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Earth Orbit Raise Design for the ARTEMIS Mission

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ARTEMIS is a mission to send two spacecraft from Earth orbit to libration orbits around the Moon Lagrange points and then into lunar orbit. Lunar flybys were used early in the mission to send the spacecraft into low-energy lunar transfers which were designed to arrive in the libration orbits for minimal ΔV. ARTEMIS began by raising the Earth orbits of each spacecraft to achieve the planned lunar flybys. Spacecraft configuration and operation constraints made the Earth orbit raise phase of the mission a significant mission design challenge by itself. This paper describes the process used to find trajectories that achieved mission goals and the resulting series of Earth orbits that culminated in successful lunar flybys.

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Nomenclature

$\bar{\theta}$ half angle of thruster pulse width, radians
ΔV thrust induced change in velocity, meters/second
V_∞ Ballistic velocity at infinity relative to a specified central body kilometers/second

I. Introduction

ARTEMIS, the Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun mission, has now arrived safely in lunar orbit, after long journeys by each of its two probes. The two spacecraft designated P1 and P2, took almost two years from their initial Earth orbits to reach the target lunar orbits. Ironically, the part of each journey that was initially considered the easiest turned out to be the most difficult of all. The preliminary mission design[1] modeled the orbit raises as a single impulsive maneuver to reach a lunar flyby(s) that would begin the low-energy transfers to Earth-Moon libration orbits. The ARTEMIS probes P1 and P2 successfully completed the orbit raise and reached the libration orbits. The P1 and P2 spacecraft completed their respective libration orbits and then entered lunar orbits on June 27 and July 17, 2011, respectively.

ARTEMIS is an extension of THEMIS,[2] the Time History of Events and Macroscale Interactions during Substorms mission. THEMIS consisted of five identical spacecraft in varying sized Earth orbits designed to make simultaneous measurements of the Earth's electric and magnetic environment. THEMIS observed geomagnetic storms resulting from the solar wind's interaction with the Earth's magnetosphere. THEMIS was meant to answer the age old question of why the Earth's aurora can change rapidly on a global scale. The goal of the ARTEMIS[3] mission extension is to deliver the field and particle measuring capabilities of two of the THEMIS spacecraft to the vicinity of the Moon. The ARTEMIS mission required transferring two Earth orbiting THEMIS spacecraft on to lunar orbit, transfers that began by raising the geocentric orbit apogees to the Moon's orbit. This paper describes the processes that resulted in successful orbit raise designs for both spacecraft and expands on the corresponding section of the ARTEMIS mission design paper.[4]

I.A. Overview of ARTEMIS

The two THEMIS spacecraft that were chosen to become ARTEMIS were the two outermost spacecraft in the THEMIS constellation, called P1 and P2, with P1 traveling in an eccentric four-day orbit around

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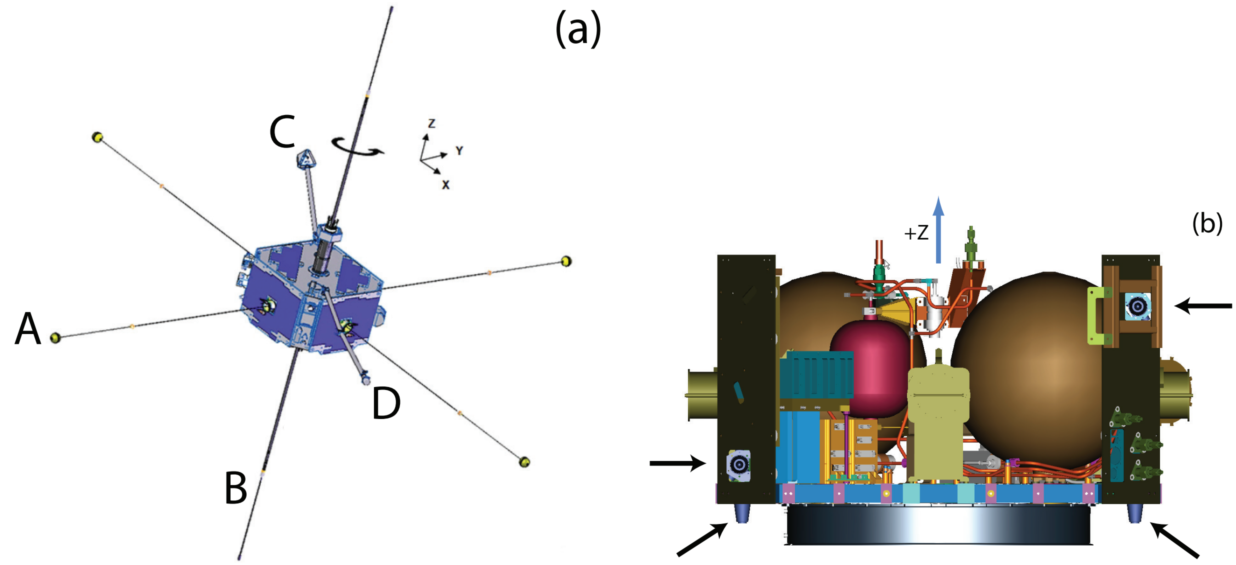

Figure 1. THEMIS/ARTEMIS probe configuration. The probe buses were manufactured by ATK Space Systems (formerly Swales Aerospace), and the instruments were manufactured under the leadership of the University of California, Berkeley with both US and international collaborators. (a) On-orbit configuration with booms deployed, adapted from Auslander et al. (2008): A – four 20 m long radial EFI booms; B – two 5 m long axial EFI booms; C – 1 m long SCM boom; D – 2 m long FGM boom (*http://www.nasa.gov/images/content/164405main_THEMIS-Spacecraft_bus2.jpg*), (b) probe bus schematic. Black arrows indicate locations of the 4.4 N hydrazine thrusters. Blue arrow indicates spin axis. *(Used by kind permission from figure 3 of Sweetser et al.,[4] ©Springer Science+Business Media B.V. 2012.)*

Earth and P2 in a two-day orbit around Earth, with perigees at about 1860 km and 3330 km altitude respectively. All of the THEMIS spacecraft were identical and their configuration is shown in figure 1. They were launched together with identical propellant loads, so that the spacecraft could be assigned to operational orbits according to how well they were performing after launch and initial on-orbit checkout.

These spacecraft were designed to be as simple as possible to minimize their cost. They are spin stabilized with 20 meter long electric field booms as well as several shorter magnetometer booms. Each spacecraft has only four 4.4 N thrusters, mounted so that two can control the spin rate up or down and two can be used to precess the spin axis. When either pair is fired together they can also provide translational thrust, in the spin plane by pulsing the spin-rate control thrusters and in the +Z direction by firing the precession thrusters. This configuration meant that there was no way to perform a maneuver with ΔV in the -Z direction, which is toward ecliptic north as the mission has been flown.

As a consequence of filling the tanks on all the spacecraft with enough propellant to meet the requirements of the most demanding orbit, the two outermost spacecraft were estimated in the ARTEMIS proposal to NASA to contain enough propulsive capability at the end of the THEMIS primary mission to change their velocity by a total ΔV of 300 m/s and 450 m/s respectively (estimates which have since been refined to 320 m/s and 467 m/s respectively). Direct transfers of the two spacecraft P1 and P2 to large lunar orbits would require impulses totaling 370 m/s and 500 m/s respectively. Actual total ΔVs required for simple transfers are actually larger than 370 m/s and 500 m/s as a result of gravity and steering losses. Therefore simple transfers were ruled out. The only possible transfers were lengthy low-energy lunar transfers. The orbit raise phase targeted lunar gravity assists to reach the low-energy transfer orbits.

