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Navigating THEMIS to the ARTEMIS Low-Energy Lunar Transfer Trajectory

Daniel Cosgrove[1], Sabine Frey[2], Jeffrey Marchese[3], Brandon Owens[4], Swapan Gandhi[3], and Manfred Bester[5]
Space Sciences Laboratory, University of California, Berkeley, CA, 94720

David Folta[6], Mark Woodard[7], and Dennis Woodfork[8]
NASA Goddard Space Flight Center, Greenbelt, MD, 20771

THEMIS – a NASA Medium Explorer (MIDEX) mission – is a five-spacecraft constellation launched in February 2007 to study magnetospheric phenomena leading to the aurora borealis. During the primary mission phase, completed in the summer of 2009, all five spacecraft collected science data in synchronized, highly elliptical Earth orbits. Both mission design and efficient navigation and flight operations during the primary mission resulted in appreciable fuel reserves. Therefore, an ambitious mission extension, ARTEMIS, became feasible. ARTEMIS involves transferring the outer two spacecraft from Earth to lunar orbits where they will conduct measurements of the Moon's interaction with the solar wind and its crustal magnetic fields. Earth departure of these two spacecraft is accomplished by successively raising the apogees of their orbits until lunar perturbations become the dominant forces significantly altering their trajectories. This orbit raise sequence requires over forty maneuvering events, with multiple lunar approaches and fly-bys, before setting the two spacecraft on low-energy transfer trajectories to lunar orbit in February and March 2010. This paper addresses overcoming the navigation and operational challenges presented by the ARTEMIS mission, consisting of two spacecraft that were not designed to leave Earth orbits.

Nomenclature

ΔV	=	change in velocity
O	=	observed orbital change
C	=	calculated orbital change
N	=	number of pulses in a sun-synchronous maneuver event

I. Introduction

TIME History of Events and Macroscale Interactions during Substorms[1] (THEMIS), a NASA Medium Explorer mission was launched on February 17th, 2007 aboard a Boeing Delta-II rocket. THEMIS's prime mission was the study of the magnetospheric processes responsible for auroral sub-storm onset with a constellation of five identical spinning spacecraft, designated THEMIS A to THEMIS E. The THEMIS prime mission was completed in July, 2009. Two ambitious mission extensions, THEMIS-Low and Acceleration, Reconnection, Turbulence, and Electrodynamics of the Moon's Interaction with the Sun (ARTEMIS) were approved to continue the THEMIS legacy.

[1] Navigation Lead, UCB/SSL, 7 Gauss Way, Berkeley CA 94720-7450.
[2] Mission Design Lead, UCB/SSL, 7 Gauss Way, Berkeley CA 94720-7450.
[3] Flight Dynamics Analyst, UCB/SSL, 7 Gauss Way, Berkeley CA 94720-7450.
[4] Flight Dynamics Analyst, UCB/SSL, 7 Gauss Way, Berkeley CA 94720-7450, Member AIAA.
[5] Director of Operations, UCB/SSL, 7 Gauss Way, Berkeley CA 94720-7450, Senior Member AIAA.
[6] Aerospace Engineer, NASA GSFC, Building 11, Room S116, Greenbelt, Maryland 20771, Senior Member AIAA.
[7] Aerospace Engineer, NASA GSFC, Building 11, Room C002, Greenbelt, Maryland 20771.
[8] Aerospace Engineer, NASA GFSC, Building 11, Room C002, Greenbelt, Maryland 20771, Member AIAA.

While planning the THEMIS mission it became apparent that the P1 and P2 spacecraft had a limiting factor to their operational lifetime. Due to the low orbital inclinations expected during the final year of the nominal THEMIS mission THEMIS B and THEMIS C, also known as Probe-1 (P1) and Probe-2 (P2), respectively, were on a course to experience solar eclipses in excess of 8-hours in March of 2010. The spacecraft systems were designed to withstand eclipses, or shadows, of no longer than 4-hours. This shadow scenario would necessitate initiation of spacecraft re-entry requirements before it could impact the viability of spacecraft disposal.

By the THEMIS end-of-mission the spacecraft would not have enough available fuel to perform the required inclination change to avoid these long shadows and extend their operational lifetime. However, the THEMIS science team and the mission design team at the Jet Propulsion Laboratory at the California Institute of Technology (JPL) determined that the P1 and P2 spacecraft would have enough fuel to enter a lunar orbit through a low-energy transfer and thus ARTEMIS was born.

The mission will send the two outermost spacecraft of the constellation to lunar orbit through a low-energy transfer trajectory that capitalizes on gravity assists to minimize fuel expenditure. These trajectories will also make these spacecraft the first ever to be placed into Lissajous orbits around collinear lunar libration points[2] in the late summer and early fall of 2010 and culminate with stable lunar orbits in the spring of 2011.

The ARTEMIS baseline trajectory[3] can be divided into four phases. The Earth-Orbiting Phase, which transfers the spacecraft from the THEMIS end-of-mission orbits to lunar flyby events through a series of apogee increasing maneuvers. The Trans-Lunar Phase connects the flyby events to an insertion into a Lissajous orbit. The spacecraft will then spend approximately 6 months in the Lissajous Orbit Phase in which they orbit the Lunar Lagrange points #1 (LL1) and #2 (LL2). The final phase is the Lunar Orbit Phase where the spacecraft will be inserted into stable lunar orbits. The focus of this paper is the operational challenges and the results of the execution of the Earth-Orbiting Phase of the ARTEMIS trajectory.

