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ARTEMIS Mission Design

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Abstract

The ARTEMIS mission takes two of the five THEMIS spacecraft beyond their prime mission objectives and reuses them to study the Moon and the lunar space environment. Although the spacecraft and fuel resources were tailored to space observations from Earth orbit, sufficient fuel margins, spacecraft capability, and operational flexibility were present that with a circuitous, ballistic, constrained-thrust trajectory, new scientific information could be gleaned from the instruments near the Moon and in lunar orbit. We discuss the challenges of ARTEMIS trajectory design and describe its current implementation to address both heliophysics and planetary science objectives. In particular, we explain the challenges imposed by the constraints of the orbiting hardware and describe the trajectory solutions found in prolonged ballistic flight paths that include multiple lunar approaches, lunar flybys, low-energy trajectory segments, lunar Lissajous orbits, and low-lunar-periapse orbits. We conclude with a discussion of the risks that we took to enable the development and implementation of ARTEMIS.

Keywords ARTEMIS · THEMIS · low-energy transfer · Lissajous orbits · lunar science · lunar mission · heliophysics · magnetosphere

1 Introduction

Time History of Events and Macroscale Interactions during Substorms (THEMIS) is a very successful NASA Explorer mission launched in February of 2007 to advance our understanding of magnetic substorms, a space weather phenomenon in the Earth's magneto-

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sphere (Angelopoulos 2008). The mission consists of five identical Earth-orbiting spacecraft (probes) equipped with particle and field instruments (Harvey et al. 2008). As of the time of this writing, the baseline mission science objectives have been achieved, and all five probes (and their instruments) are fully functional.

In February 2008 ARTEMIS, the Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun mission, was proposed to the NASA Heliophysics Senior Review (Angelopoulos and Sibeck 2008) as an extension to the THEMIS mission. It was approved for development in May of that year. The ARTEMIS mission proposed to send the two outermost THEMIS probes, P1 and P2 (also referred to as THEMIS-B and THEMIS-C), to lunar orbits by way of two circuitous transfers that take about one and a half years each. The goals of the mission as proposed in 2008 were to use the Moon as an anchor for the ARTEMIS probes to conduct studies of Earth's magnetotail and solar wind from approximately 60 Earth radii and to study the lunar wake and its refilling as a function of the upstream solar wind. ARTEMIS two-point measurements open a new vantage point to phenomena previously studied by single-spacecraft missions. In particular, when solar wind measurements are made simultaneously by one probe in the lunar wake and the second from various locations just upstream of the lunar wake, accurate comparisons of wake phenomena with upstream variations can be made.

The ARTEMIS proposal represented the combined efforts of the THEMIS science team led by the PI at UCLA, the THEMIS Mission Operations team led by the Mission Operations Manager at the University of California Berkeley's Space Science Laboratory (UCB-SSL), the NASA Goddard Space Flight Center (GSFC), and the Jet Propulsion Laboratory at the California Institute of Technology (JPL). Two earlier reports (Broschart et al. 2009; Woodard et al. 2009) describe the preliminary mission design as proposed in 2008; portions of this paper are taken from those reports. This paper presents the evolution of the trajectory design to the trajectory being flown today, only a few months prior to lunar orbit insertion.

Numerous challenges were inherent to the ARTEMIS mission's trajectory design because of the constrained capabilities of the THEMIS probes. Limited fuel remained after the THEMIS baseline mission was completed. Thruster configuration limits thrust directions to one hemisphere. Additionally, an on-off thruster duty cycle imposed due to the spinning of the probe bus restricts effective thrust to less than a newton in the spin plane, i.e., for maneuver directions near the ecliptic plane. Maneuvers cannot be done in shadow because accurate pulse timing relies on sun-sensor data. Telecommunications with the probes were limited to a range of about two million kilometers. Finally, the probes can only withstand up to a 4-hour shadow. Had nothing been done at the end of the THEMIS baseline mission, long eclipses ($>$8 hr) would have neutralized P1 by March 2010 (Angelopoulos 2010). This became a very significant driver for proposing the ARTEMIS mission.

In Section 2 we describe the capabilities and orbit configuration of the THEMIS probes at the end of their baseline mission. In Section 3 we outline the history of the ARTEMIS mission design concept as it followed the mission's programmatic evolution. Section 4 outlines the science goals and orbit design goals of the mission. The remainder of the paper describes the design of the trajectories that are taking P1 and P2 from eccentric, high-altitude Earth orbits into lunar orbits that satisfy the science objectives. Figure 1 shows the ARTEMIS trajectory design used to send P1 and P2 from their respective Earth orbits at the start of ARTEMIS maneuvers into lunar Lissajous orbits. Section 5 presents the most up-to-date ARTEMIS mission design. Section 6 describes the current mission status, including ongoing trade studies. Section 7 is a retrospective on the challenges and enabling attributes of the mission design effort.

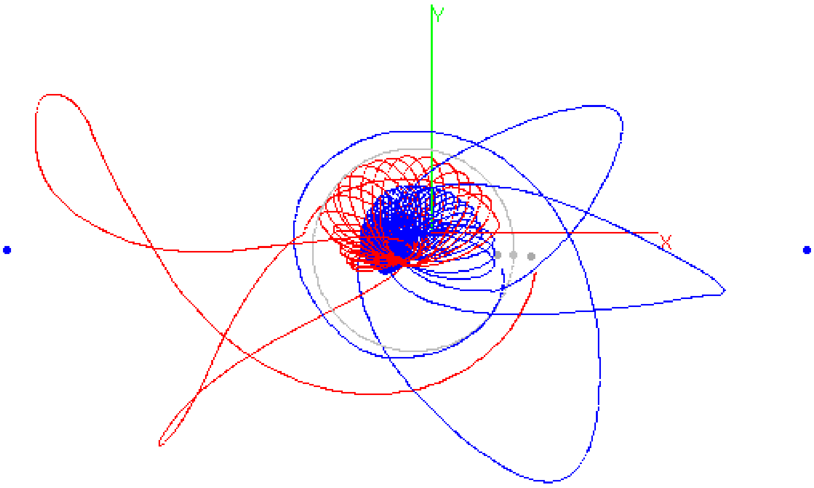

Fig. 1 ARTEMIS trans-lunar trajectories in the ecliptic plane. The coordinate frame here rotates such that the Sun is always to the left. The red line shows the P1 trajectory; the blue line shows the P2 trajectory. The Earth is at the center of the figure, and the Moon's orbit is shown in gray. The blue dots are the Sun-Earth L1 and L2 Lagrange points; the gray dots are the Moon and the Earth-Moon L1 and L2 points at a particular epoch.

2 Spacecraft Overview

On February 17th, 2007, the five THEMIS probes were launched on a Delta-II 7925 rocket into a 1.3-day Earth orbit with perigee at 437 km altitude and apogee at ~87500 km altitude (Angelopoulos 2008). Based on initial on-orbit data – in particular, better link margin performance – THEMIS-B was assigned to a 4-day orbit and designated "P1", and THEMIS-C was assigned to a 2-day orbit and designated "P2". THEMIS-D, E, and A were assigned to 1-day orbits, becoming P3, 4 and 5, respectively, per the mission design plan (Frey et al. 2008) required to achieve THEMIS mission science goals (Figure 2) (Angelopoulos 2008). After 29 months in orbit, the two outermost probes, P1 and P2, were called on to journey to the Moon as part of the ARTEMIS mission.

The five THEMIS probes were identical at launch with 134 kg mass (including 49 kg of hydrazine monopropellant). Each measures approximately $0.8 \times 0.8 \times 1.0$ meters (Harvey et al. 2008). On orbit, each has deployed a number of instrument booms and is spin-stabilized at ~20 RPM. Figure 3(a) shows a THEMIS probe with booms deployed. Figure 3(b) shows a schematic of the bus design. The blue arrow, which indicates the spin vector, shall be referred to as the probe $+Z$ direction.

Each probe has four thrusters, nominally 4.4 N each, with locations indicated by the black arrows in Figure 3(b). Two provide axial thrust (acceleration in $+Z$ direction) for large ΔV maneuvers and attitude control. The other two provide tangential thrust in the spin plane for small ΔV maneuvers and spin rate control. Note that the probes cannot apply acceleration in the $-Z$ direction. During the nominal THEMIS mission, P1 and P2 were flown with the $-Z$ axis close to the ecliptic north pole, i.e., in an "upside-down" configuration relative

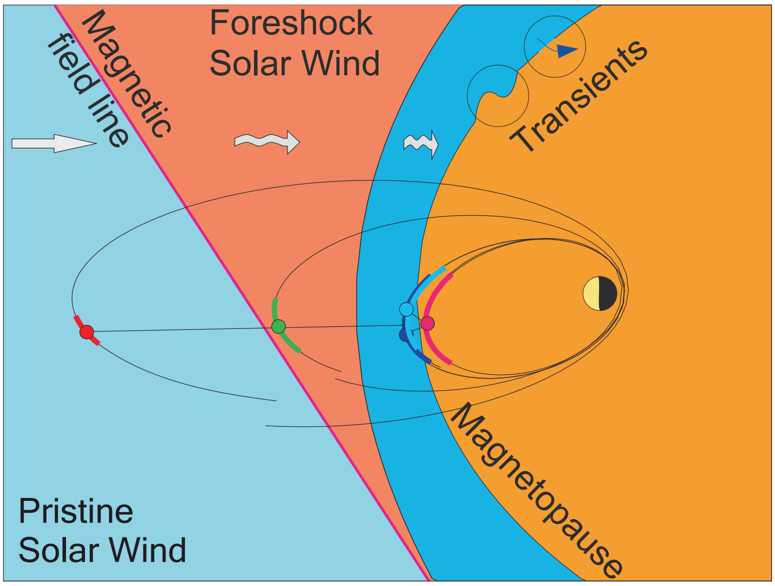

Fig. 2 THEMIS mission orbit configuration. Filled circles represent THEMIS probe locations during a day-side conjunction (Pink: P1 4-day orbit, Blue: P2 2-day orbit, Red: P3 1-day orbit, Green: P4 1-day orbit, Black: P5 1-day orbit). The orbit geometries are indicated by black lines.

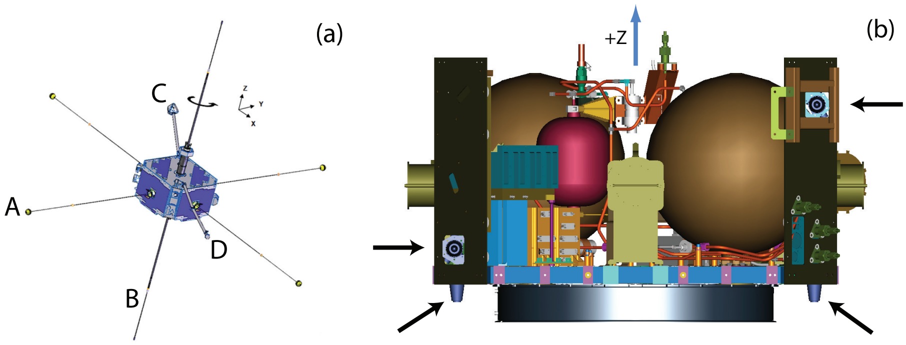

Fig. 3 THEMIS/ARTEMIS probe configuration. The probe buses were manufactured by ATK Space Systems (formerly Swales Aerospace), and the instruments were manufactured under the leadership of the University of California, Berkeley with both US and international collaborators. (a) On-orbit configuration 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.

to ecliptic north and opposite the inner three probes. This was done to aid the main orbit correction maneuvers in the second year of THEMIS, which were designed to counteract lunar perturbations on the orbit plane (Frey et al. 2008). ARTEMIS would maintain the same orientation, as it is quite fuel-intensive to impart spin-axis changes to the probes. Thus, maneuvers towards ecliptic north could not be included in the ARTEMIS trajectory design. At launch, each probe had 960 m/s total ΔV capability (Harvey et al. 2008). At the start of

ARTEMIS maneuvers the remaining ΔV (approximately 320 m/s for P1 and 467 m/s for P2) were available for the ARTEMIS trajectory design. Due to fuel tank depressurization (Sholl et al. 2007; Frey et al. 2008), each thruster is expected to produce between 2.4 N and 1.6 N force during the ARTEMIS mission.

