AAS 11 - 510

ARTEMIS Mission Overview: From Concept to Operations

David Folta [1] & Theodore Sweetser [2]

ABSTRACT

ARTEMIS (Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun) repurposed two spacecraft to extend their useful science (Angelopoulos, 2010) by moving them via lunar gravity assists from elliptical Earth orbits to L_1 and L_2 Earth-Moon libration orbits and then to lunar orbits by exploiting the Earth-Moon-Sun dynamical environment. This paper describes the complete design from conceptual plans using weak stability transfer options and lunar gravity assist to the implementation and operational support of the Earth-Moon libration and lunar orbits. The two spacecraft of the ARTEMIS mission will have just entered lunar orbit at this paper's presentation.

INTRODUCTION

The Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun (ARTEMIS) mission repurposed two in-orbit NASA spacecraft to extend their useful science investigations.[1,2,3] ARTEMIS uses simultaneous measurements of particles and electric and magnetic fields from two different trajectories to provide three-dimensional perspectives of how energetic particle acceleration occurs near the Moon's orbit, in the distant magnetosphere, and in the solar wind. The two spacecraft denoted P1 and P2, are from NASA's Heliophysics Time History of Events and Macroscale Interactions during Substorms (THEMIS) constellation of five satellites shown in Figure 1 that were launched in 2007 and successfully completed their mission.[4] The ARTEMIS mission moved two spacecraft in the outer-most elliptical Earth orbits and, with lunar gravity assists, re-directing the spacecraft to both the Earth-Moon L_1 (EM L_1) and L_2 (EM L_2) libration point orbits via transfer trajectories that exploit the multi-body dynamical environment.[5,6] After the Earth-Moon libration point orbits were achieved and maintained for several months, both spacecraft were inserted into elliptical lunar orbits. The current baseline is a multi-year mission with departure maneuvers that began in June 2009, targeted multiple lunar flybys in February and March of 2010 that eventually place the spacecraft on the transfer trajectory.

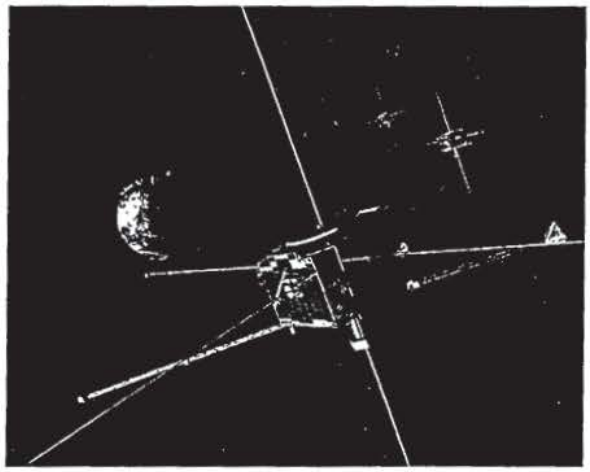

Figure 1. The ARTEMIS Spacecraft

(1) NASA/Goddard Space flight Center, Code 595, Greenbelt, MD, 20771, david.c.folta@nasa.gov
(2) Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA 91109, Theodore.h.sweetser@jpl.nasa.gov

The P1 spacecraft entered Earth-Moon Lissajous orbit August 25th 2010 and P2 followed on October 22nd 2010. The trajectory design encompassed the entire near Earth environment; Cis-lunar, Sun-Earth, Earth-Moon libration, and lunar orbits. Both spacecraft have been successfully inserted into stable elliptical lunar orbits, one posigrade and the other retrograde were they will remain for the foreseeable future.

The ARTEMIS concept avoided extended shadow durations while in their Earth elliptical orbits that would have lead to the demise of these spacecraft due to re-entry requirements. The concept moved these spacecraft to a region near the Moon in order to improve the baseline for in-situ measurements of the Earth environment. Unfortunately, the limited remaining fuel did not support a direct transfer from the Earth elliptical orbits to lunar orbits. To overcome this problem a unique design was fashioned that incorporated apogee raising, lunar gravity assists, the favorable use of perturbations in Sun-Earth weak stability regions to raise perigee, and the utilization of the Earth-Moon libration orbits to reach the intended lunar orbits.

This paper details the complete design process from the conceptual plan using multi-body transfer options and the eventual use of lunar gravity assist to the implementation and operational support of the design. We discuss the impacts and limitations of the design with respect to the spacecraft constraints in terms of restricted Delta-velocity (Δv) directions and limitations of the propulsion system. We also present information on propellant usage and the sensitivity of controlling these unique orbits. We discuss the factors that contributed to the project's resounding success despite the high risks of the proposed implementation.

The ARTEMIS mission is a collaborative effort between NASA's Goddard Space Flight Center's (GSFC) Navigation and Mission Design Branch (NMDB), the Jet Propulsion Laboratory (JPL), and the University of California at Berkeley (UCB) Space Sciences Laboratory (SSL). JPL provided the initial concept and the reference transfer trajectory from the elliptical orbit phase through libration orbit insertion and the details of the lunar orbit phase. GSFC's NMDB provided the operational trajectory design to complete the mission and the maneuver and navigation support from pre-lunar gravity assists and Sun-Earth transfers to Earth-Moon libration maintenance and lunar orbit insertion. The UCB SSL Mission Operations Center (MOC) provides spacecraft operations support (command, telemetry, maneuver planning, and daily monitoring and maintenance) of all spacecraft. Tracking, telemetry, and command services are provided using the S-band frequency via various networks, including the Berkeley Ground Station (BGS), the Universal Space Network (USN), the NASA Ground Network (GN) and Deep Space Network (DSN). The UCB SSL uses a GSFC software package for maneuver planning and navigation support via the THEMIS GSFC-UCB mission support partnership.

Mission Concept and Design

The ARTEMIS mission was approved in May 2008 by NASA's Heliophysics Senior Review panel as an extension to the THEMIS mission.[3] The proposal encompassed a baseline design which showed that two of the THEMIS spacecraft could be placed into lunar orbit with the remaining fuel onboard. While the amount of fuel required to reach the lunar orbit was available, it was insufficient to insert into a lunar orbit, so an alternate trajectory design process began. This alternate design was first investigated at JPL using software tools that could model the Earth elliptical orbits, the multi-body environment for the transfer and Earth-Moon libration orbits, and the lunar orbits.