The mission implementation and operation represents a joint effort between JPL, NASA Goddard Space Flight Center (GSFC), and the University of California, Berkeley, Space Sciences Laboratory (SSL).

II. Summary of Spacecraft Design

A. Instrument Suite

The THEMIS spacecraft are spin-stabilized probes with an approximate dry mass of 79 kg. 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. Spacecraft navigation and propulsion is further assisted by a body mounted Sun Sensor Assembly (SSA).

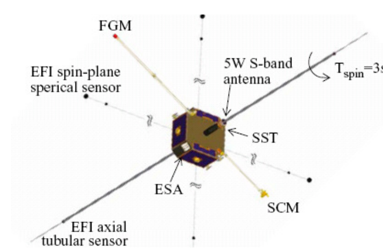


Figure 1. THEMIS spacecraft with deploy instruments.

The ARTEMIS mission will utilize these instruments to measure the lunar wake, the distance interactions of the Earth's magnetotail, and the undisturbed solar wind while maximizing opportunities for concurrent measurements with the other three THEMIS spacecraft still operating near the Earth.

B. Propulsion System

The Reaction Control System[4] (RCS) is a hydrazine blowdown system pressurized with helium. The RCS consists of two spherical propellant tanks, a single use pyro-actuated re-pressurization tank, and four 4.5 N thrusters.

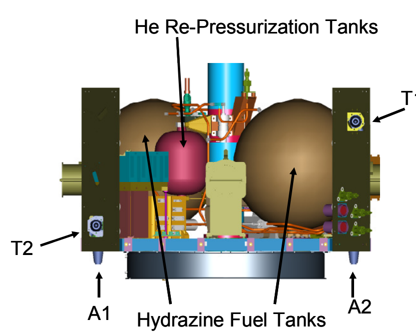


Figure 2. THEMIS RCS system and thruster locations.

Two of the thrusters, A1 and A2, point parallel to the spacecraft's spin-axis. When these two thrusters are fired in a continuous mode they provide a ΔV in the +Z spin-axis direction. These events are referred to as axial maneuvers. Attitude precession maneuvers are performed by firing either A1 or A2 in a sun-synchronous pulsed-mode.

The other two thrusters, T1 and T2, are mounted in a radial direction. When T1 and T2 are fired simultaneously in a sun-synchronous pulsed-mode these thrusters provide ΔV in a radial direction, perpendicular to the spacecraft's spin-axis. Firing either T1 or T2 alone, with thrust pulses phased 180 degrees from each other provides control over the spacecraft's spin rate while minimizing torque on the EFI wire booms.

By combining the axial and radial maneuvers into a series of burns executed in short succession the spacecraft can thrust in any direction in the southern ecliptic hemisphere without performing an attitude precession to point the thrusters in a specific direction. However, there is a small vectorization ΔV penalty for performing maneuvers in this manner, but the ΔV cost incurred by performing an attitude precession is prohibitively higher due to the spacecraft's large angular momentum.

All types of sun-synchronous pulsed maneuvers require accurate measurements of the sun's phase relative to the spacecraft and an accurate measurement of the spacecraft's current spin rate. This necessitates that any of the pulsed maneuver types be performed in full sunlight.

Maneuver execution commands are provided to the spacecraft through an Absolute Time Sequence (ATS) table, which contains most maneuver associated commands and operations. However, as a safety measure the propulsion bus that controls thruster actuation may only be enabled through a command sent directly from ground operations personnel.

The hydrazine fuel load at the time of luanch was 48.780 kg for P1 and 48.810 kg for P2. By the start of the ARTEMIS mission P1 had consumed an estimated 34.233 kg of fuel performing 72 maneuvers while P2 had consumed 27.760 kg of fuel executing 62 maneuvers. Table 1 summarizes the expended fuel and ΔV, estimated remaining fuel load and ΔV, and the total number of maneuver operations on each spacecraft.

Table 1. THEMIS end-of-mission fuel load and total maneuver count.

	P1	P2
Initial Fuel Load [kg]	48.780	48.810
Expended Fuel [kg]	34.233	27.670
Remaining Fuel [kg]	14.547	21.140
Total Expended ΔV [m/s]	712.162	552.381
Remaining ΔV [m/s]	307.195	449.931
Attitude Precessions	23	10
Spin Rate Changes	22	20
ΔV Maneuvers	27	32
Total Executed Maneuvers	72	62

The two hydrazine fuel tanks are serviced by two independent heater systems to maintain fuel temperatures within a range of 13°C to 21°C. The on-off cycling of these services is controlled automatically by thermostats that can not be activated by ground command. The fuel tank temperature and pressure can have a significant impact on maneuver performance when thermal conditions assumed during maneuver preparation do not match the observed conditions at execution[5].

III. Ground Systems Overview

While improving the THEMIS ground systems and operational environment to incorporate the required changes for successfully implementing the ARTEMIS mission great care was taken to reuse many of the working tools and systems already in-place. Only making incremental changes to these tested systems would ensure the smallest possible risk to established operational protocols and the spacecraft themselves[6-8]. There were several key systems that required more substantial improvements discussed in the following sections.

