ARTEMIS Operations from Earth-Moon Libration Orbits to Stable Lunar Orbits

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ARTEMIS – an extension of the NASA Medium Explorer mission THEMIS – is a two-spacecraft mission that completed a two-year lunar transfer by way of multiple lunar gravity assists and a low-energy trans-lunar trajectory that culminated into EM L_1 and EM L_2 lunar libration point orbits and finally stable prograde and retrograde elliptical lunar orbits in July 2011. Between the two spacecraft 170 maneuvers events were executed from the start of the mission extension to the final lunar placement with a 100% successful execution rate. This paper addresses the ARTEMIS mission's operational challenges and lessons learned during the low-energy Trans-Lunar phase, the Earth-Moon Libration Orbit phase, and Lunar Orbit phase.

Nomenclature

N	=	number of thruster pulses
P_w	=	thruster pulse width
ΔV	=	magnitude of velocity change
I_{sp}	=	specific impulse

I. Introduction

Acceleration, Reconnection, Turbulence, and Electrodynamics of the Moon's Interaction with the Sun (ARTEMIS), an ambitious mission extension of Time History of Events and Macroscale Interactions during Substorms[1] (THEMIS), a NASA constellation of five spin stabilized spacecraft launched on February 17, 2007 to study the magnetospheric processes responsible for auroral sub-storm onset, has successfully completed the transfer of two spacecraft from their final THEMIS mission orbits to stable lunar orbits.

The ARTEMIS baseline trajectory[2] was divided into four operationally unique phases. The Earth-Orbiting phase—detailed in Ref. 3—slowly increased the apogee of both spacecraft from the THEMIS end-of-mission orbit to target multiple lunar gravity assist events. The Trans-Lunar phase encompassed the low-energy trajectory connecting the gravity assist events to insertion into a lunar libration orbit. The spacecraft then spent approximately 10 months in the Libration Orbit phase executing many stationkeeping maneuvers to maintain their trajectories around the Earth-Moon L1 and L2 Lagrange points, being the first spacecraft ever to occupy a lunar libration orbit.[4] At the conclusion of libration operations, the spacecraft transitioned into the mission's Lunar Orbit phase by entering into stable prograde and retrograde lunar orbits. In total, the ARTEMIS mission operations team executed 170 maneuvers between the two spacecraft during the implementation of this trajectory.

The operations involved in the Trans-Lunar, Libration Orbit, and the Lunar Orbit phases are the focus of this paper.

II. Summary of Spacecraft and Ground Segment Systems

A. Spacecraft Design

1. Instrument Suite

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The instrument suite consists of an Electrostatic Analyzer (ESA), a Solid State Telescope (SST), a boom mounted Flux Gate Magnetometer (FGM), a boom mounted Search Coil Magnetometer (SCM), and an Electric Field Instrument (EFI) mounted on two axial booms and four spin-plane wire booms that extend 20 meters in one direction and 25 meters in the other. The spacecraft also has a Sun Sensor Assembly (SSA) that provides important data used for spacecraft navigation and propulsion. Figure 1 shows the fully deployed spacecraft with these instruments labeled.

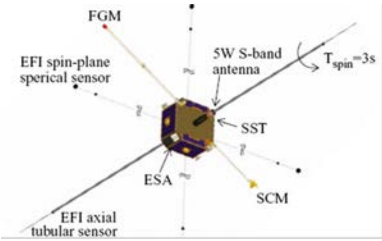


Figure 1. THEMIS spacecraft with deployed instruments.

The ARTEMIS mission is utilizing these instruments to measure the plasma properties of the lunar wake in the solar wind, the undisturbed solar wind outside of the Earth's magnetosphere, and the interactions of the Earth's magnetotail with the solar wind, adding a new dimensional scale to the continued measurements of the remaining THEMIS spacecraft. In lunar orbit, it is also measuring the plasma properties in the vicinity of crustal magnetic anomalies on the Moon's surface during low altitude periselene transits.

2. Propulsion System

The spacecraft—referred to as P1 and P2 or THEMIS B and THEMIS C, respectively—are spin stabilized and started the ARTEMIS mission rotating at a rate of approximately 20 RPM. Their attitudes were oriented within 10 degrees of the ecliptic south pole. Their Reaction Control System[5] (RCS) is a hydrazine (N_2H_4) blow-down system pressurized with helium (GHe) and has four 4.5 N thrusters.

Two of the thrusters point parallel to the spacecraft's spin-axis (A1 and A2 in Figure 3), which provide thrust in an axial direction when fired together. Attitude precession maneuvers are performed by firing either of these axial thrusters in a sun-synchronous pulsed-mode.

The other two thrusters (T1 and T2 in Figure 3) are mounted in a tangential (radial) direction and when fired simultaneously, in a sun-synchronous pulsed-mode, they provide thrust perpendicular to the spacecraft's spin-axis. Firing either thruster alone provides control over the spacecraft's spin rate. For a small vectorization penalty, the axial and radial maneuver types can be executed in short succession, but not simultaneously, to produce a combined thrust constrained in the spacecraft's lower hemisphere. It is not possible for the spacecraft to generate a thrust in the upper hemisphere.

The hydrazine fuel load at the time of launch was 48.780 kg for P1 and 48.810 kg for P2. By the completion of the ARTEMIS lunar insertion sequence, the standard bookkeeping method for fuel mass estimation[6] indicated that P1 had 5.153 kg of fuel remaining after executing 143 maneuvers. P2 had 3.401 kg of fuel remaing after executing 162 maneuvers. Both of these values include the spacecraft's portion of maneuvers during the THEMIS mission.

