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ORBIT DETERMINATION OF SPACECRAFT IN EARTH-MOON L1 AND L2 LIBRATION POINT ORBITS

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The ARTEMIS mission, part of the THEMIS extended mission is the first to fly spacecraft in the Earth-Moon Lissajous regions. In order to effectively perform lunar Lissajous station-keeping maneuvers, the ARTEMIS operations team has provided orbit determination solutions with typical accuracies on the order of 0.1 km in position and 0.1 cm/s in velocity. The ARTEMIS team utilizes the Goddard Trajectory Determination System (GTDS), using a batch least squares method, to process range and Doppler tracking measurements from the NASA Deep Space Network (DSN), Berkeley Ground Station (BGS), Merritt Island (MILA) station, and United Space Network (USN). The team has also investigated processing of the same tracking data measurements using the Orbit Determination Tool Kit (ODTK) software, which uses an extended Kalman filter and recursive smoother to estimate the orbit. The orbit determination results from each of these methods will be presented and we will discuss the advantages and disadvantages associated with using each method in the lunar Lissajous regions. In addition, we used the Orbit Determination Error Analysis System (ODEAS) to perform covariance analyses using various tracking data schedules. From this analysis, it was determined that 3.5 hours of DSN TRK-2-34 range and Doppler tracking data every other day would suffice to meet the predictive orbit knowledge accuracies in the Lissajous region.

INTRODUCTION

ARTEMIS is the first mission to achieve orbit around an Earth-Moon Lagrangian point.[1] The ARTEMIS mission comprises two identical spacecraft, referred to as ARTEMIS P1 and ARTEMIS P2, or simply P1 and P2. Each spacecraft is spin-stabilized with its spin axis offset several degrees from the ecliptic South direction. Each spacecraft carries a set of fields and waves instruments and particle instruments designed to conduct studies of Earth's magnetotail and solar wind from approximately 60 Earth radii and to study the lunar wake and its refilling as a function of the upstream solar wind. ARTEMIS two-point measurements open a new vantage point to phenomena previously studied by single-spacecraft missions.[2]

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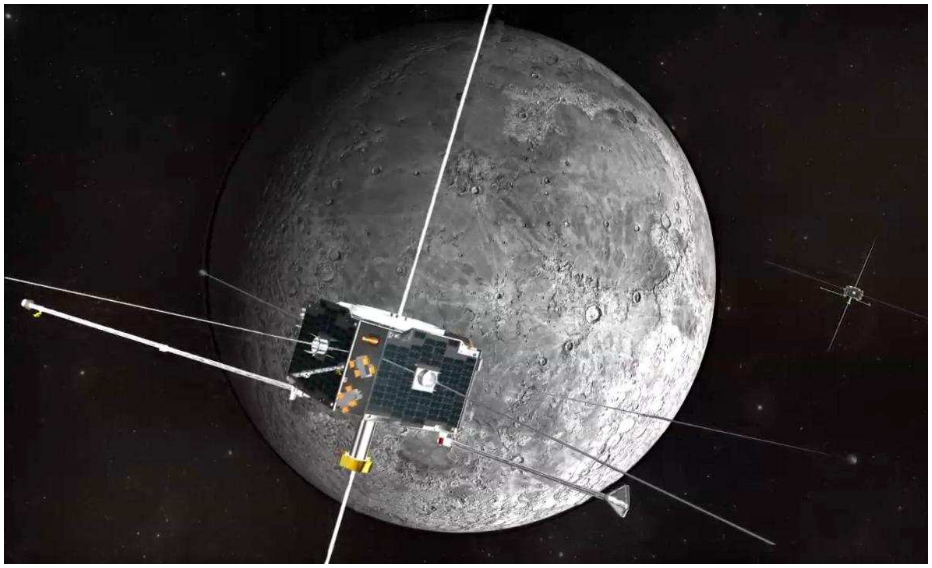

Figure 1. Artist's rendering of ARTEMIS spacecraft in lunar environment (Courtesy NASA SVS)

The ARTEMIS mission is an extension of the THEMIS mission. The five THEMIS probes were launched in February 2007 and the constellation conducted magnetospheric science out to 30 Earth radii. Each spacecraft was launched with a dry mass of 76.72 kg and approximately 49.00 kg of propellant. The Flight Operations Team (FOT) at UC-Berkeley Space Science Laboratory (UCB/SSL) can execute thruster commands to perform orbit and attitude maneuvers.[3] At the completion of the primary THEMIS mission in July 2009, a sequence of orbit-raising maneuvers were initiated to begin the transfer of THEMIS-B and THEMIS-C from high Earth orbits to Earth-Moon Lissajous orbits.[4] These two probes were designated as ARTEMIS P1 and P2, respectively. After many months and many carefully planned orbit maneuvers, P1 was inserted into a Lissajous orbit about the Earth-Moon L2 libration point in August 2010 and P2 was inserted into a Lissajous orbit about the Earth-Moon L1 point in October 2010.[5]

The Earth-Moon L1 and L2 libration points are unstable equilibrium points. Once the Lissajous orbits were achieved, the ground team had to perform frequent small station-keeping maneuvers (SKMs) to keep each spacecraft orbiting around its libration point. To effectively plan each orbit maneuver, analysts at the NASA Goddard Space Flight Center (GSFC) provided an impulsive maneuver plan to UCB, who in turn translated the impulsive maneuver into a finite maneuver using detailed models of the spacecraft, its attitude, and its orbit state. These maneuvers were typically performed weekly and were typically less than 10 cm/s, with the smallest maneuver executed being only 1.17 cm/s. To effectively and accurately plan such small maneuvers, GSFC and UCB required accurate knowledge of the pre-maneuver spin-axis attitude and orbit state (position and velocity) as well as accurate orbit force models for orbit propagation to the next maneuver event. The required 3σ RSS orbit knowledge is <1 km and <1 cm/s.

