AAS 12-181

MASS EJECTION ANOMALY IN LISSAJOUS ORBIT: RESPONSE AND IMPLICATIONS FOR THE ARTEMIS MISSION

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On October 14, 2010, a 0.092 kg instrument sphere unexpectedly detached from the first spacecraft to ever orbit an Earth-Moon libration point. At the time of the anomaly, the spacecraft—which was one of two spacecraft dedicated to the ARTEMIS mission—had been in orbit about the L2 Earth-Moon libration point for less than two months and its operations team was still adjusting to the intricacies of Lissajous orbit operations. Nevertheless, a quick response was required to prevent the spacecraft from falling out of Lissajous orbit and set it up for several months of continued Lissajous orbit operations followed by an insertion into a retrograde lunar orbit. In this paper, the actions of this team and the changes to the spacecraft's flight characteristics are described for the benefit of future Earth-Moon libration point orbiting missions. Specifically, the authors detail how the response affected the prospects of preventing the spacecraft's fall out of Lissajous orbit. These details are then used to form recommendations on spacecraft design, attitude control, and tracking schemes that can mitigate the impact of mass ejection or errant thrust events in Lissajous orbit. Additionally, the authors present several of the dynamics phenomena observed during the event in the form of "homework" or "exam" problems that could be used for teaching, interviewing, or testing science and engineering students and professionals.

INTRODUCTION

Orbits around collinear libration points in "three-body systems"—particularly the L1 and L2 libration points—are among the most unstable orbits that have been attempted in spaceflight. Accordingly, discrete perturbations to these orbits (e.g., errant thrusts, mass ejection events, etc.) pose a greater risk of casting the spacecraft into trajectories from which a desirable orbit cannot be recovered. Therefore, special consideration should be given to reducing the risks posed by discrete perturbations through spacecraft design, selection of operating attitudes, and the frequency at which such spacecraft are tracked and allowed to downlink data.

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As it turns out, the first spacecraft to ever orbit an Earth-Moon libration point—P1 of the Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun (ARTEMIS) mission[2]—encountered the dangers associated with a discrete orbit perturbation on October 14, 2010 when a 0.092 kg sphere separated from one of its instrument booms and imparted a 5.67 cm/s ΔV on the spacecraft. The L1 and L2 points in the Earth-Moon system provide a particularly weak potential field for orbiting due to the relatively small mass and short orbit period of the secondary body relative to other bodies in the solar system, and thus the operations team had to respond to the event within days to prevent a significant deviation from the spacecraft's desired trajectory. The details of this event are described throughout this paper with the intent of informing discrete orbit perturbation risk mitigation efforts for future libration point orbiting spacecraft. The discussion expands on previous reports of the event[1] through providing a detailed timeline of the event and ensuing analysis, laying out the relevant equations used in the analysis, and applying the lessons of the event to design concepts for future Earth-Moon libration point orbiting spacecraft.

Description of the THEMIS/ARTEMIS Electric Field Instrument (EFI)

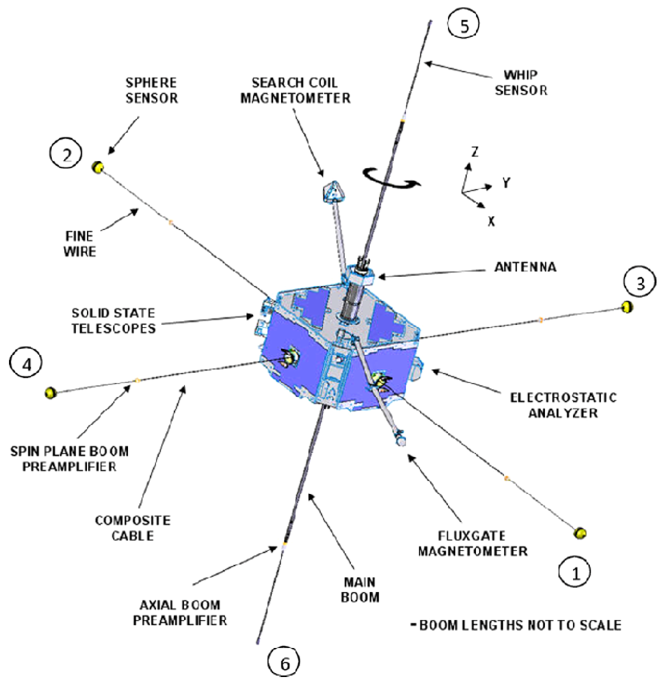

Figure 1. THEMIS/ARTEMIS Spacecraft Antenna and Boom Layout.[3]

The THEMIS/ARTEMIS spacecraft are all equipped with an instrument suite that consists of an Electric Field Instrument (EFI) mounted on two axial booms and four spin-plane wire booms that extend approximately 20 meters in ±Y direction and 25 meters in the ±X direction as shown in Figure 1.[3] The first 22 meters of the ±X booms are made out of braided Kevlar attached to a spool in the main spacecraft body on one end and a 0.043 kg pre-amplifier on the other end. The next 3 meters are made out of fine wire (weighing 1 g/m) attached to a spool in the pre-amplifier on one end and a 0.092 kg metallic sphere on the other end. The ±Y booms are similarly constructed, but the braided Kevlar section is approximately 5 meters shorter.