The low-energy transfers saved some ΔV in the orbit raises that began the mission by going propulsively to orbits that did not reach all the way to the Moon's orbit. The trajectories then took advantage of distant lunar approaches over several months to raise the spacecraft orbit the rest of the way to the lunar flybys. More ΔV was saved at lunar arrival since low-energy transfers result in rendezvous with the Moon with an arrival state that is already elliptical (though with an unstably high eccentricity) instead of a hyperbolic approach with a V_∞ of about 0.8 km/s. Thus the initial proposal design for the transfers budgeted 121 m/s and 214 m/s for the respective orbit raises (including allocations of 20 m/s and 28 m/s for gravity and steering losses), and lunar orbit insertions of 94 m/s and 121 m/s respectively (including allocations of 7 m/s and 12 m/s for gravity and steering losses). In the proposal, there was still predicted at least a 20% propellant margin on each spacecraft. This budget included the relatively small deep-space maneuvers in the low-energy transfers, allocations for trajectory correction maneuvers, and for orbit maintenance during a period planned to be spent in unstable Earth-Moon libration orbits.

Figure 2 shows the designs of the low-energy transfers that were developed for the ARTEMIS proposal to NASA headquarters. It is important to note that these transfers begin with the two probes already in orbits approaching the lunar orbit. At that stage of mission formulation the trajectory designers modeled the initiation of ARTEMIS as a single periapsis maneuver on each spacecraft to raise their apoapsis enough to achieve the needed lunar approaches and gravity assists. It was assumed at the time that it would be easy to implement this periapsis maneuver as a series of smaller periapsis maneuvers on several orbits to satisfy constraints on the capability of the small thrusters—easy both in the design process and in being able to do the orbit raises for about the same amount of propellant as a single impulsive maneuver. This assumption proved to be dramatically wrong when the time came to design that series of maneuvers.

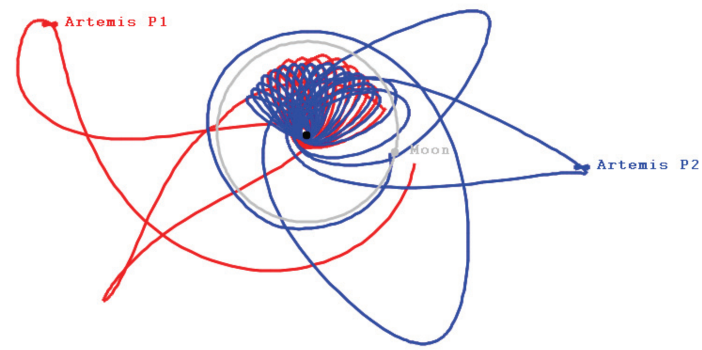

Figure 2. ARTEMIS trans-lunar trajectories in the ecliptic plane. The coordinate frame here rotates such that the Sun is always to the left. The red line shows the P1 trajectory and the blue line shows the P2 trajectory. The Earth is at the center of the figure and the Moons orbit is shown in gray. Spacecraft labels correspond to the planned positions on June 25, 2010. *(Used by kind permission from figure 1 of Broschart et al.[1])*

The entire ARTEMIS mission was divided into phases: the orbit-raise phase, the low-energy transfer phase, the lissajous phase, and the lunar orbits phase. Each phase was distinct in its trajectory characteristics and required a different approach for trajectory design and navigation planning. After the preliminary design was completed for the NASA proposal (except, as noted above, for the orbit-raise phase), and faced with limited time and resources for mission development, we made a strategic decision to freeze the trajectory architecture and allow detailed design to proceed on each of the phases in parallel by constraining the boundary states of the phases to those in the preliminary design. Thus the design of the series of maneuvers to raise the initial orbits of P1 and P2 had the goal of arriving into predetermined lunar gravity assists to begin the low-energy transfers.

I.B. The challenges of the ARTEMIS orbit raise design

There are many reasons why the trajectory design for the orbit raise phase of both P1 and P2 turned out to be significantly more difficult than anticipated. The reasons include the spacecraft maneuvering constraints, the highly constrained ΔV budget, the required precision phasing to reach the proposed low-energy transfers to the Moon, the significantly different initial states assumed in the original proposal when designing the orbit raise and trans-lunar trajectories, the single impulse by which the original proposal approximated the orbit raise when in reality many small burns are required, the fact that optimal design of highly elliptical transfers is numerically difficult, and the fact multiple lunar approaches during a long slow orbit raise are unavoidable creating a very complex design space.

Since both spacecraft are spinning, thrusters in the spin plane must operate in a pulsed mode matching the spin frequency to achieve a net inertial burn direction. The majority of the orbit raise required maneuvers in the spin plane. The pulse size was limited by the loss of efficiency due to thrust cancellation described by

$$efficiency = \frac{sin(\bar{\theta})}{\bar{\theta}}. \tag{1}$$

The efficiency in equation (1) is defined to be the fraction of the net ΔV generated by the thruster after vector cancellation due to the varying burn direction divided by the total ΔV produced by the thruster. The angle $\bar{\theta}$ in equation (1) is one half of the angle the spacecraft sweeps through during a pulse. For a pulse sweep size of $\bar{\theta}$ near zero degrees, the efficiency is $\frac{sin(\bar{\theta})}{\bar{\theta}} \to 1$. Obviously, long pulses greatly reduce the efficiency and are unacceptable given a limited ΔV budget. Short pulses, on the other hand, reduce the time averaged thrust magnitude resulting in either longer burn time and larger gravity losses or more maneuvers on distinct perigee passages.

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A further constraint on spin plane thruster pulse durations is due to certain propellant loads being susceptible to fuel slosh resonances. Fuel slosh resonances required burn pulse sizes to be even lower than those required for acceptable efficiencies for part of the P2 orbit raise. In this way, slosh resonance reduced the net thrust magnitude. This made the design of the P2 orbit raise significantly more challenging to end the orbit raise in time to rendezvous with the planned low-energy transfer to the Moon.

A limitation of both spacecraft is that thrusting can only occur in full sunlight with at least two minutes of margin for attitude acquisition. Unfortunately, the Earth's shadow covers perigee for most of the orbit raise time period resulting in no thrusting allowed near perigee. This limitation increased gravity losses over those accounted for in the proposal. To mitigate this loss shorter maneuvers were implemented requiring more maneuvers over all. However this compounded the difficulty of reaching the low-energy transfer interface on time.

The vast majority of the orbit raise maneuvers were performed using the spin plane thrusters. However, due to the thruster alignment, spin plane thrusting results in spin axis precession. In this way the direction (the spin plane) of future maneuvers is impacted by the current maneuver. The design had to account for this precession and monitor it for possible attitude constraint violations.

For both spacecraft, individual commanded burns are limited to a net inertially fixed direction. This means the burn pulses in a multiple pulse (one pulse per revolution) burn must remain centered about a fixed inertial direction in the spin plane. This limitation results in a reduction in efficiency due to thrust steering losses. Individual commanded burns are required to be separated by a minimum of two minutes of thrust off time. Thrust direction variation during a single perigee passage can be roughly approximated by commanding more than one burn each separated by at least two minutes. However breaking up a single perigee maneuver into many smaller maneuvers to reduce steering losses resulted in increasing gravity losses due to the 2 minute down time. A fairly simple analysis demonstrated more than three individual commanded burns during a single perigee passage was generally not advantageous. This is because the reduction in steering losses for more than about three burn directions does not exceed the increase in gravity losses that result. In addition the operational complexity of multiple commanded burns increases risk.

An important aspect of the propulsion system used on the ARTEMIS spacecraft is that the thruster specific impulse and thrust magnitude drop significantly with declining propellant loads (system pressure). This fact did not directly influence the trajectory design because minimizing this effect is equivalent to minimizing the orbit raise ΔV. The orbit raise phase ΔV budget for P1 was ≤ 110 m/s and for P2 ≤ 240 m/s.