Because the spacecraft is spinning the effective thrust of a sideways burn is further reduced, so a maneuver in a particular direction in the spin plane is performed by pulsing the thrusters on and off during each revolution. With a 60 deg pulse duration , the thrusters are on only one-sixth of the time (16.7% duty cycle). Because thrusters are swinging through an arc, the thrust in the desired direction is further reduced to 95.5% effective thrust; with a 40 deg duty cycle the thrusters average only one-ninth thrust, but lose only 2% in efficiency averaged through the arc of each pulse. Only the second reduction in each case influences the effective I_{sp}, so a 40 deg duty cycle would be preferred to a 60 deg one except that lower thrust means longer burns during periapse passages, which would increase gravity losses.

The thermal and power systems have been designed to withstand shadowing from the Sun for up to three hours (Harvey et al. 2008). It was demonstrated in March of 2009, however, that a 4-hour shadow is survivable with appropriate precautions. This limit is therefore being used as the maximum allowable shadow duration for the ARTEMIS mission design, where "shadow" is defined to be less than 50% sunlight.

3 ARTEMIS Concept Development

The baseline THEMIS mission design included the expectation that P1 would experience inordinately long (>8 hr) shadows by March 2010. Although the apoapse altitude of the P1 orbit could have been reduced to prevent this, THEMIS scientists and JPL mission designers came up with the idea of sending P1 "up" instead of "down" in 2005. With THEMIS instrumentation, compelling science could be conducted near or at the Moon with a single probe. According to initial trajectory studies, a direct transfer from P1 Earth orbit to a 1500 km altitude by 18000 km radius polar orbit at the Moon would require $\sim$500 m/s of ΔV (not including margin or losses associated with long thrust arcs). This was well beyond P1's expected ΔV capability at the end of the baseline mission.However, the remaining fuel appeared sufficient to transfer P1 from its Earth orbit to the desired eccentric polar lunar orbit by way of a lunar swing-by and low-energy transfer (Chung et al. 2005). When initiated by a lunar swing-by, this type of transfer does not require any less ΔV to leave Earth, but saves essentially all the ΔV cost of getting into a Lissajous orbit around one of the Earth-Moon Lagrange points. It does this by using solar gravity tidal perturbations to make the three-body energy change on the trajectory that would otherwise have to be done propulsively at arrival near the Moon. The fuel reserves on P2 offered similar capability, suggesting the possibility of sending two THEMIS probes to the Moon.

With the encouraging initial trajectory design results in hand, proposals for funding to support a detailed design study of low-energy trans-lunar trajectories, feasibility studies related to the THEMIS hardware, and optimization of the remaining THEMIS mission for P1 and P2 were made in 2006 and 2007. Although these proposals were not selected for funding, the science team continued concept development as time permitted.

In the summer of 2007, internal JPL funding became available to support an Explorer program Mission of Opportunity proposal for a THEMIS mission extension that would become ARTEMIS. A team from the JPL Inner Planets Mission Analysis group was convened to design trajectories to the Moon for P1 and P2. Building on the work done in 2005, the

JPL team (working closely with the THEMIS science and mission operations teams) developed a workable trajectory within THEMIS probe constraints that provided the opportunity for a highly rewarding scientific mission. This formed the baseline trajectory of the current ARTEMIS mission. Midway through this preliminary design effort, NASA headquarters advised the ARTEMIS team that the new mission would be more appropriately proposed as an extended mission for THEMIS, rather than as a mission of opportunity. At around the same time, the mission operations team at UCB-SSL was augmented by navigators and maneuver designers at GSFC who contributed operations experience with Lissajous and translunar orbits to the design effort.

The complete preliminary design for the extended mission was presented to the Senior Review Board for the Heliophysics Division in February 2008 (Angelopoulos and Sibeck 2008); approval to proceed with detailed design was given in May of that year. The preliminary trajectory design that was presented to the Senior Review Board is described in Broschart et al. (2009). This paper is an update of that earlier design paper; the design described there has changed significantly since the approval to proceed. As was understood at the time, the series of Earth orbits leading up to the initial lunar flybys needed to be significantly redesigned. More recently, a number of changes have been made in the science operations phase of ARTEMIS.

In 2009 it was recognized that significant additional scientific benefits from ARTEMIS could be obtained for the Planetary Division of NASA's Science Mission Directorate. The team was invited to propose an amendment to its Heliophysics plan that addressed Planetary objectives. The proposal was returned by NASA/HQ, and the invitation was re-extended for submission in the 2010 Senior Review cycle, so both Heliophysics and Planetary aspects of the ARTEMIS proposal could be evaluated by a joint panel. ARTEMIS/Heliophysics was given the go-ahead to continue operations in June 2010. The ARTEMIS/Planetary decision, though delayed until December 2010, was also positive. The 2008 preliminary design of ARTEMIS's lunar orbits needed to be modified to accommodate planetary objectives by lowering altitude periapses, raising inclinations, and adjusting the lines of apsides for better overlap of measurements with those of NASA's Lunar Atmosphere and Dust Environment Explorer (LADEE) mission.

The Planetary Division's decision to execute the planetary objectives of the mission came only 3 months prior to the baseline ARTEMIS lunar orbit insertion (originally slated for April 2011). This did not leave sufficient time for performing the necessary lunar orbit optimization to meet the expanded science objectives. Therefore, the team decided to postpone insertion to June-July 2011 to enable further study of the planetary aspects of the investigation. This postponement in turn entailed modifications to both the Lissajous phase and the transition to lunar orbits.

The ARTEMIS science objectives and the characteristics of orbits that would satisfy them (for both Heliophysics and Planetary Divisions of the Science Mission Directorate) as proposed and accepted by the 2010 Senior Review were described in Angelopoulos (2010). The revised mission design described in this paper represents the most up-to-date ARTEMIS orbit execution plan.

4 ARTEMIS Science Goals

Angelopoulos (2010) gives a comprehensive overview of ARTEMIS mission science objectives and describes how the mission design and operations are structured to meet them. Here we describe aspects of the mission that drive mission design.

Each probe is equipped with a suite of five particle and field instruments used to study geomagnetic substorm activity during the nominal THEMIS mission. These instruments include a Fluxgate Magnetometer, a Search Coil Magnetometer, an Electric Field Instrument, an Electrostatic Analyzer, and a Solid State Telescope (Angelopoulos 2008). This instrumentation suite allows the probe to measure the 3D distribution of thermal and superthermal ions and electrons and the AC and DC magnetic and electric fields to study the interaction between the Earth's magnetic field and the Sun's magnetic field and solar wind. By expanding the spatial extent of THEMIS's multiple, identically-instrumented spacecraft, ARTEMIS allows us to study plasmoids in the magnetotail, particle acceleration and turbulence in the magnetotail and the solar wind. Furthermore, ARTEMIS will study lunar wake formation and evolution for the first time with two identical, nearby probes, thereby resolving spatio-temporal ambiguities. The aforementioned heliophysics objectives of the mission can be addressed by inter-spacecraft separations and wake downstream crossings that are initially as large as 20 Earth radii and are progressively reduced to 1000 km or less. This goal is achieved initially by having the ARTEMIS probes at large separations in Lissajous orbits around two (and later one) of the Earth-Moon Lagrange points, and subsequently by insertion of probes into lunar orbits with $\sim$ 18,000 km apoapse radius and highly variable angular separation between their line of apsides.

ARTEMIS also offers a unique opportunity to contribute to planetary science. From its unique orbits ARTEMIS will study the "sources and transport of exospheric and sputtered species; charging and circulation of dust by electric fields; structure and composition of the lunar interior by electromagnetic (EM) sounding; and surface properties and planetary history, as evidenced in crustal magnetism. Additionally, ARTEMIS's goals and instrumentation complement LRO's *[Lunar Reconnaissance Orbiter's]* extended phase measurements of the lunar exosphere and of the lunar radiation environment by providing high fidelity local solar wind data. ARTEMIS's electric field and plasma data also support LADEE's prime goal of understanding exospheric neutral particle and dust particle generation and transport" (Angelopoulos 2010).

To achieve these objectives, ARTEMIS requires both high- and low-altitude measurements by one spacecraft, while the other measures the pristine solar wind nearby. Low periapses are very important in increasing the ability of ARTEMIS to measure sputtered ions and crustal magnetism in situ. For this reason periapse altitudes less than 50 km are highly desired. Additionally, the latitude of periapsis is an important consideration for lunar crustal magnetism – increased periapsis latitude provides opportunities for covering a larger portion of the lunar surface. A latitude greater than 10 deg (goal 20 deg) is highly desirable. Finally, conjunctions with LADEE at the dawn terminator necessitate that one of the ARTEMIS probes have its periapsis positioned near the dawn terminator and pass through periapse close to the time of LADEE passage through that region. These design considerations have been incorporated into the current planning for the upcoming lunar orbit insertions (**LOI**s).

5 ARTEMIS Trajectory Design

Figure 1 shows the ARTEMIS trajectory design that sent P1 and P2 from their respective orbits at the end of the THEMIS primary mission to insertion into lunar Lissajous orbit. The P1 trajectory is shown in red, and the P2 trajectory is shown in blue. The design succeeded in meeting both the trajectory constraints imposed by the probe capabilities and the requirements derived from the science objectives.

In the following subsections, the trajectory is broken up into phases for detailed discussion. These include the Earth orbit phase, the trans-lunar phase, the Lissajous orbit phase, and the lunar orbit phase. An integrated timeline of the events for P1 and P2 in these four mission phases can be found in Table 4.

5.1 Earth Orbit Phase Trajectories

When the preliminary design was being developed to show the feasibility of ARTEMIS, the orbit raise did not appear to present any particular challenge, so this phase was simplified to a single impulsive velocity increase at perigee, followed by a number of Earth orbits including lunar approaches that modified the orbit and culminated in the lunar flyby that begins the low-energy transfer to the Moon. This simplification allowed one track of the design effort to focus most strongly on the lunar flyby and transfer; the series of finite orbit raise maneuvers (**ORM**s) to raise the Earth orbit could be developed later in parallel on a separate design track.

Figure 4 shows the ARTEMIS P1 trajectory from the end of the nominal THEMIS mission through the first close lunar flyby. In the figure, the red line represents the ARTEMIS P1 trajectory starting with its orbit at the end of the THEMIS primary mission, and the gray circle indicates the Moon's orbit. The plot is centered on the Earth and shown in the Sun-Earth synodic coordinate frame, which rotates such that the Sun is fixed along the negative X axis (to the left) and the Z axis is aligned with the angular momentum of the Earth's heliocentric orbit. As time passes, the line of apsides of P1's geocentric orbit rotates clockwise in the main figure. The insert in the bottom left shows P1's motion out of the ecliptic plane, where the largest plane change was caused by a lunar approach in December 2009. The labels on the plot provide information about key events during this phase of the mission.

The design of the P2 Earth orbits phase was similar, as shown in Figure 5, but lasted two months longer because it started from a smaller Earth orbit and a longer series of finite maneuvers needed to be included to raise the orbit.

As we gradually came to realize, the reference trajectory design for the Earth orbit phase of both P1 and P2 would turn out to be significantly more complex than a simple series of maneuvers to replace the preliminary design's impulsive orbit raise maneuver. This complexity stemmed from: (1) probe operational constraints, (2) the tight ΔV budget, (3) the precision phasing required to reach the designed low-energy transfers to the Moon, and (4) the actual initial states for ARTEMIS P1, P2 in the summer of 2009. These actual states ended up significantly different from the initial states that were predicted in 2005-2007; this change was due to deterministic orbit-change maneuvers that occurred in 2008, mid-way through the THEMIS mission, to improve science yield for the second THEMIS tail season (Figure 6 shows this difference for the P1 orbit). As expected, the actual orbit raise required perigee burns on multiple orbits due to the small thrust capability. The design of these burns was challenging because generally an optimal design of highly elliptical transfers is numerically difficult, and because lunar approaches created a complex three-body design space.