A. Orbit Determination

During the THEMIS nominal mission orbit determination (OD) was performed by the operations team at SSL utilizing two-way Doppler measurements from a selection of NASA Ground Network (GN) and United Space Net (USN) ground stations along with the Goddard Trajectory Determination System[9] (GTDS), a NASA/GFSC government off-the-shelf (GOTS) software package. To account for the increased distances between the Earth and the spacecraft involved in the ARTEMIS mission the JPL Deep Space Network (DSN) was integrated into active operations in early 2009[10]. The inclusion of the DSN network also provided the opportunity to activate the ranging capability of the spacecraft's communication system, thus generating additional data for the OD analysis process. Day-to-day OD activities for this first phase of the ARTEMIS mission continue to be supported by SSL with critical period support during lunar flyby events provided by the Flight Dynamics Facility (FDF) at GSFC.

B. Attitude Determination

Spacecraft attitude is calculated using the Multi-mission Spin-Axis Stabilized Spacecraft[11] (MSASS), a NASA/GSFC GOTS software package for attitude determination (AD). In its original configuration this system utilized data from the SSA and FGM instruments, particularly the high magnitude magnetic field measurements from near perigee, and Spinning-Spacecraft Kalman Filter[12]. However, the usefulness of the magnetic field data and the reliability of the Earth magnetic field models for AD decrease dramatically as the perigee attitude increases.

When the spacecraft leave the near Earth region data from the FGM will become unusable for AD. This necessitated a new method that uses only sun vector measurements from the SSA. To accomplish this task a new module for the MSASS system was developed by J. Hashmall, at A.I. Solutions, Inc. under a NASA Mission Operations and Missions Services contact. The new system called Fuzzycones is a maximum likelihood algorithm for combining SSA data into vectors and angle measurements from the spin-axis into an orientation of that axis[13].

C. Finite Maneuver Planning

Finite maneuver planning is performed by SSL using the General Maneuver Program[14] (GMAN), a NASA/GFSC GOTS software package. GMAN is implemented in the operations process as a subsystem to the Mission Design Tool (MDT), an extensive suite of Interactive Data Language (IDL) programs developed in-house at SSL. These software packages were used in this configuration for the THEMIS mission and planned over 300 successfully executed maneuvers. The finite maneuver planning system provides the capability to simulate long series of maneuvers quickly while dynamically utilizing in-flight performance data[15].

During the flyby targeting, and future phases of the mission, maneuver targeting will be performed at GFSC using a combination of the General Mission Analysis Tool[16] (GMAT), a NASA/GFSC GOTS software package, and STK/Astrogator[17] developed by Analytical Graphics, Inc. (AGI). Final maneuver simulation and operational preparations will continue at SSL using the targeting data provided.

IV. ARTEMIS Orbit Raise Maneuver Sequence

At the beginning of the ARTEMIS Earth-Orbit Phase P1 occupied the largest of the THEMIS orbits with a period of 4-days, a perigee of 1,936 km, and an apogee of 195,703 km. P2 held the second largest orbit with a period of 2-days, a perigee of 3,201 km, and an apogee of 117,438 km. Figure 3 illustrates the general configuration of these orbits. The spacecraft spin-axis orientations at this time were approximately 8 degrees from an ecliptic south attitude and with a spin period of approximately 3 seconds.

The Earth-Orbit Phase of the ARTEMIS trajectory had two goals to accomplish; raise the apogee of both spacecraft through a series of orbit raise maneuvers (ORM) to near lunar altitudes and to target the lunar gravity assists needed to transition into the Trans-Lunar trajectories through a series of flyby targeting maneuvers (FTM). Both ORM and FTM events are designated serially from 1 (ORM1, ORM2, etc).

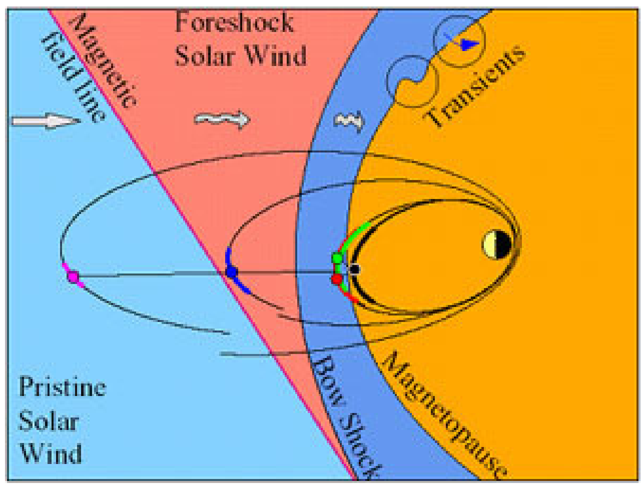


Figure 3. THEMIS end of mission orbit configuration. P1 is colored magenta and P2 is blue. The red, green and black are the other three THEMIS spacecraft.

A. Maintaining Orbital Phasing with ORM Baseline

The goal of the ORMs is to increase the altitude of apogee and the orbital period. The ORM sequence for both spacecraft can be highly sensitive to phasing with the mission baseline. As ORM maneuvers are executed navigation and performance errors could accumulate in the spacecraft's phase and prevent proper targeting of the flyby events. Subsequent maneuvers in the sequence would have to be adjusted to remove these errors. This issue is analogous to challenges faced during the operation of the THEMIS mission where five spacecraft were required to maintain proper phasing with ground based observatories located in central Canada[18].