Another trajectory design challenge is associated with maneuvers being limited to the spacecraft spin plane. The usual intuition of 1 burn allowing targeting of 3 orbital elements so that 2 burns separated in time allows for the targeting of 6 orbital elements is not applicable. In fact, even 3 separated burns can fail to provide 6 element targeting when all maneuvers are confined to a single plane. The goal of the orbit raise maneuvers was to rendezvous with the proposed low-energy transfers to the Moon. These rendezvous required fixed time six-state targeting.

The final challenge of the orbit raise design was the result of THEMIS science planning decisions that could not be anticipated at the time of the ARTEMIS proposal. The THEMIS science decisions resulted in unanticipated maneuvers and therefore the actual initial orbits at the start of the orbit raise phase were significantly different than those assumed in the proposal. For example, see figure 3 which shows the dramatic difference between the proposal and actual initial orbit raise orbits for the P2 spacecraft. Both the orbit planes and longitude of the ascending nodes for both spacecraft were significantly different than those initially assumed in the proposal. Propulsive correction of the initial condition differences was prohibitive given the limited ΔV budget. Instead, lunar interactions near the end of the orbit raise were relied on to compensate for these differences so that the rendezvous with the proposed low-energy transfers to the Moon could still be achieved.

II. Orbit raise design strategy

The P1 and P2 orbit raise designs were constructed using Mystic software.[5] The optimization algorithm used in Mystic is the Static Dynamic optimal Control or SDC algorithm.[6] Mystic was originally designed for low-thrust (typically electric propulsion) mission design. Mystic has been used for mission and maneuver design for NASA's solar electric Dawn Discovery mission to the giant asteroid Vesta and the dwarf planet Ceres.[7–10] The low-thrust optimal control formulation of Mystic turns out to be well suited to the ARTEMIS

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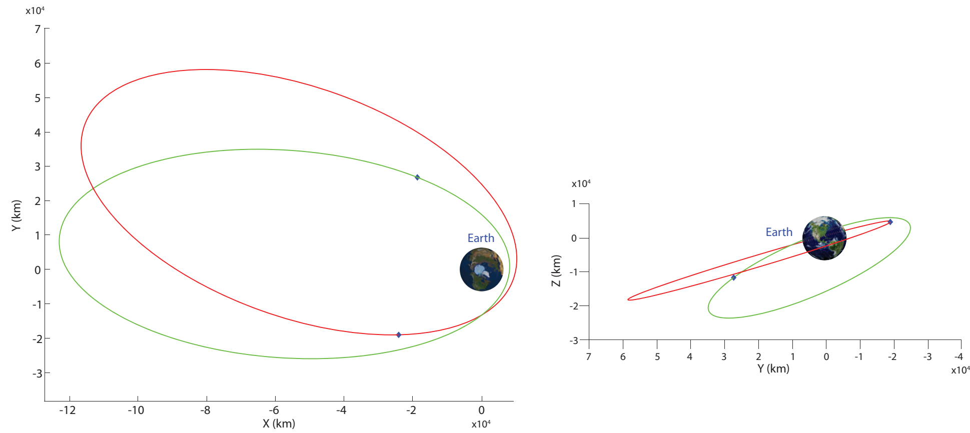

Figure 3. The assumed initial orbit of the "P2" spacecraft at the start of the orbit raise. The initial orbit assumed when ARTEMIS was proposed is green, the actual orbit at the beginning of the orbit raise is red. Both an ecliptic projection and an ecliptic edge on view are provided. ***(Used by kind permission from figure 6 of Sweetser et al.,[4] ©Springer Science+Business Media B.V. 2012.)***

trajectory design problem. The chemical propulsion systems on the ARTEMIS spacecraft have relatively low net thrust magnitudes. ARTEMIS trajectory design therefore requires finite burn modeling. Typical orbit raise maneuvers for both P1 and P2 are limited to order 15 m/s near perigee to avoid unacceptable gravity losses. Maximum total burn times around perigee were on the order of 1 hour. Thrust magnitudes declined during the ascent because the reduced propellant reduced the system pressure. Mystic was also able to accommodate the mission constraints outlined above, for example, the requirement that thrust direction must be inertially fixed for the duration of each finite burn.

The limited ΔV budget and the complex (often treacherous) design space resulting from numerous lunar approaches during the orbit raise phase made simple design strategies impossible. The slow ascent required to reach close lunar flybys requires many intermediate orbits with increasingly strong lunar interactions. As a result, the design problem has a large number of ΔV local minima. The best local minima take advantage of weak and intermediate lunar interactions more often than not through careful phasing of each intermediate orbit. Simple orbit raise strategies that ignore intermediate orbit phasing invariably required more than the very limited available ΔV. Distant lunar interactions that provided as little as the propulsive equivalent of 1 m/s were sought.

A simpler design strategy was applied for intermediate orbits smaller than those that can have lunar perturbation with the propulsive equivalent of about 1 m/s. For these initial ascent orbits the goal was to minimize propellant usage while still ascending on a schedule that would make rendezvous with the low-energy transfer portion of each spacecraft's trajectory reachable.

In order to provide some robustness against missed burns at perigee, burn passes were allowed only as often as every other perigee passage. This would, in principle, allow an emergency maneuver to be designed during the next unused perigee passage to recover the reference trajectory. Of course, the reference could not be exactly recovered due to the phase error introduced by the missed maneuver. It is likely that several subsequent planned maneuvers after the emergency maneuver would also need to be redesigned to regain the required favorable distant lunar interactions. Sometimes it proved advantageous to separate burns by more than a single periapsis passage to take advantage of or avoid certain lunar interactions. Most perigee burns were divided into two burn arcs, one on either side of the Earth's shadow, since the spacecraft are unable to maneuver in shadow. The duration and pointing of each burn was optimized subject to a propellant minimization objective and the objective of achieving favorable future distant lunar interactions.

Some form of non-local optimization was required due to the enormous number of local minima in total transfer ΔV that exist as a result of repeated opportunities for lunar interactions. Specifically, most transfer local minima exceed the ΔV budget. The globally optimal solution for each spacecraft was deemed far too computationally complex to find. Instead, the method used was to combine local optimization with a branch and bound approach.[11] The branch and bound approach is particularly useful in eliminating large portions of the design space when applied to the problem in forward time order. Each maneuver opportunity represented a possible branch point, for example, whenever it involved a choice in the number of orbits until the time of a possible lunar interaction (which occur monthly, when the Moon passes over the apogee point). Any set of possible transfers after the first significantly detrimental lunar interaction in a candidate branch was discarded. Moderate set backs due to lunar interaction were tolerated if within the next 6 to 12 revolutions around Earth a very advantageous interaction (or interactions) occurred more than making up for the original setback.