During the refinement of the orbit design, it was recognized that several factors conspired to further complicate the development of the reference trajectory:

1. Earth's shadow covers perigee for much of the orbit raise season, prohibiting thrusting at/near perigee. The design necessitated splitting most perigee burns into two (A and B) burn arcs bracketing the shadow, further increasing burn arc length and gravity losses.
2. The initial propellant load of $\sim 50\%$ for P2 forced a large fraction of the maneuvers to be performed at a lower duty cycle (shorter pulse) due to the propellant load being near

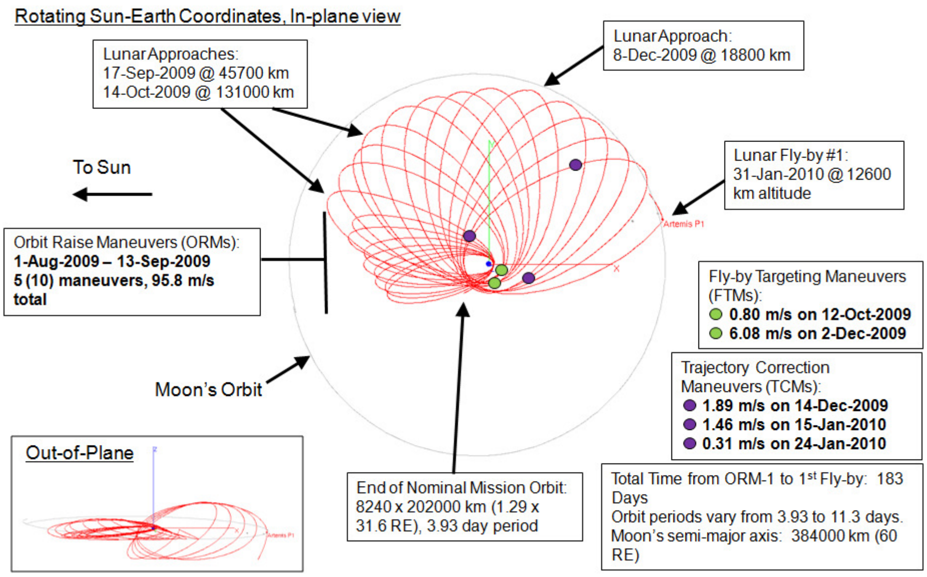

Fig. 4 Earth orbit portion of the P1 trajectory design. Distances quoted are ranges measured from the center of mass of the Earth or Moon.

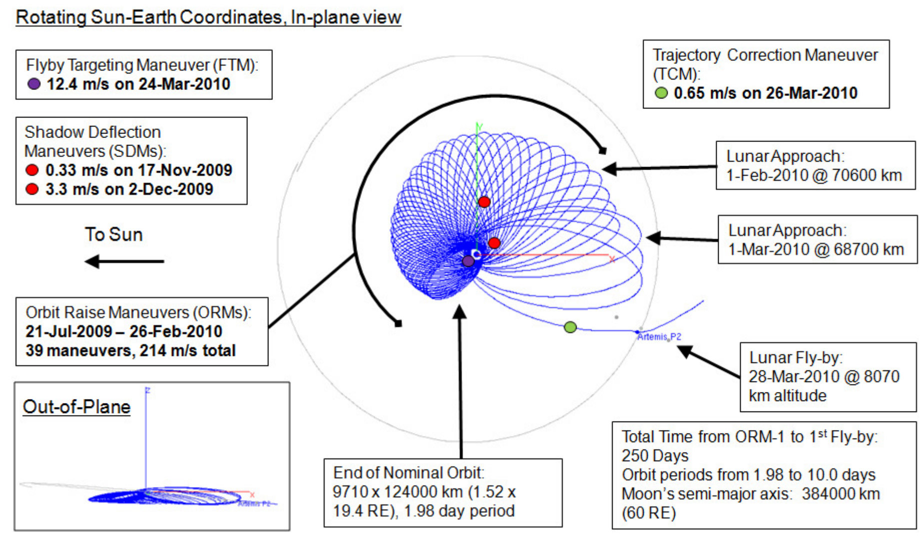

Fig. 5 Earth orbit portion of the P2 trajectory design. Distances quoted are ranges measured from the center of mass of the Earth or Moon.

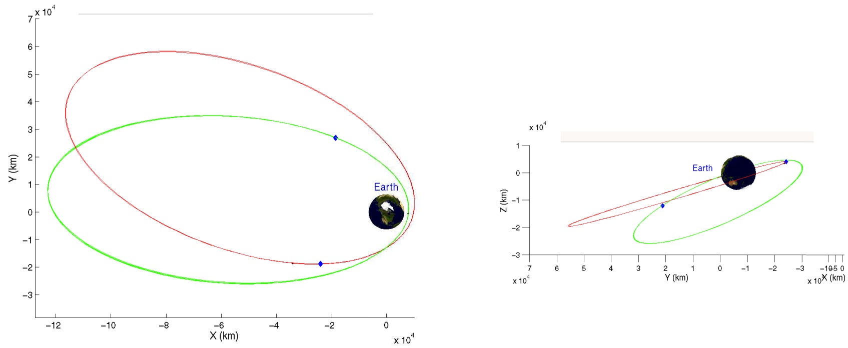

Fig. 6 a). Initial orbit of the Earth orbit portion of the P1 preliminary trajectory design. The initial condition for ARTEMIS P1 predicted when ARTEMIS was proposed is in green; the actual starting orbit is in red. b). End-on view of a).

a "slosh resonance" (Sholl et al. 2007; Auslander et al. 2008; Frey et al. 2008). This further exacerbated gravity losses, necessitating more maneuvers to obtain the same total orbit-raise ΔV. This was addressed by starting the ORM sequence for P2 as early as July 20, 2009.

3. Side thrusting for orbit-raise maneuvers also results in a small reorientation (precession) of the spin axis due to a small offset of the thrust direction relative to the probe center of mass. The cumulative effect of side thrusting has been significant spin-plane precession of the probes in directions that either violated operational constraints or increased losses from vector-thrusting. Spin axis reorientation maneuvers were included in the mission design to account for that effect.
4. Thrust restrictions due to the absence of "up" thrusting capability posed a non-traditional restriction to the mission design. The usual intuition that 1 burn allows targeting of 3 elements and 2 burns separated in time allows for the targeting of 6 elements is not correct for ARTEMIS. In fact, even 3 separated burns can fail to provide 6-element targeting when all maneuvers are confined to a single plane.

5.1.1 Orbit-raise Design Process

The P1 and P2 orbit-raise designs were constructed using Mystic software (Whiffen 1999, 2006). Mystic was able to accommodate all mission constraints outlined above. However, the complex (and often treacherous) design space resulting from numerous lunar approaches during the orbit-raise phase made simple design strategies impossible. To provide some robustness against missed burns, and sufficient tracking data for orbit/maneuver reconstruction, perigee maneuvers were double-spaced, i.e., two orbits apart. On occasion it proved advantageous to separate burns even farther to take advantage of or avoid strong lunar interactions. Most perigee burns were divided into and modeled as two separate burn arcs, one on either side of the Earth's shadow. The duration and pointing of each burn was fully optimized using Mystic, with the constraint that the end states of this phase would be on the translunar trajectories already designed.

Several different end-to-end orbit-raise strategies were thus attempted for both P1 and P2, with the desired translunar injection as a goal and the initial ARTEMIS state as a starting point as early as needed, i.e., with an ascend start date unrestricted by THEMIS science considerations. The strategy that proved most successful for the P1 trajectory was to first optimize sets of burns on three double-spaced perigees to reach an orbital period of 131 hours. From states near this point 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 would be best to rejoin the low-energy transfer. To address this problem, a large number of ballistic trajectories were used as initial guesses for targeting and optimization. Different families were organized based on the number of Earth revolutions. A computer cluster was used for this compute-intensive process. Trajectories that were found to be feasible or nearly feasible were then further refined by moving the time of rejoining the low-energy transfer to successively later dates.

5.1.2 P1 and P2 Orbit-raise Designs

The P1 low-energy transfer began with a pair of lunar flybys separated by only 14 days–see Figure 7. To minimize the ΔV cost of getting onto the designed translunar trajectory, it seems desirable to match these flybys as closely as possible, though exact matching does not seem to be necessary. Intuitively, re-joining the low-energy transfer at later times would provide increasing efficiency, since a longer time would allow a lower rendezvous velocity. It was expected (and found) that re-joining much beyond the second lunar flyby provided diminishing returns. The final total effective ΔV for P1's Earth orbit phase as actually flown was 106.3 m/s (compare this to the 107.8 m/s final allocation and the 125 m/s conservative estimate in the ARTEMIS proposal (Angelopoulos and Sibeck 2008) from a single-impulse Earth departure, which included 24 m/s for gravity and steering losses and trajectory correction maneuvers (**TCM**s)). The final design maneuvers are given in Table 1, along with trajectory correction maneuvers designed by the mission operations team during the execution of this phase.

The P2 orbit design was more complex than the P1 design because P2 begins in a much smaller orbit. A process similar to the P1 design process was used to develop the P2 orbit-raise design. Very careful planning of distant lunar approaches was necessary to stay within the allocated ΔV budget, which was was more constraining for P2 than for P1. The P2 orbit raise required 42 burns, counting each split maneuver as two burns (see Table 2). The method used was a branching process. Each orbit-raise maneuver was designed several times to reach different orbital periods (different period = different "branch"). Subsequent maneuvers reaching longer periods were designed for each branch. The most promising branches were continued; poorly performing branches were abandoned. Poorly performing branches often led to situations in which 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 that provided maneuver savings as little as 1 meter per second early in the orbit raise were sought. The final few orbit-raise maneuvers required very careful planning to maximize the positive influence of the Moon.

A major additional complication of the P2 trajectory design occurred shortly before the first ORM, when a check for eclipses found an unacceptably long passage through Earth's shadow just after the ORMs and before the first lunar flyby. Additional shadow-deflection maneuvers (SDMs) were added to change the orbit plane to reduce the time in shadow and then change the orbit plane back to return to the planned flyby conditions. These SDMs solved the problem without requiring a complete redesign of the series of ORMs, though at

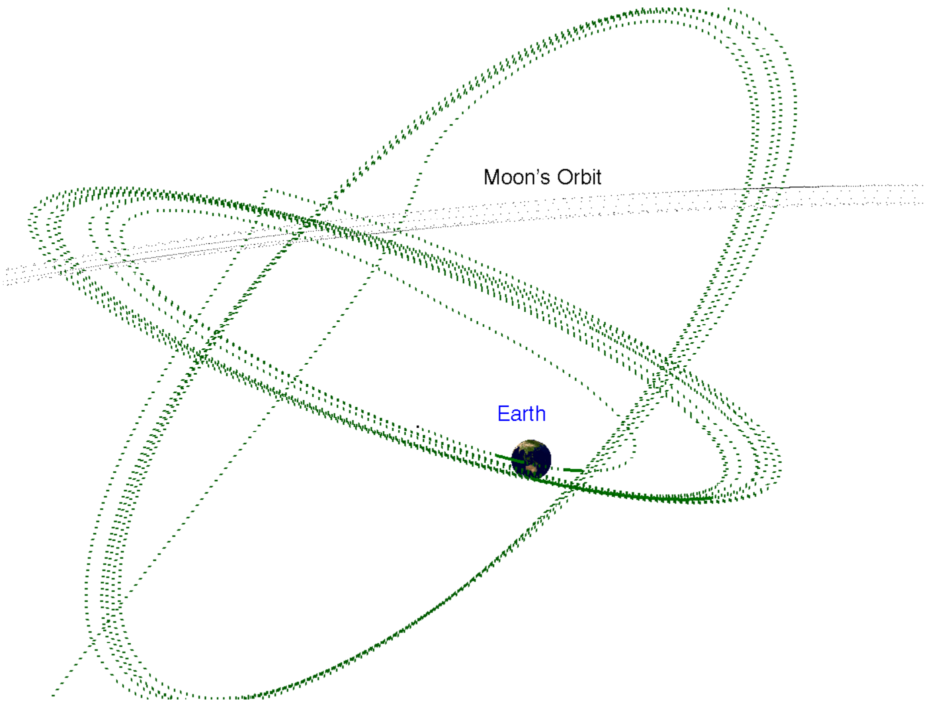

Fig. 7 P1 low-energy transfer showing 2 flybys from an oblique viewpoint.

a cost of 11 m/s in additional ΔV. Even with these maneuvers added, the final total effective ΔV for P2's Earth orbit phase as actually flown was 255.5 m/s (compare to the 264.7 m/s finally allocated and the 219 m/s originally estimated (Angelopoulos and Sibeck 2008) from the single-impulse Earth departure, which included 33 m/s for gravity and steering losses and TCMs). The final design and trajectory correction maneuvers are given in Table 2.