We will discuss spacecraft orbit determination in the Earth-Moon L1 and L2 libration point regions. Since ARTEMIS was the first mission to fly in these regions and several months of tracking measurements were collected from Earth-based ground stations, the results presented in this paper are original and define appropriate guidelines for designing future navigation systems that will operate in the vicinity of lunar libration points. The analysis and methods used for ARTEMIS orbit estimation provides techniques that can be carried over to similar missions.

ORBIT COVARIANCE ANALYSIS

ARTEMIS orbit determination (OD) is performed by processing radiometric S-band tracking data on the ground. ARTEMIS tracking data comes from a variety of ground-based tracking stations, including the Deep Space Network (DSN), the Universal Space Network (USN), and the Berkeley Ground Station (BGS).[6] The DSN provides range and Doppler data in the TRK-2-34 format. The USN provides range and Doppler data in the Universal Tracking Data Format (UTDF). The BGS provides Doppler data in the UTDF format; the link margin at BGS is generally not sufficient to provide reliable range data to a spacecraft near lunar distance and is not used for ARTEMIS orbit determination in the L1 and L2 regions. Table 1 provides more details on the ARTEMIS ground stations and their tracking capabilities.

Table 1. ARTEMIS Tracking Stations during Lissajous Operations.

Network	Antenna ID Acronyms	Location	Dish Diameter	Data Format; Types
DSN	DS24, DS27 DS34, DS45 DS54, DS65	Goldstone, California Canberra, Australia Madrid, Spain	34.0 m 34.0 m 34.0 m	TRK-2-34: range & Doppler
USN	USHS USPS	South Point, Hawaii Dongarra, Australia	13.0 m 13.0 m	UTDF: range & Doppler
BGS	BRKS	Berkeley, California	11.0 m	UTDF: Doppler

Prior to ARTEMIS Lissajous orbit insertion, GSFC needed to assess whether the baseline tracking schedule would be sufficient to meet the OD requirements (<1 km and <1 cm/s, 3σ RSS.) The baseline tracking schedule for each ARTEMIS spacecraft is as specified in Table 2. Note that for lunar and deep space missions, it is important to receive a good balance of Northern and Southern hemisphere coverage to ensure accurate orbit knowledge in the cross-track or "Z" direction.

Table 2. Nominal Tracking Schedule during Lissajous Operations.

Network	Pass Frequency	Pass Duration
DSN	1 every other day, alternating North/South	3.5 hours
BGS	2 per day	45 minutes
USN	1 per week	30 minutes

To determine if the nominal tracking schedule would meet the OD requirements, GSFC performed a covariance error analysis using the Orbit Determination Error Analysis System (ODEAS)[7]. ODEAS is a general purpose linear error analysis program that can be used for various orbit determination scenarios, including the lunar libration point regions. ODEAS models systematic error sources, such as the uncertainties in the orbital dynamics and measurement process, and random errors such as measurement noise and state process noise. By defining a tracking schedule and the expected accuracies of the orbital dynamics and measurement process, ODEAS provides the magnitude and characteristics of the errors that can be expected in an orbit estimation process. ODEAS can evaluate the use of either a sequential or batch processing mode. A Kalman filter and optimal smoother are supported in sequential processing, and a weighted least-squares method is supported in batch mode. Operationally, UCB uses a batch weighted least-squares method for orbit determination.

An 8-day tracking arc with a total of four 3.5 hour DSN passes was simulated in ODEAS, consistent with the nominal tracking data plan. The results are shown in Figures 2 and 3, with the DSN tracking events indicated by the vertical grey line. The tracking data arc is expected to end within 48 hours of the maneuver time to allow the operations team sufficient time to perform orbit determination, create the impulsive and finite maneuver plans, send the maneuver command load to the spacecraft, and execute the maneuver during a real-time contact. The plots in Figures 2&3 represent a 9-day definitive orbit plus a 2-day predictive orbit, i.e., the error components on Day 11 represent the expected orbit determination errors at the time of the maneuver. It can be seen in this case the OD requirements of <1 km position error and <1 cm/s velocity error at the maneuver time are satisfied. The velocity error, which is of utmost concern for maneuver planning, exceeds the requirement with a safety margin of greater than 2:1. This result suggests that the nominal DSN tracking data plan is valid for ARTEMIS maneuver planning. Inclusion of BGS and USN tracking data into the OD arc will further improve the solution accuracies.

