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AN OVERVIEW OF THE ARTEMIS I NAVIGATION PERFORMANCE

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The goal of NASA's Artemis Program is to explore the Moon and beyond. The Artemis I Mission which flew in late 2022 was the uncrewed test flight whose goal was to exercise the entire navigation system in an extended duration flight and evaluate its performance over the entire mission, from pre-launch to post-landing. This paper provides an overview of the Artemis I navigation system architecture, examines the reasoning behind the design, and showcases the navigation performance. In particular, it presents systems-level performance of the navigation filters, sensors, and fault detection/isolation/recovery (FDIR). Also provided are glimpses into the lessons-learned during the flight, the run-up to the mission, and the post-flight analyses. Of particular interest are the four navigation Extended Kalman Filters (EKFs): the Atmospheric EKF (ATMEKF), the Earth Orbit EKF (EOEKF), the Attitude EKF (ATTEKF) and the Cislunar EKF (CLEKF). Whereas only the ATMEKF is a coupled translation/rotation filter, the other three are either translation-only (EOEKF, CLEKF) or rotation-only (ATTEKF). Also presented is an overview of the Optical Navigation system performance.

INTRODUCTION

The long-awaited Artemis I launch occurred on November 16, 2022 at 6:47:44 UTC, sending the Orion spacecraft on a 25.5 day mission to a Distant Retrograde Orbit (DRO) in the Earth-Moon system. The goal of the mission was to test and to stress all of the subsystems (with the exception of the ECLSS system). There were various maneuvers throughout the mission, particularly during the lunar flybys. There were various Flight Test Objectives (FTOs) and Developmental Flight Test Objectives, each of them to validate Level 1 requirements (the FTOs) and the subsystem requirements (DFTOs). All of these were designed to ensure the vehicle performed as required and to gather data to evaluate the performance of the spacecraft system.

This paper will devote itself to describing, first, the mission and then proceed to giving an overview of the navigation system. It will describe the performance of the navigation system during the various phases of flight from pre-launch to post-lading. As of this writing much of the data is still in raw form, so some of the results presented have glitches, gaps, or other features. The grammatically awkward phrase, 'Never Don't Navigate', was the adopted motto of the Navigation System. Finally, a few concluding comments are offered.

MISSION OVERVIEW

On November 16, 2022, the Artemis I mission was launched from the Kennedy Space Center using a Space Launch System (SLS) Block 1 configuration. The goal of the mission was to test

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and evaluate the performance of the launch vehicle (SLS) and the Orion spacecraft through all phases of the flight, culminating in a Skip Entry on December 11, 2022. After separation from the SLS, the Interim Cryogenic Propulsion Stage (ICPS) performed the TransLunar Injection (TLI). The destination was a Distant Retrograde Orbit (DRO) which necessitated an Outbound Powered (Lunar) Flyby (OPF) burn. Several days after the OPF, Orion inserted itself in a DRO by means of a Distant Retrograde Insertion (DRI) burn. Sixteen days into the mission, the spacecraft performed a Distant Retrograde Departure (DRD) burn, followed by a Return Powered Flyby (RPF) burn, the final of the four major (deterministic) burns. Sprinkled liberally throughout the mission were several trajectory correction burns: 6 on the outbound leg, 3 in the DRO, and 6 on the return leg. These mission milestones are shown in Figure 1.