5.2 Trans-Lunar Phase Trajectories

The trans-lunar phase of the ARTEMIS trajectory for each probe extended from the first close lunar flyby to insertion into the target Lissajous orbit.

Figure 8 shows the trans-lunar phase of the ARTEMIS trajectory for P1. The trajectory is shown in the same Sun-Earth synodic coordinate frame used in Figures 4 and 5. In the figure the trajectory begins on the right side of the plot with "Lunar Fly-by #1". The P1 trajectory made use of a "back-flip", wherein the first lunar fly-by set up a second lunar fly-by on the opposite side of the Moon's orbit $\sim$14 days later. The back-flip can be seen clearly in the out-of-plane view insert in the bottom left of Figure 8 and the beginning of it is shown in Figure 7. This second flyby raised the apogee significantly, throwing the probe out beyond the Moon's orbit towards the Sun. This began the low-energy trajectory leg for P1, which is characterized by significant gravitational perturbation imparted on the probe by the Sun. This low-energy trajectory had two deep-space legs that included one relatively small deep-space maneuver (DSM). After the second leg, the orbit perigee had been raised to lunar

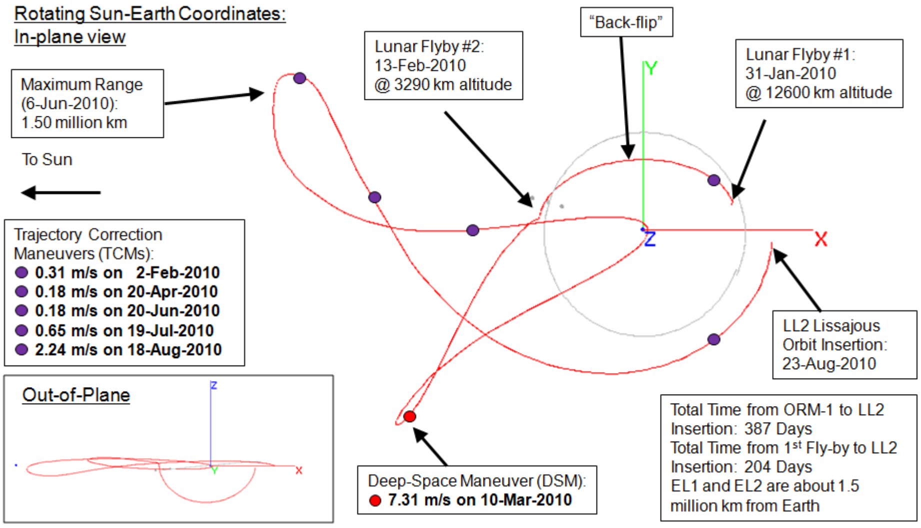

Fig. 8 Trans-lunar portion of the P1 trajectory design. Distances quoted are ranges measured from the center of mass of the Earth or Moon.

distance, and the phasing with the Moon's orbit was such that the probe moved into a lunar Lissajous orbit around lunar Lagrange point #2 (LL2) without requiring any deterministic insertion maneuver. By the time P1 reached the Lissajous orbit in August of 2010, 387 days had elapsed since the start of ARTEMIS maneuver operations.

Figure 9 shows the trans-lunar trajectory for P2. The P2 trajectory only included one lunar fly-by, which sent the probe away from the Sun and beyond the Moon's orbit into a region where the perturbative influence of solar gravity is significant. P2 followed a low-energy trajectory that included three deep-space legs before entering a lunar Lissajous orbit around lunar Lagrange Point #1 (**LL1**) without any deterministic thrusting. The P2 trajectory included three deep-space maneuvers (DSM), one relatively large, totaling 30.4 m/s. P2 arrived in Lissajous orbit about 2 months after P1, requiring a total of 458 days since the start of ARTEMIS maneuver operations to reach this stage.

5.2.1 Transfer Trajectory Implementation

As the transfer trajectory was flown, correction maneuvers were required to adjust for earlier maneuver execution and probe pointing and implementation errors, as well as navigation errors. These maneuvers, called trajectory correction maneuvers (TCMs), encompassed the statistical maneuvers along the transfer. TCMs in addition to DSMs were inserted in each of the P1 and P2 designs.

We allocated 4% of the total propellant budget of each probe to perform any required TCMs along the way to control the energy to keep P1 and P2 near their appropriate outgoing trajectories. Since the two probes had already completed their primary mission in a highly elliptical Earth orbit, propellant was extremely limited. Thus, with the unique operational constraints, accomplishment of the transfer goals with the minimum propellant cost was the highest priority. To implement the mission design, our trajectory simulations use a full ephemeris model with point-mass gravity representing Earth, Moon, Sun, Jupiter, Sat-

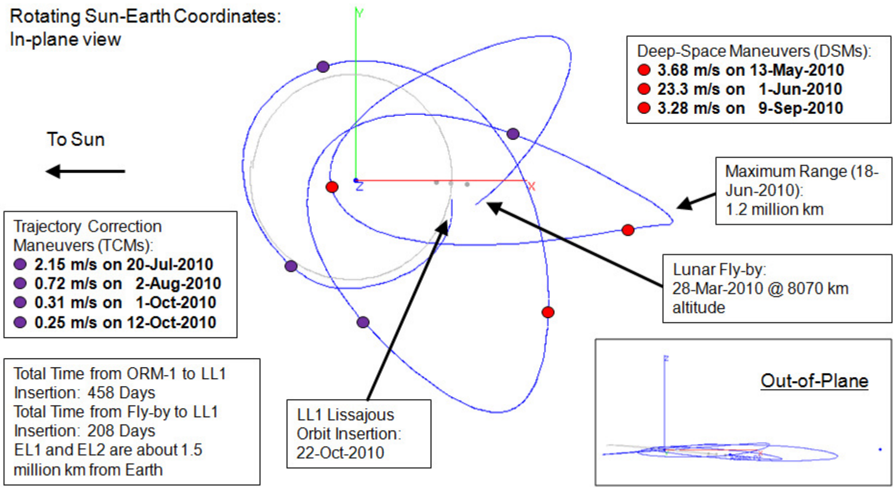

Fig. 9 Trans-lunar portion of the P2 trajectory design. Distances quoted are ranges measured from the center of mass of the Earth or Moon.

urn, Venus, and Mars. Also included is an eighth degree and order Earth potential model. The solar radiation pressure force is based on: (1) the measured probe area, (2) the probe estimated mass (from bookkeeping), and (3) the coefficient of reflectivity determined by navigation estimation. The same models with estimates for the mass usage and coefficient of reflectivity were used in the mission design process that determined the reference trajectory. The trajectory propagations in operations were based on a variable step Runge-Kutta 8/9 and Prince-Dormand 8/9 integrator. Initial conditions used throughout the planning process correspond to the UCB-delivered navigation solutions using the DSN and the UCB tracking system. Software tools used in this process include the General Mission Analysis Tool (GMAT) developed at GSFC as an open source, high-fidelity tool with optimization and MATLAB connectivity and AGI's STK/Astrogator suite.

To compute actual commanded maneuver ΔV requirements, we used two numerical methods: differential corrections (DC) targeting using central or forward differencing and an optimization method using the VF13AD algorithm from the Harwell library. A DC process provided *a priori* conditions. Equality constraints were incorporated for DC application; nonlinear equality and inequality constraints were employed for optimization. These constraints incorporated both the desired target conditions in the Earth-Moon system and probe constraints on the ΔV direction and relationship between the spin axis and the ΔV vector.

The end goal of the transfer phase was to achieve the Earth-Moon Lissajous insertion conditions necessary for a minimal energy insertion into the Earth-Moon L2 or L1 Lissajous orbits. The goals were defined in terms of states expressed in Earth J2000 coordinates. These targets were held constant over the entire mission design and implementation process once the reference translunar transfer had been designed. Although a baseline trajectory was defined to design the mission, the adaptive strategy used in operations required exactly matching this baseline only at the end of the transfer.

5.2.2 Navigation Uncertainties

Throughout the transfer trajectory implementation process, navigation solutions were generated at a regular frequency of once every three days with the exception of post-maneuver navigation solutions, which were made available as soon as a converged solution was determined. The rapid response was to ensure that the maneuver had performed as predicted and that no unanticipated major changes to the design were necessary. The RSS of the uncertainties were on the order of tens of meters in position and below 1 cm/s in velocity. As a conservative estimate for maneuver planning and error analysis, 1σ uncertainties of 1 km in position and 1 cm/s in velocity were used. These accuracies were obtained using nominal tracking arcs of one three-hour contact every other day. The Goddard Trajectory Determination System (GTDS) was used for all navigation estimations.

5.2.3 Trajectory Design During Operations

The transfer trajectory implementation approach used the numerical methods discussed above augmented by dynamical systems theory for verification and to gain knowledge of the transfer dynamics. The probes were targeted to the libration point orbit insertion locations knowing full well that maneuver execution and navigation errors would push the path off the "baseline" design. A correction maneuver scenario was planned that would essentially shift the trajectory, such that the new path would be consistent with a nearby manifold. It was decided to use a forward-integrating numerical optimization process that included probe constraints to calculate optimized ΔVs. This procedure permitted minimization of the ΔV magnitude, variation of the ΔV components in direction, as well as variation of the maneuver epoch, while incorporating the nonlinear constraint on the probe ΔV direction relative to the spin axis.

Originally, it was envisioned that errors in navigation and maneuvers could lead to the need for an unobtainable correction in an "up" direction with respect to the ecliptic plane. Fortunately, experience with trajectory design on other missions that incorporate weak stability regions near Sun-Earth libration orbits and near the ecliptic plane showed us that we could allow upward ΔV corrections to be delayed until an equivalent magnitude but opposite direction (downward) ΔV location could be found in the long-duration transfer. These locations were then used to correct the trajectories without any upward maneuver component to achieve the final Earth-moon insertion targets.

As the TCMs were performed, the path essentially jumped from the vicinity of one local transfer manifold to another at a slightly different energy level. The number of optimized TCMs was very low and their magnitudes quite small, considering the sensitivity of the dynamics and uncertainties of the OD solutions.

5.2.4 Maneuver Design

To target to the desired Earth-Moon Lissajous conditions, a VF13AD optimizer was used. We optimized each maneuver to determine the minimal ΔV location. To determine an *a priori* maneuver location and to achieve an intuitive feel for the maneuver results, a DC process was first performed using planned DSN coverage. For P1, the first four TCMs were completed in Earth-centered elliptical orbit or during lunar gravity-assist targeting. Maneuver execution errors are small, only a few percent. These errors are a function of actual start time with respect to a sun pulse of a spinning spacecraft, tank temperatures, attitude knowledge, and general propulsion system performance.