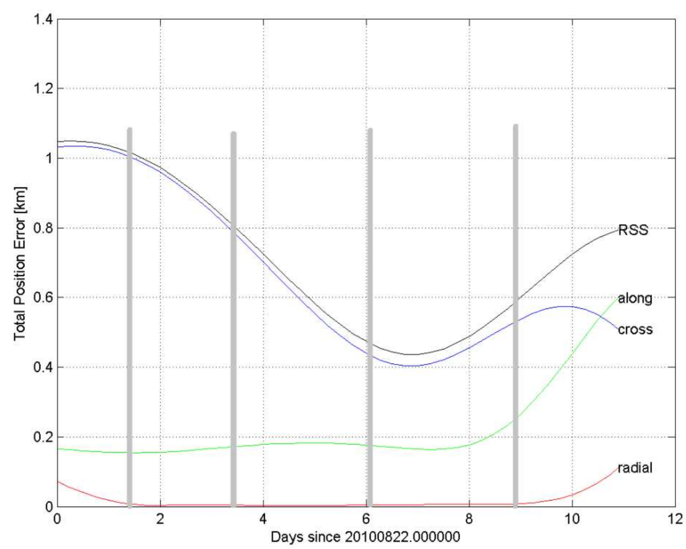

Figure 2. ARTEMIS 3σ Position Errors from Covariance Analysis

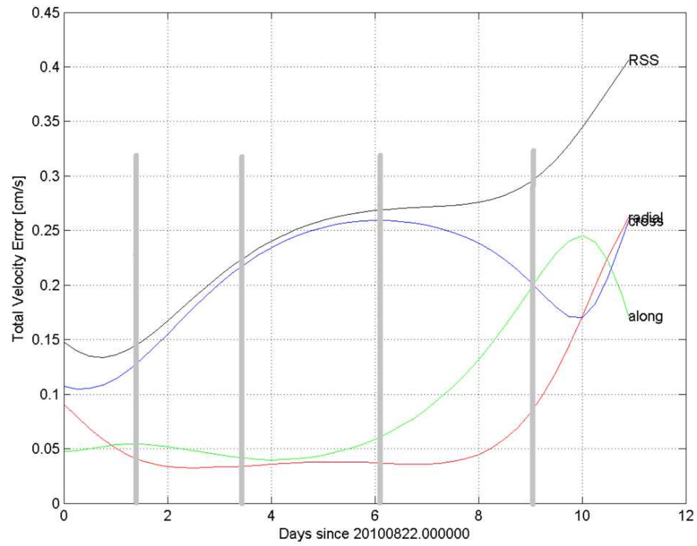

Figure 3. ARTEMIS 3σ Velocity Errors from Covariance Analysis

OPERATIONAL ORBIT DETERMINATION

UCB performs ARTEMIS orbit determination on a daily basis and the daily orbit solutions are typically delivered by 14:00 UTC. UCB provides orbit solutions via email to members of the ARTEMIS support team. These activities are automated through integrated IDL and UNIX scripts and cron daemon processes which, at the core, employ the Goddard Trajectory Determination System (GTDS).

Orbit determination through GTDS uses a batch-weighted least-squares method to estimate the ARTEMIS orbit from received tracking data. The estimator solves for spacecraft position, velocity, and solar radiation coefficient, C_R, at the solution epoch and provides the corresponding covariance matrix. The solution epoch is targeted near the end of the epoch. This sacrifices some accuracy (and convergence stability) in the overall solution in exchange for better resolution near the most recent observations, since GTDS minimizes the solution residuals near the target epoch.

The Flight Dynamics Facility (FDF) at GSFC performs less frequent orbit determination using GTDS in a backup role to UCB. UCB and GSFC use consistent force modeling parameters for orbit determination, orbit prediction and maneuver planning during Lissajous operations. These parameters are specified (for the ARTEMIS mission) in the *ARTEMIS Models and Constants* document[8] and are summarized in Table 3. Mission-supporting orbit propagations are performed using a Runge-Kutta 7/8 integration method. Orbit state vectors and ephemerides transferred between GSFC & UCB are referenced to Earth-Centered True-of-Date (TOD) coordinates and are time-tagged relative to Universal Time Coordinated (UTC).

The differential corrector used by GTDS (for orbit determination) defines a single “central-body” for which a non-point-mass potential (in the form of zonal harmonics) can be used. All other “non-central” bodies are considered as point masses. During the Lissajous orbital phase of the ARTEMIS mission, orbit determinations performed by UCB and FDF used the Earth as the central-body with parameters defined in Table 3. The non-central bodies were Moon, Sun, Mars, Jupiter, Saturn, Mercury, and Venus.

Table 3. Orbit Force Model Parameters.

Parameter	Source/Value
Solar, Lunar & Planetary Ephemerides	DE-421
Earth GM	398600.4356 km^3/s^2
Earth Gravity Model	GGM02C, 8x8
Lunar GM	4902.8000 km^3/s^2
Lunar Gravity Model	LP150Q, 8x8
Solid Earth Tide Effects	Not included
Solar & Planetary GM	DE-421
Solar Radiation Pressure	Included
Force from Solar Flux	4.570×10^{-6} $N\text{-}AU^2/m^2$
AU	149597890.0 km
Spacecraft Area (flat plate model)	0.95 m^2
Solar Radiation Coefficient, C_R	Solved-for in OD

ORBIT DETERMINATION ACCURACY

Orbit determination accuracy is dependent on the quality and quantity of tracking measurements, fidelity of the orbit force models, and the estimation techniques used.

Figures 4 and 5 show the results of a series of daily batch least squares orbit solutions across several maneuvers. The maneuvers are not modeled in the batch estimator, so a new tracking arc is begun immediately following the completion of each maneuver. As additional tracking passes are collected, daily OD solutions are generated with GTDS. The orbit accuracy typically improves each day leading up to the next orbit maneuver. This convergence in orbit accuracy is illustrated in Figures 4 and 5 by the parameter V_ε, which is the root-sum-square (RSS) uncertainty of the in-plane velocity at the solution epoch. It can be seen that V_ε is generally in the range of 0.01 to 0.001 cm/s prior to each station-keeping maneuver (SKM) and thus, the velocity uncertainty of the orbit state is often several orders of magnitude less than the magnitude of the SKM. Such a low level of velocity uncertainty provides the team with a high degree of confidence that the maneuver plan will achieve its objectives of maintaining orbit stability until the next weekly SKM is planned and executed.