As a related effect to this issue ORM maneuvers could not be fixed in time. As the difference in orbital period between the baseline and actual trajectories grew the locations of the ORMs needed to move to maintain efficient use of ΔV and to correct the accumulating error.

The strategy employed for ARTEMIS was to adjust downstream ORM target states to correct for the errors introduced upstream. For example, if ORM1 underperformed the spacecraft's orbital period would be smaller than the desired target orbit. When calculating and preparing ORM2, the timing of this event would be adjusted such that the maneuver would occur at the same mean anomaly (MA) as originally intended, but at a different (i.e. earlier) time. Furthermore, ORM2's target apogee would be adjusted upward such that any accumulated phase difference between the actual trajectory and the baseline would be reversed or corrected by ORM3. ORM3's target state would be adjusted in the same manner to correct for errors introduced during ORM2's execution.

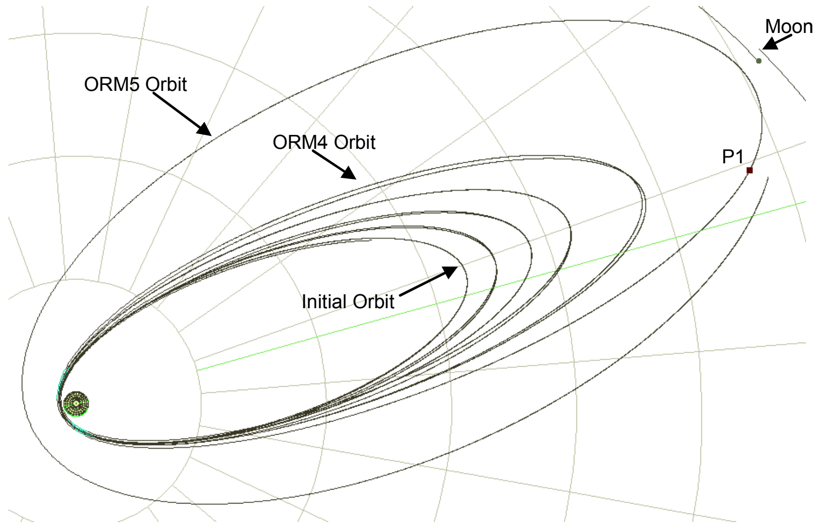


Figure 4. P1 ORM Sequence in Earth Inertial coordinates viewed from the North. The grid represents the equatorial plane.

Table 2. Baseline and Actual ORM Cost.

	Baseline ORM Cost Magnitude [m/s]	Actual ORM Cost [m/s]
P1	96.698	99.168
P2	230.370	231.455

B. P1's Orbit Raise Maneuver Sequence

P1's orbit raise maneuver sequence consisted of five radial thrust ORMs starting on August 1st, 2010. Due to the time of year, P1's orbital alignment was phased relative to the sun such that the majority of its near perigee trajectory was shadowed by the Earth. The requirement that sun-synchronous maneuvers be performed in full sunlight necessitated that all of P1's ORM maneuvers be split into two burn events that bracketed these shadows. The split ORMs are designated such that the pre-shadow maneuvers are labeled A and the post shadow maneuvers are labeled B (e.g. ORM1A and ORM1B).

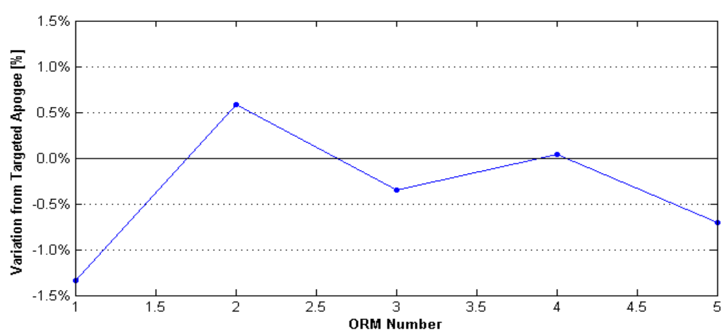


Figure 5. P1 ORM variation from achieved to targeted apogee.

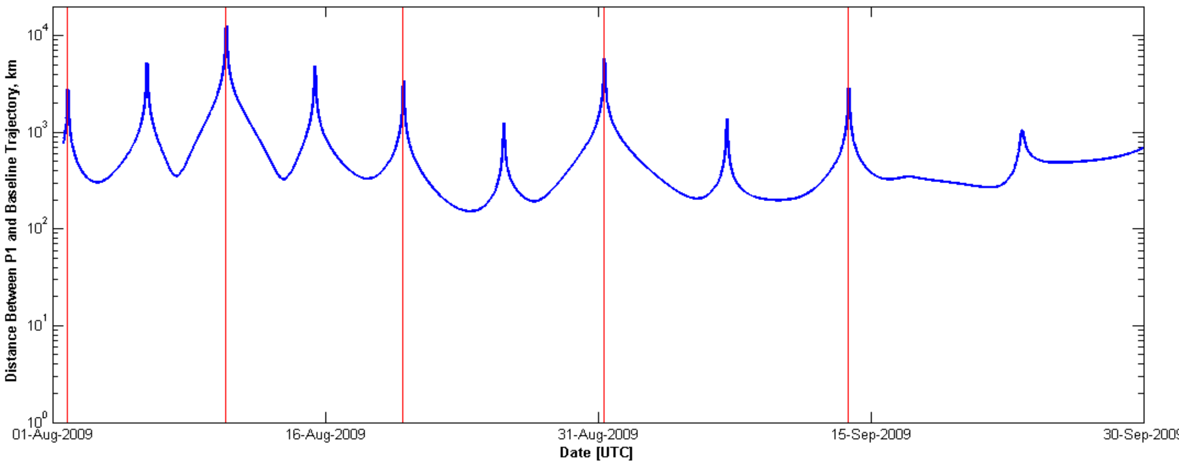


Figure 6. Distance between P1's actual path and the baseline trajectory. The red vertical lines indicate executed ORMs, starting with ORM1.