The complete approach as shown in Figure 2 encompassed;

- Elliptical Earth Orbits
  - Raise apogee of P1 and P2 orbits
  - Utilize multiple lunar approaches to increase the spacecraft energy
- Trans-lunar trajectory
  - Use lunar fly-by(s) to send spacecraft towards the Earth-Sun Lagrange points
  - Free insertion into Earth-Moon Lissajous orbit
- Lissajous orbit phase
  - Place P1 and P2 into complementary Lissajous orbits and transfer to lunar orbit
- Lunar orbit phase
  - Insert P1 and P2 into long-term stable, complementary Lunar orbits
- Gravity Models and Assumptions (from start to lunar orbit insertion)
- Earth J2 and 8X8 models and point mass bodies: Sun and Moon
- Deterministic, impulsive maneuvers as needed to attain target goals
- 20x20 Lunar gravity and finite burns for lunar orbits

ARTEMIS takes advantage of 4-body dynamics to minimize Δv. The design raised the elliptical orbit apogee just enough for a lunar encounter. The lunar flyby provided the change in energy and direction to attain a transfer beyond lunar distance (but still loosely captured by the Earth-Moon system). For P1, one lunar flyby was to setup another flyby 180 degrees away across the lunar orbit. Once in the Sun-Earth environment, solar gravity perturbations raised perigee to lunar orbit distance for the flyby. After the timing is achieved with respect to the lunar position, the design approached the Moon along the Earth-Moon line, so that Earth gravity perturbations reduce lunar-relative energy. The Earth-Moon dynamics allow ARTEMIS to enter, exit, and cross between Lissajous orbits around EM L_1 and EM L_2 with no or very small maneuvers. After lunar capture, subsequent periselene maneuvers reduce aposelene to the desired altitude for the science.

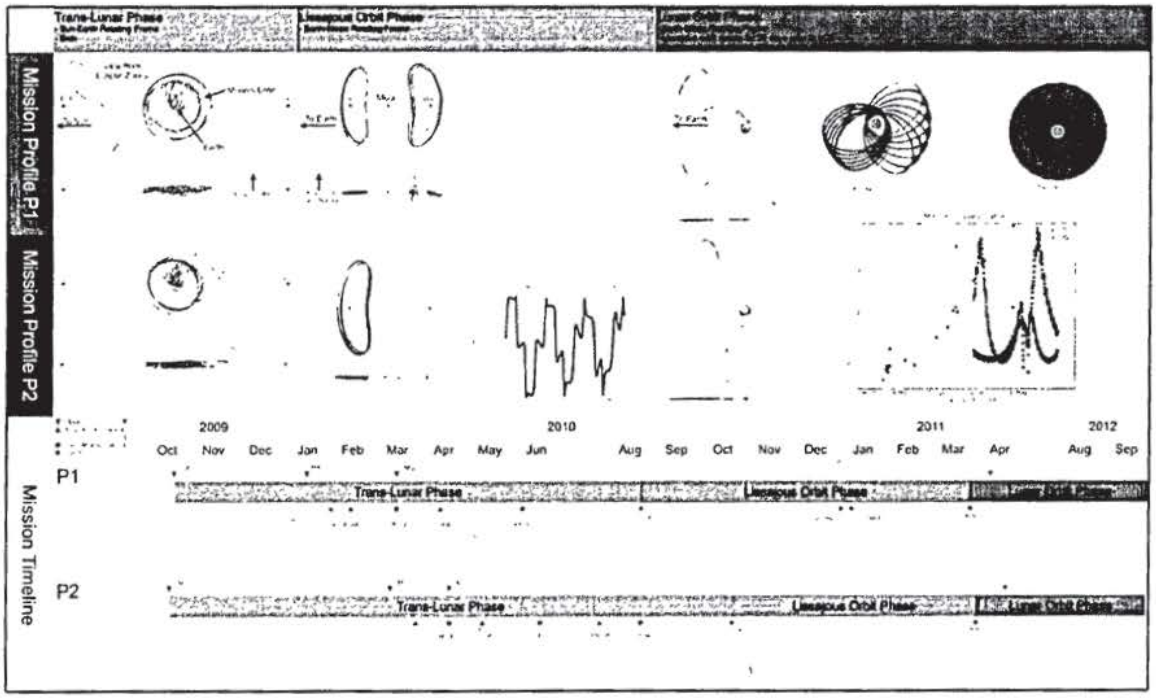

Figure 2. ARTEMIS Concept Design

Software Tools

The software used to design this original concept was a combination of JPL's LTool and Mystic.[7] LTool's trajectory design objects and differential corrector were used to find a trajectory candidate that comprised the multi-body environment, assumed impulsive maneuvers, but ignored spacecraft thruster constraints. LTool applied successively tighter Δv constraints to reduce total Δv. In this design process one can check eclipse times. Once this placeholder design was generated impulsive orbit raise maneuver (ORM) are replaced with series of finite burns. We also had to estimate the Δv cost of meeting thruster constraints and to add trajectory correction maneuvers (TCMs) to find a Δv-99 solution, which represents a statistical maximum Δv requirement to fly the mission.. In the original concept, biased maneuvers in the –Z Sun-Earth rotating System direction were added at TCM locations to allow over-performance of some maneuvers. To meet required updates and higher fidelity models, analysis was switched to the Mystic software. Mystic minimized the total Δv with thruster constraints and bias maneuvers as necessary. Once the mission was given permission to proceed, GSFC became involved with the verification, mission design, and maneuver planning. At that time, a models and constants document was written to coordinate design efforts. GSFC and JPL software included full ephemeris models (DE421 file) along with third body perturbations including a solar radiation pressure acceleration based on the spacecraft mass and constant cross-sectional area (e.g. cannon ball model). A potential model for the Earth with degree and order eight was used. The operational plans are based on a variable step Runge-Kutta 8/9 or PrinceDormand 8/9 integrator. The libration point locations are also calculated instantaneously at the same integration interval. To compute maneuver requirements in terms of Δv, different strategies involve various numerical methods: traditional Differential Correction (DC) targeting with central or forward differencing, optimization using the Mystic Optimizer and the VF13AD optimizer from the Harwell library, and the Analytical Graphics Inc (AGI) /Satellite toolkit (STK) SQP optimizer. For the corrections scheme, equality constraints are incorporated, while for the optimization scheme, nonlinear equality and inequality constraints are employed. The software employed to met spacecraft constraints and orbit goals for operational trajectory design and maneuver planning included GSFC's General Mission Analysis Tool (GMAT) (open source s/w), Analytical Graphic Inc's (AGI) STK/Astrogator, and GSFC's Goddard Maneuver program (GMAN).[8] . GMAN is a high fidelity propulsion modeling software that incorporates moments of inertia, thruster placement, and models spacecraft kinematics and dynamics of the spinning ARTEMIS spacecraft. GMAN has been used successfully over 30 years to model spinning spacecraft kinematics and is being used for THEMIS support.