We observe through the data (in Figures 4 and 5) that V_ε decreases (i.e., the solution accuracy increases) in log-linear fashion as tracking hours increase until about 30-35 hours of data (over ~6 days) are received. The subsequent 'knee' in the $\log(V_\varepsilon)$ curve was recognized as a transition to convergence. With regard to the C_R data in Figures 4 and 5 it can be seen that, for the most part, the a priori values held reasonably well until convergence in V_ε. Once transition to convergence was begun, some drift in C_R was incurred.

At this point, a note about C_R is in order. Allowing the GTDS differential corrector to solve for C_R obviously has specific ramifications on the solutions generated. Not having to specify C_R is convenient in that one avoids introducing error in the knowledge of this quantity. Variations in solar activity and errors in knowledge of the spacecraft's presented cross-sectional area all contribute to this uncertainty. But in allowing this value to be calculated, the differential corrector has an additional degree of freedom in which to aggregate error due to unmodeled forces. This places some significance on giving the differential corrector a reasonable initial (‘a priori’) value. UCB performed an extensive investigation to determine optimal ‘a priori’ values (for P1 and P2), the details of which are beyond the scope of this paper and are subjects for further development.

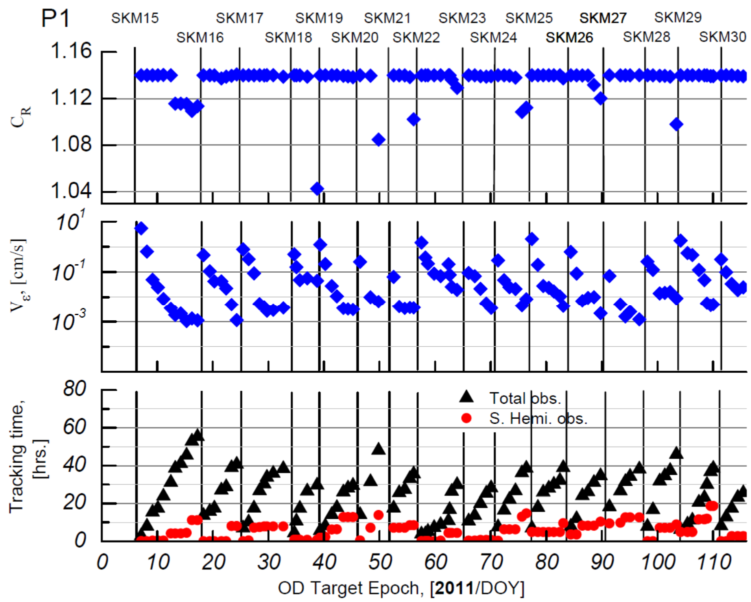

Figure 4. P1 Batch Least-Squares Orbit Determination Trending

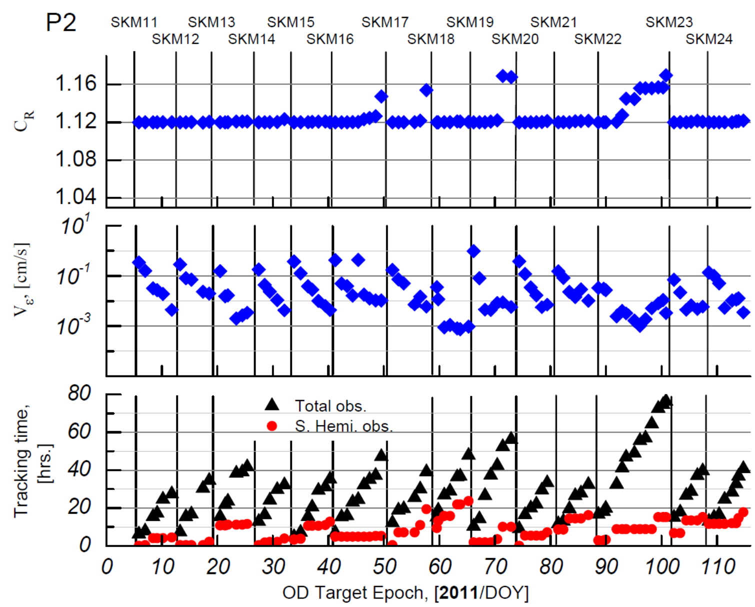

Figure 5. P2 Batch Least-Squares Orbit Determination Trending

A somewhat *indirect* metric for evaluating the accuracy of the orbit determination is the maneuver delta-v magnitude error in the ongoing Lissajous station-keeping maneuvers. This metric is obtained by producing pre- and post-maneuver state vectors, propagating them to the maneuver center time, and comparing their velocity difference at that time to the targeted delta-v for the maneuver. The delta-v magnitude error (i.e., the difference between the targeted delta-v magnitude and the observed delta-v magnitude) is due to two (most probable) sources of error: orbit determination error and maneuver execution error. Accordingly, this metric can only serve as an “upper-limit” for the orbit determination error—the relative proportion of delta-v magnitude error due to maneuver execution and orbit determination error is unknown. As shown in Table 4, the delta-v magnitude error decreased throughout Lissajous Orbit operations, as the FOT discovered and implemented techniques to reduce maneuver execution error. Over 17 most recent maneuvers in the dataset, the average and median delta-v magnitude errors were 1.012 mm/s and 0.7697 mm/s, respectively.

Table 4. Delta-V Magnitude Error Statistics for Several Subsets of Lissajous Station-keeping Maneuvers.