Description of the THEMIS/ARTEMIS fuel tanks

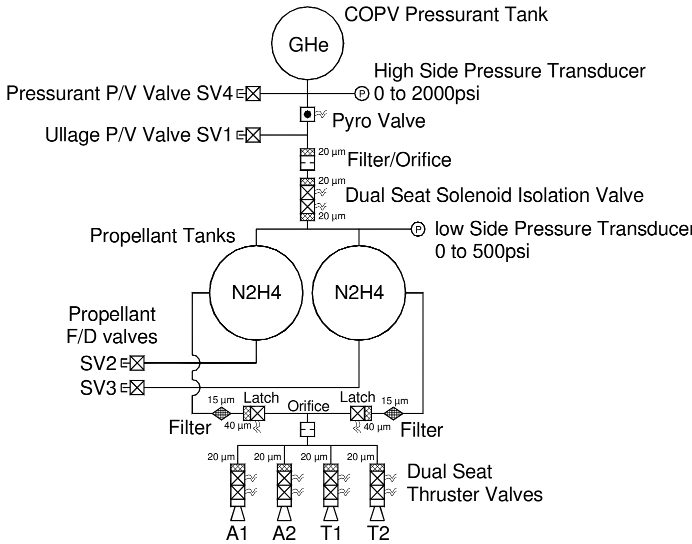

Figure 2. THEMIS/ARTEMIS Spacecraft Reaction Control Subsystem.[4,5]

The design of the monopropellant reaction control systems onboard each THEMIS/ARTEMIS spacecraft is depicted in Figure 2. The fuel, hydrazine (N_2H_4), is stored in two tanks linked up-stream of the fuel flow through a valve-less ullage line and downstream of the fuel flow through a valved fuel line. The fuel pressurant, helium (GHe), is stored in one Composite Over-wrapped Pressure Vessel (COPV) linked to the ullage line between the fuel tanks through a one-shot, pyro-actuated valve and pair of solenoid isolation valves. The tanks are offset equal and opposite distances from the geometric center of the spacecraft; the XYZ-coordinates for the centers of Tanks 1 and 2 are (0.2043m, -0.2043m, 0.321m) and (-0.2043m, 0.2043m, 0.321m), respectively.

Throughout THEMIS and ARTEMIS operations, the two latch valves of the fuel line have been left open while the solenoid isolation valves between the ullage line and pressurant tank are only opened for minutes at a time between maneuver events if the pressure differential is great enough to ensure mass flow from the pressurant tank. Thus, during a thrust operation, fuel from the two tanks combines downstream of two latch valves and passes through an orifice on its way out of the thrusters in use. At all other times, the fuel is free to flow from one tank to the other through the fuel line.

For further information on the THEMIS spacecraft reaction control system, see (References 4 and 5).

ANOMALY TIMELINE

Though the separation event and the transient dynamic conditions that it created lasted for only a few minutes, it would take months before the full impact of the event on the mission would be understood. The relevant events and data involved in this process are presented in the remaining paragraphs of this section.

The anomaly occurs

At 06:04:56 Coordinated Universal Time (UTC) on October 14, 2010, the instrument sphere and an estimated 3 meters of fine wire on the end of P1's –X EFI boom detach at the pre-amplifier near the crimp of the spool. The braided Kevlar line attaching the spacecraft to the pre-amplifier contracts slightly due to the release of tension in the line and spacecraft's spin rate increases by approximately 0.003 RPM. The change in mass properties of the spacecraft causes the spacecraft spin axis to shift primarily in the +X direction, which in turn causes fuel to migrate primarily from Tank 1 to Tank 2. Oscillations of the EFI booms and fuel migration between the two fuel tanks create transient accelerations on the spacecraft for several minutes. As later analysis has shown, after a few minutes, the wet spacecraft center of mass settles at XYZ-coordinates of (2.401 cm, 0.278 cm, 26.547 cm)—its location before the separation was (0.470 cm, -0.470 cm, 26.305 cm).

Through conservation of linear momentum, the separation event imparts an approximate ΔV of 5.67 cm/s on the spacecraft and 51.9 m/s on the sphere. Because the spacecraft is in Lissajous orbit around the L2 Earth-Moon libration point at the time of the anomaly, its trajectory is significantly affected by the ΔV, creating the need for an emergency stationkeeping maneuver. The sphere escapes the Earth-Moon system and enters a heliocentric orbit with a periapsis of 1.500×10^8 km and an apoapsis of 1.666×10^8 km.

At the time of the anomaly, the spacecraft is not in communication with any ground stations and therefore, there is no indication in the ARTEMIS Mission Operations Center (MOC) that an anomaly has occurred on the spacecraft.

Initial post-anomaly tracking data is collected

From approximately 12:45:00 UTC – 13:30 UTC on October 14, Deep Space Network (DSN) Station 45 tracks ARTEMIS P1. Data from the spacecraft is not downlinked to the ground due a power outage at the location of a data relay between the DSN network and the University of California, Berkeley (UCB) MOC. The tracking data is sent to UCB well after the daily orbit solutions have been run and it is automatically ingested into the tracking database.

Stationkeeping maneuver calibration reveals anomalous tracking data

At approximately 14:00:00 PT on October 14, a flight dynamics analyst performs a calibration run on P1's fifth Lissajous stationkeeping maneuver (P1 SKM 5), which occurred on October 9, 2010. The calibration process[6,7] involves the generation of an orbit solution through the Goddard Trajectory Determination System (GTDS) and several maneuver reconstruction iterations utilizing the General Maneuver (GMAN) software package. As the analyst runs through the procedure, the orbit solution converges, but the maneuver reconstruction iteration loop exceeds the maximum number of iterations and is automatically halted. This result surprises the analyst because the calibration runs from the two previous days had converged on reasonable values for the thruster scale factor. More detailed analysis of the orbit solution reveals that the DSN 45 tracking data produces significant residuals when compared to the orbit solution. Three possible conclusions can be drawn from this data: 1) the tracking data was corrupted by the spacecraft, ground station, or UCB, 2) the software for the calibration was somehow misconfigured, or 3) the spacecraft had experienced an unaccounted for perturbation to its orbit. Because the first two conclusions are more likely, the analyst closely examines his calibration software configuration and asks a DSN scheduler at UCB to file a discrepancy resolution request with DSN 45 before he leaves for the day. He also checks the spacecraft telemetry database for signs of an orbit perturbation, but telemetry from the time of the anomaly has still not been downlinked from the spacecraft.