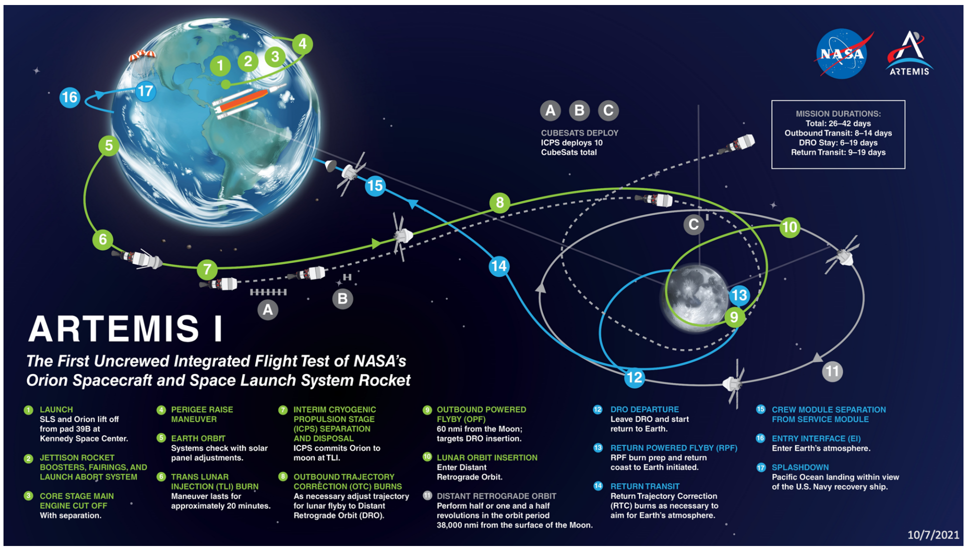

Figure 1. Artemis I Mission Overview

NAVIGATION ARCHITECTURE

The Artemis I onboard navigation architecture is presented in Figure 2. The navigation sensors include 3 Inertial Measurement Units (IMUs), 2 GPS Receivers, 2 Star Trackers (STs), and 1 Optical Navigation (OPNAV) camera. The star trackers and the optical navigation camera are mounted together on an optical bench on the Crew Module Adapter (CMA). The IMUs were derived from the Honeywell Miniaturized Inertial Measurement Units (MIMUs) family modified to accommodate the Orion high-g environments, particularly ascent aborts. The OIMUs consist of three accelerometers and three gyros along with the associated electronics and firmware. The 12-channel GPS Receivers, which provided pseudo-range measurements, were originally based upon the Boeing 787 GPSRs but were modified with the Goddard Navigation tracking IP. The STs, which provided attitude quaternions, were provided by Jena-Optronik and had space heritage. In addition, there is a Pixelink optical navigation camera used for optical navigation. The Star Trackers and the Optical Navigation Cameras were mounted on an optical bench which was in turn mounted on the Crew Module Adapter (CMA).

The navigation architecture includes a set of navigation filters configured to operate with specific sensors.[1] There are four types of Extended Kalman Filters (EKFs) that are part of the Navigation