Maneuver Set	Number of Lissajous Orbit Station-keeping Maneuvers	Minimum Targeted ΔV Magnitude (m/s)	Maximum Targeted ΔV Magnitude (m/s)	Average ΔV Magnitude Error (m/s)	Median ΔV Magnitude Error (m/s)
2011/075 – 2011/139	17	0.01329	0.2788	1.012×10^{-3}	7.697×10^{-4}
2011/006 – 2011/139	33	0.01173	0.2961	1.105×10^{-3}	7.697×10^{-4}
2010/315 – 2011/139	45	0.01173	0.3485	1.174×10^{-3}	9.025×10^{-4}
2010/237 – 2011/139	57	0.01173	2.562	2.505×10^{-3}	1.215×10^{-3}

In April 2011, the maneuver team elected to waive off one of the planned weekly P2 station-keeping maneuver as the required maneuver magnitude was much less than 1 cm/s. The maneuver was delayed another week and this allowed the team to collect and assess the accuracies of daily orbit determination solutions with as few as two and as many as fourteen days of tracking data. Each daily orbit solution was trended as to the 3σ uncertainty reported by the GTDS batch least squares estimator, and the required delta-v magnitude of the delayed maneuver. The results indicated that the optimal batch was approximately ten days, where the RSS OD velocity uncertainty was as low as 0.002 cm/s. The associated position uncertainty was 19.9 m. Increasing the batch size increased the OD uncertainty, however the effect of the increased uncertainty on the maneuver magnitude was negligible.

P1 SPACECRAFT ANOMALY

On October 14&15, 2010, UCB and FDF noticed unusually high residuals on both range and range rate for the P1 GTDS orbit solution, as illustrated in Figures 7 and 8. The range rate residuals for three successive DSN passes were on the order of -0.06 ± 0.01 cm/s, and the range residuals indicated a drift rate of the same magnitude. GSFC consulted with UCB, and upon receipt of the full telemetry data, the FOT reported unexpected readings including a sudden spin rate increase, a sudden change in the Sun angle measured by the digital Sun sensor, and loss of science data from the –X axis Electric Fields Instrument (EFI). UCB analyzed that all of these phenomena were consistent with the loss of the EFI sphere at the end of the –X wire boom. UCB and GSFC concluded that the fine wire at the end of the –X boom had snapped and the ejection of the EFI sphere had imparted an opposing delta-v to the spacecraft of approximately 5.6 cm/s.

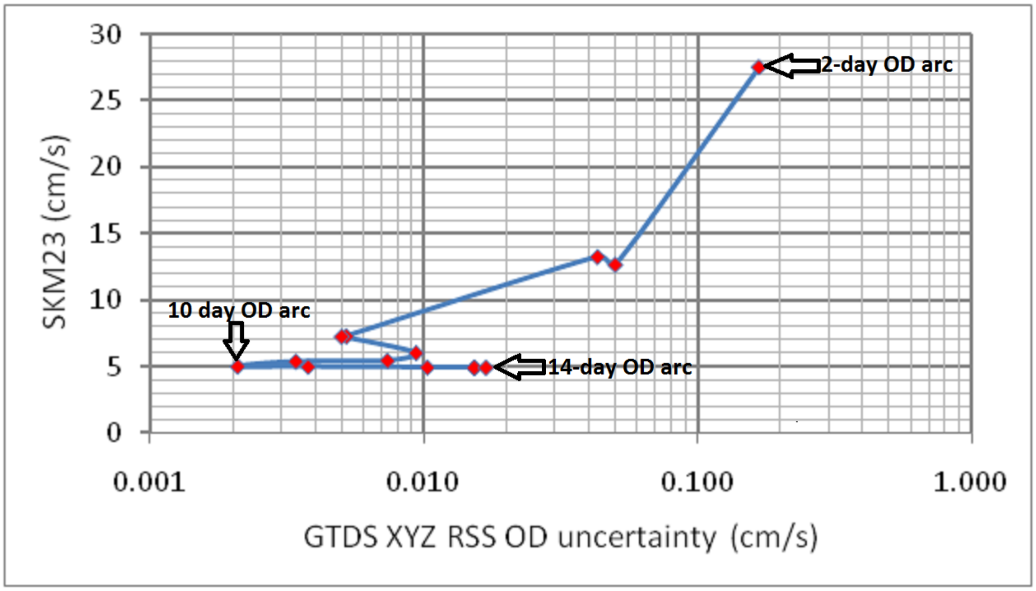

Figure 6. Orbit Uncertainty and Maneuver Trending across Variable Tracking Data Arcs