Navigation

Since the navigation solution is provided both by the UCB team and the GSFC Code 595 Flight Dynamics Facility, we were able to plan maneuvers with confidence.[9,10] The observed navigation uncertainty was significantly smaller than the values used in the pre-flight assessment. The tracking of P1 and P2 was accomplished using the Deep Space Network (DSN), Universal Space Network (USN), and the antenna at UCB. More information on navigation can be found in reference 9.

The Goddard Trajectory Determination System (GTDS) was used for all operational navigation solutions. The GTDS least squares solution uncertainty, found from the comparison of overlap regions of the navigation solution, was estimated to be below 100 meters and 0.1 cm/s in all phases of the mission. Originally, 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. We believe that the observed uncertainty value were optimistic and that it actually ranges in the 100s of meters and tens of cm/s. Additionally, it was difficult to ascertain the correct navigation error near maneuvers since both maneuver performance and attitude uncertainty was at the limit of observability.

Throughout the transfer trajectory implementation process, navigation solutions were generated at a regular frequency of once every three days, while in the orbit raise and Lissjaous orbits daily solutions were generated. Post-maneuver navigation solutions 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. These accuracies were obtained using nominal tracking arcs with alternating north and south stations.

ARTEMIS SPACECRAFT OVERVIEW

Each ARTEMIS spacecraft is spin-stabilized with a nominal spin rate of roughly 20 RPM. Spacecraft attitude and rate are determined using telemetry from a Sun sensor (SS), a three-axis magnetometer (TAM) used near Earth perigee, and two single-axis inertial rate units (IRUs). The propulsion system on each spacecraft is a simple monopropellant hydrazine blow-down system. The propellant is stored in two equally-sized tanks and either tank can supply propellant to any of the thrusters through a series of latch valves. Each observatory was launched with a dry mass of 77 kg and 49 kg of propellant, supplying a wet mass of 126 kg at beginning of life.

Each spacecraft has four 4.4 Newton (N) thrusters – two axial thrusters and two tangential thrusters. The two tangential thrusters are mounted on one side of the spacecraft and the two axial thrusters are mounted on the lower deck, as seen in Figure 3. The thrusters fire singly or in pairs – in continuous or pulsed mode – to provide orbit, attitude, and spin rate control. Orbit maneuvers were implemented by firing the axial thrusters in continuous mode, the tangential thrusters in pulsed mode, or a combination of the two (beta mode). Since there are no thrusters on the upper deck, the combined thrust vector is constrained to the lower hemisphere of the spacecraft.

ARTEMIS Spacecraft Maneuvers Constraints

The ARTEMIS spacecraft are pointed within five degrees of the south ecliptic pole. These spacecraft can implement a Δv (thrust direction) along the spin axis towards the south ecliptic pole direction or in the spin plane, but cannot produce a Δv in the northern hemisphere relative to the ecliptic. While the axial thrusters were used when necessary, these thrusters are not calibrated as well as the tangential (radial) thrusters. The pointing constraint limited the location of maneuvers so most maneuvers were performed in a radial direction. For the lunar gravity assist and the multi-body dynamical environment, the trajectory was optimized using a nonlinear constraint that placed the Δv in the spin plane. The maneuver epoch was also varied to yield an optimal radial maneuver magnitude.

Figure 3. ARTEMIS Spacecraft

In addition to the direction of maneuvers, another 'error' source also resulted in some interesting maneuver planning. This is the fact that, as a spinning spacecraft, a maneuver will be quantized into ~ 0.7 cm/s (1/2 thrust arc) intervals with a start time that is dependent upon the Sun pulse in each spin. This meant that there was a finite maneuver accuracy that could be achieved that was dependent upon the Δv magnitude for each maneuver. Maneuvers were quantized by varying the maneuver epoch, but DSN coverage often led to this method not being easily enacted. Thus many early maneuvers were executed with the associated errors from spin pulse and timing. Later in the mission, the UCB operations team updated the onboard software to permit a variable spin-pulse to more accurately match the required Δv, reducing the uncertainty in the Δv per pulse to less than 1%.

TEAM DISCIPLINES

As mentioned in the background, ARTEMIS is a team effort and the process to plan and execute maneuvers demonstrates how that team process worked. Upon receipt of the daily orbit determination solution, a maneuver was computed for possible maneuver locations to meet the tracking and command load schedule. Even if an optimal Δv was found that minimized fuel use, the epoch of the maneuver needed to be contained within a schedule tracking pass for the upload and verification in real time of the maneuver execution. This meant that there were epochs and maneuver locations that did not meet true optimal placement, but rather the minimal Δv for the station contact. Maneuver plans were then generated as part of the optimization and the transmitted to UCB SSL for further processing within the GMAN program to target these optimal ΔVs. GMAN output was then sent to GSFC for verification of the maneuver plan and for an initial estimate of the next maneuver, since navigation and performance errors would result in the orbit eventually escaping. In a weekly setting, full team meetings were held for a complete presentation and discussion of trajectory design analysis. These analyses included; maneuver Δv estimates and contingency plans, Monte Carlo analysis, optimization results, navigation accuracies, spacecraft status, and tracking and contact schedules. The team not only discussed the analysis, but also cooperated in serving as reviewers of the plans as they related to the original concept design and support operations.

MISSION PHASES

With the conceptual design in place the mission support in place, the team of UCB-SSL, JPL, and GSFC began the endeavor to make this mission a reality. Beginning in July of 2009, the design was restructured to begin the inclusion of the spacecraft constraints and to include high fidelity operational software that could model all the multi-body regions that ARTEMIS would survey and transit.

Orbit Raise Maneuvers

Orbit Raise Maneuvers (ORMs) were required for each of the spacecraft.[11] The P1 spacecraft was initially in a higher apogee orbit of 120,000 km than the P2 spacecraft at 80,000 km. With a low thrusting capability, multiple apoapsis raising maneuvers were executed to achieve the lunar flyby conditions necessary to place the spacecraft on their respective transfer trajectories. The number of maneuvers in the ORM sequence was a function of the current apoapsis, the propulsion system capability, station coverage, eclipse avoidance, and overall efficiency with long duration maneuvers near periapsis. Several ORMs were executed as two maneuvers separated by the Earth shadow cone. All these orbital constraints along with the operational constraints of the ARTEMIS spacecraft resulted in the need to begin the orbit raise sequence in the appropriate time to permit the final apoapsis distance and timing to arrive at the Moon for the flyby. There were several intermediate gravity assists with distances on the order of 50,000 km to just over 11,000 km, during the ORM sequence. The perturbations due to these encounters were accounted in the overall design. The P2 spacecraft had more fuel onboard since the THEMIS design moved the most efficient spacecraft, P1, to the higher apoapsis, thus P1 had less fuel.