Figure 4 is an illustration of P1 trajectory from July 24th, 2009 to September 27th, 2009 as viewed from North of the equatorial plane; the direction of travel is counterclockwise. The locations of the ORMs are near perigee and the inner most orbit is the initial state at THEMIS end-of-mission. The Moon and its trajectory are shown in the upper right hand corner. After ORM5 P1 experienced significant lunar perturbations that further increased apogee and provided a large increase in its perigee altitude.

The total P1 ORM sequence used 99.168 m/s of ΔV, Table 2., which was 2.6% more than predicted in the baseline trajectory due to performance errors. Figure 5 shows these performance errors as calculated by Eq. (1) where O is the observed change in apogee and C is the calculated change in apogee.

$$error = \frac{O - C}{O} \tag{1}$$

Figure 6 shows the distance between P1's observed trajectory and the baseline. The sharp spikes correspond to passages through perigee where variations in distance are magnified due to relative phase differences. The baseline trajectory was generated using an orbit solution created many months before the start of the ORM sequence. Deviations between this orbit estimate and the actual spacecraft state on August 1st, 2009 is the cause of the initial offset in figure. It is clearly seen that after the execution of ORM1 the relative distance begins to increase until ORM2. This maneuver was retargeted to adjust for and remove this error, which is the source of the decreasing trend until ORM3. The overall variation in distance at apogee was bounded within a range 400 km to 1000 km.

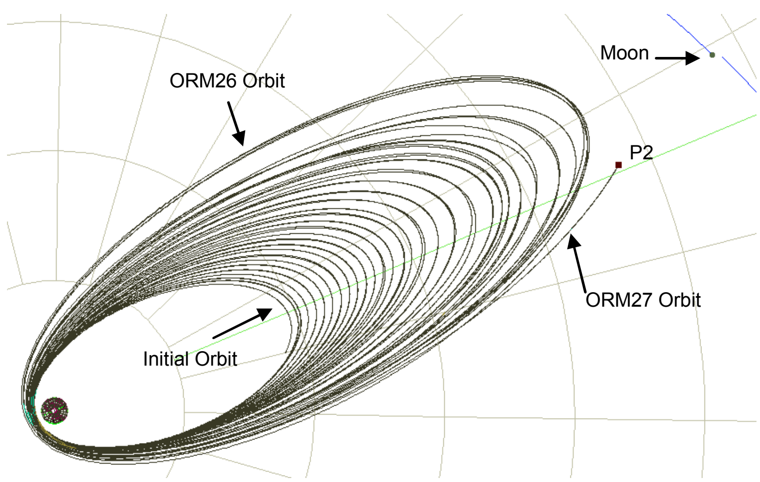


Figure 7. P2 ORM Sequence in Earth Inertial coordinates. The grid represents the equatorial plane.

C. P2 Orbit Raise Maneuver Sequence

The P2 ORM sequence is more extensive, consisting of 27 ORM events, due to the significantly smaller orbital size at the start of the ARTEMIS Earth-Orbit Phase. The spacecraft was also experiencing Earth shadows near perigee, which necessitated the segmenting of ORM3 through ORM14 into pre- and post-shadow maneuvers, increasing the total number of burn events to 39.

P2's trajectory from July 21st, 2009 through March 1st, 2010 is shown in Figure 7.

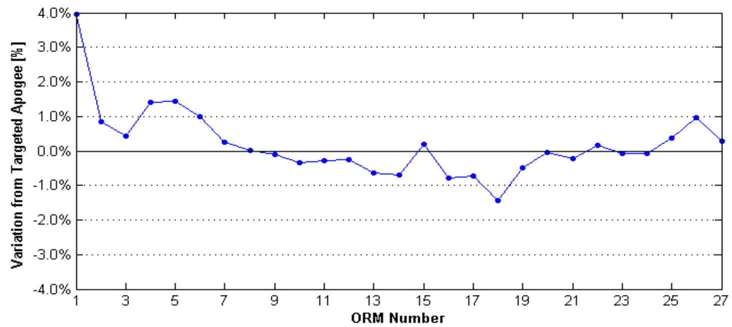


Figure 8. P2 ORM variation from achieved to targeted apogee.

A larger than expect performance error was observed on the execution of ORM1. After adjusting the finite maneuver model calibration process with the data gathered from ORM1 typical performance errors dropped to ±1.5%. Figure 8 illustrates P2 ORM performance as calculated by Eq. (1).