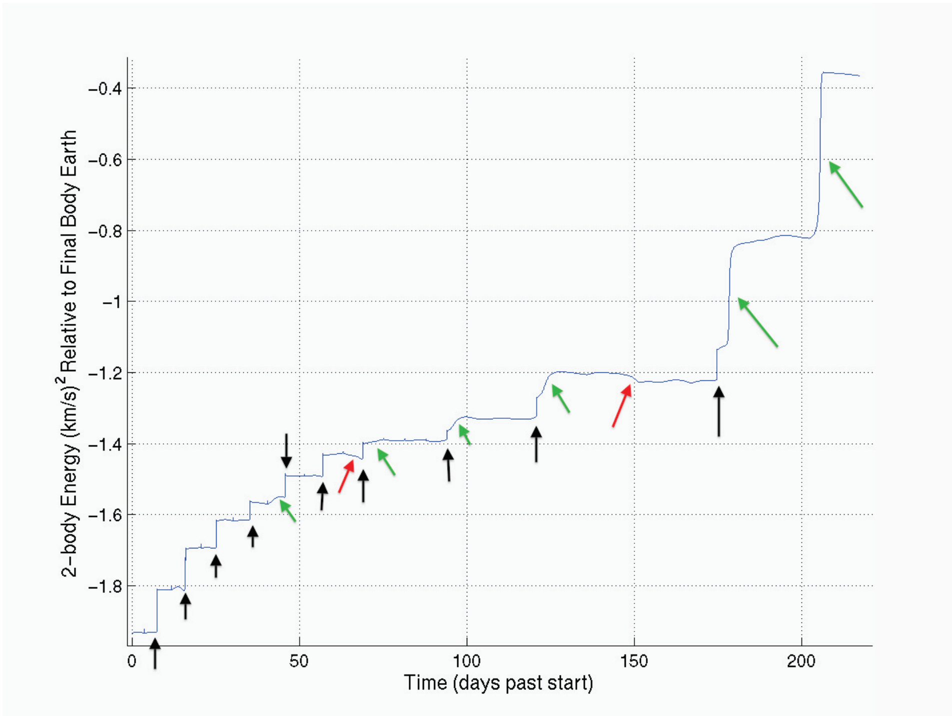

Figure 4. An example of the orbital energy evolution for a candidate transfer for the P2 spacecraft. Advantageous (energy increasing) lunar interactions are indicated with green arrows. Disadvantageous (energy decreasing) lunar interactions are indicated with red arrows. Maneuvers are indicated with black arrows.

Candidate lunar interactions were graded based on the increase (or decrease) in Earth relative orbital energy imparted to the spacecraft. Once the orbit reached high altitudes favorable Moon-induced orbital plane change was considered in addition to orbital energy gains. For both spacecraft large orbital plane changes were required in order to reach the targeted conditions for the lunar gravity assists. These plane changes could not be performed propulsively so lunar interactions were utilized. Figure 4 illustrates a single candidate trajectory which achieves high altitudes with the help of (mostly) advantageous lunar interactions. The essentially instantaneous increases in energy in this figure are maneuvers. The 10 maneuvers are indicated by the black arrows in figure 4. Advantageous (energy increasing) lunar interactions are indicated with green arrows. Disadvantageous (energy decreasing) lunar interactions are indicated with red arrows. This transfer was one of the top performing transfers for P2. Although there are two disadvantageous lunar interactions, enduring them allows vastly more advantageous lunar interactions later on. Very strong interaction can occur above an orbital energy of about -1.2 km^2/s^2.

II.A. Applying Branch and Bound Optimization

Beginning with the first possible perigee centered maneuver time, multiple candidate maneuvers were optimized to maximally increase orbital energy subject to several different fixed total maneuver durations. Each optimized maneuver defines one "branch". The future lunar interactions after each of these optimized maneuvers or branches was investigated through propagations. Two types of propagations were used for each maneuver branch. The first is a ballistic propagation of at least 12 Earth revolutions. Each branch's propagation was analyzed to see if significant advantageous or disadvantageous interactions existed in the near term (next 4 to 6 revolutions) and if these interactions lead to advantageous or disadvantageous interactions later. A similar test was applied to an extension of each of the maneuver branches assuming a simple control law maneuver of varying size is applied at the next alternating perigee or the perigee after that. The thrust direction control law resulted in thrust parallel to mean spacecraft-Earth relative velocity direction

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(maximizing orbital energy increase).

If either the purely ballistic or control law propagated extensions to the initial branch maneuver demonstrated that moderate to relatively strong advantageous lunar interactions exist then this branch was continued. If not, then the branch was abandoned. When it was decided that a branch should be continued, the trajectory was then numerically optimized including current and previous maneuvers. The optimized trajectory then became the basis for new branches. In practice, relatively few new branches were created at each maneuver stage. For each extended branch, local optimization was used to introduce maneuvers or optimize the crude control law maneuvers along with previous maneuvers up to this point as an initial guess. The objective was to maximize orbital energy by using maneuvers and by achieving phasing to maximize the net effect of lunar interactions.

Once a high apogee was obtained for a given branch so that lunar interactions were potentially very strong, it was found that the branch extensions based on the ballistic propagation alone could be used as initial guesses for local optimization to excellent effect. In these large orbits, the problem becomes one primarily of local optimization because the small maneuvers possible have a smaller effect on future phasing with the Moon. Often families of initial guesses would be constructed based on the total number of revolutions around the Earth. For the initially higher P1 spacecraft, this regime was encountered after only three perigee passages with burns. For the much lower P2 spacecraft, the branch and bound process was much more lengthy, and this regime was not encountered until about 22 to 25 perigee passages with burns had occurred.

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II.B. P1 Orbit-raise Design

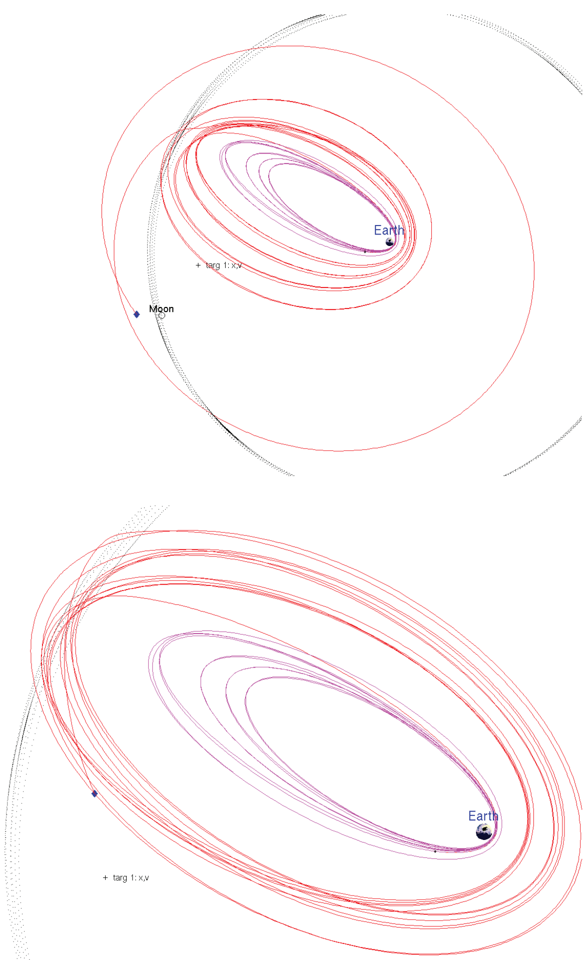

Figure 5. Four members of the family of 13 revolution ballistic transfers before optimization and targeting. The magenta portion of each trajectory corresponds to the fixed, first three orbit raise maneuvers. The red portion of each trajectory corresponds to the unique initial guess for each family member. All trajectories are ecliptic projections.

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Several different strategies were attempted for the P1 orbit raise design. The strategy that proved successful for the P1 trajectory was to first optimize sets of burns on three alternating perigees to reach an orbital period near 131 hours. Lunar interactions in orbits having significantly below this orbital period were deemed too small for any practical concern[a]. From states near this orbital period forward there existed a tremendous number of possible paths involving differing lunar interactions, numbers of Earth revolutions, plane changes, and node changes over the next 140 days of ballistic propagation. It was not at all obvious which of these many paths might be feasible, and then which feasible path is the best to rejoin the low-energy transfer. To address this problem, a large number of trajectories were used as initial guesses for targeting and optimization. Different families of initial guess trajectories were organized based on the number of orbital revolutions around Earth between lunar interactions.