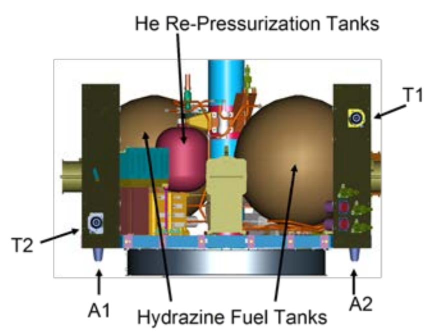


Figure 2. RCS system and thruster locations.

B. Ground Systems Overview

During the ARTEMIS mission the operations team reused many of the tools and systems already successfully utilized for THEMIS.[7-10] A short description of the key flight dynamics related systems are discussed in the following sections.

1. Orbit Determination

During all phases of the ARTEMIS mission, orbit determination (OD) was performed by the operations team at the University of California, Berkeley's Space Sciences Laboratory (SSL) using the Goddard Trajectory Determination System[11] (GTDS) and two-way Doppler and ranging measurements collected from a selection of ground stations from NASA's Near Earth Network (NEN), the Universal Space Network (USN), the Deep Space Network (DSN), and SSL's own 11-meter antenna (BGS). Parallel OD support during critical mission phases, up to the start of nominal lunar operations, was provided by the Flight Dynamics Facility (FDF) at GSFC.

Ephemeris modeling was performed using the GTDS's differential corrector with the DE421 planetary ephemeris, an 8 x 8 Earth gravity potential model, and a flat plate model of the spacecraft for solar radiation pressure estimation. In lunar orbit operations, the central body of integration was switched to a 70 x 70 lunar gravity potential model (LP150Q).

2. Attitude Determination

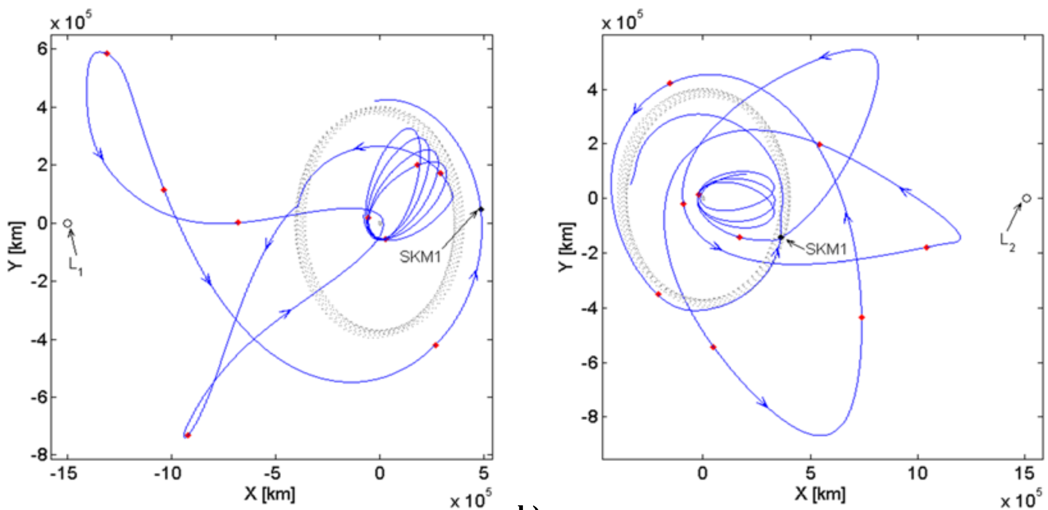
Sun-only ground based attitude determination (AD) is performed with the Multi-mission Spin-Axis Stabilized Spacecraft[12] (MSASS) suite, a GSFC GOTS software package with the Fuzzycones maximum likelihood algorithm developed for ARTEMIS operations by J. Hashmall et al.[13]

3. Maneuver Planning

The initial transfer trajectory and lunar orbiting maneuver designs were created at NASA's Jet Propulsion Laboratory (JPL) using the LTOOL[14] and MYSTIC[15] software packages. The lunar gravity assist and transfer trajectory correction maneuvers, and the libration stationkeeping maneuvers, were designed by NASA's Goddard Space Flight Center (GSFC) using a combination of the General Mission Analysis Tool[16] (GMAT) and Analytical Graphics, Inc.'s STK/Astrogator.[17]

Finite maneuver planning, during all mission phases, was performed at SSL using the General Maneuver Program[18] (GMAN) to leverage the extensive experience and calibration history gained during the mission.[19,20]

III. Trans-Lunar Phase



a) b)

Figure 4. The definitive Trans-Lunar trajectories in a Sun-Earth rotating frame, with the Sun in the negative X direction. The DSMs and TCMs are marked in red. SKM1 marks the start of the Libration Orbit phase. The Sun-Earth libration points are labeled as L1 and L2 respectively. The dotted trajectory shows the Moon's orbit during this time period. a) P1 b) P2

A. Summary

The P1 spacecraft successfully started its trans-lunar trajectory after the completion of its final lunar gravity assist event on February 13, 2010. Figure 4a shows the definitive path the spacecraft took on its way to its insertion into orbit around EM L_2. The spacecraft executed one deep space maneuver (DSM) and required 4 trajectory correction maneuvers (TCM) between the two approaches to the Earth-Sun L_1 Lagrange point and its return to the near-lunar region in August 2010. Total executed ΔV during this mission phase was 7.3 m/s for the single DSM and a total of 3.3 m/s for TCMs.

Figure 4b illustrates the definitive trajectory of P2 through its trans-lunar phase. On its three excursions to Earth-Sun L_2, the spacecraft executed a total of 3 deterministic DSMs, one of which was a combined thrust from both the spacecraft's axial and radial thruster pairs, and 4 TCMs. The DSMs totaled 30.2 m/s and the TCMs 3.4 m/s.