The distance between P2's trajectory and the mission baseline is shown in Fig. 9. The sharp spikes in the trend are passages through perigee where variations in distance are magnified due to relative phase differences. The observed distances at apogee remained bounded within a range of 100 km to 1000 km. The sharp drop in deviations on approximately February 26th, at the location of ORM27, was caused by a re-optimization of the baseline trajectory to better avoid the upcoming shadow. Since the baseline was changed at this point the variation between

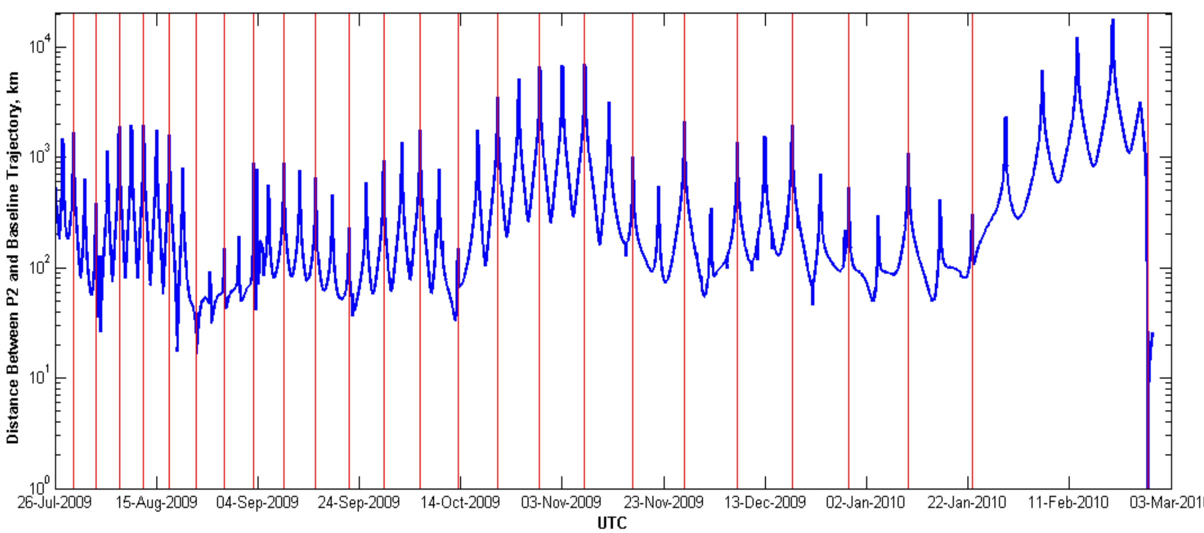


Figure 9. Distance between P2's actual path and the baseline trajectory. The red vertical lines indicate executed ORMs.

the observed and planned path was reduced.

V. ARTEMIS Flyby Targeting Maneuver Sequence

With the completion of the ORM sequence the two ARTEMIS spacecraft were on course to navigating the lunar flybys that will result in their transfer to the Trans-Lunar phase of the mission. To complete the targeting of these flyby events both spacecraft required a series of flyby targeting maneuvers (FTM) to properly connect the final ORM orbital states to the trans-lunar injections.

A. P1 Flyby Targeting Maneuver Sequence

P1's original baseline trajectory had three deterministic FTM maneuvers scheduled, which would target the three lunar flyby (FB) events listed in Table 3. After the completion of the P1 ORM sequence the deterministic FTMs were retargeted by GFSC to achieve the proper B-plane conditions at FB2 and accounting for deviations from the baseline in the final ORM5 orbit state. Furthermore, after the retargeting the number of FTMs was reduced to a series of two burns with the FTM2 maneuver being canceled. FTM1 was a vectorized maneuver with a combined axial and radial thrust component totaling 0.825 m/s. FTM3 was a radial maneuver of 6.300 m/s.

On December 8th, 2009 P1 experienced it first lunar flyby event, passing within 18,576 km of the Moon and receiving a significant inclination change as a result. Figure 10a illustrates the P1 FB0 event.

While FB0 was successful the achieved state differed from the desired baseline. To account for the new orbital conditions, and to provide the control needed to reach the ultimate goal of a proper trans-lunar transfer through the sensitive path of FB1 and FB2, called the "backflip" by operators, it became apparent that a series of addition trajectory correction maneuvers (TCM) were needed to provide the ability to adjust the spacecraft's course during the flyby period to minimize maneuver execution errors and remove navigation errors.

Table 3. P1 Lunar Flyby Events.

Event	Flyby Time [UTC]	Flyby Distance [km]
FB 0	2009-Dec-08 01:19:24	18576.693
FB 1	2010-Jan-31 08:19:55	14131.836
FB 2	2010-Feb-13 09:54:37	5020.054

Three TCM maneuvers where executed to target through FB1 to the desired B-plane targets of FB2. A 1.940 m/s TCM1 was executed on December 14th, 2010 followed by a 1.481 m/s TCM2 on January 15th, 2010 and a 0.323 m/s TCM3 on January 24th, 2010 consuming an estimated 3.744 m/s of the 10 m/s allocated in the ΔV budget to cover trajectory corrections from Earth orbit through Lissajous insertion.

FB1 occurred on January 31, 2010 and marked the start of the P1 backflip, where the spacecraft traveled under the Earth and lunar orbital plane, meeting the Moon on the opposite side of its orbit roughly two weeks later at FB2 (see Figure 10b). Due to the intense sensitivity of this series of events TCM4, a 0.119 m/s maneuver was executed approximately 2 days after FB1 to perform the final adjustment before FB2. FB2 was achieved with a B-plane magnitude error of -5.5 km and a B-plane angle error of 4.3×10^{-3} degrees and P1 is now safely beginning its long journey through the Trans-Lunar transfer.