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 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. 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.

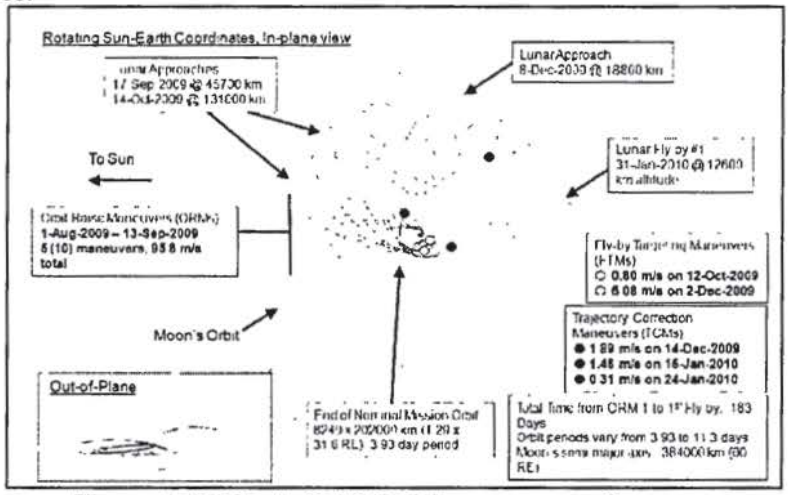

Figure 4. P1 Earth Orbit Raising maneuver Sequence

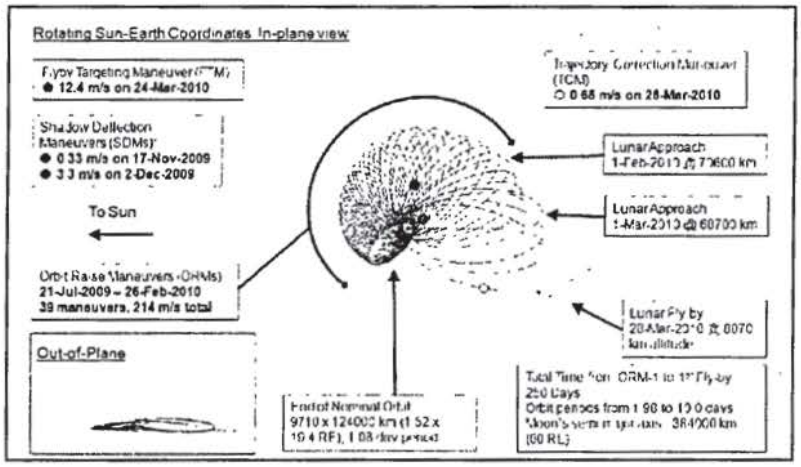

Figure 5. P1 Earth Orbit Raising maneuver Sequence

Lunar Gravity Assist

As part of the final transfer trajectory design, lunar gravity assist were required. These flybys targeted an encounter with the lunar B-Plane. The B-Plane parameters are shown in Table 2.

Table 1. B-plane Target Parameters

	B-Magnitude	B-Angle	Epoch
P1 – First Flyby	**21659.211**	**-96.44**	**January 31, 2010 08:10:13**
P1 –Second Flyby	**11931.05**	**-11.73**	**February, 15 2011 09:54:34**
P2 – Single Flyby	**16933.33**	**-6.23**	**March, 28 2010 07:58:11**

** B-plane reference Vector is lunar orbit normal

Once the ORM sequence was completed, several revolutions of the Earth elliptical orbit were made available for correction maneuvers and for Flyby Targeting maneuvers (FTMs) before the lunar gravity assist. These maneuvers were optimized as a total maneuver sequence to minimize the overall targeting Δv.

Transfer Trajectory

Following the first close lunar gravity assist, the P1 spacecraft flies under the Earth and performs a second gravity assist roughly 13 days later, as seen in the Sun-Earth rotating frame in Figure 6. A Deep Space Maneuver (DSM1) was performed 33 days later. DSM1 targets through Earth peraispsis and to the Earth-Moon libration insertion state. Following the Earth periapsis, the P1 spacecraft once again transfers into the general vicinity of the Sun-Earth L_1 Lagrangian point. This region is also identified as a "weak stability boundary" region. At the final bend in the P1 trajectory, the spacecraft is at a maximum range of 1.50 million km from the Earth. At this point, the trajectory begins to fall back towards the Earth-Moon system on an unstable Lissajous manifold. A second deep space maneuver (DSM2) originally modeled was not required to target the Earth-Moon L_2 Lagrangian point. A small Lissajous insertion orbit maneuver was performed to insert P1 into the proper L_2 Lissajous orbit. The P2 translunar trajectory uses a single lunar swingby and three deep space maneuvers, two Earth periapses, and the Lissajous orbit insertion maneuver.[12,13]

For both P1 and P2, we allocated 4% of the total propellant budget to perform any required trajectory correction maneuvers (TCMs) along the way. The trajectory design focused on achieving the Earth-Moon libration insertion conditions to permit the final stage of the ARETMIS mission, which includes a Lissajous orbit with a transfer to a high eccentric lunar orbit. The flyby targets were required to enable the energy to place the ARTEMIS spacecraft near the appropriate outgoing manifolds. Since the two spacecraft were originally designed for a different mission, a highly elliptical Earth orbit, and were already flying, fuel was (and is) extremely limited. Thus, with the unique operational constraints, accomplishment of the transfer goals with the minimum cost in terms of fuel is the highest priority. The total Δv for this phase of the mission was ~ 11 m/s for P1 and ~33 m/s for P2.