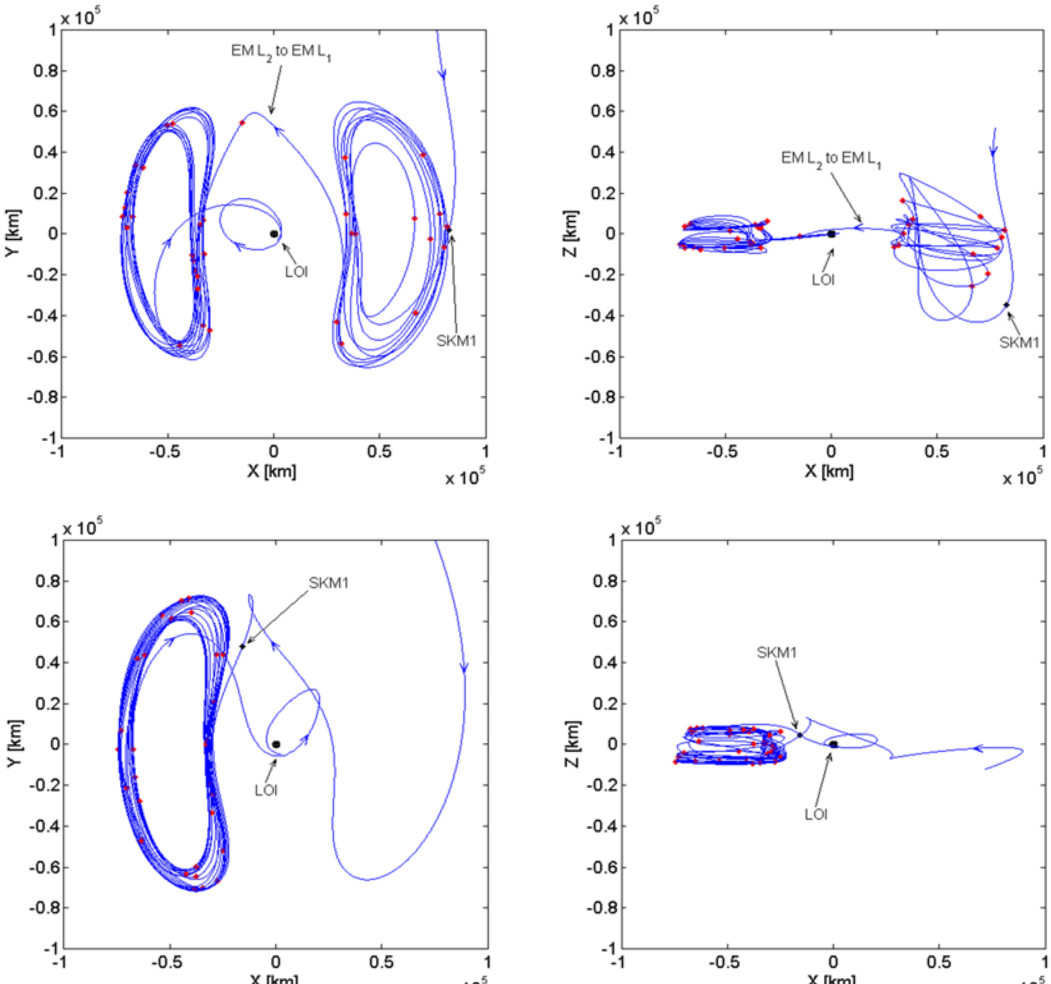
B. Operational Challenges

1. Data Downlink During Distant Excursions

During trans-lunar trajectories, P1 and P2 reached maximum ranges of approximately 1.5 million and 1.2 million kilometers, respectively. Significant telemetry data downlink volume was impossible at the 4 kbps maximum data rate due to extreme distances from the Earth. Thus, it became a priority to only retrieve spacecraft engineering data to monitor its health and safety. However, a limited amount of science data was stored onboard for downlink once the spacecraft returned to the Earth-Moon region.

In lunar orbit operations, the ARTEMIS mission has begun to use the DSN 70-meter antenna network in a receive-only configuration to reduce the mission's usage of the extensively subscribed 34-meter network. While this configuration does not provide Doppler tracking and ranging data it does allow the return of large data volumes in a shorter period of time due to the increased link margin from the larger dish size and sensitive receivers of these antennas. It would have been a benefit to ARTEMIS's data return capability to have integrated these 70-meter antennas before the trans-lunar phase of the mission, which would have increased the possible data rate from 4 kbps to 32 kbps at the maximum ranges observed. However, at the time of execution of this phase the mission scope did not include these 70-meter assets.

IV. Libration Orbit Phase



a)
b)

Figure 5. The definitive Libration Orbit trajectories in a Moon fixed frame with the Earth in the negative X direction. SKMs are marked in red with SKM1 labeled. a) P1 b) P2

A. Summary

The nominal ARTEMIS libration orbit phase was originally planned to cover a 6 to 8 month period ending with both spacecraft entering lunar orbit in April, 2011. However, the overall libration stationkeeping ΔV costs incurred in actual operations were significantly lower than expected from the conservative estimates generated during mission inception. Due to this fortunate circumstance, the operations team redesigned the lunar orbit insertions to occur in June and July, 2011, extending the time spent in this scientifically and navigationally interesting region.

P1's libration orbit phase began on August 24, 2010 with its insertion into a quasi-halo libration orbit[21] where it began stationkeeping activities. For the first 4 months, the spacecraft orbited EM L_2 until it was transferred to EM L_1 in January, 2011. It orbited this point until a lunar transfer initiation maneuver was executed to start its descent toward the Moon in June, 2011. During its 10 month stay in Earth-Moon libration orbit, 36 stationkeeping maneuvers were executed totaling 8.7 m/s.

On October 22, 2010 the P2 spacecraft was inserted into a quasi-halo libration orbit around EM L_1. This spacecraft executed 31 stationkeeping events in the 8 months of libration orbit operations, totaling 4.5 m/s.