More anomalous tracking data collected

From approximately 02:30:00 UTC to 03:20:00 UTC on October 15, the BGS ground station tracks the spacecraft. Because the link margin with the station is low, telemetry from the time of the anomaly is not downlinked, but two-way Doppler data is collected. After the pass, the tracking data is automatically ingested into the tracking database.

Daily orbit determination runs reveal anomalous tracking data from multiple ground stations

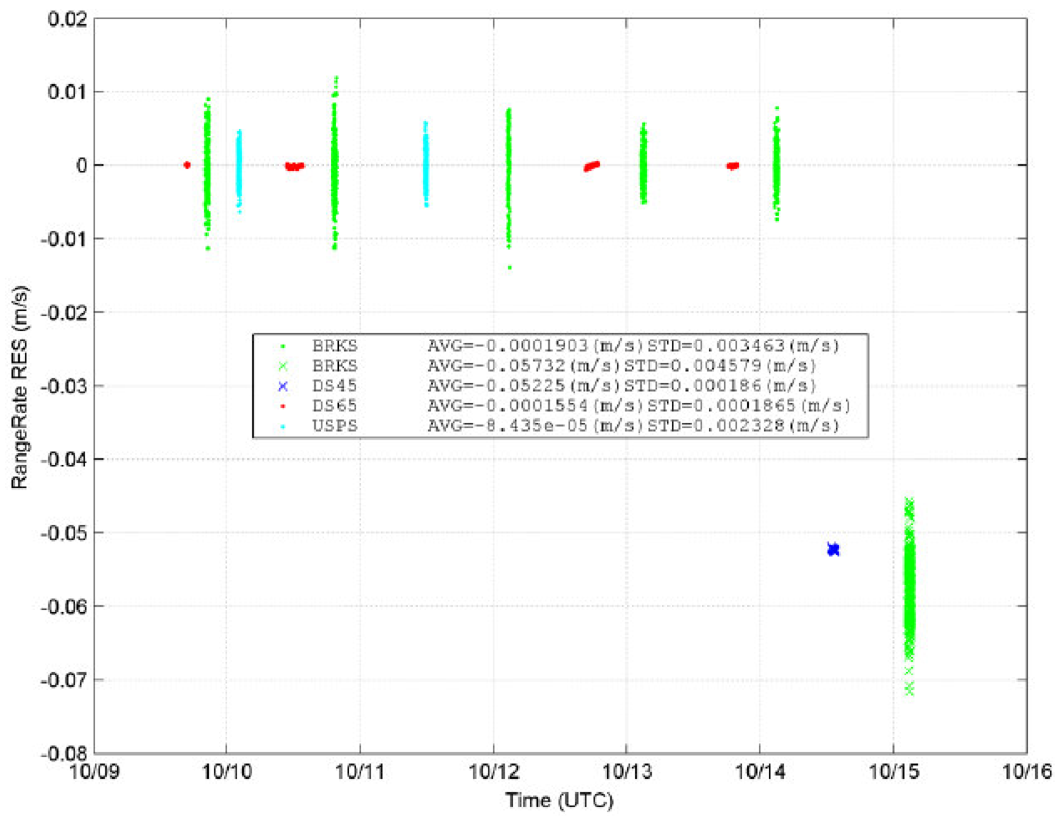

Figure 3. Range rate residuals for the GSFC FDF orbit solution generated on Oct. 15, 2010 (source: Patrick Morinelli, NASA GSFC).

By approximately 14:00 UTC on October 15, both the NASA Goddard Space Flight Center (GSFC) Flight Dynamics Facility (FDF) and UCB run orbit solutions with the new tracking data. The FDF notes significant residuals on both the DSN 45 and BGS tracking data and sends the range rate residual plot in Figure 3 to UCB. UCB also notes range rate residuals in its orbit solution. Because range rate residuals are observed in tracking data from two separate stations on separate networks, the leading theories for the cause of the anomaly are: 1) corruption of the tracking data due to a change in the spacecraft transmitter performance or 2) an unaccounted for perturbation to the spacecraft's orbit.

Spacecraft health and status telemetry indicates dynamic disturbance to spacecraft

Telemetry from the time of the anomaly is downlinked through DSN Station 65 during a contact from 17:00:00 UTC to 20:20:00 UTC on October 15. It is loaded into UCB's telemetry analysis database[8] shortly after the end of the contact and is analyzed immediately.