Figure 5 provides an example of some of the family of 13 revolution ballistic transfers that were created using a simple control law for future orbit raise maneuvers. These and many other trajectories including families from 10 to 16 revolutions were used as initial guesses for targeting and optimization. Different revolution families occur as the result of orbital period changes from lunar interactions. All families allow a fixed 140 days of propagation. This duration corresponds to the time when the low-energy transfer has to be reached. A computer cluster was used for this compute-intensive optimization process. Trajectories that were found to be feasible or near feasible were then further refined by moving the targeting (or joining) of the low-energy transfer to successively later dates. An example of the refinement of an 11-revolution transfer is provided in figure 6. Often the first step of the refinement process depicted in figure 6 is prohibitively

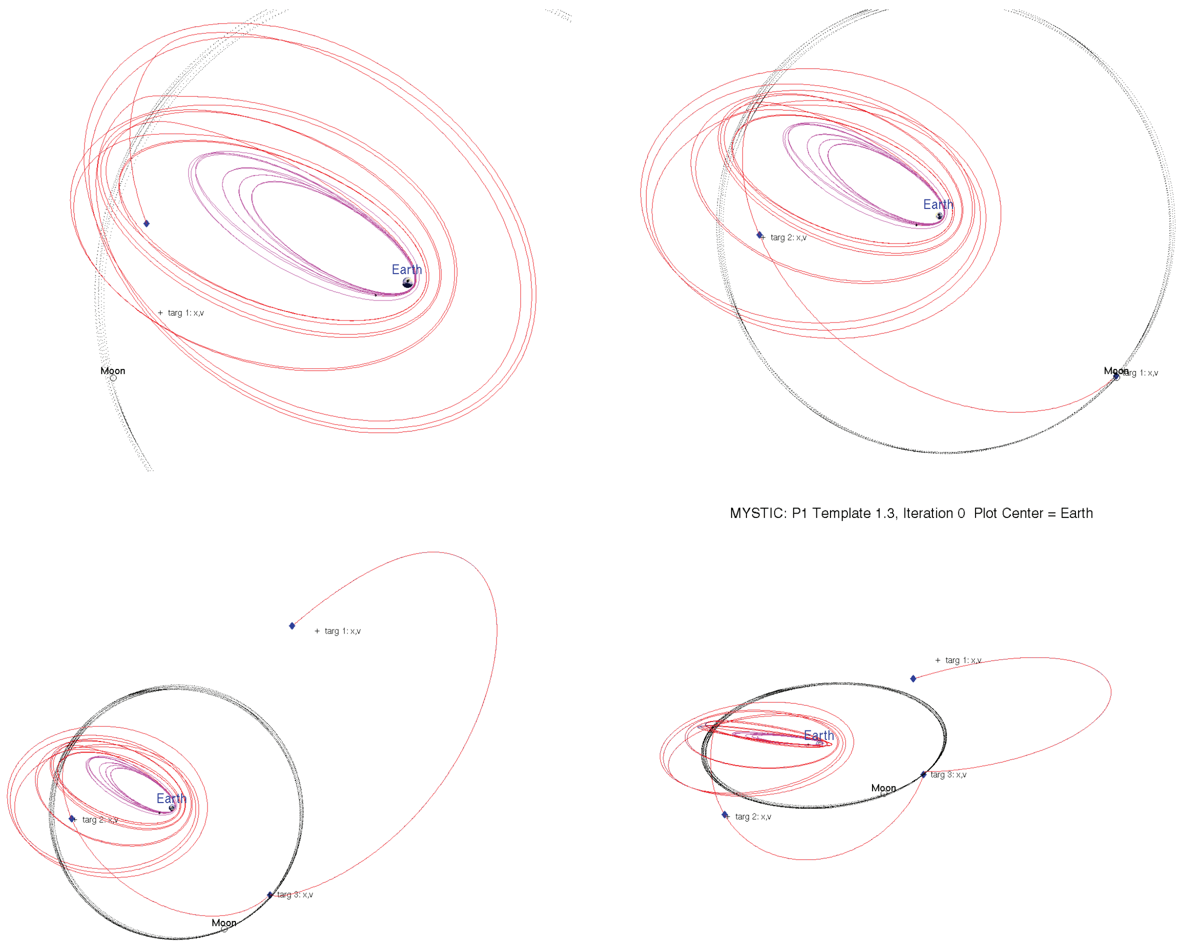

Figure 6. Steps in the refinement of a candidate P1 transfer from the 11 revolution family. The upper left trajectory targets a state just after the first of two lunar flybys that lead to the low-energy transfer. The upper right trajectory targets the second lunar flyby. The lower two trajectories target a state well into the low-energy transfer. All but the lower right trajectory are ecliptic projections. The lower right trajectory is an oblique view.

[a]This is in contrast to the method used for the P2 spacecraft. Lunar interactions below an Earth orbital period of 131 hours were carefully considered because P2 had an even more constrained timeline and ΔV budget than P1.

expensive from a ΔV stand point and is abandoned.

The key to reaching the P1 low-energy transfer was a pair of lunar flybys separated by only 14 days. For any transfer to remain within the ΔV budget, it was necessary to match these flybys closely. The possibility of rendezvous with the low-energy transfer by a means that did not include the double lunar flyby was ruled out as likely much too expensive in terms of ΔV. Exact matching is not necessary. Intuitively, re-joining the low-energy transfer at later times will provide ever increasing efficiency. However, it was expected (and was found) that re-joining much beyond the second lunar flyby provided diminishing returns because it was necessary to target the two preceding lunar flybys closely in order to arrive in a state near the low-energy transfer. The final design required only 103.5 m/s and the executed maneuvers required a total of 102.7 m/s (compared to the budget of 110 m/s) to reach the lissajous injection state at the end of the deep space low-energy transfer. Figure 7 illustrates the large plane change required during the transfer for P1. The large plane change was essential to set up the two lunar flybys that occur 14 days apart.

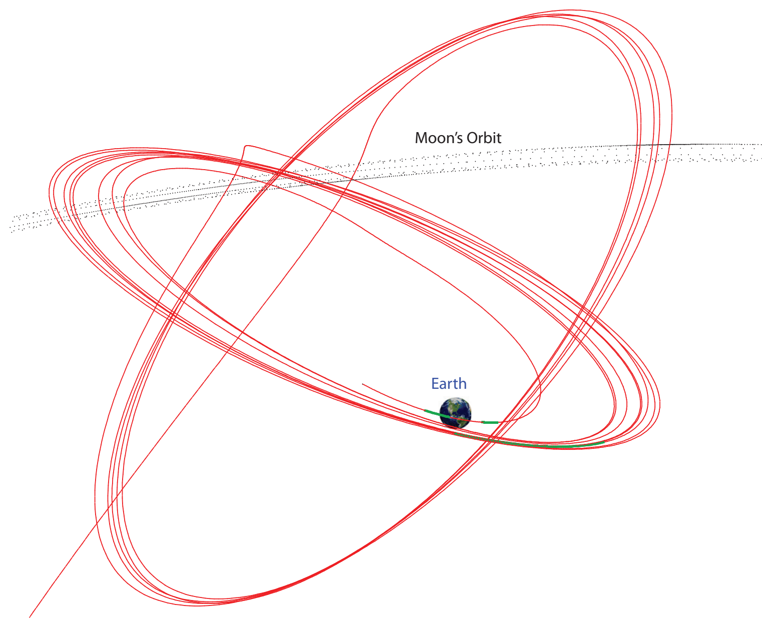

Figure 7. An oblique view of the final orbits of P1 during the orbit raise transfer. ***(Used by kind permission from figure 7 of Sweetser et al.,[4] ©Springer Science+Business Media B.V. 2012.)***

II.C. P2 Orbit-raise Design

The P2 orbit design was more complex than the P1 design due to the fact P2 began in a much smaller orbit, was more time constrained, and had a more constrained ΔV budget. A process somewhat different from the P1 design process was used to develop the P2 orbit raise design. Figure 8 indicates P2's initial orbit before the orbit raise and the target low-energy transfer. Very careful planning of distant lunar approaches was necessary to keep under the allocated ΔV budget. The P2 orbit raise ended up using 42 burns.