The above ΔV values also contain a vectorized maneuver, executed on each spacecraft, used to control the evolution of the libration orbit's Z-axis amplitude. These maneuvers required an out of orbit plane component along with the usual in plane stationkeeping component. Without these events the spacecraft's Z-axis excursions would have been too great to allow an insertion into a lunar orbit with the desired inclination.

The definitive trajectory for both spacecraft is shown in Figure 5. Figure 6 shows the SKM cost for both probes as a function of maneuver number.

B. Operational Challenges

1. Stationkeeping Operational Process

The maintenance of the libration orbits required frequent interaction between the mission operations team at SSL and the GSFC navigation team. Typical stationkeeping maneuvers were executed once every 7 to 14 days, or once per 1 to 2 libration point revolutions. Stationkeeping maneuvers were designed to counter the natural perturbations that would cause the spacecraft to depart the libration point region, either towards the Moon or towards the Earth. The stationkeeping maneuver targeting strategy was to continually select and maneuver toward targets designed to extend the libration orbit downstream by several revolutions. This was accomplished by bounding the spacecraft velocity at the libration trajectory's X-axis crossing in Earth-Moon rotating coordinates.[21]

In practice, the stationkeeping targeting process required near daily communication between the teams at SSL and GSFC. The process started with SSL generating a daily orbit determination solution utilizing tracking data collected since the last maneuver event, and an attitude solution as frequently as the available data permitted. These navigational products were then processed by the GSFC team into an impulsive stationkeeping maneuver ΔV vector that was located inside a previously scheduled DSN ground station contact as close to the desired Earth-Moon X-axis crossing as possible. This impulsive maneuver plan was provided to SSL for processing into the finite maneuver plan with the GMAN program while incorporating maneuver calibration data[19,20] to estimate the expected RCS performance of the event. Results from these finite simulations were made available to GSFC for validation and verification. If the results were unsatisfactory, the entire process was iterated again until the desired maneuver goals were met.

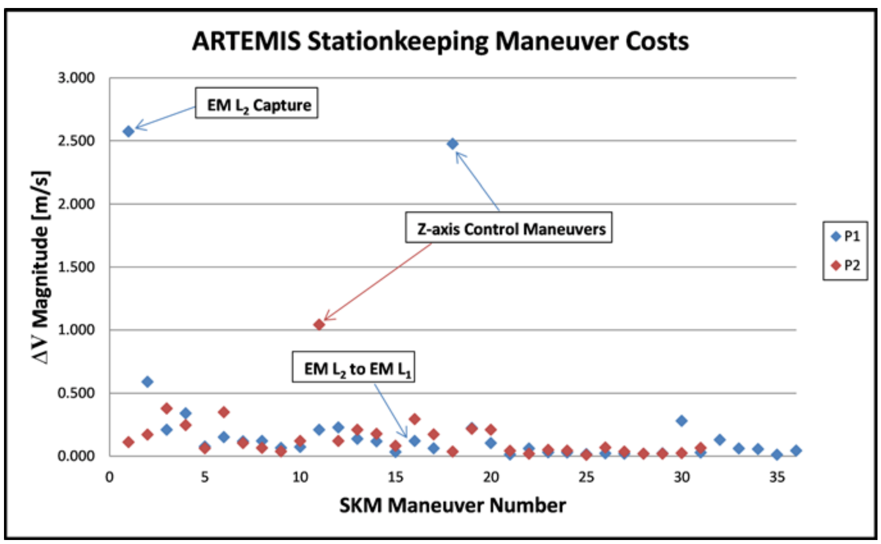


Figure 6. The stationkeeping maneuver ΔV cost trend.

The risks associated with the multiple data handovers between the teams at SSL and GSFC were mitigated through defined data formats and automated systems where possible. In particular, the daily automated OD system at SSL could generate OD solutions at any time, allowing navigation products to be generated and transmitted while reducing the need for active personnel. Furthermore, the GMAN simulations at SSL are highly automated by integrated systems that process the input maneuver targets from GSFC, navigational data, and maneuver calibration data into full finite maneuver simulations and spacecraft ready command sheets. These automated systems were an absolute necessity for the small navigation team to accurately target the stationkeeping maneuvers at the needed frequency.

2. Scheduling Stationkeeping Maneuvers on Long Lead-Time Ground Station Assets

The vast majority of stationkeeping maneuver operations were executed with the support of DSN's 34-meter antenna network. Scheduling negotiations for time on this network can start as far as 12 weeks in advance, which presented a significant challenge when the ideal libration stationkeeping maneuver time was not known until at most 2 weeks before the event.

To mitigate this problem, the operations team would schedule ground station time as close to the predicted X-axis crossings as possible. These ground station contacts then constrained future stationkeeping maneuvers to conform to these previously scheduled times. The operations team gained additional flexibility through trading contacts amongst P1 and P2 because these types of schedule changes did not require negotiations with other missions.

In the rare circumstances where stationkeeping events needed to occur outside of the previously planned ground station schedule (see *P1 Mass Loss Anomaly* below), the operations team was able to acquire time from negotiating with other accommodating missions, however, these types of requests were not typical and kept to a minimum.

While this scheduling strategy worked well for ARTEMIS operations it did, in theory, increase the overall ΔV expenditure of the mission because stationkeeping maneuvers did not generally occur at their most optimal locations. Future missions to Earth-Moon libration orbit that expect a high execution frequency of stationkeeping maneuvers would greatly benefit by planning for ground station assets that can easily be scheduled with a turnaround much shorter than the intended maneuver rate.