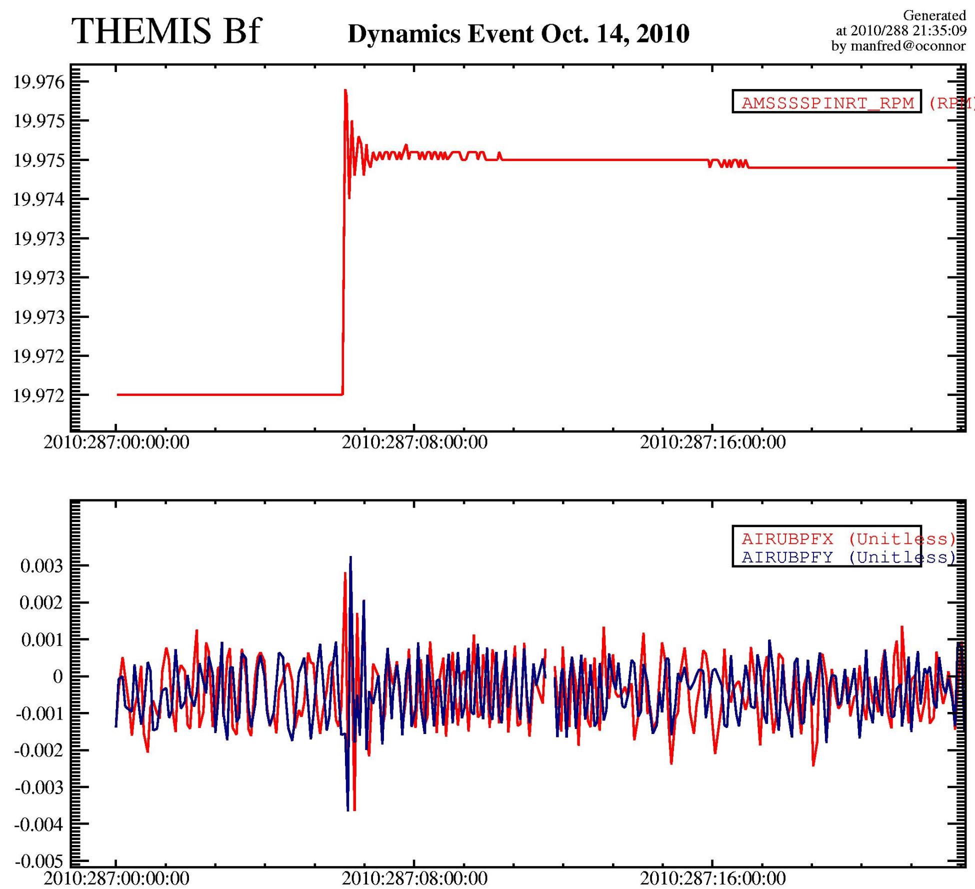

Figure 4. Spin rate (top panel) and accelerometer (bottom panel) data around the time of the anomaly (i.e., 06:04:56 UTC).

As shown in Figure 4 the change in spacecraft spin rate and accelerations measured by the spacecrafts accelerometers indicate that a dynamic disturbance has occurred on the spacecraft. After some analysis, the flight dynamics group speculates that a piece of the spacecraft (particularly the EFI booms) may have detached from the spacecraft and asks the instrument teams to evaluate the integrity of their instruments.

Doppler data indicates shift in spacecraft center of mass

High rate Doppler data is collected by the BGS station from approximately 22:25:00 UTC to 22:45:00 UTC. The modulation of the data indicates that the transmitter is coning around the spin axis. The radius of coning is approximately 2.4 cm. This number is used as the initial estimate of the shift of the spacecraft's center of mass.

Instrument data reveals degradation in EFI –X performance

Analysis of EFI data shows that the sensor potentials on the –X booms changed drastically after the anomaly while potentials on the other booms remained unchanged. This signature is consistent with the electrical severing of the –X sphere from the pre-amplifier along with some portion of the fine wire connecting it to that device.

The ΔVs imparted on the spacecraft and sphere are determined

From the conservation of linear momentum calculations (refer to the Analysis section), it is determined that the ΔV imparted on the spacecraft is approximately 5.49 cm/s to 5.67 cm/s depending on the amount of fine wire that separated along with the sphere. Trajectory analysts at GSFC use Satellite Tool Kit (STK) to propagate a pre- and an early post-anomaly orbit solution to the anomaly time and determine the observed ΔV. Their analysis shows that the imparted ΔV ranges from 5.65 cm/s to 5.78 cm/s depending on the targeting criteria used. This result suggests that the fine wire separated at the pre-amplifier (EFI measurements would later confirm that there was no fine wire remaining on the boom after the anomaly).

A contingency stationkeeping maneuver is executed

Typical stationkeeping maneuvers are executed on or near the spacecraft's transit across the Earth-Moon line, which would not occur again until October 25th, 2011. The magnitude of an event at that time rose to 2.95 m/s, a large expenditure when compared to the normal costs of less than 25 cm/s per event. To avoid this increase the operations team moves forward with a contingency stationkeeping event. A 15.2 cm/s maneuver is executed at 13:59:50 UTC on October 18, 2010. The spacecraft, which normally spins down during a side-thrust maneuver, experiences a drop in its spin rate that is nearly 15 times greater than normal due to the change in the spacecraft center of mass and moments of inertia.

Fuel tank thermal cycles indicate expected shift in fuel mass

After the anomaly, the rate of fuel heating in Tank 1 during electric heater cycling increased while the rate of fuel heating in Tank 2 decreased. This result is consistent with a shift in fuel mass from Tank 1 to Tank 2. After collecting several weeks of thermal cycling data, analysts are able to use a thermal gauging mass estimation process[4], to determine that approximately 1.25 kg of fuel shifted from Tank 1 to Tank 2.

Spacecraft mass properties are recalculated and updated in GMAN

After several weeks of iteration by analysts at UCB, it is determined that as a result of the anomaly, the spacecraft's dry center of mass moved approximately in XYZ-coordinates from (0.300 cm, -0.300 cm, 25.9 cm) to (3.303 cm, -0.2975 cm, 25.87 cm). These numbers, along with the spacecraft's dry moments of inertia, are updated in the spacecraft data file used to run GMAN. Due to the dependence of the fuel mass distribution among the two tanks on the spacecraft center of mass, an iterative routine is developed over this same period of time to allow GMAN to converge on a wet spacecraft center of mass. In this routine, GMAN initially estimates the wet spacecraft center of mass based on the dry center of mass, the total fuel mass, and an assumed equal distribution of fuel mass among the tanks. With each successive iteration the fuel mass distribution is updated, which then leads to an update in the center of mass location. The use of these numbers and the iterative routine causes GMAN to converge on a wet spacecraft center of mass location and fuel mass distribution that is in close agreement with both the observed Doppler modulation and fuel tank thermal cycling data.