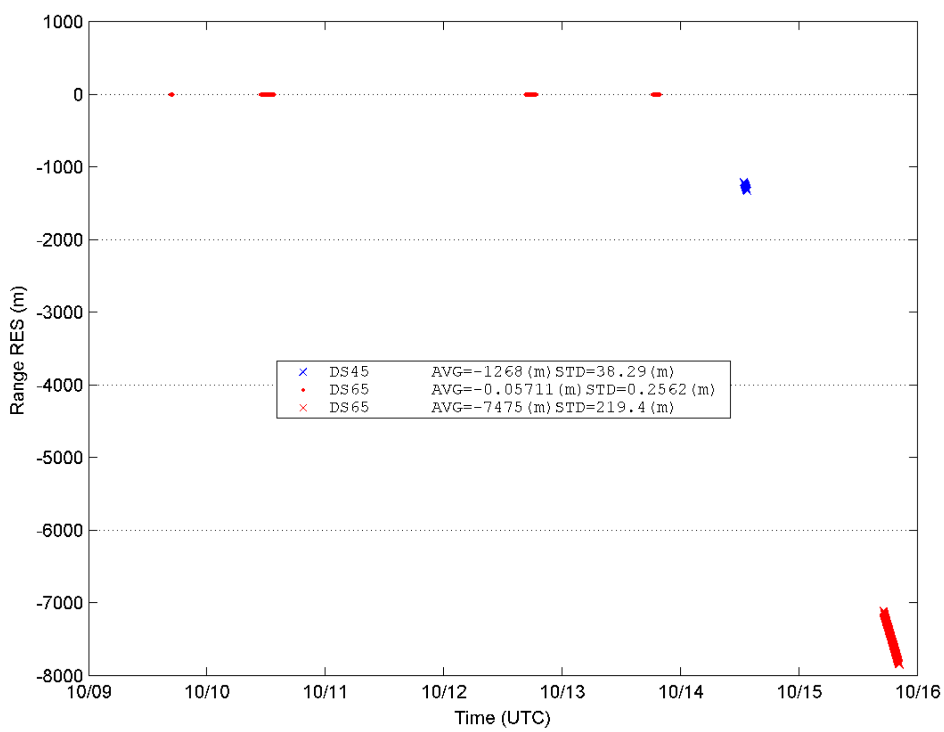

Figure 7. Range Residuals Before & After P1 Anomaly

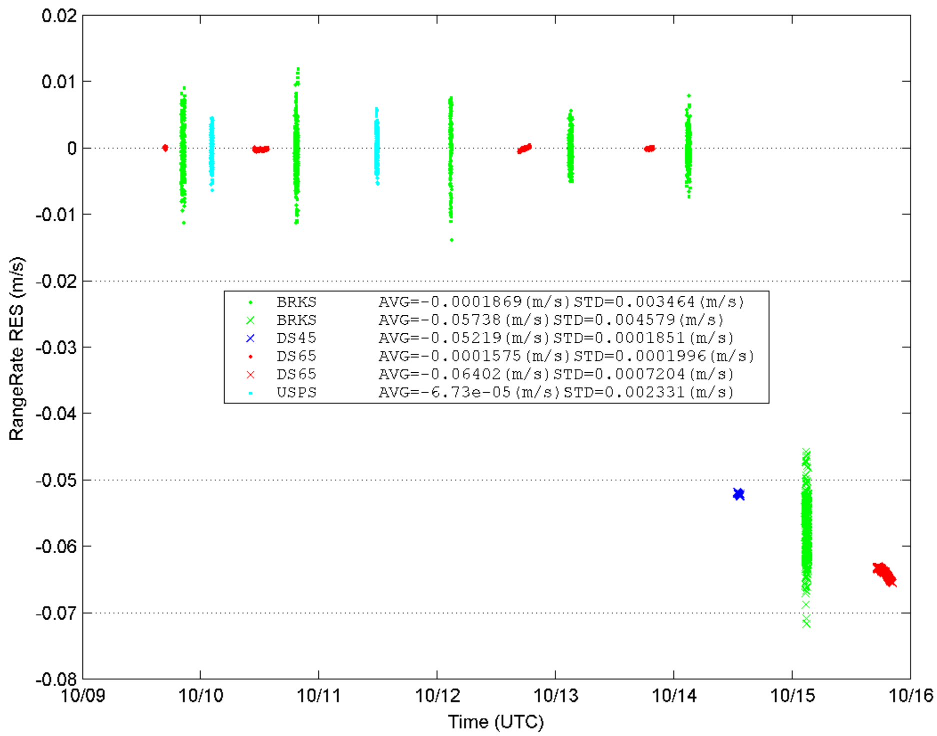

Figure 8. Range Rate Residuals Before & After P1 Anomaly

Personnel from Analytical Graphics Incorporated (AGI) were consulted and they were able to assist in the anomaly resolution. Using their Orbit Determination Tool Kit (ODTK), they were able to ingest pre- and post-anomaly DSN tracking data into ODTK and perform orbit determination across the event using ODTK's Kalman filter estimator and recursive smoother. By propagating a pre-anomaly orbit state forward, a post-anomaly orbit state backward, and differencing the two propagations, they found that the minimum distance between the two propagations occurred on October 14, 2010 at 05:42:49 UTC as shown in Figure 9. This is our best estimate of the time that the EFI sphere separated from the spacecraft wire boom.