3. Refinements in Finite Maneuver Targeting Using a Variable Thruster Pulse Width

The radial maneuver thruster firing mode used for stationkeeping events originally constrained the sun-synchronous thruster pulses to use either a 40 or 60 degree firing arc, or pulse width (P_w). These pulse widths were selected for use during THEMIS to balance the relationship between efficiently executing radial maneuvers in a reasonable amount of time, reducing the ΔV inefficiency caused by large pulse widths, and to avoid possible fuel resonance scenarios at specific fuel tank fill levels. This constraint limited the possible ΔV magnitudes of stationkeeping maneuvers into quantized steps of approximately 2 cm/s. This problem was previously encountered by ARTEMIS while targeting the lunar gravity assist events[3] and would result in an increase of overall stationkeeping ΔV costs because each executed maneuver would either overshoot or undershoot the desired magnitude.

After the significant fuel expenditures taking P1 and P2 from their THEMIS orbits to libration insertion, the fuel tank fill levels were low enough that the resonance scenarios were no longer of concern, thus lifting the constraint on radial thrusting pulse width. With this constraint removed, the navigation team at SSL implemented a new algorithm in the finite maneuver targeting system that allowed finer control of the ΔV magnitude by varying the pulse width for each stationkeeping maneuver. By applying this variation, any ΔV magnitude could be targeted down to a pulse width the size of the thruster actuation limit (50 ms).

This pulse width variation was accomplished simply by simulating an initial event at the 60 degree pulse width and then comparing the resultant ΔV magnitude ($|\Delta V_i|$) to the desired magnitude ($|\Delta V_t|$). Any difference was divided by N_i, the number of thruster pulses needed to generate $|\Delta V_t|$, converted into a pulse width adjustment (ΔP_{w_i}), and then applied to the targeted pulse width as shown in Eq. (1) and Eq. (2).

$$P_{w_{i+1}} = P_{w_i} + \Delta P_{w_i} \tag{1}$$

$$\Delta P_{w_i} = P_{w_i} \frac{\left(\frac{|\Delta V_t|}{|\Delta V_i|} - 1\right)}{N_i} \tag{2}$$

To take maneuver calibration data—which is based on the accumulated thrust time—into account, this process was executed iteratively until the prescribed adjustment (ΔP_{w_i}) was less than 0.05 degrees, which is half the minimum pulse width variation allowed by GMAN. The smallest events executed with this method were a 1.2 cm/s maneuver on P1 and 1.0 cm/s maneuver on P2 (one pulse at 32.58 and 32.55 degrees, respectively).

Future missions to this region should plan to have thrust magnitude control fine enough such that desired stationkeeping maneuver magnitudes can easily be achieved. This level of control authority is of particular importance to missions employing spinning spacecraft where the primary method for stationkeeping thrust generation will use a similar pulsed mode.

4. Attitude Determination Frequency

THEMIS performed attitude determination using data from the spacecraft's FGM and SSA sensors. When ARTEMIS was envisioned, the FGM's usefulness for attitude determination was limited only to the Earth-Orbit phase. This limitation necessitated the creation of Fuzzycones, a sun-only attitude determination process that utilizes SSA sensor data.

The ARTEMIS SSA sensor measures the sun angle in digital increments of 0.125 degrees. At the time the sun transitions from one digital bin to the next sun angle is known to a higher degree of accuracy. Data generated from these bin transitions is used along with regular sun angle data to meet the mission required attitude knowledge error of less than 1 degree.[13]

The bin transition rate is directly related to the angle between the ecliptic pole and the spacecraft's attitude. The closer the attitude is to the ecliptic pole the longer it takes to observe bin transitions. Due to this relationship the operations team encountered periods of time where the needed bin transition frequency was lower than the stationkeeping maneuver frequency. For such cases, it was impossible to generate an attitude solution from unperturbed data between maneuver events.

The navigation of the spacecraft in libration orbit was not significantly impacted by the resultant decrease in attitude knowledge accuracy because the average stationkeeping maneuver was small enough that the effect of the resultant torque on the attitude was minimal. Therefore, it was possible to generate a sufficient Fuzzycones attitude solution spanning small maneuver events.

It is expected that challenges similar to this one could be easily avoided by spacecraft specifically designed for libration operations, namely one that included additional attitude sensors whose usefulness are not tied to their proximity to the Earth (e.g. star tracking sensors).

5. P1 Mass Loss Anomaly

On October 14, 2010 P1 experienced an anomalous 5.7 cm/s ΔV due to the separation of a 0.092 kg sphere from the end of one of its 25 m EFI instrument wire booms. This mass loss occurred approximately 3 days before the next planned stationkeeping maneuver, and initial analysis on the direction and effect of the anomalous ΔV showed that a significant portion of the magnitude went into a beneficial direction for libration orbit maintenance.

Sufficient orbit solution accuracy was achieved post-anomaly because additional tracking data collected in response to the event. Therefore, the operations team was able to go forward with the planned stationkeeping maneuver. However, the ΔV of that maneuver increased to 15.8 cm/s, up from the pre-anomaly cost of 11.0 cm/s.

It is now believed that this mass loss was the result of a micrometeoroid impact that corresponded to the arrival of the Orionid Meteor Shower. A detailed discussion of the timeline of the anomaly and the related analysis can be found in Ref. 22.

6. Orbit Determination Convergence

The short turnaround time between stationkeeping maneuvers presented two challenges in the OD: 1) determining when an OD solution had adequately converged for planning the subsequent stationkeeping maneuver and 2) optimizing the distribution of data received from various tracking sites. As discussed in Ref. 23, convergence of the solar radiation coefficient—which was used as a solve-for parameter in GTDS—ultimately provided an indication of the quality of the OD solution and distribution of tracking data amongst the tracking sites used for ARTEMIS.

V. Lunar Orbit Phase

A. Summary

On June 17, 2011 P1 executed a 3.2 m/s lunar transfer initiation (LTI) maneuver. This maneuver was then followed by a 0.5 m/s TCM on June 22 to target the periselene altitude and direction of approach needed for insertion into its desired retrograde lunar orbit. The lunar orbit insertion (LOI) maneuver was executed on June 27 as three segmented maneuvers to reduce steering loss inefficiencies. The LOI burn lasted 2.5 hours and expended 50.3 m/s.