Lunar orbit insertion

On June 6, 2011, ARTEMIS P1 effectively exits Lissajous orbit of Earth-Moon L1—it had transferred there from L2 in January, 2011—by performing its last stationkeeping maneuver. Twenty-one days later, on June 27, 2011, the spacecraft is inserted into a highly elliptical lunar orbit. The spacecraft spins down by 2.56 RPM over the course of the maneuver due to the shift in the spacecraft's center of mass and change in moments of inertia—for comparison, P2 only spun down by 0.823 RPM during its lunar orbit insertion. While the stationkeeping maneuvers following the anomaly helped the flight dynamics analysts characterize and predict the spin down fairly well, an unrecognized peculiarity in GMAN led to a roughly 6% overburn on the insertion. In GMAN, the thruster I_{SP} drops as the spacecraft spin rate drops throughout a maneuver segment, causing GMAN to underestimate thruster performance throughout the maneuver segment. Thus, the unprecedented spin down throughout the lunar orbit insertion ultimately led GMAN to underestimate the maneuver performance and prescribe a maneuver that was roughly 6% longer than necessary.

Second anomaly occurs

At approximately 21:28:00 UTC on August 27, 2011, another spin rate increase occurs on the spacecraft. Data from the –X EFI indicates that most (or perhaps all) of the pre-amplifier has detached from the –X boom. The ΔV imparted on the spacecraft is approximately 1.8 cm/s, however, because the spacecraft is in a much stronger potential field, its trajectory is not altered in a significant way and thus there is no need for a contingency maneuver. In the following weeks, new moments of inertia and a new dry spacecraft center of mass are determined for use in GMAN's spacecraft data file.

ANALYSIS

The equations behind several of the analyses performed throughout the anomaly timeline are derived in this section.

The ΔV imparted on the spacecraft

At the time of the anomaly, the spacecraft was rotating at a rate θ—measured in radians per second—and the sphere had a rotation radius, r.[*] Thus, the tangential velocity of the sphere is given in Eq. (1).

$$V_T = r\theta \tag{1}$$

If we assume that the sphere and fine wire separated from the spacecraft cleanly (i.e., without a significant loss or addition of energy from another colliding body or the severing of the line), conservation of linear momentum tells us that the momentum of the spacecraft and sphere before separation is equivalent to the momentum of the spacecraft and sphere after the separation. This relationship is expressed in Eq. (2) where the reference frame is defined such that the tangential velocity of the sphere is parallel to the x-axis. Additionally, the velocity of the sphere (and accompanying fine wire) after the separation will be the tangential velocity of the sphere added to the initial velocity of the spacecraft, as expressed in Eq. (4).

[*] The notations used for equations in this paper are listed in Notation section.

$$P=(m_1+m_2)\overline{V}=m_1\overline{V}_{1f}+m_2\overline{V}_{2f} \tag{2}$$

$$\overline{V}_{1f}=\Delta V_1\hat{x}+\overline{V} \tag{3}$$

$$\overline{V}_{2f}=V_T\hat{x}+\overline{V} \tag{4}$$

Substitution of Eq. (3) and Eq. (4) into Eq. (2) yields Eq. (5). By rearranging Eq. (5) and substituting Eq. (1) into it, one gets Eq. (6), which relates the ΔV imparted on the spacecraft to the mass ratio of the spacecraft and sphere, the rotational radius of the sphere, and rotational velocity of the spacecraft.

$$(m_1+m_2)\overline{V}=m_1\Delta V_1\hat{x}+m_2V_T\hat{x}+(m_1+m_2)\overline{V} \tag{5}$$

$$\Delta V_1=-\frac{m_2V_T}{m_1}=-\frac{m_2r\theta}{m_1} \tag{6}$$

At the time of the anomaly, the spacecraft's rotation rate was 2.092 radians/s and the sphere's radius of rotation was 24.8 m. The mass ratio of the spacecraft and sphere (plus fine wire) depends on the sever point of the fine wire and thus ranges from 0.001058 to 0.001092. Assuming that the effect of the fine wire mass on the rotation radius is negligible and plugging these values into Eq. (6) yields a ΔV ranging from 5.49 cm/s to 5.67 cm/s.

The preceding analysis should serve as the basis for completing Problem 1 in the Appendix.

Migration of the spacecraft C.G. and fuel mass

The rotation of the spacecraft creates a centripetal acceleration of the fuel away from the spacecraft center of mass. Because the fuel tanks are spherical, the fuel accumulates into the approximate shape of spherical caps in each tank with apexes of curvature at the points of the tanks that are furthest away from the center of mass in the plane of rotation. The volume of a spherical cap is provided in Eq. (7). Thus, the mass of the fuel in each tank is the density of hydrazine, ρ_f, multiplied by the volume of the fuel in each tank as shown in Eq (8) and Eq (9).

$$V_{cap}=\frac{1}{3}\pi h^2(3R-h) \tag{7}$$

$$m_{f1}=\frac{1}{3}\pi\rho_f h_1^{\,2}(3R-h_1) \tag{8}$$

$$m_{f2}=\frac{1}{3}\pi\rho_f h_2^{\,2}(3R-h_2) \tag{9}$$

Because fuel is allowed to flow from one tank to the other, the fuel surface can settle at equal distances from the spacecraft center of mass even if the two tanks are not the same distance from the spacecraft center of mass. In other words, the height—and thus volume—of fuel in one tank can differ from that of the other tank, but the fuel surface in each tank is the same distance from the spacecraft's center of mass. If one assumes that the outlets of the tanks are the furthest points away from the center of mass on the spin plane, then the height of fuel can be expressed as shown in Eq. (10) and Eq. (11).

$$h_1 = \| \bar{r}_{O1} \| - R_S \tag{10}$$

$$h_2 = \| \bar{r}_{O2} \| - R_S \tag{11}$$

$$R_S = \| \bar{r}_{O1} \| - h_1 = \| \bar{r}_{O2} \| - h_2 \tag{12}$$

If either h_1 or h_2 is known, one can use Eq. (10) or Eq. (11) and Eq. (12) to determine the fuel height in the other tank and plug these numbers into Eq. (8) and Eq. (9) to determine the total amount of fuel on the spacecraft. Similarly, if one is given the total amount of fuel on the spacecraft, one can iterate on values for h_1 or h_2 to plug into Eq. (8) or Eq. (9) and Eq. (12) until one achieves the appropriate total fuel mass.