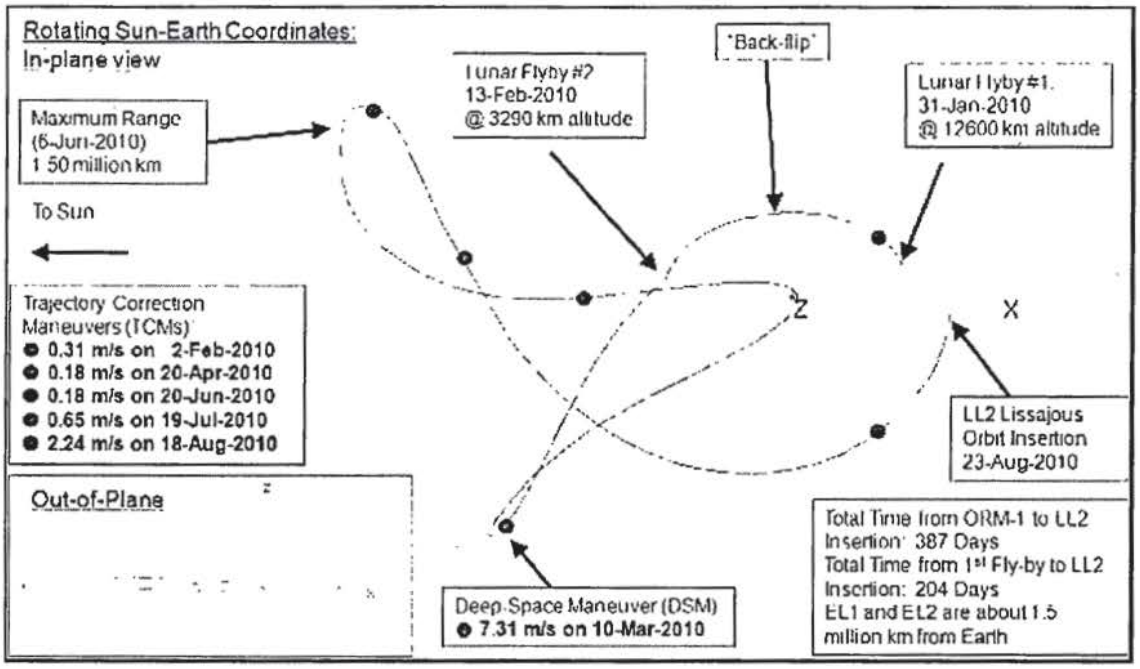

Figure 6. P1 Transfer Trajectory

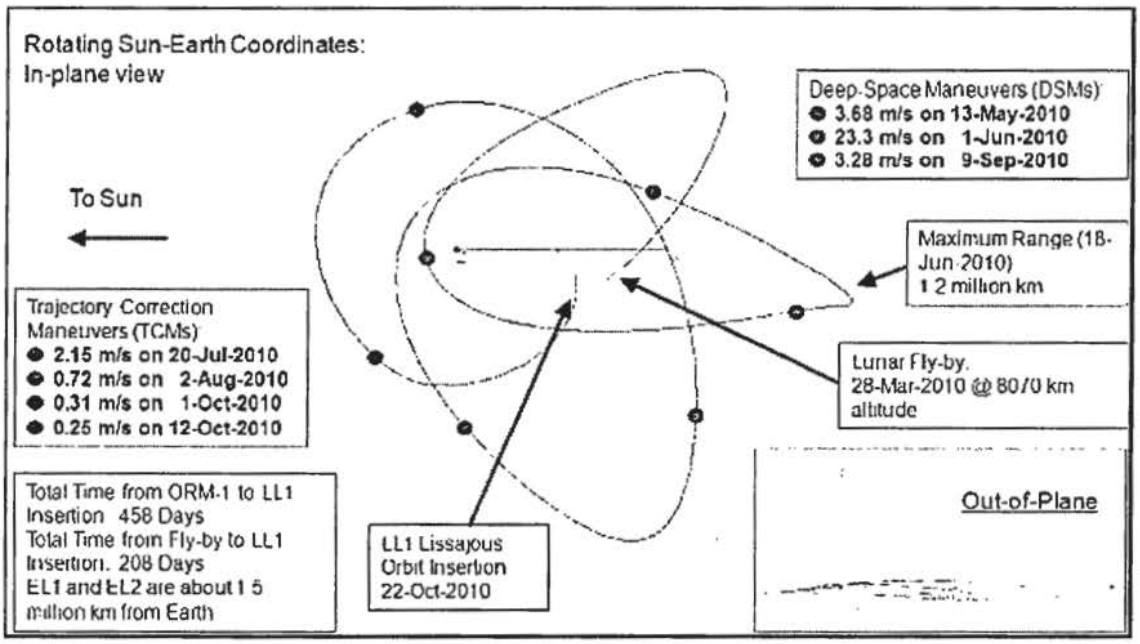

Figure 7. P2 Transfer Trajectory

Earth-Moon Libration Orbits

It was already known that any change in energy from an unstable Earth-Moon libration point orbit would result in an orbit departure, either towards the Moon or in an escape direction towards the Earth or the Sun-Earth regions. The Δv required to effect these changes are very small as are the small accelerations from solar radiation pressure, since natural perturbations will also result in these escape trajectories. To continue the orbit downstream and maintain the path in the vicinity of the libration point, this information can be exploited to selectively choose the goals that must be achieved to continue the orbit from one side of the libration orbit to the other.[14,15,16] For the method applied directly to ARTEMIS the goals are directly related to the energy (velocity) at the *x*-axis crossing to simply wrap the orbit in the proper direction, always inward and towards the libration point.

The targets used for the ARTEMIS optimal continuation method differed slightly between the EM L_2 orbit and the EM L_1 orbit.[15] The continuation maintains P1 while in orbit about EM L_2 and used two different x-axis velocities targets, depending on which side of the orbit P1 was on. For example, targets on the far side (away from the Moon) used an x-axis crossing velocity of -20m/s with a tolerance of 1 cm/s. Targets on the close side (nearer to the Moon) used x-axis crossing velocity targets of +10 m/s with a tolerance of 1 cm/s. Once in orbit about the EM L_1 orbit the P1 targets were changed to meet the ongoing operations similar to P2. These targets are +/- 10 cm/s at each crossing, a much smaller velocity target. The scheme here is to continuously target the next crossing downstream. Up to four crossings were used as the decrease in the stationkeeping Δv after meeting the third crossing target was usually below 0.1 cm/s and therefore unachievable by the spacecraft propulsion system. Depending on the location of the maneuver wrt the lunar radius, the Δv also varied from maneuver to maneuver.

Note that the P1 spacecraft was required to perform a transfer from EM L_2 to EM L_1. This transfer occurred once the z-amplitude of the L_2 orbit became 'planar' and therefore would result in a more planar L_1 Lissajous orbit as well. This planar design was needed to meet the goal of a low inclination lunar orbit once it was transferred to lunar orbit at the end of the Lissajous phase.

Figures 8 and 9 show the P1 and P2 Earth-Moon Lissajous orbits in a rotating system as viewed from above and in the Earth-Moon plane. The stationkeeping Δv required for this 11-month phase of the mission was only 3.99 m/s for P1 and 3.24 m/s for P2, well below pre-mission estimates between 25 to 50 m/s.