After the completion of the LOI event, P1 executed 5 period reduction maneuvers (PRM) which decreased the semi-major axis to lessen the amplitude of periselene altitude fluctuations that result from the Earth's perturbing forces, stabilizing the orbit. The lunar orbit geometry achieved at the conclusion of the PRM sequence on September 8, was approximately 1760 by 25260 km. It took P1 a total of 768 days from departing its THEMIS orbit to reach this state.

P2 followed a similar insertion plan as P1 except it was targeted to insert into a prograde lunar orbit. It executed a series of 2 LTIs and a single TCM, totaling 1.1 m/s, between June 21 and July 4, 2011. Its 3 segment LOI event occurred over 3 hours on July 17 and expended 73.1 m/s.

This spacecraft also executed a sequence of 5 PRMs between July 23 and November 7, 2011, totaling 45.3 m/s. The prograde lunar orbit geometry at this time was approximately 2700 by 31800 km. P2's trajectory took 839 days to reach completion. The lunar trajectories for both spacecraft are shown in Figure 7.

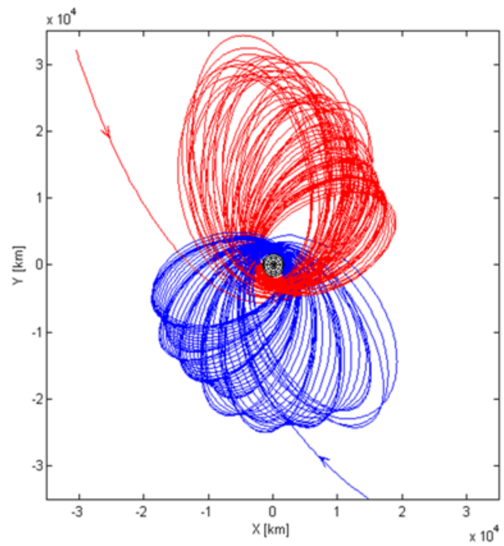


Figure 7. The definitive lunar orbits of P1 (blue) and P2 (red) through the PRM maneuver sequence shown in rotating frame with the Sun in the –X direction.

B. Operational Challenges

1. Mitigating Lunar Orbit Insertion Risk

ARTEMIS's lunar orbit insertion[24] (LOI) maneuver represented a significant risk to the success of the mission. A large enough under performance of the LOI would have either resulted in the spacecraft failing lunar capture and departing the near-lunar region, or it would have achieved an orbit with an aposelene that was too high to remain safely stable once influenced by the Earth's perturbing forces (which would ultimately result in an impact on a subsequent periselene). Mitigating these risks became a major driving goal for the operations team.

All maneuvers are executed out of the spacecraft's absolute time sequence (ATS) table, which is stored in volatile memory. Therefore a central concern with the LOI execution was the chance of a rare* spacecraft reset interrupting the maneuver during a period where operators at SSL would not be able perform recovery operations and initiate, or restart, thrusting without loss of ΔV.

Several steps were taken by the operations team to reduce the impact of this type of LOI interrupting event:

1) Maneuver commands were preloaded on the spacecraft and the propulsion bus was enabled 2 days in advance of the LOI to ensure that in the event of a major interruption of service at the SSL Mission Operations Center (MOC) the spacecraft would be able to initiate the maneuver sequence independently. These commands were then updated daily as the navigation team refined the LOI event when new navigation data became available.
2) Onboard fault detection and correction (FDC) mechanisms that could abort a maneuver were disabled to further reduce the risk of interrupting a LOI segment.
3) Continuous ground station tracking through DSN began at LOI – 12 hours to allow the operations team to immediately respond to any health and safety concerns and to provide the navigation team with additional tracking data and continuous RCS temperature measurements for final maneuver targeting adjustments.
4) Continuous tracking of the spacecraft was provided by two DSN antennas during the maneuver itself, with one primary antenna for command and telemetry and a secondary antenna as a "hot backup". Using two antennas at DSN support Level 3 ("important event") was deemed more efficient than one antenna at Level 2 or Level 1 ("critical event").
5) Operational procedures were developed to allow the restart of a LOI burn segment outside of the ATS in the eventuality of a mid-maneuver spacecraft reset. These procedures would ensure that in the case of an interruption, the whole LOI segment would not be lost.
6) Navigation procedures were developed to provide near real-time updated ephemerides of the predicted insertion orbit as spacecraft conditions evolved or within minutes of a maneuver interrupting event. (see *Lunar Orbit Insertion Confirmation Process* below).
7) Multiple LOI maneuver sequences were developed by the navigation team to account for possible deviations between predicted and actual RCS system temperature, which would impact the expected performance of the maneuvers. By having these backup maneuver sequences ready, the operations

* In over 5 years of on-orbit operations 16 resets that could have impacted maneuver execution have occurred across all 5 of the original THEMIS spacecraft. ARTEMIS P2 has experienced 2 of these resets and P1 has had none.

team could easily respond to an unexpected fuel tank heater cycle, an event over which the operators have no direct control.

In practice, both LOI sequences executed near perfectly, and safely delivered P1 and P2 to lunar orbit except with one exception in P1's maneuver performance outlined below.

2. *Limitations in Targeting and Simulating Lunar Centered Maneuvers*

Another limitation that the mission operations team was required to overcome was that GMAN, the finite maneuver simulation software, is designed to simulate maneuvers in Earth centered frames only where it uses either a desired change to specific Keplerian orbital elements or a ΔV vector as the targeting input value. During P1 and P2's Earth-Orbit phases, the operations team extensively used the Keplerian targeting functions of GMAN, mirroring the same methods used successfully during the THEMIS prime mission.[25] For the trans-lunar and libration orbit phase, Keplerian element targeting was not an option so the operations team made use of GMAN's ΔV vector targeting algorithms to great success.

