar Perturbations for 44° Inclined Orbit

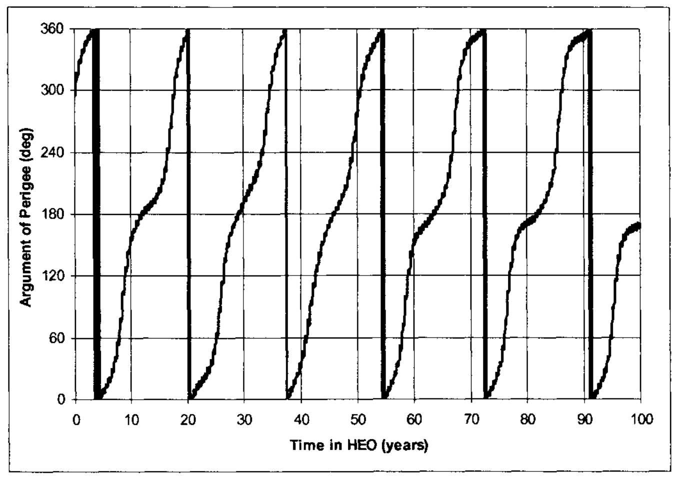

Figure 16: Evolution of Argument of Perigee with Lunar Perturbations for 6° Inclined Orbit

4 – CONCLUSIONS AND FUTURE DIRECTIONS

Approximately 150 individual trajectories were designed and examined to look for trends and general conclusions that could be used to establish guidelines for designing the Kronos baseline mission trajectory. Unfortunately, the lunisolar perturbations have greatly complicated the analysis so that it has been difficult to be as incisive as one might like. It is apparent that both the direct and phasing loop transfer techniques are feasible. We have found that suitable trajectories can found for 11 to 13 days of each month, assuming the month of June to be reasonably representative.

There are advantages and disadvantages for both the direct and phasing loop transfers. The ΔV budget for the direct transfer design is slightly higher (~ 140 to 170 m/s) because of the need to correct the 3σ launch vehicle energy errors within approximately seven hours of launch. However, it should be pointed out that these delta-V numbers represent impulsive delta-Vs. Possible finite burn losses have not been investigated. The phasing loop transfer does not require such a severe time-critical error correction maneuver but it does introduce the necessity of performing perigee raising maneuvers at A1 for many possible launch dates because solar and lunar perturbations can drive perigee down sufficiently to result in spacecraft reentry. The ΔV budget for the phasing loop scenario is approximately 120 m/s.

No trajectories could maintain a high ecliptic inclination for the entire five-year mission lifetime. Typically, the inclination varies between 23° to 25° and 55° to 65°. For the June trajectories, judicious choice of launch date and PLEP orbit parameters, however, can give trajectories that maintain perigee radius above the geosynchronous radius for the entire mission lifetime. It should be noted that even though perigee radius and ecliptic inclination oscillates between certain limits, these trajectories are actually very stable. One case was propagated for 250 years to determine if it would eventually leave the HEO orbit and perhaps reenter the Earth's atmosphere. It did not; in point of fact, this trajectory maintained a perigee radius greater than 7 R_E for the first 28 years.

No matter what choice of PLEP targets is used, it appears impossible to avoid eclipses. The Sun must pass through the orbit plane twice per year regardless. Eclipses with the spacecraft near perigee will be less frequent and of shorter duration, but when the Sun is eclipsed with the spacecraft near apogee, shadow periods of several hours' duration were observed.

Several HEO orbits with ¼ lunar period orbits were examined and showed essentially the same behavior as the ½ lunar period orbits. The only significant difference was the approximately four to five times greater PAM ΔV needed to establish the proper phasing with the Moon (References 1 and 6).

It would be desirable to take this analysis further in several respects. Launch window and error analysis should be performed for more than the few cases examined so far in order to pin down the ΔV budgets more precisely. Any possible finite burn losses should also be investigated. The maximum eclipse duration should be determined to see if it might impact spacecraft design or conflict with mission requirements. Further analysis of the effects of the lunisolar perturbations should be performed to better predict optimal launch opportunities and initial orbital elements.

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