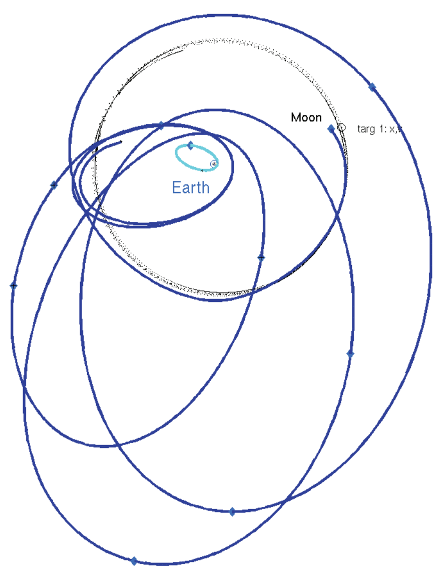

Figure 8. P2's initial orbit before the orbit raise (blue-green) and the target low-energy transfer (blue). The objective is to transfer to a rendezvous with the low-energy transfer. The trajectories are shown in an ecliptic projection.

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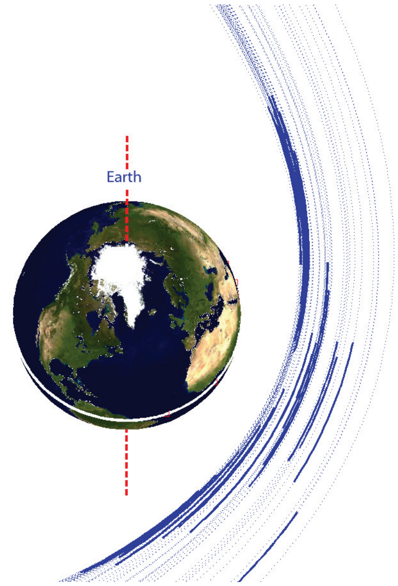

Figure 9. P2's final design burns near perigee. Burns are indicated by solid lines. Early (low periapsis) burns were spilt around the Earth's shadow. Later (higher periapsis) burns did not require splitting because the Earth's shadow no longer covered perigee.

The method used was again the branch and bound process. Each orbit raise maneuver was designed to optimally reach several different orbital periods (different period = different "branch"). Subsequent maneuvers reaching longer periods were designed for each branch. The most promising branches were continued while poor performing branches were abandoned. Poorly performing branches often led to situations where lunar interactions reduced the orbit period or required long periods without maneuvers to avoid disadvantageous lunar interactions. Highly performing branches ended up with advantageous distant lunar interactions early on. Distant lunar interactions were sought that provided maneuver savings as little as 1 meter per second. The final few orbit raise maneuvers required very careful planning to maximize the positive influence of the Moon. Another important factor used in evaluating lunar interactions for a given branch was whether or not the interaction changed Earth perigee significantly. Large changes were avoided because they either resulted in impacting trajectories or increased gravity losses.

Early on in the transfer, all perigee maneuvers were split into two burns, one on either side of the Earth's shadow. The duration of each maneuver around shadow was optimized for maximum efficiency. Later in the orbit-raise phase the Earth's shadow moved away from perigee so single burns were used. Figure 9 illustrates the burns near perigee. Burns are indicated as solid lines. The perigee naturally increased in altitude as the orbit raise proceeded when the objective of maximizing energy is applied locally to one or only a few burns. This result increases the global gravity losses because higher-altitude burns are less efficient at increasing orbital energy. To reduce this loss intermediate state targets were used to keep periapsis lower.

The best branch to rendezvous with the low-energy transfer unfortunately entered Earth umbra for a duration deemed too long for the spacecraft to survive. The unexpected shadow occurred at the large Earth distance of 300,000 km. To avoid umbra, two maneuvers were required in locations far from perigee to effect a small plane change. The final reference trajectory is illustrated in figure 10. The two shadow deflection maneuvers and the location of the problem shadow are both highlighted in figure 10. The final orbit raise design required a total ΔV of 239.6 m/s compared to the budget of 240 m/s. Of the designed 239.6 m/s, 12.1 m/s of the total ΔV was required for the shadow deflection maneuvers.

III. Conclusion

Both spacecraft entered the low-energy transfer on time and within the ΔV requirements. When the design process was started it was not at all clear if a solution existed that had an acceptable ΔV given the numerous challenges outlined above. Most designs that were completed or near completed exceeded the ΔV budget. Many grossly exceeded the budget. The design process required high-precision integrated propagation (including multi-body gravity and gravitational harmonics) and nonlinear optimization due to

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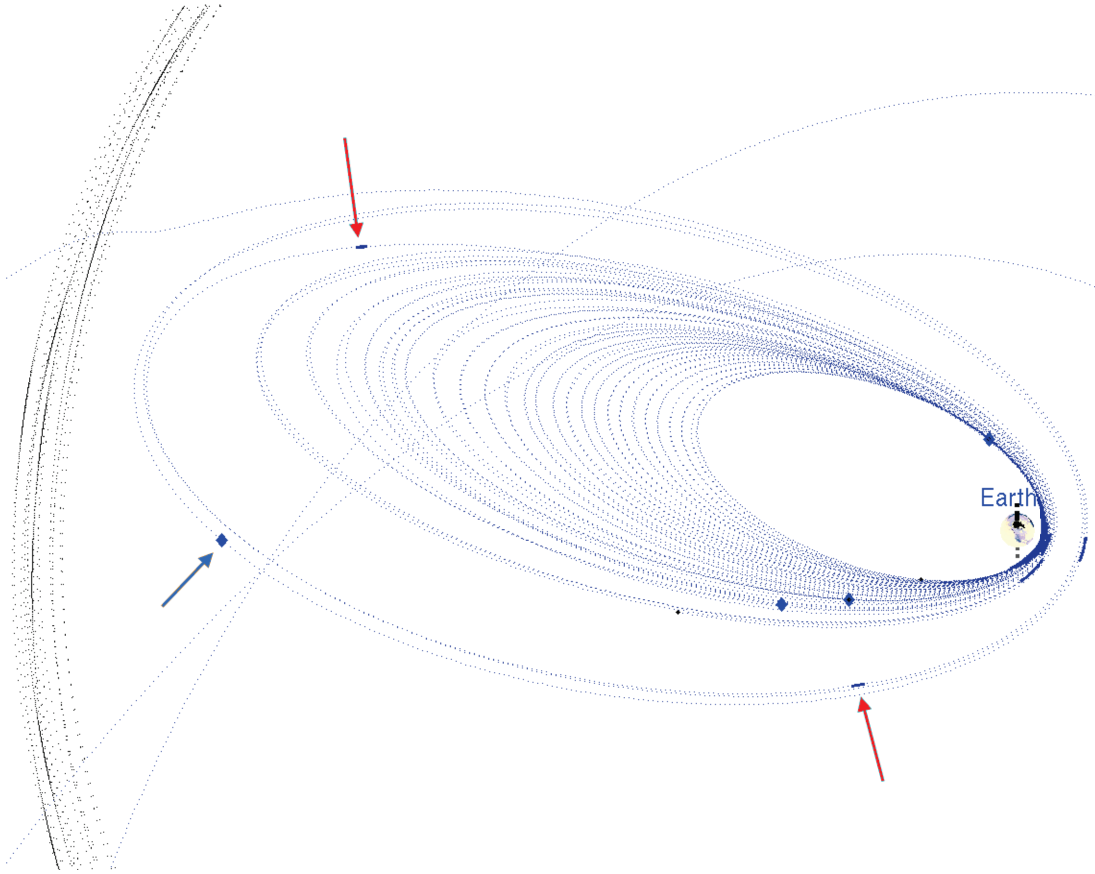

Figure 10. P2's final design burns. Burns are indicated by solid lines. Two "shadow" deflection maneuvers (indicated by red arrows) were required to avoid a lengthy shadow passage. The location of the shadow is indicated by a blue arrow. The trajectory is shown in an ecliptic projection.

the complex dynamical regime of the Earth - Moon system. The fact that the maneuver plan for both spacecraft was successfully implemented as designed validated both the numerical optimization tool set Mystic and the modeling. The careful modeling of the nonlinear dynamics and the complex spacecraft behavior was essential to predict the ultimate ΔV used during execution accurately.