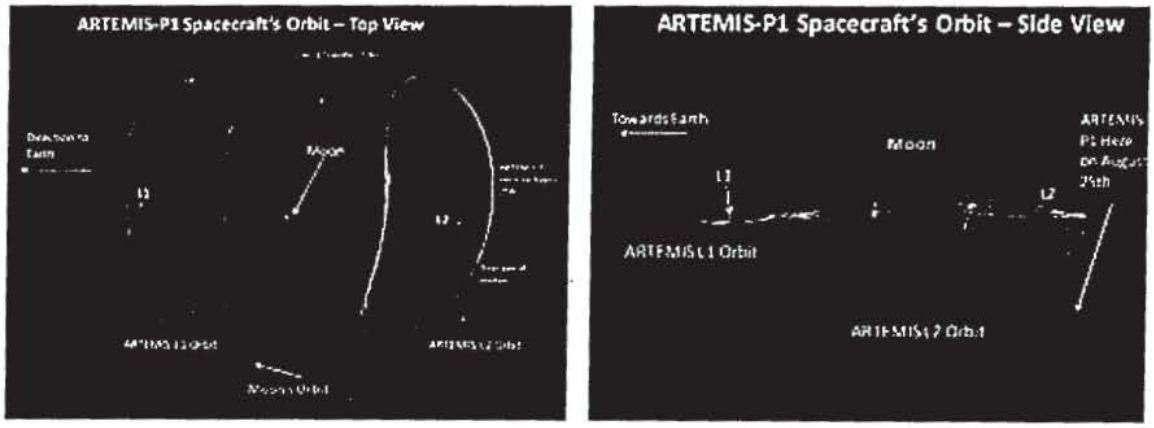

Figure 8. ARTEMIS P1 Lissajous Orbit Viewed from +Z and Y-Z in rotating coordinates

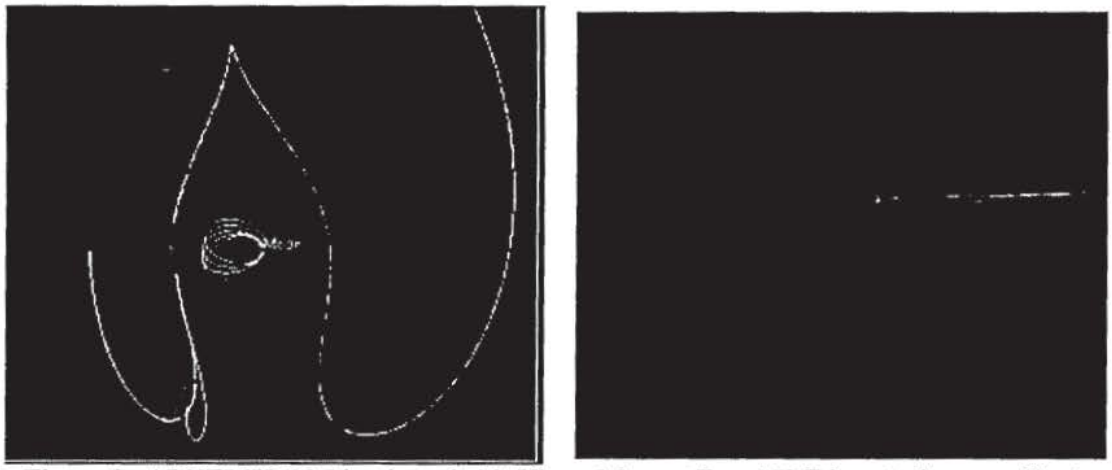

Figure 9. ARTEMIS P2 Lissajous Orbit Viewed from +Z and X-Z in rotating coordinates

Lunar Orbit Insertion

The final phase of the mission began with the transfer of both spacecraft from their respective Lissajous orbits into lunar orbits, P2 posigrade, P1 retrograde in true anomaly motion.[17] These transfers can be seen in figure 8 and 9, but are highlighted in Figures 10 and 11 for resolution. To achieve these transfers, small maneuvers placed the spacecraft onto unstable Lissajous manifolds that naturally flow towards the Moon. With the z-amplitude and its evolution fixed from previous stationkeeping maneuvers, which included an axial maneuver to jump Lissajous orbits to extend the mission from and end date of April to June and July.

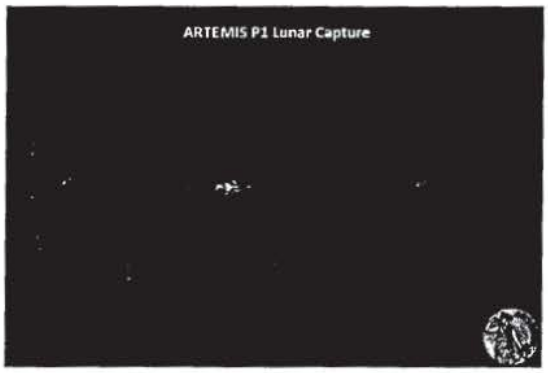

Figure 10. P1 Transfer Orbit

Figure 11. P2 Transfer Orbit

For P1, we designed and executed a 2.43 m/s maneuver on June 18th and a 0.50 m/s on June 22nd 2011 to align the trajectory to achieve the lunar insertion periapsis target of 1850 km altitude on June 27th 2011 at 15:15:00 UTC with a tolerance below 10 m and 5 seconds. For the P2 spacecraft, we designed and executed a 0.35 m/s maneuver on June 21st and a 0.63 m/s maneuver on June 28th and a small trajectory correction maneuver in early July of 8 cm/s to align the P2 trajectory for its lunar insertion periapsis target of 3800 km altitude on July 17th 2011at 22:38:00 UTC.

In addition to the nominal planning for the transfers, the P1 design incorporated a contingency plan that would return P1 into the EM L_1 orbit if the Lunar Orbit Insertion maneuver did not occur. A similar design for P2 did not apply as the retrograde orbit which permits such a design, is not available with P2 posi-grade. To affect this contingency plan, the nominal target epoch was changed from approximately 19hrs on June 27th to approximately 15hrs. This change in time increased the departure Δv from a minimal Δv that flows onto the natural manifold. The P1 contingency trajectory is shown in Figure 12 and displays not only the return to the Lissajous orbit, but a natural progression back to another lunar encounter.