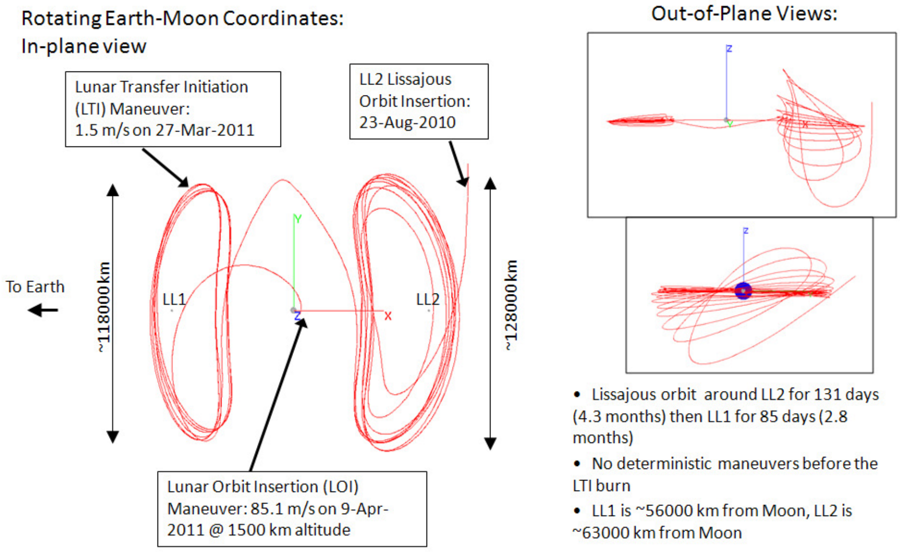

Fig. 10 Lissajous orbit phase of the P1 trajectory. Distances given are ranges measured from the lunar center of mass unless otherwise specified.

It should be noted that maneuver execution errors, current navigation errors, and subsequent maneuvers to correct for these errors along with small mis-modeled perturbations can lead not only to late or early arrival times at the prescribed Lissajous insertion location, but also may contribute to out-of-plane effects and may result in trajectories that intersect with the Moon. Clearly, the trajectory is very sensitive to such small variations. But that sensitivity also implies that small corrections can alter the trajectory design significantly and allow low ΔV cost orbit control, assuming sufficiently frequent tracking for orbit reconstruction.

5.3 Lissajous Orbit Phase Trajectories

The Lissajous orbit phase of ARTEMIS has permitted repeated observations of the distant lunar wake. For the first $\sim$ 1.5 months of this phase (from August 22 to October 2, 2010), P1 was alone at the Moon in orbit around the LL2 point while P2 was still en route. P2 then arrived, making a partial orbit around LL2 on its way to Lissajous orbit at LL1. For about the next 2.3 months, P1 orbited LL2 while P2 orbited LL1, and then P1 also crossed over to orbit LL1. During this phase, the trajectories permit 16 independent observations of the lunar wake when crossing behind the Moon on the anti-Sun side, observations of the distant Earth magnetotail once per month when the Moon's orbit passes through it, and observations of the pristine solar wind when out of the influence of both. These two-point measurements were made at separation scales up to $\sim$100000 km when the probes were in orbit around different Lagrange points and up to $\sim$50000 km when both orbit LL1. Distant magnetotail measurements can also be correlated with concurrent measurements from THEMIS-A, THEMIS-D, and THEMIS-E in low-Earth orbit.

Figure 10 shows the P1 trajectory during the Lissajous orbit phase. In this figure, the Moon is at the origin and the trajectory is drawn in the Earth-Moon synodic coordinate

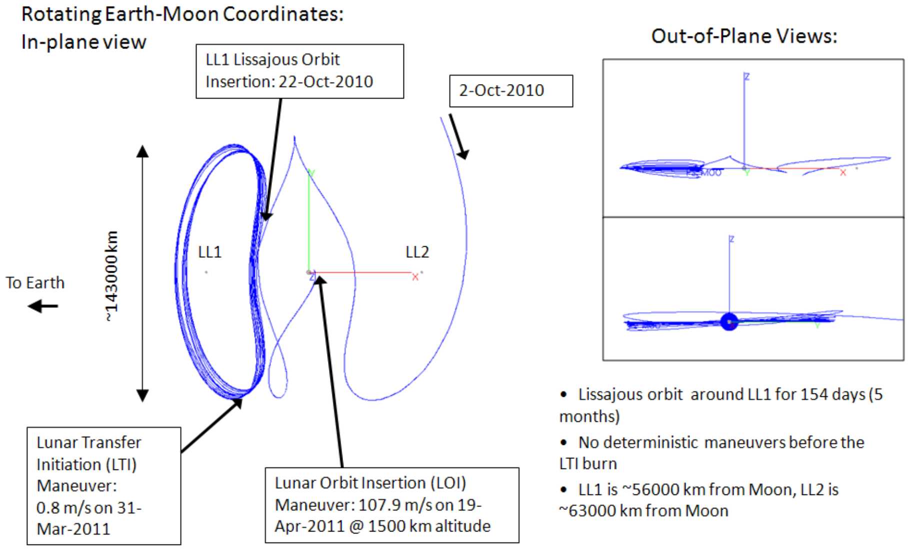

Fig. 11 Lissajous orbit phase of the P2 trajectory design. Distances given are ranges measured from the lunar center of mass unless otherwise specified.

frame, which rotates such that the Earth is always to the left along the negative X axis. The Z axis is aligned with the angular momentum vector of the Moon's geocentric orbit. The main figure on the left side shows the view looking down on the geocentric orbital plane of the Moon, and the two insets show perspectives from within the Moon's orbital plane. The LL1 and LL2 points are marked in the figure.

P1 entered Lissajous orbit around LL2 on August 23, 2010 without a deterministic maneuver. Although the initial Lissajous orbit was somewhat inclined with respect to the Moon's geocentric orbit plane, the orbit flattened after a few orbits (see Figure 10 inserts). After $\sim$131 days in orbit around LL2, P1's trajectory followed an unstable orbit manifold along a 10-day heteroclinic connection to a Lissajous orbit around LL1 (Howell et al. 1997; Koon et al. 2000). Although this transfer required no deterministic ΔV for initiation or insertion, in practice weekly station-keeping maneuvers (**SKM**s) were required to maintain the Lissajous orbit. P1 will have spent 154 days orbiting LL1 before executing a small maneuver to depart from Lissajous orbit on June 14, 2011. The probe descends to an 1850 km periselene altitude, where the lunar-orbit insertion (**LOI**) maneuver is executed, beginning the lunar orbit phase on June 27, 2011. At the time of LOI, P1 will have operated for 707 days since the beginning of the ARTEMIS mission.

Figure 11 shows the P2 trajectory during the Lissajous orbit phase. P2 entered Lissajous orbit around LL1 on October 22, 2010. As with P1, this insertion was achieved without any deterministic ΔV because the incoming trans-lunar trajectory approached on the stable manifold of this particular Lissajous orbit. P2 will have stayed in this nearly planar Lissajous orbit for about 8.5 months before initiating descent to a $\sim$3800 km altitude periselene on July 5, 2011. The LOI maneuver for P2 begins on July 17, 2011, at which time P2 will have been operating for 727 days since the end of the nominal THEMIS mission.

After P1 and P2 entered their Lissajous orbits, a project decision was made to extend the Lissajous phase from April to July. This required adding axial components to SKM18 on February 1, 2011, for P1 and to SKM11 (January 4), SKM13 (January 18), and SKM15 (February 1) for P2. These axial burns, which directly affected the Z velocity of the probes in the Earth-Moon frame, were needed to prevent the Z axis components of the Lissajous orbit states from oscillating too much. These oscillations otherwise grow to uncontrollable levels before the transition to lunar orbits despite ARTEMIS's stationkeeping process.

The Lissajous orbit phase of the ARTEMIS mission is particularly exciting because the ARTEMIS probes are the first to fly in a lunar Lissajous orbit. Flying these orbits continues to be a challenge for operations and maneuver design teams because Lissajous orbits are inherently very unstable; small, unavoidable deviations from the Lissajous orbit are amplified to problematic proportions (Howell and Keeter 1995) after approximately one revolution ($\sim$14 days). This leaves little room for error in the operations. Because of this instability, correction maneuvers need to be executed about weekly to keep the probes in orbit. So even though these orbits require no deterministic ΔV, orbit maintenance ΔV is required.

5.3.1 Stationkeeping

There are many stationkeeping methods to chose from: classical control theory or linear approximations of Farquhar (1971) and Hoffman (1993), who provided analysis and discussion of stability and control in the Earth-Moon collinear L1 and L2 regions; Renault and Scheeres (2003) offered a statistical analysis approach; Howell and Keeter (1995) addressed the use of selected maneuvers to eliminate the unstable modes associated with a reference orbit; and Gómez et al. (1998) developed and applied the approach specifically to translunar libration point orbits. Folta et al. (2010) presented an analysis of stationkeeping options and transfers between the Earth-Moon locations and the use of numerical models that include discrete linear quadratic regulators and differential correctors.

The ARTEMIS stationkeeping method uses maneuvers performed at optimal locations to minimize the ΔV requirements while ensuring continuation of the orbit over several revolutions downstream. There are no reference trajectories to plan against, so other methods such as linear (continuous) controllers are impractical. Likewise, other targeting along the X axis or Y axis is more costly or cannot be attained without violating probe constraints. Goals in the form of energy achieved, velocities, or time at any location along the orbit can be used, but our goal is defined in terms of the X velocity component at the X axis crossings. This assumes selection of a velocity that can be related to the orbit energy at any particular time. To initialize the analysis, a DC scheme is used, based on the construction of an invertible sensitivity matrix by numerical sampling of orbital parameters downstream as a consequence of specific initial velocity perturbations (Folta et al. 2010). The orbit is continued over several revolutions by checking the conditions at each successive goal then continued to the next goal. This allows perturbations to be modeled over multiple revolutions.

The targeting algorithm uses an impulsive maneuver with variables of either Cartesian ΔV components or ΔV magnitude and azimuth angle within the ARTEMIS spin plane. Target goals are specified uniquely for each controlled orbit class because LL1 and LL2 dynamics differ slightly. The velocity target chosen is specifically set to continue the orbit in the proper direction. Targeting is then implemented with parameters assigned at the X-Z plane crossing such that the orbit is balanced and another revolution is achieved. Each impulsive maneuver is targeted to the X component of the velocity at the third X axis crossing after the maneuver; the maneuver supplies velocity (energy) in a direction that subsequently continues the libration point orbit. Additionally, the VF13AD1 optimizer is used to minimize the stationkeeping ΔV by optimizing the direction of the ΔV and the location (or time) of the maneuver. Included in the DC and optimization process are constraints required to keep the ARTEMIS maneuvers in the spin plane.

Given the constraints of the ARTEMIS mission orbit, probe maneuvers are currently planned at a seven-day frequency to ensure a stable navigation solution while minimizing the ΔVs and staying within the ARTEMIS ΔV budget. The maneuvers were originally planned to occur at or near the X axis crossings and to use a continuation method to maintain the orbit. The location of each maneuver has been relaxed to permit a user-friendly operational schedule. Orbital conditions have been and will continue to be set to permit the energy or velocity at the crossings to continue the orbit for at least 2 revolutions.