Table 1 on page 13 and table 2 on page 14 show the times and magnitudes of the maneuvers planned and actually executed by the two ARTEMIS spacecraft in the process of leaving Earth orbit. The naming convention for the maneuvers listed in both tables are Orbit Raise Maneuvers: ORM, Flight Trim Maneuvers: FTM, and Shadow Deflection Maneuver: SDM. The ORMs are all executed at or near perigee. The FTMs are specifically designed for Lunar flyby targeting, and the SDMs occur far from perigee for the purpose of reducing the impact of a very distant Earth shadow on the P2 spacecraft. The "Reference ΔV" column in both table 1 on page 13 and table 2 on page 14 are the designed maneuvers using the methods described in this paper. The "Executed ΔV" columns in these tables are estimates of the actual applied maneuvers. The differences between the reference and executed ΔV are partly a result of maneuver execution error—each maneuver was sequentially redesigned before execution to account for delivery errors from previous maneuvers. A further difference resulted from the required discretization of all radial maneuvers into an integer number of spin pulses. The reference and executed ΔV totals are remarkably close given the system complexity and the statistical component of the long and complex ascent plans. The trajectories that were flown are shown in figure 11 and figure 12 on the following page.

Neither spacecraft missed a maneuver and therefore none of the unplanned perigee passages were required for make-up maneuvers. It is a credit to the maneuver and operations team that the long and complex orbit raise design was implemented so accurately and without incident.

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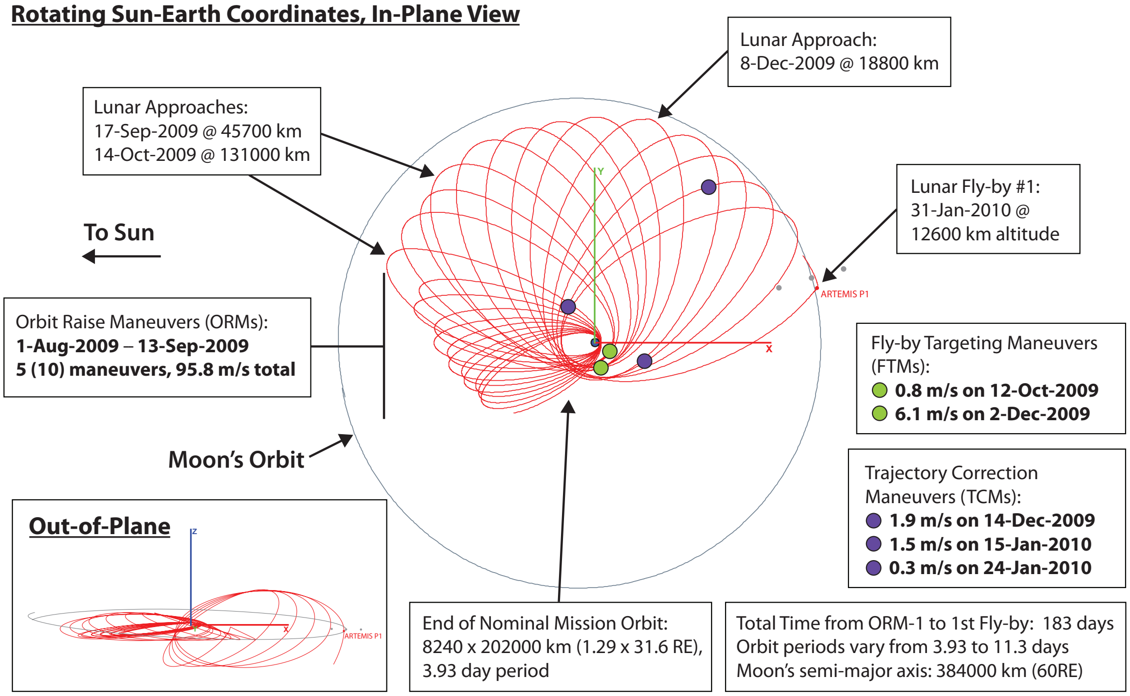

Figure 11. Earth orbit portion of the P1 trajectory design. The view is from ecliptic north toward the Earth and rotates to keep the Sun toward the left. Distances quoted are ranges measured from the center of mass of the Earth or Moon. *(Used by kind permission from figure 4 of Sweetser et al.,[4] ©Springer Science+Business Media B.V. 2012.)*

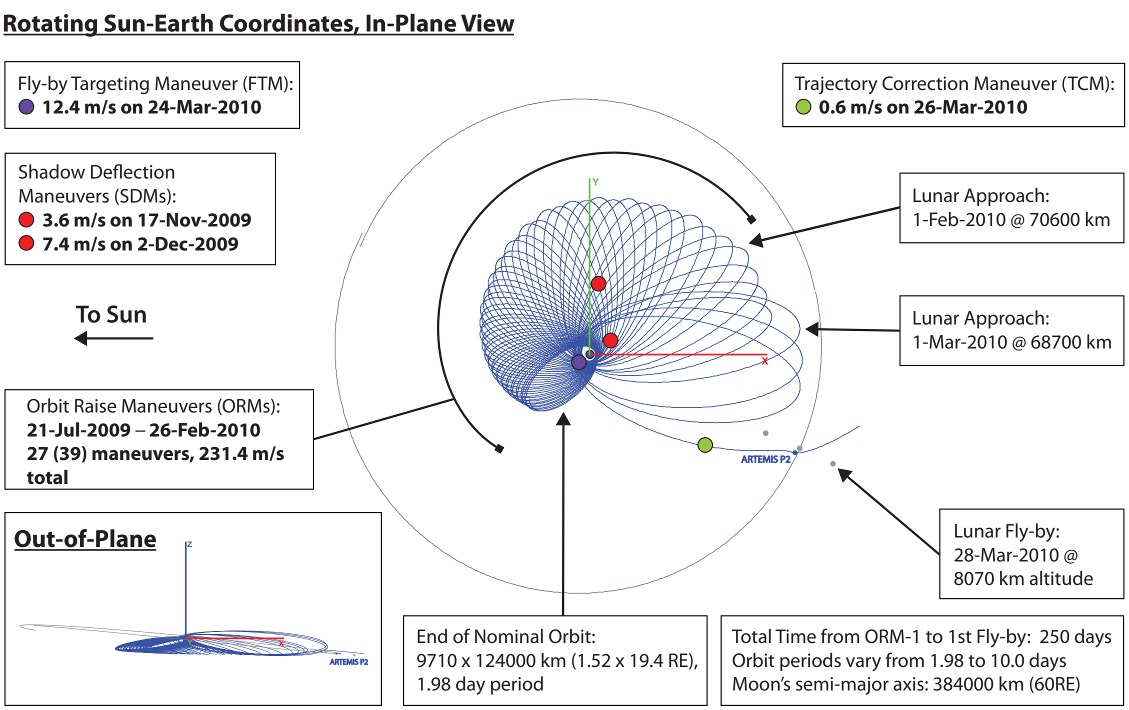

Figure 12. Earth orbit portion of the P2 trajectory design. The view is from ecliptic north toward the Earth and rotates to keep the Sun toward the left. Distances quoted are ranges measured from the center of mass of the Earth or Moon. *(Used by kind permission from figure 5 of Sweetser et al.,[4] ©Springer Science+Business Media B.V. 2012.)*