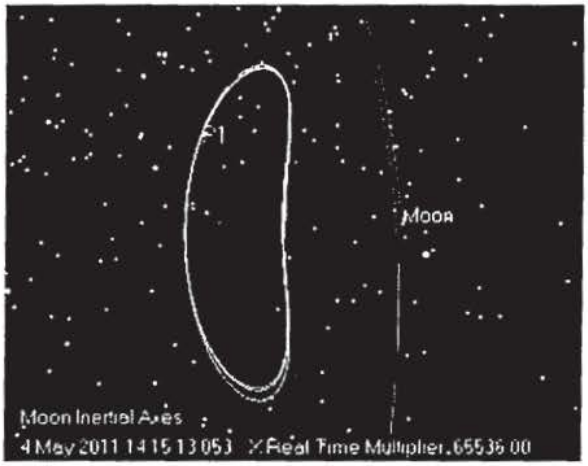

Figure 12. P1 Contingency Trajectory

Final Lunar Orbits

On June 27th and July 17th, respectively, the P1 spacecraft executed a ~2.5 hr long-duration Lunar Orbit Insertion (LOI) maneuver of 50.24 m/s and the P2 spacecraft executed a ~3 hr LOI of 71.85 m/s.[18] These maneuvers completed the capture of each spacecraft into its lunar orbit. P1 is in a retrograde lunar orbit and P2 is in a posigrade lunar orbit. In the weeks following, a series of four maneuvers decreased both periapsis and apoapsis altitudes. The final orbit provides the science team with multiple low-altitude lunar surface observations over the next several years to allow characterization of several known lunar crustal magnetic anomalies. ARTEMIS will also provide collaborative science observation with LADEE during its science campaign in 2013. Table 2 provides the lunar orbit data for P1 and P2.

Table 2. Initial Osculating Elements – Moon Fixed Coordinates

	P1	P2
sma (km)	10900	10312
eccentricity	.78	0.78
Inclination (deg)	172.5	17.6
Node (deg)	175	259
Apoapsis radius (km)	17830	16687
Periapsis range (km)	100-1200	100-4600

Considering the third body orbit perturbations, mostly the acceleration due to the Earth's apparent location in a rotating system, both the orbit periapsis and inclination will vary. Figure 13 shows a sample variation in the inclination and lunar latitude over five years.

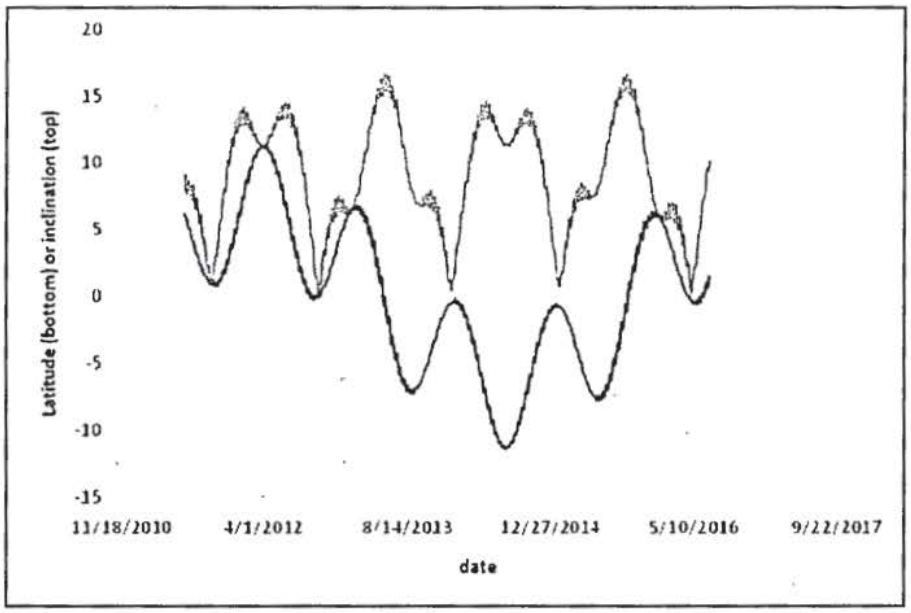

Figure 13. Evolution of P1 Lunar Inclination and Latitude over Five Years

We know 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 spacecraft, which is most effective when the spacecraft 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 Lissajous stationkeeping 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. 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 spacecraft into opposing orbits 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 spacecraft separations (from ~150 to ~30000 km) and geometries to be achieved during this phase.

Mission Maneuver Summary

The maneuvers to successfully complete this mission varied significantly by mission phase. The variability of the space environment and the types of targets archived for each phase of the mission (Lunar gravity assist, Libration insertion, Apoapsis radius, etc) was a concern at the beginning of the mission. Using the limited amount of Δv onboard each spacecraft, a budget was laid out during the early concepts. There was much uncertainty in the estimated Δv in the area of the statistical maneuvers to correct for the previous maneuver errors and for anticipated navigation uncertainties which were tied to an indefinite (at that time) tracking schedule. Shown in Table 3 are the pre-mission budget and executed Δv and the delta in these values. A negative delta indicates that a savings occurred. The top portion of this table shows the deterministic Δvs and that the budget was underestimated. The lower portion shows that the statistical Δv estimates were accurate, meaning that

our analysis and assumptions were correct. The total Δv (deterministic plus statistical) was underestimated by ~ 2.6% for P1 and 6.1% for P2. The increase in the Lunar Orbit Insertion (LOI) and Periapsis Lower Maneuver (PLM) contributed mostly as these Δv estimates were based on an early design which changed as we executed maneuvers. The Lissjaous orbit maintenance and the trajectory correction maneuvers (TCMs) in the transfer were much smaller than anticipated. We attribute this to the earlier planning and analysis which allowed us to place maneuvers in the optimal locations. Note that the Lissajous Δv for P1 includes the transfer from EM L_2 to EM L_1. Additionally, a large margin was held and at the completion of the lunar insertion maneuvers, we have substantial Δv for lunar orbit and attitude maintenance.