Once entering lunar orbit, maneuver targeting based on Keplerian elements would once again be feasible, however the GMAN software has no ability to change its central body of influence from the Earth to the Moon. As a result the targeting process for the LOI and PRM sequence maneuvers was further complicated by the need to not only continue targeting maneuvers with precalculated ΔV vectors, but to provide these ΔV vectors in an Earth centered reference frame. While the frame conversions between Moon and Earth centered coordinates are easy to implement, the procedures required extensive testing and verification to mitigate a potentially dangerous risk from such a simple problem. The only other possible solution would be direct software modification of GMAN, but this alternative was viewed as too costly and a greater potential risk.

Furthermore, FlatSat, the THEMIS spacecraft simulator used by ARTEMIS to verify all ATS command tables containing maneuvers, utilizes a dynamics modeling system that contains an Earth gravity model, but no lunar modeling. The procedures that validate data generated by FlatSat against the GMAN finite simulations—confirming the ATS contained the expected maneuver command structure—could not rely on trajectory data for confirmation, and thus required additional modifications to provide command verification using thruster firing phase and duration only.

While it is expected that a mission initially designed for lunar, or near-lunar, operations would not encounter these types of limitations these were significant challenges the ARTEMIS team had to overcome to ensure this very unique mission was successful with these limited resources.

3. *Lunar Orbit Insertion Confirmation Process*

In order to mitigate the effects of an LOI interrupting event, the operations team created a maneuver confirmation process by which a probe's subsequent dynamic state could be predicted so that contingency operations planning could start almost immediately.

Design of the confirmation process was based on the existing maneuver reconstruction process which is integral to ARTEMIS/THEMIS post-maneuver recovery operations. While the reconstruction uses previously recovered telemetry data to re-enact the maneuver, the confirmation acquires data by immediate operator intervention.

As each segment was executed, the confirmation operator monitored the probe and its performance through the real-time telemetry. At the completion of each LOI segment, the operator used the observed data— including temperature, pressure, ignition time, spin-rate and number of thruster pulses —to generate the resulting state. That state was used to generate an ephemeris estimating a new effective trajectory that could then be used in the event of an unsuccessful continuance of LOI.

If a maneuver segment was interrupted or failed to complete, the confirmation simulation could be tailored to match the observed maneuver performance. In the case of segment interruption, a new segment could be created to reflect the thrusting gap created by the recovery and reigniting process. In this manner, a reasonable approximation for the trajectory could aid in recovering the resulting orbit.

4. *P1 Mass Loss Anomaly and Lunar Orbit Insertion Overburn*

When the spacecraft lost 0.092 kg of mass from the end of its 25 m wire boom, the center of mass moved closer toward the spacecraft's spin-up thruster. As a result, when executing a radial maneuver the spacecraft now spins down at a rate 15 times greater[22] than previously observed due to the new relationship between the center of thrust and center of mass.

This increased spin change had an unforeseen impact on GMAN's planning of this spacecraft's LOI maneuver. GMAN uses internal algorithms to interpolate between measured pre-launch specific impulse (I_{sp}) and thruster force data to generate performance curves for these factors at the RCS's current state. When a significant spin change occurs during a simulation, these algorithms appear to overestimate the decrease in I_{sp} during maneuvers as a result of fuel consumption and subsequent pressure drop, ultimately resulting in an overall underestimation of maneuver performance. GMAN then counterbalances this spurious underperformance by adding more thruster pulses to the maneuver, which when executed on the spacecraft are unnecessary and cause an overburn relative to the desired target.

It did not become apparent to the operations team that P1 was experiencing this problem until the LOI because the impact of the error is directly related to the magnitude of the maneuver being executed. A maneuver of significant enough magnitude to show this error was not executed after the libration orbit phase concluded. As a result, P1's LOI was 6% larger than desired and inserted the spacecraft into orbit with a lower aposelene than planned. Upon reflection, the need for a 0.5 m/s TCM after the 3.2 m/s LTI maneuver might have been an early indication of this problem, but was initially attributed to expected performance deviation due to the actual post mass loss anomaly changes and not a simulation artifact.

Ultimately, the overall total cost of P1's LOI and PRM sequence maneuvers did not change significantly because the overburn had an effect that was beneficial to PRM's goals, but the navigation team at JPL was required to redesign the PRM sequence to compensate for the error. To prevent this problem from impacting future maneuver events, the navigation team at SSL adjusted P1's maneuver performance calibration curves to account for the simulation error.

5. *P2 Attitude Dependent Thermal Issues Encountered During the PRM Sequence*

As previously mentioned, each radial thrust maneuver generates a small torque on the spacecraft attitude as a result of a spacecraft Z-axis offset between the center of thrust and mass. While for each individual maneuver the torque is relatively small, the combination of many maneuvers may result in a significant precession. During the THEMIS mission, these attitude changes were corrected periodically with attitude precession maneuvers. However, for ARTEMIS these attitude precession events would result in significant undesired disturbances to the sensitive trajectories—especially during libration stationkeeping operations—so the requirement to correct these effects was waived.

At the start of the ARTEMIS mission, P2's attitude was orientated 7.8 degrees away from the ecliptic south pole. The combined attitude change from P2's Earth-Orbiting, Trans-Lunar, and Libration Orbit phase maneuvers, including the LOI, left the spacecraft 16.8 degrees from ecliptic south. This attitude did not initially violate any flight constraints set to ensure proper operating temperatures, but by September 2011 temperatures onboard the spacecraft began to rise as the sun become more incident on the bottom flight deck, which is the mounting surface for the majority of the spacecraft's electronics, fuel tanks, and radiator (see Figure 3). Within a short period, fuel tank and transponder temperatures violated their yellow limits and prompted the need for an immediate attitude maneuver to reduce the sun angle and cool the spacecraft down. Afterwards, the spacecraft temperatures stabilized and then began to cool as the seasonal variations of the sun angle trended upward.