At the time of the anomaly, the total fuel mass of the spacecraft was 9.45±1.5 kg.* The XYZ-coordinates for the tank outlet locations were (0.3344m, -0.3344m, 0.321m) and (-0.3344m, 0.3344m, 0.321m) for Tank 1 and Tank 2, respectively. Using a center of mass location of (2.401 cm, 0.278 cm, 26.547 cm) and a hydrazine density of 1011.715 kg/m^3 one would calculate the fuel mass in Tank 1 as 3.463 kg (with an upper bound of 4.145 kg) and the fuel mass in Tank 2 as 5.988 kg (with an upper bound of 6.805 kg).

The preceding analysis should serve as the basis for completing Problem 2 in the Appendix.

The observation of a center of mass shift in the Doppler data

The THEMIS/ARTEMIS spacecraft's spin axis is nominally aligned and centered on its antenna mast. With the loss of the sphere and the shift in the center of mass this alignment is no longer true. From the point of view of the ground station the two-way Doppler signal now shows a modulation caused by the antenna motion around the new spin-axis. The radius of the antenna's path about this axis can be determined with Eq. (13).

$$R_A = \frac{\upsilon}{\theta \cos \beta} \tag{13}$$

Where υ is the observed modulation amplitude, θ is spacecraft rotational rate, and β is the angle between the ground station and the spin plane.

Analysis of high rate two-way Doppler data recorded at the BGS ground station shows clear indication of the shift. A radius of 2.4 cm was observed over multiple β angles. It is important to note that this method of measuring the spin-axis displacement can not differentiate between a pure translation, where the antenna mast and spin-axis remain parallel, or a tilt between the two. However, in this case the mass loss occurred in the spin plane and therefore the change in center of mass would primarily be a translation.

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* Refer to (Reference 4) for a summary of fuel mass estimation errors.

DISCUSSION

The events and equations described above provide a basis for discussion on the cause of the mass ejection anomalies, the implications of the anomalies for the ARTEMIS mission, and the implications of the anomalies for future libration point orbiting missions. In the remainder of this section, the information provided in the previous sections is explored to address these topics.

The cause of the mass ejection anomalies

There are two theories on the cause of the mass ejection anomalies: 1) both the EFI pre-amplifier and sphere were severed from the spacecraft by one or more micrometeoroid impacts or 2) the fine wire and braided Kevlar wires holding the sphere and pre-amplifier, respectively, failed separately due to the stresses of the space environment on the boom. The conservation of linear momentum calculations mentioned above do not account for the added momentum of a micrometeoroid, yet they are in close agreement with ΔV observations, suggesting that if a micrometeoroid impact did occur, it had a negligible impact on the total momentum of the system.[*] However, the fact that the anomalies occurred nearly a year apart on the same boom casts doubt on the second theory (i.e., a single impact event damaging both the pre-amplifier and fine wire is more likely than two random wire failures on the same boom). Furthermore, the first anomaly occurred during the second day of the Orionid Meteor Shower.

The implications of the mass ejection anomalies for the ARTEMIS mission

Though the trajectory disturbances to ARTEMIS P1 in Lissajous and lunar orbit were ultimately corrected, the mass ejection anomalies have several long-term implications for the ARTEMIS mission. First, as mentioned earlier, the spacecraft's spin down rate during maneuvers has increased dramatically. This increase in the spin down rate could create a need for spin up maneuvers and ultimately limit the ΔV available for lunar orbit operations. Additionally, the change in the spin down rate affects GMAN's modeling of maneuver performance and thus invalidates the thruster scale factor calibration curve; particularly for large maneuvers.[6]

Also, the shift in fuel mass from Tank 1 to Tank 2 will alter the fuel management strategy for the ARTEMIS mission. The fuel in Tank 1 will deplete before the fuel in Tank 2. Therefore, if it is decided to hold off decommissioning of the spacecraft until after Tank 1 is fully depleted, the latch valve at the exit of the tank will have to be closed in order to prevent the helium pressurant from escaping through the its outlet during a maneuver.

The implications of the mass ejection anomalies for future libration point missions

Future missions to libration point orbits could encounter mass ejection anomalies (or errant thrust events) similar to or worse than those encountered during the ARTEMIS mission depending on the design and operating characteristics of the spacecraft. Table 1 lists the mass ratios, rotation radii, and rotational velocities that would cause specific ΔVs on a spinning spacecraft with a mass that could be ejected from the end of a boom. For example, if the ARTEMIS P1 sphere-to-spacecraft mass ratio velocity were increased to 6.224×10^{-2}—and its sphere rotation radius and spacecraft rotational velocity were left at their actual values for the anomaly—the ΔV imparted on

[*] The micrometeoroid's affect on the total linear momentum of the system in the plane of the spacecraft's velocity could have been minimized by a small micrometeoroid mass, the melting of the fine wire (i.e., conversion of the micrometeoroids kinetic energy to thermal energy), or the angle of impact (e.g., the impact could have been roughly normal to the spacecraft velocity vector).

the spacecraft would have been as large as the ΔV required for the Lissajous orbit departure maneuver.