5.4 Lunar Orbit Phase Trajectories

Most scientific observations of the lunar wake occur during the lunar orbit phase, which nominally lasts 2 years. For initial analysis, both probe orbits were modeled as nearly equatorial with roughly 1500 km altitude periselene and 18000 km radius aposelene. Consideration of these initial conceptual orbits allowed us to understand their evolution with time and to analyze how well they would serve to meet the required science measurements. The main driver of change in these orbits is Earth's perturbing gravitational influence. We found that the eccentricity of each orbit oscillates over time so the periselene altitude of a retrograde orbit varies by several hundred kilometers and that of a prograde orbit by as much as a thousand kilometers. This change in eccentricity is driven by the tidal force of Earth's gravity on the probe, which is most effective when the probe is farthest from the Moon, i.e., at apoapse of the lunar orbit. Because the orbit orientation changes much more slowly than the Moon goes around the Earth, the interaction of the probe velocity vectors and the direction of the tidal acceleration at apoapse results in a biweekly oscillation in periapse altitude, with the lowest periapses occurring around lunar longitudes of 90 deg and 270 deg for P1 and 0 deg and 180 deg for P2. Also, the axial burns included in SKMs in January and February of this year, which were required to extend the Lissajous phase, were tuned to achieve higher inclinations of the lunar orbits than originally designed so that low-altitude periapse latitudes could be raised to provide coverage of magnetic anomalies in the crust in support of planetary science (see Figure 12).

Another effect of Earth's perturbation on the orbits is to cause the ecliptic longitude of the periapse to change in the same direction as the orbital motion by about 100 deg per year, so that putting the probes into opposing orbits, e.g., P2 prograde and P1 retrograde, would maximize the relative motion of their lines of apsides. The combination of this apsidal motion with the significant eccentricity of the orbits enables observations at a wide range of probe separations (from $\sim$150 to $\sim$30000 km) and geometries to be achieved during this phase. Figure 13 shows the range from the Moon's center in the anti-Sun direction of the lunar wake crossing observation opportunities for these conceptual orbits as a function of time. Note the large number of potential measurements, the variety of down-Sun ranges, and the variety of relative geometries of P1 and P2.

5.4.1 Finite-Burn Maneuvers for Lunar Orbit Insertion and Period Reduction

As early as 2008, the lunar orbit insertion (LOI) was modeled as both a finite-burn maneuver and an impulsive-burn maneuver so potentially large ΔV penalties could be understood. Because of the small size of the tangential thrusters, the transition to operational lunar orbit needs to be split into a number of fairly long maneuvers that incur significant gravity losses

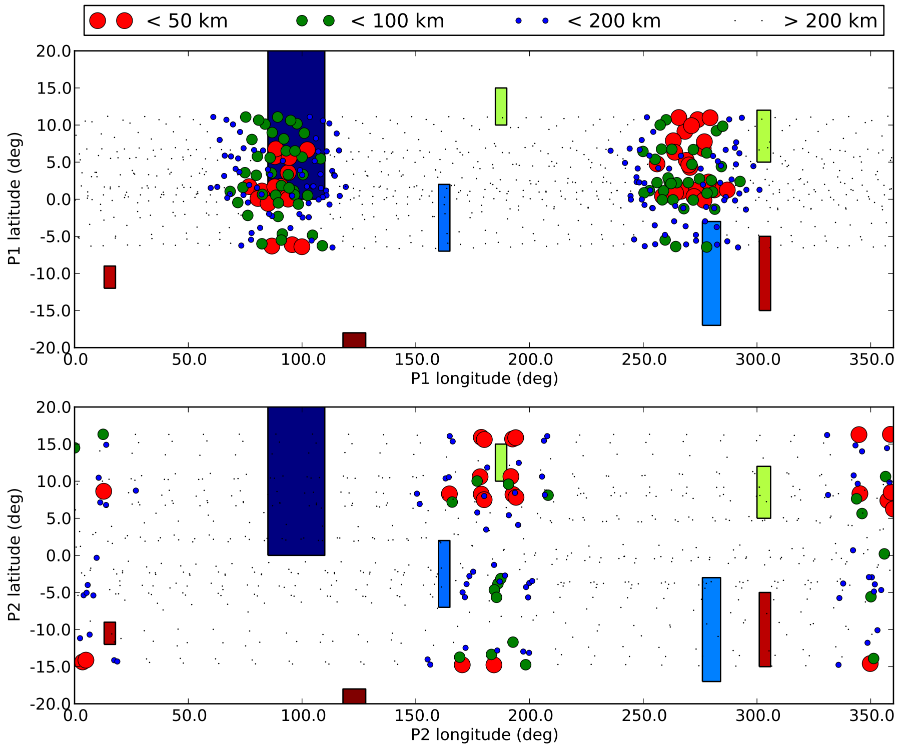

Fig. 12 Periapse locations at the Moon during the four years of lunar orbits. Periapse altitudes are affected by perturbations due to Earth's gravity so that they are lowest at constrained longitudes for P1 and P2. Orbital dynamics are such that the periapse latitudes of the lunar orbits are correlated with the arguments of periapse. The lunar orbits have been tuned to give good coverage of certain magnetic anomalies in the lunar crust (shown as shaded rectangles).

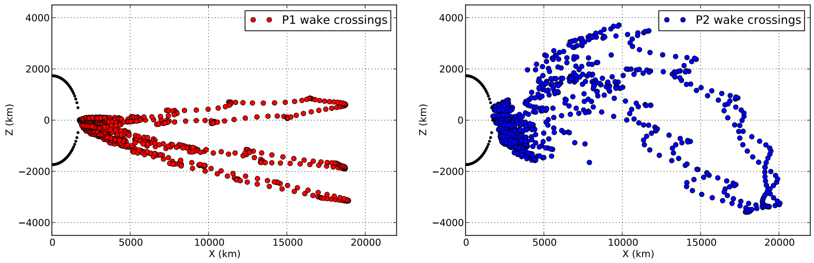

Fig. 13 P1 (red) and P2 (blue) lunar wake observation opportunities during LOI and the lunar orbit phase, with the Moon's limb indicated by black dots. Each red or blue point is a plane crossing of the respective lunar orbit, where the Sun is on the $-X$ axis.

on the longest maneuvers and thrust inefficiencies due to probe rotation. Despite gravity losses arising from long arcs, a significant ΔV has to be applied at the first periapsis to capture into a low enough lunar orbit that Earth-gravity perturbations do not cause the probe to impact at subsequent periselenes. Analysis to date shows that this requires an LOI for P1 that lasts between 135 and 140 minutes if the LOI is modeled as an anti-velocity burn. The geometry of P2 arrival is different; it requires that the initial periapse be much higher at around 3800 km and that the LOI be significantly longer at around 173 minutes. The P2 LOI duration has been further increased to about 205 minutes to improve the eclipse phasing for the PRMs.

An initial LOI design using finite burns for both P1 and P2 served as a guide to the expected ΔV required for LOI. Both designs made use of a $+/-30$ deg pulse width, making them 95.5% efficient in imparting ΔV in a specified direction. Thrust was modeled to be centered around the anti-velocity vector. The P1 design incurred only a 4.8 m/s finite-burn penalty on the 85.1 m/s single-impulse LOI design. The P1 finite-burn design consisted of an LOI followed by five PRMs on succeeding periapses, with almost all the finite-burn loss being gravity losses on the LOI portion. The P2 design incurred a 9.2 m/s finite-burn penalty relative to the 107.9 m/s single-impulse LOI solution; this penalty was proportionately larger than P1's because the P2 LOI occured at a much higher altitude. The P2 LOI was followed by 9 subsequent PRMs.

Recalling that the ARTEMIS thrust cannot be dynamically steered along an anti-velocity direction without significant redesign of its thruster operations software, we opted to segment the LOI finite burn into a series of three constant-direction maneuvers separated by a minimum of three minutes. For fuel efficiency purposes, all the PRMs should be small and performed around periselene. By keeping these maneuvers less than half an hour each, gravity and steering losses were found to be around 1% in high-fidelity modeling of an orbit in the spin plane. An analytic estimate using an orbit inclined by 20 deg found that the additional steering losses amounted to only another 2%.

These PRM maneuvers, however, still need careful attention because of the biweekly oscillation in probe periselene altitude. This can cause the series of PRMs to result in a high probe periapse. P2's periapse ended up around 1500 km in an early simulation because as the period of the orbit is reduced, the influence of Earth's gravity is also reduced, and subsequent oscillations in periapse altitude have a reduced amplitude. As a result, period reduction done at a high altitude periapse will raise the periapse altitudes of all subsequent orbits whose smaller oscillations cannot push the periapses back down as low as periapses before the maneuver. Thus, PRMs should only be performed on orbits with particularly low periapses. This constraint does have an upside, though — thrusting at the lower periselene altitudes allows for improved PRM efficiency. Another consideration in the design of the sequence of PRMs is to have at least three days between PRMs to reduce stress on the mission operations team. Sequences of PRMs successfully incorporating all of the above considerations were designed for the April transitions to lunar orbit for both probes.

As discussed above in subsection 5.3, the mission design was recently changed to perform a later transition from Lissajous orbits to lunar orbits, bring the periselenes as close to the surface as LOI uncertainties allow, and increase the inclination to somewhere between 20 and 30 deg. This has involved a redesign for the LOI and PRMs that continues to be refined at the time of this writing. This new baseline design has P1 departing Lissajous orbit around June 14 for a lunar periapse at an altitude of 1850 km at about 3 PM (UTC) on June 27 and P2 departing Lissajous around June 28 for a lunar periapse at an altitude of 3800 km at about 11 PM on July 17. Figure 14(a)-(b) shows the new baseline LOI trajectories (and subsequent lunar orbits) in the Earth-Moon synodic coordinate frame with the Earth fixed

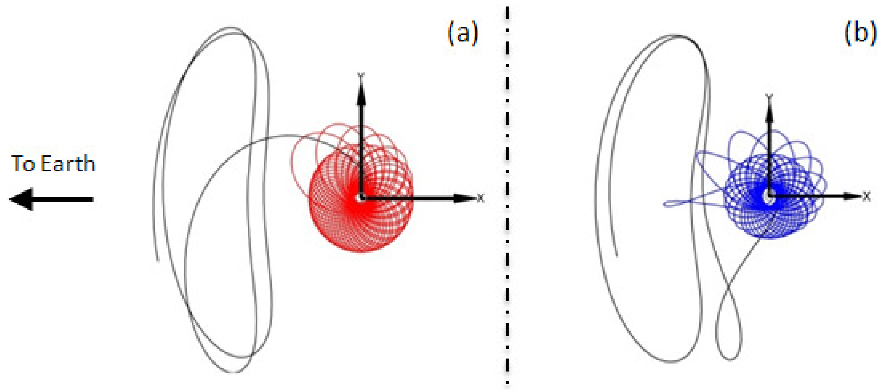

Fig. 14 LOI and low-lunar orbit trajectories for (a) P1 and (b) P2 in the rotating Moon-centered frame. LL1 Lissajous orbits shown for scale.

on the negative X axis. With high-fidelity modeling of gravity and steering losses for segmented finite-burn maneuvers, P1 takes 98.8 m/s characteristic ΔV and P2 takes 126.9 m/s in the current baseline design (as of May, 2011), where *characteristic* ΔV is the ΔV the thrusters would have provided if the spacecraft weren't spinning.

The baseline periapse altitudes during the first six months in lunar orbit are shown in Figure 15. As the figure shows, the absence of secular drift in the P1 orbits means that the lowest early periapses can be targeted quite close to the surface, and the current baseline has succeeded in doing so without using planetary-science enhancement burns (PEBs) at apoapse to lower the periapse. Because the current baseline still leaves P2 with a relatively small number of very low periapses, we are continuing to adjust the PRM sequence for P2 and are likely to include a PEB to lower periapse and increase the number of very low periapses. Later in the lunar orbit phase, additional PEBs are planned to control the periapse altitude up or down or to make other orbit changes to optimize science measurements.

Care was taken in the arrival design to ensure that very long eclipses do not occur during the early, longer-period lunar orbits. Once the science operations orbit is achieved, very long eclipses are not an issue; in fact, this was a major determinant in the size of the science operations orbit. More recently, concern has been raised about the age of the batteries on board the probes; they will be about eight years old when long eclipses occur again in the summer of 2013. To allow for age-diminished energy storage capacity, the baseline design of the LOI and PRMs for P1 has been tuned to reduce the total duration of later eclipses (combined umbral and penumbral) to less than about three and a half hours, and P2 is being adjusted likewise. Figure 16 shows the eclipse history for three and a half years in lunar orbit for the baseline design as of May, 2011.