The downstream PRM maneuvers were redesigned to account for the orbital disturbance caused by the attitude maneuver and were executed without further interruptions.

VI. Budgeted ΔV Versus Realized Expenditure

Table 1 contains a summary of the budgeted and realized ΔV expenditure for both spacecraft. The non-deterministic TCMs from all phase of the mission are listed in a separate row. Positive values indicate remaining budgeted ΔV and negative indicate over-expenditure.

The increased ΔV cost in P1's Trans-Lunar phase was due to the retargeting of that spacecraft's single DSM maneuver to incorporate both the original deterministic DSM and a needed TCM to correct the variation in the Trans-Lunar injection state from the baseline trajectory. The overall cost of this event was less than performing both maneuvers separately.

P2 over-expended ΔV in both its Earth-Orbiting and Trans-Lunar phase. The root cause of this increased cost was the need for a series of shadow deflection maneuvers (costing 11.8 m/s) designed to reduce a dangerously long Earth shadow during the later portion of the Earth-Orbiting phase.[3] The impact of these maneuvers on P2's lunar gravity assist event and its injection into the Trans-Lunar phase also increased the ΔV cost of that phase.

Both P1 and P2 had over-expenditures in their Lunar Orbit phases as the result of retargeting their lunar insertion to allow for the extended libration orbit operations. P1's costs also reflect the increase due to the 6% LOI overburn discussed in *P1 Mass Loss Anomaly and Lunar Orbit Insertion Overburn*.

Of particular note is the significantly lower ΔV expenditure realized during the Libration Orbit phase for both spacecraft, even with the additional 2 – 3 months of libration orbit operations. This difference in actual to budgeted costs is attributed to the very conservative stance taken when estimating the budget for this phase. In fact, when removing the ΔV costs of libration capture, P1's EM L_2 to EM L_1 transfer and mass loss anomaly correction, and the

Table 1. The realized ΔV expenditure versus the budget. All values are in m/s.

	P1			P2		
	Budget	Realized	Difference	Budget	Realized	Difference
Earth-Orbit	103.7	102.9	0.8	245.7	253.3	-7.6
Trans-Lunar	4.8	7.2	-2.4	15.1	27.1	-12.0
Libration Orbit	15.0	8.9	6.1	12.0	4.6	7.4
Lunar Orbit	93.4	96.2	-2.8	119.9	121.4	-1.5
Trajectory Correction Maneuvers (All Phases)	15.0	7.0	8.0	14.0	4.1	9.9
Total	231.9	222.3	9.6	406.7	410.6	-3.9

Z-axis control maneuvers for lunar inclination targeting, the total ΔV expended for maneuvers with purely a stationkeeping goal was 3.7 and 3.3 m/s for P1 and P2, respectively. This is a surprisingly small 0.31 m/s and 0.41 m/s per month for libration stationkeeping.

VII. Current Mission Status

ARTEMIS is continuing to collect solar wind, lunar wake, magnetospheric, and crustal magnetic anomaly data. Over the next few years the operations team will be executing approximately two maneuvers per year, per spacecraft, to control their periselene altitudes and enhance the low altitude transits while maintaining a safe distance from the surface. The team also plans to bring at least one of the ARTEMIS spacecraft into the near-terminator exospheric region in conjunction with NASA's upcoming Lunar Atmosphere and Dust Environment Explorer (LADEE).

In early May 2012, P2 will execute a short series of maneuvers to slow its spin rate down to P1's rotational rate, with a goal of matching their spin synchronized data collection volumes and enhancing science operations. This maneuver sequence is being designed using a new spin control maneuver scheme that will not only spin the spacecraft down, but provide a beneficial attitude change to further mitigate the post-LOI thermal conditions mentioned above and to take the place of an altitude control maneuver.

VIII. Conclusion

After 170 maneuver events and several unexpected occurrences ARTEMIS is safely in lunar orbit. The trajectory was very challenging—but exceedingly rewarding—to implement. Many of the challenges encountered by the ARTEMIS mission were a result of taking two spacecraft created for Earth orbit into regions of space not encompassed in their original design. It is hoped that the experiences of the ARTEMIS team will help further the goals of future missions to the Moon and that the navigation data collected by the first spacecraft ever to continuously orbit Earth-Moon libration points will be of benefit to continued exploration of that interesting region of space.

Appendix A
Acronym List

ARTEMIS	Acceleration, Reconnection, Turbulence, and Electrodynamics of the Moon's Interaction with the Sun
ATS	Absolute Time Sequence
BGS	Berkeley Ground Station
DSM	Deep Space Maneuver
DSN	Deep Space Network
EFI	Electric Field Instrument
EM L_1	Earth-Moon Libration point 1
EM L_2	Earth-Moon Libration point 2
ESA	Electrostatic Analyzer
FDF	GSFC's Flight Dynamics Center
FDC	Fault Detection and Correction
FGM	Flux Gate Magnetometer
GOTS	Government Off-the-Shelf
GSFC	NASA' Goddard Space Flight Center
JPL	Jet Propulsion Laboratory
LOI	Lunar Orbit Insertion
NEN	Near Earth Network
ORM	Orbit Raise Maneuver
P1	ARTEMIS Probe 1 (Also known as THEMIS B)
P2	ARTEMIS Probe 2 (Also known as THEMIS C)
PRM	Period Reduction Maneuver
RCS	Reaction Control System
SCM	Search Coil Magnetometer
SKM	Stationkeeping Maneuver
SSA	Sun Sensor Assembly
SSL	University of California, Berkeley's Space Sciences Laboratory
TCM	Trajectory Correction Maneuver
THEMIS	Time History of Events and Macroscale Interactions during Substorms
USN	Universal Space Network, Inc.

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