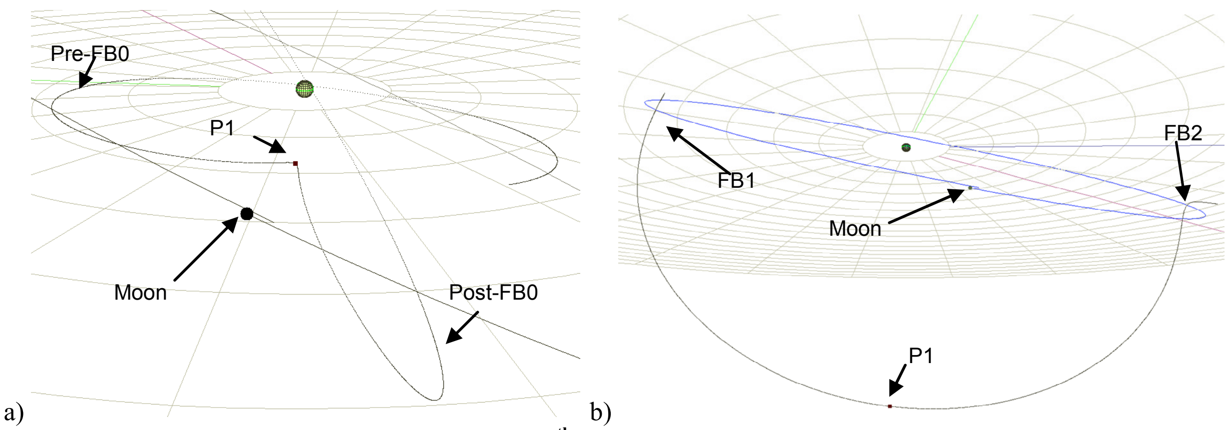


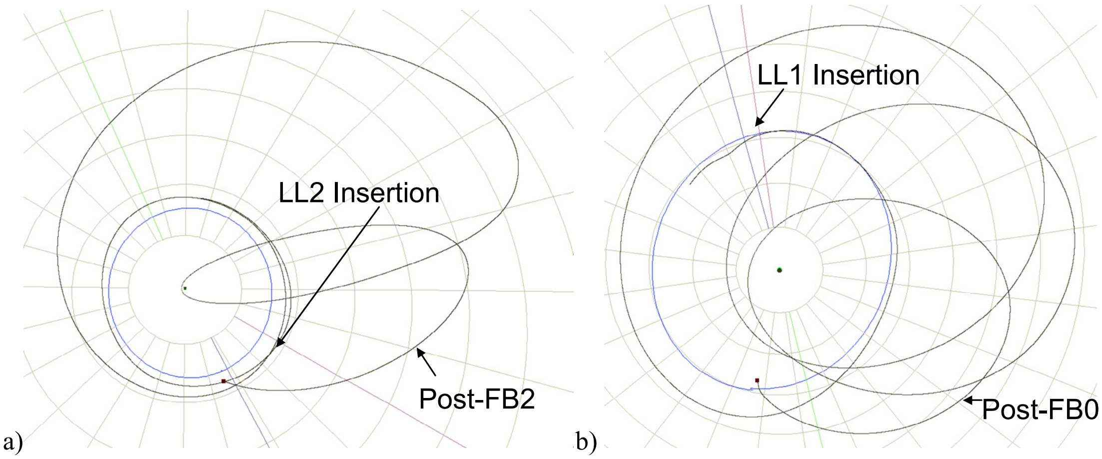
Figure 10. a) P1 Flyby 0 Event, December, 8th, 2009. b) P1 backflip, viewed from below the equatorial plane, halfway between FB1 and FB2.

B. P2 Flyby Targeting Maneuver and Shadow Deflection Sequence

P2's path to the Trans-Lunar transfer trajectory only requires one lunar flyby event. This flyby will not occur until March 28th, 2010, after the long Earth shadow event on March 22nd, 2010. Initial estimates of the shadow duration from the baseline trajectory were in excess of 10 hours. Because previous operational experience has shown that the spacecraft will remain power positive when illuminated by at least 50% sunlight it was determined that a series of two shadow deflection maneuvers (SDM1 and SDM2) between the completion of the ORM sequence and the FTM maneuver could remove the umbra and significantly reduce the penumbra to a survivable durations. At the time of writing the SDM1 maneuver has been completed and analysis of the shadow profile shows that the umbra will be completely avoided. Furthermore, P2 is also on target for a successful completion of its lunar flyby.

VI. The ARTEMIS Trans-Lunar Phase

The Trans-Lunar Phase begins for both ARTEMIS spacecraft upon the conclusion of their lunar flyby events. P1 and P2 will reach maximum ranges of 1.5 million km and 1.2 million km respectively. P1 will execute one deterministic maneuver in deep space and an insertion burn resulting in a LL2 Lissajous orbit on August 23rd, 2010.



a) b)

Figure 11. a) P1 Trans-Lunar Phase. b) P2 Trans-Lunar Phase. Lunar orbit is illustrated in blue.

P2 will be executing two maneuvers in deep space, along with its insertion burn, which will result in a LL1 Lissajous orbit on October 22nd, 2010. Figure 11 illustrates the planned trajectories of both spacecraft.