For the baseline designs (as of May, 2011), PRMs are included on periapse numbers 3, 5, 26, and 28 for P1, and 2, 20, 36, and 45 for P2. Figures 17 and 18 show the values for probe altitude, inclination, and latitude as determined at periapse for the lunar orbits projected as a consequence of the main Z-oscillation damping maneuver on January 4, 2011. Note that the maximum latitude of P1 at periapse is much less than the co-inclination of the orbit (where the co-inclination is defined as 180 deg minus the inclination and measures how far the

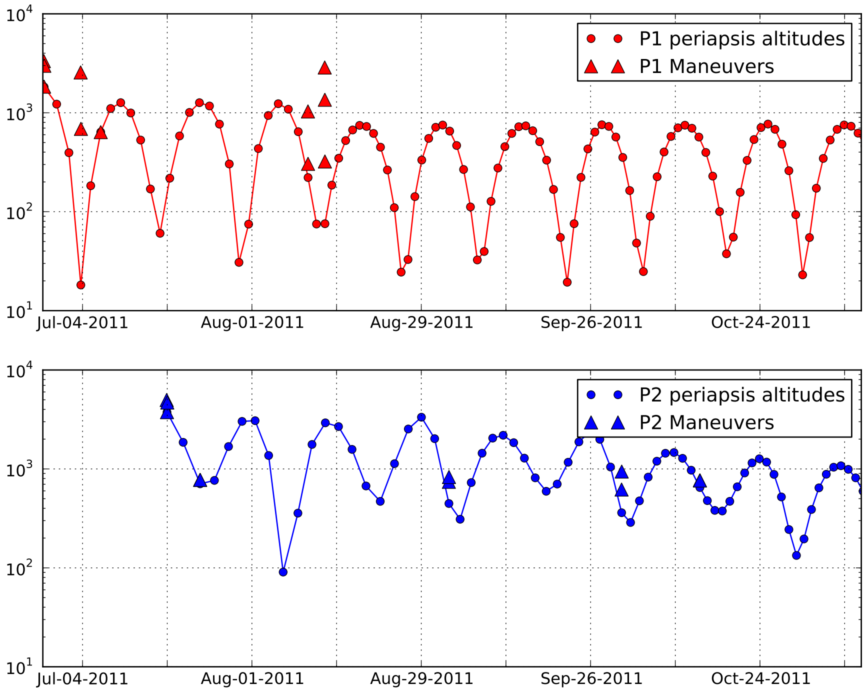

Fig. 15 Altitude at periapse for P1 in lunar orbit with LOI in June, 2011 and for P2 with LOI in July, 2011. The LOIs and most of the PRMs have been simulated as segmented finite burns at selected periapses; the altitude at the start of each segment is shown by a triangle.

P1 retrograde orbit plane is from the X-Y plane). This relationship between periapse latitude and inclination is characteristic of all the lunar orbits studied; the Earth gravity perturbations on the orbit, which make both quantities oscillate, are such that periapse latitude extremes only happen when the probe orbit plane is closest to the Earth-Moon orbit plane, i.e., has inclination nearest 0 deg or 180 deg.

The higher inclinations have added a long term component to the periapse oscillations, which means many fewer periapses are near the minimum altitude, especially for P2. This effect can be reduced by reducing the size of the lunar orbits, but doing so would also change the rate of progression of the periapses and disturb the times when the P1 and P2 periapses align with the lunar terminator to support LADEE. Once LADEE has been taken care of, the lunar orbits will be reduced as much as remaining propellant allows to significantly increase the number of very low periapse passages.

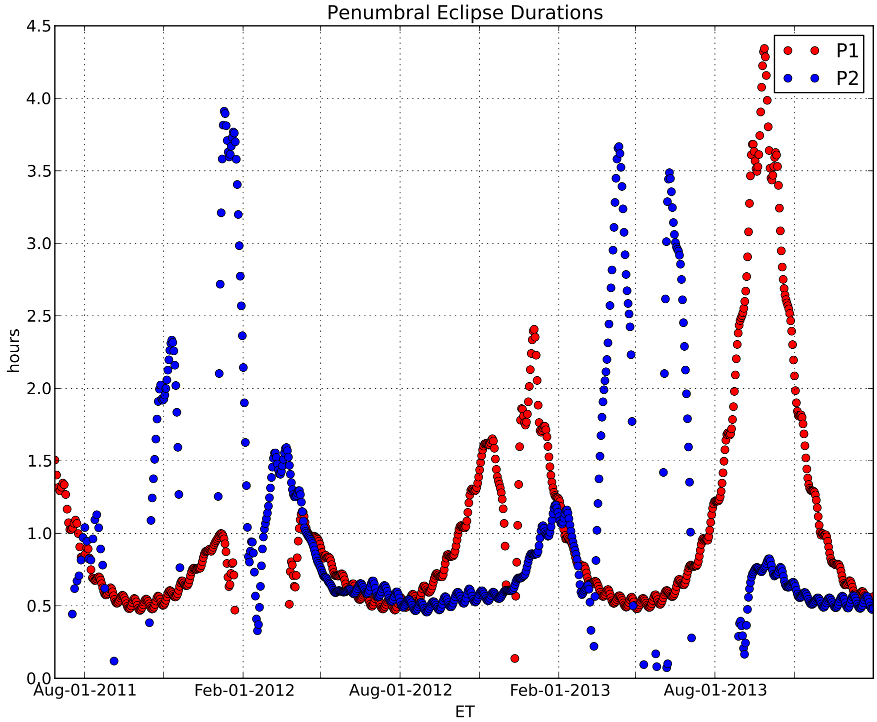

Fig. 16 Durations of eclipses for P1 and P2 while in lunar orbit. These include umbral and penumbral shadows from both the Earth and the Moon.

6 Mission Status

As of May, 2011, both P1 and P2 have successfully arrived into and maintained Lissajous orbits around the Earth-Moon L1 point. Both probes and their instruments are functioning normally. One minor surprise occurred on October 14, 2010, when a small, sudden change was observed in the velocity and spin rate of P1, which was quickly traced to the loss of the EFI sensor ball at the end of one of the four EFI booms deployed from the sides of the probe. This loss has been attributed to a micrometeorite severing the fine wire that connected the sensor ball to the end of the boom (http://www.nasa.gov/mission_pages/themis/news/artemis-struck.html). Although reduction in the number of EFI sensors will cause a slight reduction in the quality of the electric field measurements, the instrument still satisfies its science requirements. The loss of the sensor mass also shifted the probe's center of mass, which will complicate operations somewhat, especially in the management of the propellant on board, and affect the mission design because side maneuvers now have a much larger (though still small in absolute terms at -0.05 RPM per m/s) effect on the spin rate.

After the decision to delay the LOI until June/July 2011 to better accommodate planetary science objectives, it became necessary to modify both P1 and P2 Lissajous orbits by

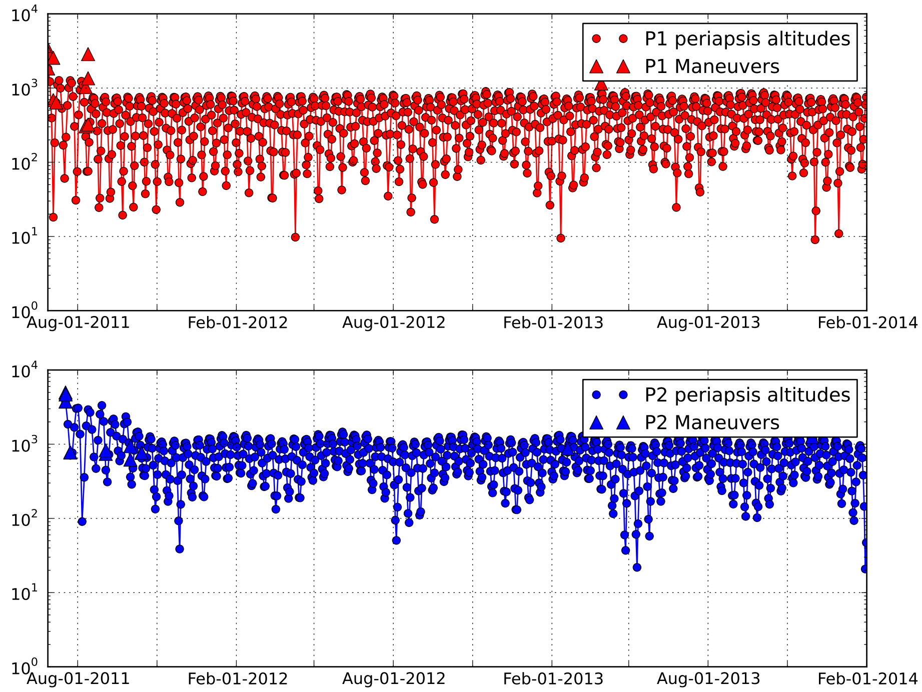

Fig. 17 Altitude at periapse for P1 and P2 in lunar orbit. The absence of secular drift simplifies the orbit design. Note the addition of a planetary-science enhancement burn (PEB) for P1 in March of 2013

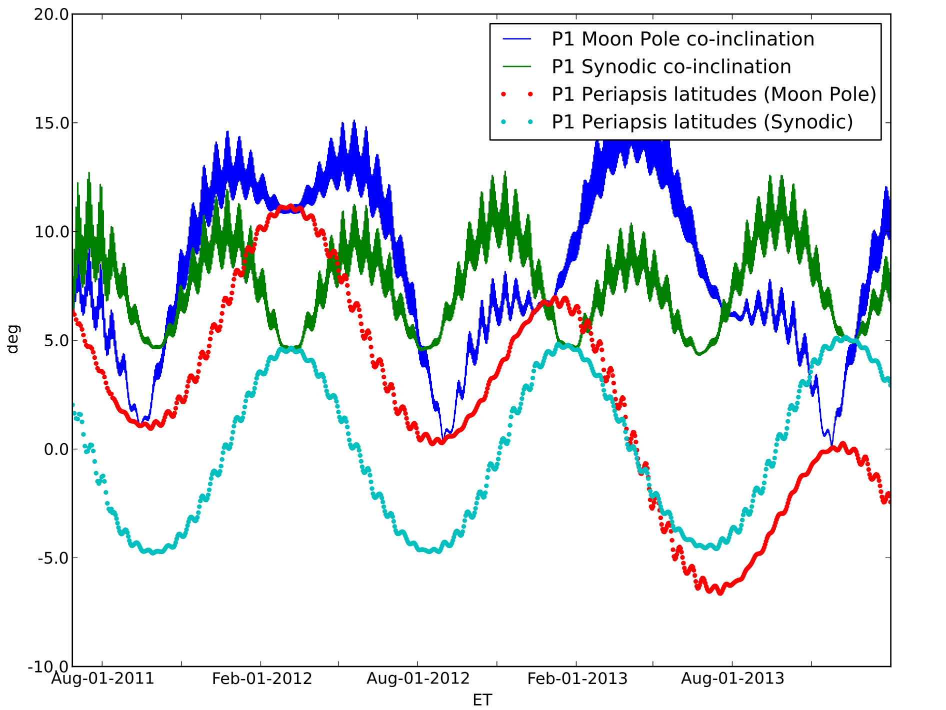

Fig. 18 Co-inclination (supplement of the inclination) and latitude of P1 at periapse in lunar orbit with LOI in June, 2011. Both quantities are measured with respect to both the lunar equator and the lunar orbit plane. The symmetry with respect to the orbit plane reflects the dominance of Earth perturbations in varying the lunar orbit elements.

damping the probes' Z axis motions (the Z axis is parallel to the pole of the Moon's orbit around the Earth). Damping this motion early was needed because the Lissajous orbits have an exponential growth of oscillation amplitudes of the Z components of the probe positions, and prolonged residence in these orbits results in large, off-equatorial oscillations and velocities at the time of LOI. The damping maneuvers were designed and successfully executed in January and February of this year. At this writing the designs of the LOI and PRM sequences of maneuvers are basically done, but the PRM durations are still being adjusted to minimize the total ΔV and optimize science return. Table 3 shows how the maneuver ΔVs added up for all of the phases, both as proposed and as experienced so far, with current estimates of future required ΔV.