Table 1. Anomaly conditions necessary to impart specific ΔVs on the spacecraft.

Ejected Object Mass-to-Spacecraft Mass Ratio	Ejected Object Rotation Radius (m)	Spacecraft Rotational Velocity (radians/s)	ΔV (m/s)
1.092×10^{-3}	24.8	2.092	0.0567 (observed)
5.416×10^{-3}	123.0	10.38	0.281 (largest stationkeeping maneuver)
1.928×10^{-2}	437.7	36.93	1.000
4.959×10^{-2}	1126.3	95.01	2.573 (Lissajous orbit insertion maneuver)
6.224×10^{-2}	1413.5	119.2	3.229 (Lissajous orbit departure maneuver)

Of the three parameters evaluated in Table 1, the ejected object mass-to-spacecraft mass ratio realistically has the most potential to cause a significant ΔV to be imparted on the spacecraft. The EFI booms on the THEMIS spacecraft are among the longest booms flown on a spinning spacecraft and thus, it is difficult (though not impossible) to imagine spinning spacecraft operating in Lissajous orbit with significantly longer booms. Additionally, there are few foreseeable applications where the spin rate of a spacecraft with fairly long booms will have to be significantly larger than 20 RPM. However, there are foreseeable applications where the end mass could be greater than 1/20th of the mass of the spacecraft (e.g., a crew module tethered or “rigidly” attached to a space station that rotates to simulate artificial gravity).

Another anomaly-related consideration for future missions is the affect of the attitude of the spacecraft (or more precisely, the spin plane of the ejected object) on the stability of the orbit. In the case of ARTEMIS P1, the spacecraft spin plane was closely aligned with what could be called its “orbit plane” and thus, separation of the sphere at any time would have produced an “in-plane” orbit perturbation.* However, if the spin plane and “orbit plane” were closer to being normal, separation occurring within a large portion of the rotation arc would produce “out-of-plane” orbit perturbations. Such perturbations would be less disruptive to the Lissajous orbit, at least in the short run.

FUTURE WORK

The mass ejection anomalies on the ARTEMIS mission provide a wide array of opportunities for further study. One could generate numerous “what-if” scenarios to show what would have happened if the separation event had occurred at different points in the orbit, spin phases, spin rates, spacecraft attitudes, ejected mass-to-spacecraft mass ratios, and ejected mass rotation radii.

* The term “orbit plane” is used loosely due to the fact that Lissajous orbits are inherently non-planar.

One could even illustrate the impact of event if the correction maneuver had occurred at different times. Such analyses—perhaps conducted through Monte Carlo techniques—could shed light on the still open questions of what tracking data collection intervals, spacecraft attitudes, spacecraft mass properties, and spacecraft dimensions are optimal for preventing a spacecraft's fall out of Lissajous orbit following a mass ejection event. Furthermore, one could potentially characterize the orbital debris hazards associated with such scenarios in order to develop safety constraints for future designs. Moreover, one could also evaluate structural designs that would minimize the risk of boom structural failure or the severing of mass from the spacecraft due to an impact from micrometeoroids or orbital debris.

CONCLUSION

The mass ejection anomalies during the ARTEMIS mission—particularly the one that occurred in Lissajous orbit—were most likely caused by a micrometeoroid impact that had little affect on the spacecraft's linear momentum in the plane of its velocity vector. Furthermore, though the event caused an instrument sphere to detach from the spacecraft and depart the Earth-Moon system, the ΔV imparted on the spacecraft was ultimately corrected, thus enabling the spacecraft to remain in Lissajous orbit for several months after the event. However, long-term impacts to the mission were not completely avoided; due to the event, a 6% overburn occurred on the lunar orbit insertion and it may someday be necessary to conduct spin up maneuvers, significantly revise the thruster scale factor calibration curve, and shut the latch valve exiting Fuel Tank 1 to prevent the loss of the helium pressurant during a maneuver.

While the ARTEMIS spacecraft were not designed with libration point orbit operations in mind, the mass ejection anomaly can inform the design and operation of future libration point orbiters. Designers of future libration point orbiters can reduce the impact of mass ejection anomalies on their spacecraft by adhering to the following design principles whenever possible:

1. Structures on spinning spacecraft in libration point orbits should be designed to minimize the possibility of structural failure or the severing of mass from the spacecraft by micrometeoroids and orbital debris.
2. Objects that could detach from a spinning spacecraft in a libration point orbit should have a much smaller mass than the spacecraft.
3. The radius of rotation for objects that could detach from a spinning spacecraft in a libration point orbit should be sufficiently small to prevent large ΔVs from being imparted on the spacecraft.
4. The revolution rate of the objects that could detach from a spinning spacecraft in a libration point orbit should be sufficiently small to prevent large ΔVs from being imparted on the spacecraft.
5. Tracking data for spacecraft in libration point orbits should be collected often enough to identify and correct dynamic disturbances between station keeping events.
6. The spin planes of objects that could detach from spacecraft in libration point orbits should be oriented normal to the "orbit plane" whenever practical to reduce the risk of the spacecraft falling out of libration point orbit due to a mass ejection event.

Though future work is needed to characterize these principles more rigorously, it is clear from the analysis in this paper that characterization of and adherence to the first two principles should take priority over that of the others. The mass ratio of the ejected object-to-the-spacecraft is perhaps the most realistic manner in which a significant ΔV will be imparted on a libration point orbiting spacecraft based on foreseeable applications for such spacecraft. Moreover, preventing the mass ejection anomaly from occurring in the first place is the only way to eliminate all of the risks associated with each of the steps of the response to a mass ejection event in a libration point orbit.