VII. Lessons Learned During the ARTEMIS Earth-Orbit Phase

Executing the ORM maneuver sequences with minimal errors was of critical importance to maintaining the proper orbital phasing to reach the lunar gravity assist events in early 2010. Even one missed thrust event would have required a complete redesign of the orbit raise process and would have been prohibitively costly to the project. Several adjustments to ground operations were made to ensure the highest possible maneuver execution success rate with the smallest possible errors.

A. Propulsion Bus Enabling Through Ground Command

Typical operations for THEMIS required enabling the propulsion bus system no later than burn start minus 30 seconds. The failure to complete this activity in time resulted in missing 4 of roughly 300 total maneuver events during the course of the original THEMIS mission. The causes varied from communication failures to ground personnel scheduling issues with 3 of the events occurring in the first few months of on-orbit operations.

With the complexity of the ARTEMIS trajectory it quickly became apparent that a single missed maneuver event would pose a significant risk to the success of the mission. Possibly requiring a complete trajectory redesign or resulting in increased health and safety risk to the spacecraft when attempting to avoid large solar eclipse events. Planning of extensive contingency events was also difficult since the required effort in exploring the probable outcomes would greatly exceed the available project resources.

However, the missed maneuver events during the THEMIS mission could have been avoided if the spacecraft's propulsion system was enabled many hours before the event instead of within minutes. This would allow the ability for operators to schedule multiple ground station assets to support the pre-maneuver systems checkout and propulsion bus enabling. The spacecraft would then execute the maneuver without further ground intervention with only a spontaneous operating system reset would be a probable point of failure in maneuver execution. Furthermore, after two and a half years of on-orbit operations the risk of an unexpected propulsive event during the additional time the system would be active was considered low and did not outweigh the benefits of significantly increasing the likelihood of a successful maneuver.

A new protocol was implemented where up to three ground station contacts are scheduled before any maneuver event to perform these critical activities, while maximizing the diversity of stations to reduce reliance on any single antenna and to increase opportunities to recover from an operating system reset. Contingency maneuvers are then planned for only the highest risk events (i.e. flyby targeting maneuvers) in the eventuality of unforeseen circumstances impacting operations.

Since the implementation of this operational adjustment there has not been a single missed maneuver out of the 75 ARTEMIS and THEMIS-Low executed events. However, on two separate occasions these additional precautions prevented possible missed maneuvers during P2's ORM sequence when command and telemetry links were not available at the time of burn start. Under the original THEMIS maneuver procedures the lack of communication during that critical period of time would have resulted in an unexecuted event.

This lesson is ultimately one of risk management and while not unique it exemplifies the need to continuously evaluate the relationship between mission goals and available resources even when established procedures have been previously successful.

B. Pulse Quantization for Small Sun Synchronous Maneuvers

Because the spacecraft only have 4.5 N thrusters for use in all maneuvers it quickly became apparent that executing very small ΔV corrections accurately would be challenging. The predominate mode of thrust execution during the ORM, FTM, and TCM events was radial sun-synchronous pulses sized to a 60 degree spin-arc. In this mode of thrust each pulse from the RCS system gives approximately 1.8 cm/s to 2 cm/s depending on the spacecraft and system pressure at the time of execution.

Only an integer number of pulses can be executed during each maneuver causing a ΔV quantization effect. If the mission design process uses a propulsion model that allows targeting of a finite amount of ΔV then the quantization effect would cause noticeable discrepancies in maneuver performance. Figure 12 illustrates the possible pulse selections for P1 FTM3 as viewed at the perigee following the maneuver event. There is no possible pulse selection that would place the spacecraft on the nominal trajectory. In this example the quantization, if not accounted for, could impart an execution error of approximately 0.154%.

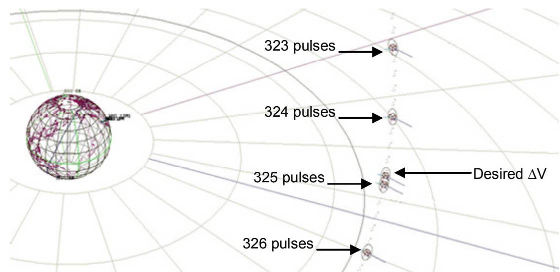


Figure 12. Maneuver performance spread for FTM3 due to pulse quantization.

As the size of a maneuver is decreased the maximum possible ΔV error due to pulse quantization would increase as shown in Eq. (2) where N is the total number of pulses.

$$\Delta V_{error} = \frac{1}{2N} \tag{2}$$

An additional step in the maneuver designing process was implemented to alleviate this systematic ΔV error. Once a maneuver candidate had been designed by the navigation team an initial simulation of the event is performed using the high fidelity GMAN maneuver model generating an estimated number of thrust pulses and the estimated achievable ΔV magnitude. This ΔV estimate is then used as an input to re-optimize the trajectory to account for expected performance deviation due to the pulse quantization effect.

VIII. Conclusion

The ARTEMIS extension to the THEMIS mission is a very ambitious endeavor that is expected to return a high scientific yield from the study of the Moon and it nearby environment. The planned layovers in LL1 and LL2 Lissajous orbits will be of further interest to the space exploration communities and the challenges faced in the implementation of their station-keeping[19] will be the subject of future publications.

To date, the implementation of the ARTEMIS trajectory has been on schedule. The P1 spacecraft is now well into its Trans-Lunar trajectory and P2 is on course for its March 28th, 2010 lunar flyby event. Both spacecraft will be arriving in their respective Lissajous orbits in August and October, 2010 respectively.

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