Table 3. Estimated and Executed P1 and P2 Δvs

	P1 Estimate (m/s)	P1 Actual (m/s)	P1 Delta (m/s)	P2 Estimate (m/s)	P2 Actual (m/s)	P2 Delta (m/s)
Deterministic Δvs						
ORMs	96.7	96.1	-0.6	204.0	204.9	36.9
FTMs	7.0	6.9	-0.1	5.7	12.4	6.7
DSMs	4.8	7.2	2.4	15.1	30.4	15.3
LTIs	1.5	2.0	0.5	0.8	2.0	1.2
LOIs	89.9	99	9.1	117.1	127.0	9.9
PLMs	0	8	8	0	12	12
Deterministic Total	200	219	19	343	425	82
Additional Δvs						
Lissajous	15	12	-3	12	3	-9
TCMs	15	7	-8	14	4.1	-9.9
Total	232	238	6	407	432	25
Available	324	320	-4	475	467	-8
Margin	92	82	-10	68	35	-33

The sources of additional Δv cost include the TLI declination penalty, TLI gravity and steering loss of 36 m/s for P1 Estimate, the LOI declination penalty of 2 m/s in the estimates, the LOI gravity and steering loss

Conclusion

An ARTEMIS mission overview that began in July of 2009 has been presented. The design transferred two spacecraft from Earth elliptical orbit to lunar elliptical orbits via a multi-body environment transfer that took two years and numerous lunar approaches and flybys, Cis-lunar orbits, low-energy trajectory legs in the Earth-Sun system, Lissajous orbits around both of the EM L_1 and EM L_2, and concluded with the insertion into lunar orbit. The constraints imposed on this design by the limitations of the spacecraft originally designed for a passive Earth-orbiting mission include; thruster orientation and capabilities, available Δv, maximum shadow capability, and maximum distance for radio telecommunication necessitated an innovative design. The flown design satisfied all mission constraints and presented a variety of dual scientific measurements 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. The knowledge that both spacecraft

would have met an untimely demise due to very long shadow events in the original Earth orbit make this accomplishment even more agreeable.

Operations for this mission were performed with a minimal staff, thus the efficiency and dedication of the entire team was paramount in achieving a successful mission. The limited team size also meant that the design itself was nearly "single string" in the absence of backup and contingency trajectories, though many Monte Carlo and optimization simulations were completed. Team knowledge was also a key component, with 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 spacecraft minus Z axis. The maneuver design team was able to design correction maneuvers 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 was the stellar performance of the two spacecraft and the mission operations team with over one hundred maneuvers successfully executed.

ACKNOWLEDGEMENTS

The authors would like to recognize and compliment the outstanding contributions of the THEMIS/ARTEMIS science team, the ARTEMIS mission design team at the Jet Propulsion Laboratory, the ARTEMIS navigation and maneuver design team at Goddard Space Flight Center, and the THEMIS/ARTEMIS navigation team and maneuver and operations team at the University of California-Berkeley Space Science Laboratory to the successful development and implementation of the ARTEMIS mission.

REFERENCES

1. V. Angelopoulos, The THEMIS Mission. Space Sci. Rev. 141, 5–34 (2008). doi:10.1007/s11214-008-9336-1
2. V. Angelopoulos, The ARTEMIS Mission. Space Sci. Rev. (2010). doi:10.1007/s11214-010-9687-2
3. V. Angelopoulos, D.G. Sibeck, 2008 THEMIS and ARTEMIS. A proposal submitted for the Senior Review, 2008 of the Mission Operations and Data Analysis Program for the Heliophysics Operating Missions;available at http://www.igpp.ucla.edu/public/THEMIS/SCI/Pubs/Proposals%20and%20Reports/-HP SR 2008 THEMIS SciTech 20080221.pdf
4. S. Frey, V. Angelopoulos, M. Bester, J. Bonnell, T. Phan, D. Rummel, Orbit design for the THEMIS mission.Space Sci. Rev. 141, 61–89 (2008). doi:10.1007/s11214-008-9441-1
5. S.B. Broschart, M.K. Chung, S.J. Hatch, J.H. Ma, T.H. Sweetser, S.S. Weinstein-Weiss, V. Angelopoulos, Preliminary trajectory design for the Artemis lunar mission, in AAS/AIAA Astrodynamics Specialists Conference, Pittsburgh, Pennsylvania, ed. by A.V. Rao, T.A. Lovell, F.K. Chan, L.A. Cangahuala. Advances in the Astronautical Sciences, vol. 134 (Univelt, Inc., San Diego, 2009). American Astronautical Society / American Institute of Aeronautics and Astronautics
6. M. Woodard, D. Folta, D. Woodfork, 2009 ARTEMIS: The First Mission to the Lunar Libration Points. Presented at the 21st International Symposium on Space Flight Dynamics, Toulouse, France

7. G.J. Whiffen, Mystic: Implementation of The Static Dynamic Optimal Control Algorithm for High-Fidelity, Low Thrust Trajectory Design, in Proceedings of the AIAA/ASS Astrodynamics Specialists Conference, Keystone, Colorado, 2006. Paper AIAA 2006-6741
8. S. Hughes et al, General Mission Analysis Tool (GMAT), Product Description, *http://gmat.gsfc.nasa.gov, August 2007*
9. M. Woodard, D. Folta, D. Cosgrove , J. Marchase, and B. Owens, "Orbit Determination of Spacecraft in Earth-Moon L1 and L2 Libration Point Orbits, AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011, Girdwood, Alaska.
10. B. Cheetam, G. Born, D. Folta, and M. Woodard, "Evaluating Orbit Determination Post-Processing Methods for Operational Artemis Data" , AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011,Girdwood, Alaska.
11. G. Whiffen, and T. Sweetser, "Earth Orbit Raise Design for the Artemis Mission", AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011, Girdwood, Alaska.
12. D. Folta, M. Woodard, T. Sweetser, S. Broschart, and D. Cosgrove, "Design and Implementation of the ARTEMIS Lunar Transfer Using Multi-body Dynamics", AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011,Girdwood, Alaska.
13. K.C. Howell, B.T. Barden, M.W. Lo, Application of Dynamical Systems Theory to Trajectory Design for a Libration Point Mission. J. Astronaut. Sci. 45(2), 161–178 (1997)
14. D. Folta, M. Woodard, and D. Cosgrove, "Stationkeeping of the First Earth-Moon Libration Orbiters: the ARTEMIS Mission" AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011, Girdwood, Alaska.
15. D. Folta, T.A. Pavlak, K.C. Howell, M.A.Woodard, D.W.Woodfork, "Stationkeeping of Lissajous trajectories in the Earth-Moon system with applications to ARTEMIS", AAS/AIAA Space Flight Mechanics Meeting, San Diego, CA, Feb. 2010
16. T. Pavlak and K. Howell, "Strategy for Long-Term Libration Point Orbit Stationkeeping in the Earth-Moon System" , AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011,Girdwood, Alaska.
17. W.S. Koon, M.W. Lo, J.E. Marsden, S.D. Ross, Heteroclinic Connections Between Periodic Orbits and Resonance Transitions In Celestial Mechanics. Chaos 10(2), 427–469 (2000). doi:10.1063/1.166509
18. T. Sweetser, S. Broschart, V. Angelopoulos, D. Folta and M. Woodard, "ARTEMIS Lunar Orbit Insertion and Science Orbit Design through 2013", AAS/AIAA Astrodynamics Specialist Conference held August 1-4 2011,Girdwood, Alaska.