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Table 1. P1 Earth orbit raise and low-energy transfer phase targeting maneuvers

Maneuver Name	Year/ Day of year	Time UTC	Reference ΔV m/s	Executed ΔV m/s
ORM1A	2009/213	20:01:04.853	8.320	8.243
ORM1B	2009/213	20:50:51.171	8.356	8.441
ORM2A	2009/222	13:01:15.050	8.391	8.389
ORM2B	2009/222	13:51:33.867	8.427	8.468
ORM3A	2009/232	07:06:30.159	8.465	8.477
ORM3B	2009/232	07:56:36.098	8.501	8.517
ORM4A	2009/243	08:12:29.465	14.124	13.901
ORM4B	2009/243	09:14:21.187	12.526	11.855
ORM5A	2009/256	18:57:06.519	13.977	13.984
ORM5B	2009/256	19:48:19.402	5.610	5.505
FTM1A	2009/285	08:38:00.635	0.203	0.203
FTM1B	2009/285	08:41:51.037	0.602	0.602
FTM2	2009/336	08:02:21.215	5.998	6.084
		Totals	**103.5**	**102.7**

Table 2. **P2 Earth orbit raise and transfer-energy transfer phase targeting maneuvers.**

Maneuver Name	Year/ Day of year	Time UTC	Reference ΔV m/s	Executed ΔV m/s
ORM1	2009/202	07:33:03.552	10.418	10.686
ORM2	2009/206	10:41:40.407	5.179	5.292
ORM3A	2009/210	15:10:44.501	2.400	2.399
ORM3B	2009/210	16:10:50.116	8.121	8.348
ORM4A	2009/215	00:46:49.019	3.295	3.324
ORM4B	2009/215	01:47:36.747	8.411	8.695
ORM5A	2009/219	15:24:58.115	3.846	3.870
ORM5B	2009/219	16:22:13.277	7.714	7.915
ORM6A	2009/224	11:22:18.944	3.841	3.903
ORM6B	2009/224	12:16:13.013	6.812	6.952
ORM7A	2009/229	12:35:08.258	3.814	3.867
ORM7B	2009/229	13:26:36.035	6.176	6.257
ORM8A	2009/234	19:27:11.635	4.160	4.267
ORM8B	2009/234	20:18:08.362	5.958	6.058
ORM9A	2009/240	08:07:08.656	3.793	3.871
ORM9B	2009/240	08:53:50.173	4.678	4.753
ORM10A	2009/246	02:16:11.659	4.623	4.729
ORM10B	2009/246	03:01:38.535	4.343	4.387
ORM11A	2009/252	02:39:19.201	4.865	5.000
ORM11B	2009/252	03:21:47.856	3.441	3.419
ORM12A	2009/258	09:02:17.385	5.408	5.586
ORM12B	2009/258	09:43:04.698	2.852	2.831
ORM13A	2009/264	22:31:20.114	5.680	5.882
ORM13B	2009/264	23:11:39.983	2.561	2.507
ORM14A	2009/271	18:34:32.874	6.818	7.092
ORM14B	2009/271	19:14:07.763	2.278	2.240
ORM15	2009/278	23:21:22.389	5.660	5.881
ORM16	2009/286	09:49:58.506	8.013	8.599
ORM17	2009/294	06:15:08.345	9.606	9.851
ORM18	2009/302	13:41:36.289	10.002	10.269
ORM19	2009/311	10:47:56.571	9.999	10.113
ORM20	2009/320	22:41:23.999	9.935	10.039
ORM21	2009/331	01:50:58.997	4.101	4.148
ORM22	2009/341	12:28:02.607	2.086	2.083
ORM23	2009/352	07:22:49.060	4.693	4.782
ORM24	2009/363	11:22:37.941	6.141	6.233
ORM25	2010/010	05:05:57.689	7.472	7.580
ORM26	2010/022	19:08:17.058	6.264	5.846
ORM27	2010/057	08:52:20.815	12.022	11.875
SDM1	2010/059	08:17:18.815	2.474	3.636
SDM2	2010/074	09:55:42.965	9.639	7.360
		Totals	**239.6**	**242.4**

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References

[1] Broschart, S. B., Chung, M.-K., Hatch, S. J., Ma, J. H., Sweetser, T. H., Weinstein-Weiss, S. S., and Angelopoulos, V., "Preliminary trajectory design for the Artemis lunar mission," *AAS/AIAA Astrodynamics Specialists Conference, Pittsburgh, Pennsylvania*, edited by A. V. Rao, T. A. Lovell, F. K. Chan, and L. A. Cangahuala, Vol. 134 of *Advances in the Astronautical Sciences*, American Astronautical Society / American Institute of Aeronautics and Astronautics, Univelt, Inc., San Diego, 2009.

[2] Angelopoulos, V., "The THEMIS Mission," *Space Science Reviews*, , No. 141, 2008, pp. 5–34.

[3] Angelopoulos, V., "The ARTEMIS Mission," *Space Science Reviews*, 2010.

[4] Sweetser, T. H., Broschart, S. B., Angelopoulos, V., Whiffen, G. J., Folta, D. C., Chung, M.-K., Hatch, S. J., and Woodard, M. A., "ARTEMIS Mission Design," *Space Science Reviews*, Vol. 165, No. 1, 2012, pp. 27–57, DOI 10.1007/s11214–012–9869–1.

[5] Whiffen, G. J., "Mystic: implementation of the Static Dynamic Optimal Control Algorithm for High-Fidelity, Low-Thrust Trajectory Design," *Proceedings of the AIAA/ASS Astrodynamics Specialists Conference*, Keystone, Colorado, 21-24 August 2006, Paper AIAA 2006-6741.

[6] Whiffen, G. J., "Static/Dynamic for Optimizing a Useful Objective," United States Patent, No. 6,496,741, Filed March 1999, Issued December 2002.

[7] Parcher, D. W. and Whiffen, G. J., "Dawn Statistical Maneuver Design for Vesta Operations," *Proceedings of the AIAA/ASS Astrodynmaics Specialist Conference*, New Orleans, Louisiana, February 13-17, 2011, Paper AAS2011-180.

[8] Parcher, D. W., "Low-Thrust Orbit Transfer Design for Dawn Operations at Vesta," *Proceedings of the AIAA/ASS Astrodynmaics Specialist Conference*, New Orleans, Louisiana, February 13-17, 2011, Paper AAS2011-183.

[9] Whiffen, G. J., "The Stability of Powered Flight Around Asteroids With Application to Vesta," *Proceedings of the AIAA/ASS Astrodynmaics Specialist Conference*, New Orleans, Louisiana, February 13-17, 2011, Paper AAS2011-186.

[10] Whiffen, G. J., "Low Altitude Mapping Orbit Design and Maintenance for the Dawn Discovery Mission at Vesta," *Proceedings of the AIAA/ASS Astrodynmaics Specialist Conference*, New Orleans, Louisiana, February 13-17, 2011, Paper AAS2011-182.

[11] Lawler, E. L. and Wood, D. E., "Branch-and-bound Methods: A Survey," *Operations Research*, Vol. 14, 1966, pp. 699–719.

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