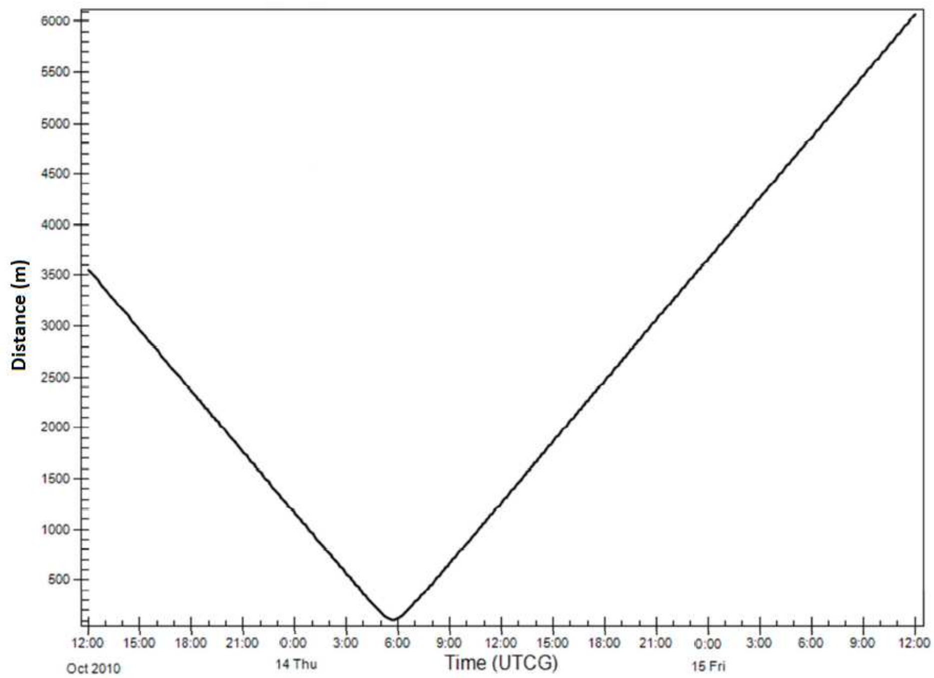

Figure 9. Ephemeris Position Difference of Pre- and Post-Event Orbit Propagations

FILTER/SMOOTHER PROCESSING

GSFC assessed the orbit determination accuracies that could be achieved using the ODTK Kalman filter and recursive smoother. The tracking data arc was selected near the beginning of the P1 Lissajous phase and included DSN TRK-2-34 tracking data before, during, and after the P1 station-keeping maneuver #2 (P1 SKM2). The tracking arc comprised a total of 11,085 range and Doppler measurements collected between 30 August 2010 06:30:50 UTC and 22 September 2010 12:59:30 UTC. P1 SKM2 was a 59.38 cm/s maneuver executed on 8 September 2010 from approximately 10:59:13 UTC to 11:00:49 UTC. This maneuver was modeled in ODTK as a 59.38 cm/s impulsive maneuver occurring at 11:00:00 UTC.

Figures 10 through 13 show the orbit determination accuracies achieved in ODTK across the tracking arc. The 3σ estimation uncertainties in the in-track, radial, and cross-track directions are plotted along the vertical axes and time along the horizontal axis. The SKM2 maneuver event is designated by a vertical line. The first 2 plots show behavior typical of a Kalman filter – the orbit solution shows initial convergence, divergence following the maneuver event, and recovergence. Steady-state performance is achieved near the end of the tracking data arc, where the RSS accuracies are estimated to be 185 m and 0.06 cm/s. The Kalman filter with recursive smoother shows improved accuracies and improved behavior across the maneuver event. The smoother RSS accuracies are identical to the Kalman filter only solution at the end of the tracking arc, as expected. However, the smoother accuracies are considerably improved (compared to the Kalman filter only solutions) across the rest of the tracking data arc.