ACKNOWLEDGMENTS

The authors would like to thank David Folta, Patrick Morinelli, Swapan Gandhi, Theodore Sweetser, Stephen Broschart, Mark Woodard, Bryce Roberts, Mark Lewis, and John McDonald for their roles in the characterization of and response to the ARTEMIS P1 mass loss anomalies.

The THEMIS and ARTEMIS missions are operated by the University of California, Berkeley Space Sciences Laboratory under NASA contract NAS5-02099.

NOTATION

h_1	Fuel height in Tank 1
h_2	Fuel height in Tank 2
m_1	Mass of the spacecraft after the separation event
m_2	Mass of the sphere and accompanying fine wire after the separation event
m_{f1}	Mass of fuel in Tank 1
m_{f2}	Mass of fuel in Tank 2
P	Total linear momentum of the spacecraft and ejected sphere
r	EFI sphere rotation radius
$\bar{r}_{O1}$	Position vector from the spacecraft center of mass to the outlet of Fuel Tank 1
$\bar{r}_{O2}$	Position vector from the spacecraft center of mass to the outlet of Fuel Tank 2
R	Radius of the spherical fuel tanks
R_A	Radius of the antenna's rotation about the spin-axis
R_S	Fuel surface radius (i.e., the fuel surface's distance from the center of mass)
$\bar{V}$	Velocity of the spacecraft prior to the separation event
$\bar{V}_{1f}$	Velocity of the spacecraft after the separation event
$\bar{V}_{2f}$	Velocity of the sphere and accompanying fire wire after the separation event
V_T	Scalar tangential velocity of the sphere as it rotated on the end of the EFI boom
V_{cap}	The volume of a spherical cap

$\hat{x}$ X-axis unit vector

β Angle between the spacecraft's spin plane and an observing ground station

ΔV Change in spacecraft velocity

ΔV_1 The velocity change imparted on the spacecraft due to the separation event

θ Rotational velocity of the spacecraft

ρ_f The density of hydrazine

υ Observed Doppler modulation due to the antenna motion about the spin-axis

APPENDIX: "HOMEWORK"/"EXAM" PROBLEMS

The analyses conducted by the authors to explain the dynamics phenomena observed following the loss of the instrument sphere represent "real-world" applications of fundamental principles of physics that are often studied or tested in university-level science and engineering courses, professional certification programs, and technical job interviews. Accordingly, they can also be applied to a classroom or interview setting in the form of "homework" or "exam" problems. Thus, the problems presented in this appendix are loosely based on the mass ejection anomaly described in this paper and can be used for teaching, interviewing, and testing engineering students and professionals. Each of these problems can be solved through the application of the equations and concepts presented in the Analysis section of this paper.

Problem 1

Given: A 90 kg spin stabilized spacecraft is in orbit around the Earth-Moon L2 libration point when a 0.1 kg sphere on one of its instrument booms detaches. The distance from the spacecraft center of mass to the sphere before separation is 25 meters and the spacecraft spin rate is 20 RPM. Assume that the instrument booms are in the spin plane of the spacecraft system.

Find: A) The magnitude of the post-separation velocity of the sphere (in an inertial frame that moves at the velocity of the spacecraft prior to the separation)

B) The ΔV imparted on the spacecraft by the departure of the sphere.

C) The sphere mass that would be required to eject the spacecraft from Lissajous orbit, before the spacecraft operators would be able to perform a trajectory correction maneuver (assume that this scenario would occur if the separation event imparted a $\Delta V \geq 2$ m/s on the spacecraft).

D) The minimum spin rate that would be required to eject the spacecraft from Earth orbit. Assume that the Earth Centered position and velocity coordinates of the spacecraft are (48089 km, -391205 km, -184050 km) and (1.17 km/s, 0.06 km/s, 0.192 km/s). Also assume that the gravitational parameter for Earth is $3.986 \text{x} 10^5$ km^3/s^2 and that the spacecraft's velocity vector is currently aligned with the spacecraft's spin plane.

Answer Key: A) 52.36 m/s, B) 5.82 cm/s, C) 3.311 kg, D) 57.2 RPM (or 5.99 radians/s)

Problem 2

Given: A spinning spacecraft at time X^- has 10 kg of hydrazine stored in two identical spherical fuel tanks (with a radius of 20 cm) with center points mounted 30 cm from the spacecraft center of mass on opposite sides of the spacecraft. The tanks are connected to each other by a shared fuel line to the thrusters and thus, fuel is able move from one tank to the other. At time X, an instrument sphere detaches from the spacecraft in a manner such that the spacecraft center of mass moves 3 cm towards Tank 1 by time X^+ (assume that fuel slosh reaches steady state by this time). The volume of a spherical cap is in Eq. (7). Assume that the volume of the hydrazine is 1 kg/L. Assume that the mass of fuel in the fuel line is negligible.

Find: A) The mass of fuel in each tank at time X^- and the percentage of the tank volume occupied by fuel.

B) The mass of fuel in each tank at time X^+ assuming that the fuel height in Tank 1 is 50 mm at that time.

C) The mass of fuel in Tank 2 when Tank 1 is fully depleted (neglect the motion of the spacecraft center of mass as the fuel in both tanks is depleted).

D) The direction of motion for the center of mass (i.e., towards Tank 1 or towards Tank 2) as the fuel is depleted.

Answer Key: A) 5 kg and 14.9% for both tanks, B) 1.44 kg in Tank 1 and 4.33 kg in Tank 2, C) 0.94 kg, D) Towards Tank 1, the extra fuel in Tank 2 pulls the c.m. away from Tank 1 and thus, as the fuel depletes, the c.m. migrates towards Tank 1.

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