7 Conclusions

The trajectory design of the ARTEMIS mission that began in July of 2009 has been presented here. The design sent two probes from Earth orbit to the Moon via a transfer that took $\sim$2 years and involved numerous lunar approaches and flybys, low-energy trajectory legs in the Earth-Sun system, and Lissajous orbits around the Earth-Moon Lagrange points on either side of the Moon, and will finally culminate with both probes in very eccentric low-lunar orbits. The constraints imposed on the design by the limitations of the THEMIS probes (which were designed for an Earth-orbiting mission)—including thruster orientation, available ΔV, maximum shadow capability, maximum distance for radio telecommunication, and thruster capabilities—necessitated an innovative design. Ultimately the design satisfied all mission constraints and offers a variety of scientific measurement opportunities that have the potential to enhance understanding of Earth-Moon-Sun interactions.

Given the challenges that the ARTEMIS mission presented and the complexity of the design needed to meet those challenges, it is notable that the cost of the mission design effort was many times less than one would estimate for a new, i.e., non-extended, full mission of comparable difficulty. One major difference is that ARTEMIS started in space with given orbits for the two probes, saving the significant cost of determining a launch period and optimal launch targets for the mission. But an even bigger factor in cost savings was acceptance of risk that is unacceptable for a more expensive mission. The THEMIS mission was already a success and completely justified the investment already made in building and launching the probes. Furthermore, the outermost two probes were forced to find a new mission because the THEMIS orbits they were in would have led to fatal shadows by now. So in a sense the only thing at risk was the cost of the ARTEMIS design itself, leading to a situation where the investment at risk was reduced by accepting a higher probability that the risk would be realized.

The primary cost-saving characteristic of the mission design process that put ARTEMIS at risk was the near absence of redundancy, both in the design process and in the products of that process. There is a certain amount of natural redundancy in the use of two probes, and indeed much of the opportunity for new science could be realized even in the absence of one. A significant opportunity would have been missed, though, without the dual measurements that have already been made by the two probes and that are planned for the remainder of the mission. On the ground, however, the design team was pared down so that at times it relied on a single person; had that person been unavailable, a different and uncertain approach for that part of the design would have been required. The limited team size also meant that the design itself was nearly "single string" in the absence of backup and contingency trajectories. The analysis that would have produced such alternative designs was most often replaced by engineering judgment that such alternatives existed and could be found if needed. Similarly, in the area of maneuver design, extensive Monte Carlo runs covering all the ways that reality could diverge from the nominal plan were replaced by experience-based estimates of when trajectory correction maneuvers might be needed and of how much ΔV capability might be needed to correct the trajectories as they were flown.

The greatest uncertainty in the design was perhaps in the area of trans-lunar trajectory corrections because these could contain only minimal ΔV components in the direction of the probe $-Z$ axis. In one of the rare instances of backup analysis, an alternative transfer that included deterministic "down" maneuvers at strategic points along the way was designed; these maneuvers could serve to enable upward corrections by reducing the size of the down maneuvers. But this alternative transfer was not used or needed, and the maneuver design team was able to design TCMs in flight that kept the probes on track to their Lissajous rendezvous. The enabling mitigation of the probe's thrust-direction constraints was that every phase of the mission, including the transfer phase, included multiple orbits of the Earth or Moon so that an up maneuver on one side of the orbit could be replaced by a down maneuver or in some cases a radial maneuver elsewhere in the orbit. Another critical factor of mission success so far has been the stellar performance of the two probes and the mission operations team: every one of the dozens and dozens of maneuvers has been executed as planned.

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Table 1 P1 Earth orbit phase maneuvers

ORM1A	2009/213	20:01:04.853	8.243
ORM1B	2009/213	20:50:51.171	8.441
ORM2A	2009/222	13:01:15.050	8.389
ORM2B	2009/222	13:51:33.867	8.468
ORM3A	2009/232	07:06:30.159	8.477
ORM3B	2009/232	07:56:36.098	8.517
ORM4A	2009/243	08:12:29.465	13.901
ORM4B	2009/243	09:14:21.187	11.855
ORM5A	2009/256	18:57:06.519	13.984
ORM5B	2009/256	19:48:19.402	5.505
FTM1A	2009/285	08:38:00.635	0.203
FTM1B	2009/285	08:41:51.037	0.602
FTM2	2009/336	08:02:21.215	6.084
TCM1	2009/348	04:51:56.825	1.886
TCM2	2010/015	12:27:38.304	1.455
TCM3	2010/024	07:00:59.591	0.311

Table 2 P2 Earth orbit phase maneuvers.

ORM1	2009/202	07:33:03.552	10.686
ORM2	2009/206	10:41:40.407	5.292
ORM3A	2009/210	15:10:44.501	2.399
ORM3B	2009/210	16:10:50.116	8.348
ORM4A	2009/215	00:46:49.019	3.324
ORM4B	2009/215	01:47:36.747	8.695
ORM5A	2009/219	15:24:58.115	3.870
ORM5B	2009/219	16:22:13.277	7.915
ORM6A	2009/224	11:22:18.944	3.903
ORM6B	2009/224	12:16:13.013	6.952
ORM7A	2009/229	12:35:08.258	3.867
ORM7B	2009/229	13:26:36.035	6.257
ORM8A	2009/234	19:27:11.635	4.267
ORM8B	2009/234	20:18:08.362	6.058
ORM9A	2009/240	08:07:08.656	3.871
ORM9B	2009/240	08:53:50.173	4.753
ORM10A	2009/246	02:16:11.659	4.729
ORM10B	2009/246	03:01:38.535	4.387
ORM11A	2009/252	02:39:19.201	5.000
ORM11B	2009/252	03:21:47.856	3.419
ORM12A	2009/258	09:02:17.385	5.586
ORM12B	2009/258	09:43:04.698	2.831
ORM13A	2009/264	22:31:20.114	5.882
ORM13B	2009/264	23:11:39.983	2.507
ORM14A	2009/271	18:34:32.874	7.092
ORM14B	2009/271	19:14:07.763	2.240
ORM15	2009/278	23:21:22.389	5.881
ORM16	2009/286	09:49:58.506	8.599
ORM17	2009/294	06:15:08.345	9.851
ORM18	2009/302	13:41:36.289	10.269
ORM19	2009/311	10:47:56.571	10.113
ORM20	2009/320	22:41:23.999	10.039
ORM21	2009/331	01:50:58.997	4.148
ORM22	2009/341	12:28:02.607	2.083
ORM23	2009/352	07:22:49.060	4.782
ORM24	2009/363	11:22:37.941	6.233
ORM25	2010/010	05:05:57.689	7.580
ORM26	2010/022	19:08:17.058	5.846
ORM27	2010/057	08:52:20.815	11.875
SDM1	2010/059	08:17:18.815	3.636
SDM2	2010/074	09:55:42.965	7.360
FTM1	2010/083	16:07:17.000	12.406
TCM1	2010/085	02:05:41.282	0.648

Table 3 ARTEMIS ΔV budget as proposed and actual (with italic values showing current estimates of ΔV to come, as of May 2011).

	P1 cost est. (m/s)	P1 cost act. (m/s)	P2 cost est. (m/s)	P2 cost act. (m/s)
ORMs	96.7	96.1	204.0	240.9
FTMs	7.0	6.9	5.7	12.4
DSMs	4.8	7.2	15.1	30.4
LTI	1.5	*2*	0.8	*2*
LOIs	89.9	*99*	117.1	*127*
Lunar orbit periapse lowering		*8*		*12*
Deterministic DV total	200	*219*	343	*425*
Sources of additional DV cost:				
TLI declination penalty	(included)	(included)	(included)	(included)
TLI grav and steering loss (w/ shadow)	(included)	(included)	36	(included)
LOI declination penalty	2	(included)	2	(included)
LOI grav and steering loss	(included)	(included)	(included)	(included)
Lissajous maintenance	15	*12*	12	*3*
TCMs (3% + 1 m/s per ORM x sqrt(n))	15	7.0	14	4.1
Total	232	*238*	407	*432*
Available DV	324	*320*	475	*467*
Margin	92	*82*	68	*35*
Liens against margin:				
Matching ORM phase to transfer phase	None	0	5	0
Precession correction in ORM phase	1	0	2	0
Lissajous maintenance increase	20	*0*	13	*0*
End-of-mission deorbit	10	*2*	64	*2*

Table 4 Integrated preliminary trajectory design timeline.

Earth-Orbit Phase	Oct 17, 2009	P2 Orbit-Raise Maneuver
Earth-Orbit Phase	Oct 20, 2009	P1 Orbit-Raise Maneuver
Earth-Orbit Phase	Jan 15, 2010	P1 Fly-by Targetting Maneuver
Earth-Orbit Phase / Trans-Lunar Phase	Jan 31, 2010	P1 Lunar Fly-by #1 (min Range = 3200 km)
Trans-Lunar Phase	Feb 13, 2010	P1 Lunar Fly-by #2 (min Range = 4500 km)
Earth-Orbit Phase	Mar 11, 2010	P2 Fly-by Targetting Maneuver
Trans-Lunar Phase	Mar 15, 2010	P1 Deep-space Maneuver (+ Local Maximum Range = 1200000 km to Earth)
Earth-Orbit Phase / Trans-Lunar Phase	Mar 28, 2010	P2 Lunar Fly-by (min Range = 21000 km)
Trans-Lunar Phase	Apr 13, 2010	P1 Earth Fly-by (min Range = 17000 km)
Trans-Lunar Phase	Apr 19, 2010	P2 Deep-space Maneuver (+Local Maximum Range = 940000 km to Earth)
Trans-Lunar Phase	May 11, 2010	P2 Earth Fly-by #1 (min Range = 86000 km)
Trans-Lunar Phase	Jun 06, 2010	P1 Maximum Range (1500000 km to Earth)
Trans-Lunar Phase	Jun 18, 2010	P2 Maximum Range (1200000 km to Earth)
Trans-Lunar Phase	Jul 27, 2010	P2 Earth Fly-by #2 (min Range = 170000 km)
Trans-Lunar Phase / Lissajous Orbit Phase	Aug 23, 2010	P1 LL2 Insertion
Trans-Lunar Phase	Aug 23, 2010	P2 Local Maximum Range (1100000 km to Earth)
Trans-Lunar Phase / Lissajous Orbit Phase	Oct 22, 2010	P2 LL1 Insertion
Lissajous Orbit Phase	Jan 01, 2011	P1 Departs LL2
Lissajous Orbit Phase	Jan 08, 2011	P1 LL1 Insertion
Lissajous Orbit Phase	Jun 14, 2011	P1 Lunar Transfer Initiation
Lunar Orbit Phase	Jun 27, 2011	P1 LOI (1850 km alt)
Lissajous Orbit Phase	Jun 28, 2011	P2 Lunar Transfer Initiation
Lunar Orbit Phase	Jul 17, 2011	P2 LOI (3800 km alt)
LADEE Science Phase	Jul 7, 2012	Beginning, for earliest LADEE launch
LADEE Science Phase	Oct 15, 2012	End, for earliest LADEE launch
LADEE Science Phase	Dec 16, 2012	Beginning, for latest LADEE launch
Lunar Orbit Phase	Dec 28, 2012	P1 End of 1.5 year Lunar Orbit Phase
Lunar Orbit Phase	Jan 17, 2013	P2 End of 1.5 year Lunar Orbit Phase
LADEE Science Phase	Mar 26, 2013	End, for latest LADEE launch

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