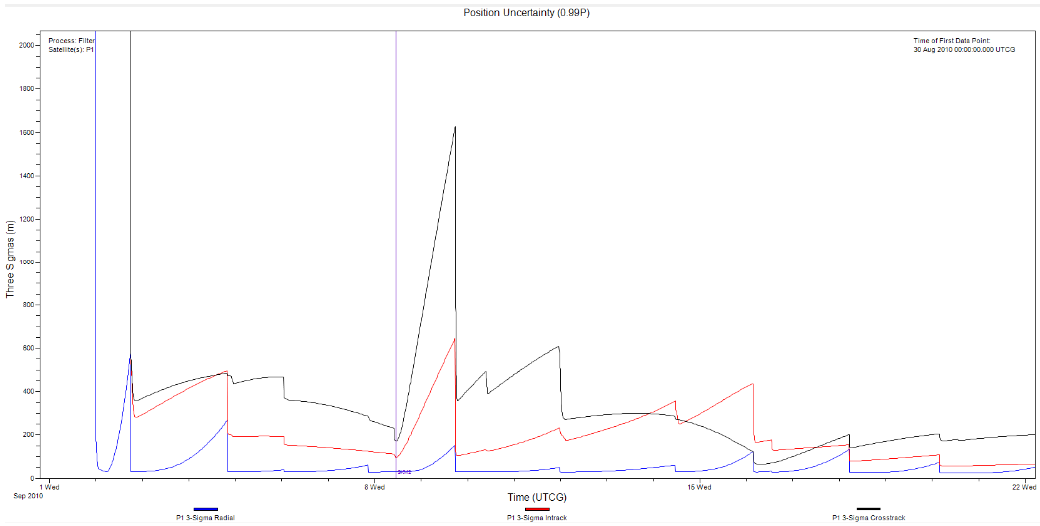

Figure 10. P1 Orbit Determination Position Uncertainties from Kalman Filter

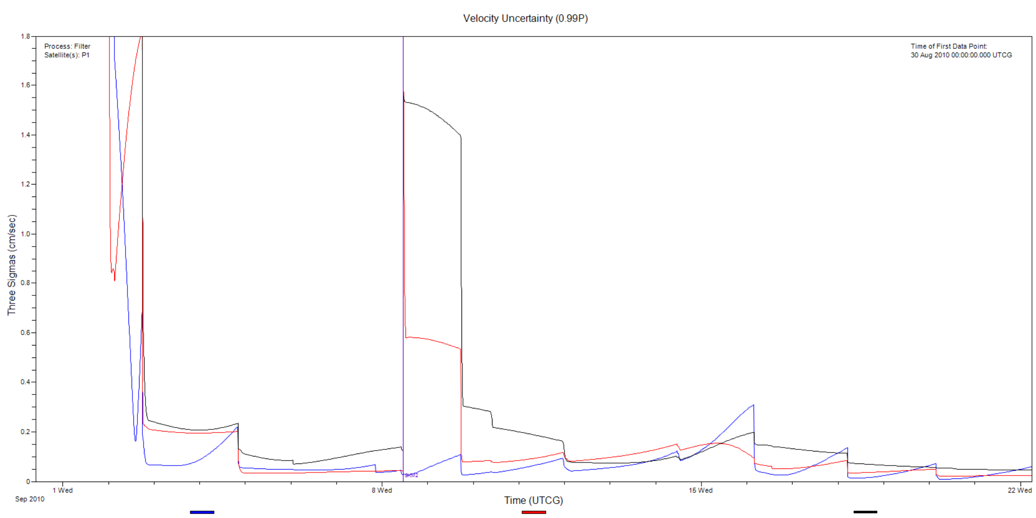

Figure 11. P1 Orbit Determination Velocity Uncertainties from Kalman Filter

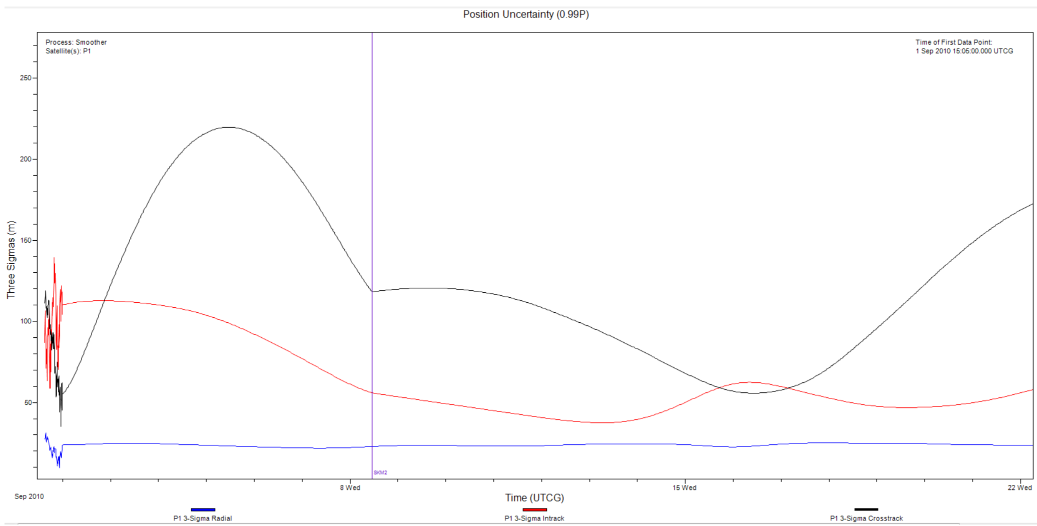

Figure 12. P1 Orbit Determination Position Uncertainties from Kalman Filter & Recursive Smoother

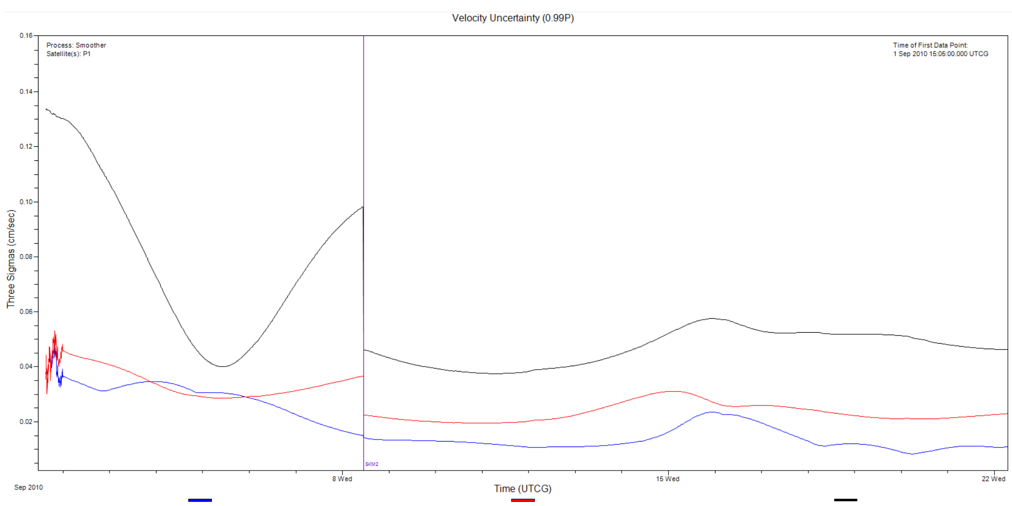

Figure 13. P1 Orbit Determination Velocity Uncertainties from Kalman Filter & Recursive Smoother

CONCLUSION

We have provided high-accuracy orbit determination results for the two ARTEMIS spacecraft obtained by processing ground tracking measurements. The techniques and analysis methods used can provide a baseline for future Earth-Moon libration point spacecraft missions, including Lissajous, Lyapunov, and Halo orbits. Orbit determination using a weighted batch least squares method has shown that orbit accuracies of approximately 20 m and 0.002 cm/s are achievable and that the optimal batch size is approximately ten days. Orbit determination using a Kalman filter and recursive smoother method has shown that this technique also works in the Earth-Moon libration point region and that station-keeping maneuvers can be modeled as impulsive maneuvers in the OD process to provide well-behaved orbit estimation across maneuver events.

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