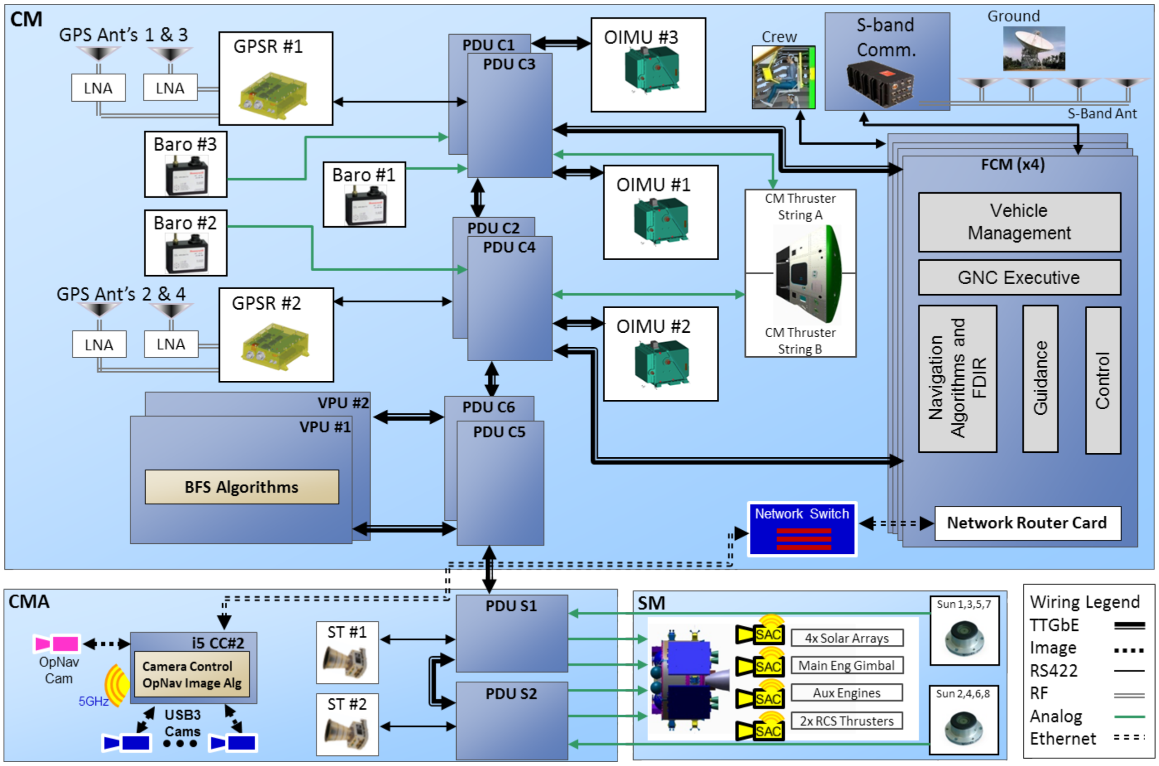

Figure 2. Orion Navigation Architecture

Subsystem: Atmospheric EKF (ATMEKF), Attitude EKF (AttEKF), Earth Orbit EKF (EOEKF), and Cislunar EKF (CLEKF). Each of these filters is modeled after a multiplicative EKF (MEKF) where the attitude error states are treated as (small) deviations from the reference attitude states and are incorporated into the attitude states in a multiplicative fashion. After each measurement cycle, the attitude state is updated and the attitude error state is zeroed out. The reference attitude state is then propagated using the gyro outputs. The Cislunar EKF, which is the filter used outside of low-earth orbit (and the GPS constellation) is a 21-state MEKF whose states are: position (3), velocity (3), attitude error (3), unmodeled acceleration (3), accelerometer biases (3), accelerometer scale factor (3), optical sensor biases (3). Since the ATTEKF is the primary filter to estimate the attitude (by processing Star Tracker measurements and gyro data), the attitude of the vehicle is not updated from the output of the CLEKF. Instead, the attitude states in the CLEKF are modeled as first-order Gauss-Markov processes with process noise strength modeled to be consistent with the expected attitude error from the ATTEKF. As well the other bias states are treated as first-order Gauss-Markov processes with appropriate process noise strength and time constants. The accelerometer bias and scale factor parameters are included in this filter to account for the effects of the accelerometer errors during powered flight maneuvers. Since the accelerometers are thresholded during coasting flight, the unmodeled acceleration states are chosen in order to include the effect of non-powered accelerations imparted on the vehicle due to residual acceleration caused by attitude deadbanding, attitude maneuvers, sublimator vents, Pressure Swing Adsorption (PSA) vents, and waste-water vents. The latter two were not a factor during Artemis I since it was be uncrewed. Finally, the presence of the Earth's atmosphere and the lunar terrain will introduce measurement biases and these effects are attempted to be captured by the optical navigation measurement states (which are modeled as slowly time varying first-order Gauss-Markov states).

The filter architecture uses a square-root-free UDU covariance factorization. The UDU factorization was chosen for all of Orion navigation filters because of its numeric stability and computational efficiency.[2] In particular computation savings were exploited in the propagation phase by partitioning the filter states into ‘states’ and ‘parameters’. The covariance associated with the ‘states’, comprising of the position and velocity, were propagated using a numerically computed state transition matrix and updated via a modified weighted Gram-Schmidt algorithm, while the covariance of the ‘parameters’ was propagated analytically using a rank-one Agee-Turner algorithm.[3] The measurements were updated one-at-a-time (assuming uncorrelated measurements) using a Carlson rank-one update.[4] In addition, the EKF architecture allowed for ‘considering’ any of the parameter states.

PRELAUNCH AND ASCENT

The prelaunch phase lasted from approximately T-5 hours to T-90 seconds, during which pad position measurements were processed. During earlier launch attempts, a few surprises involving exceedances of the IMU parity thresholds were observed. These were understood to have been caused by different heating profiles of the IMUs during power-on. The Fine-Align process worked as expected. The following plots show the residuals and the performance of the navigation system during this phase of the flight. The Pad Position residuals are presented in Figure 3.

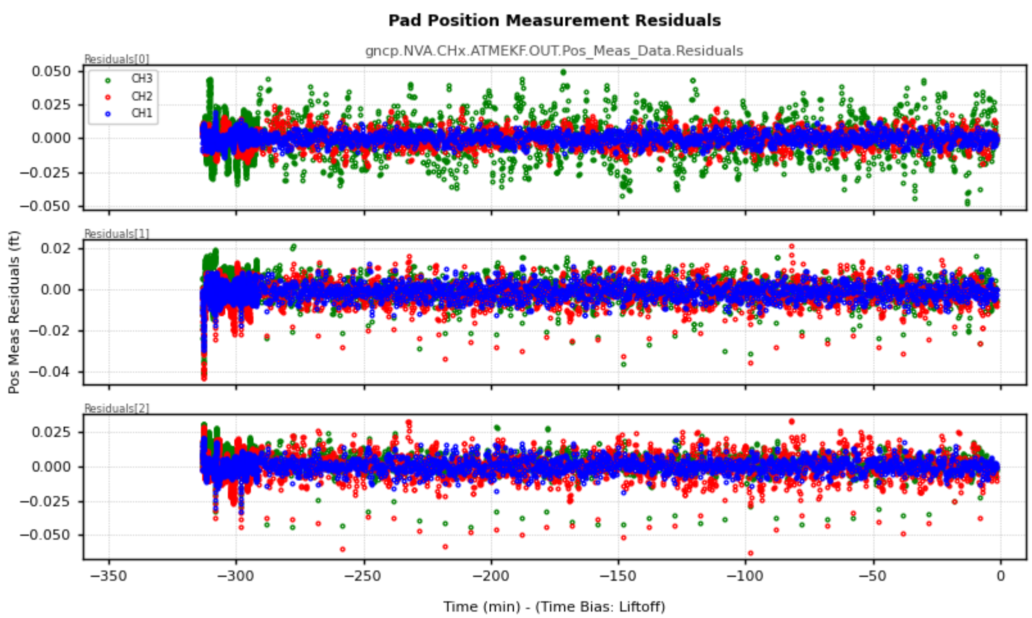

Figure 3. Pad Position Residuals

During the initial portion of the launch, when the LAS cover was shielding the GPSR antennas, no measurements were processed. Once the LAS was jettisoned, the 2 GPSRs began processing PR measurements and the filter covariance reduced rapidly, as expected. It should be noted that Orion was a passenger during this portion as the SLS was in control of the trajectory of the SLS-ICPS-Orion stack. Despite this, GPS measurements were obtained after LAS Jettison and are presented in Figure 4.

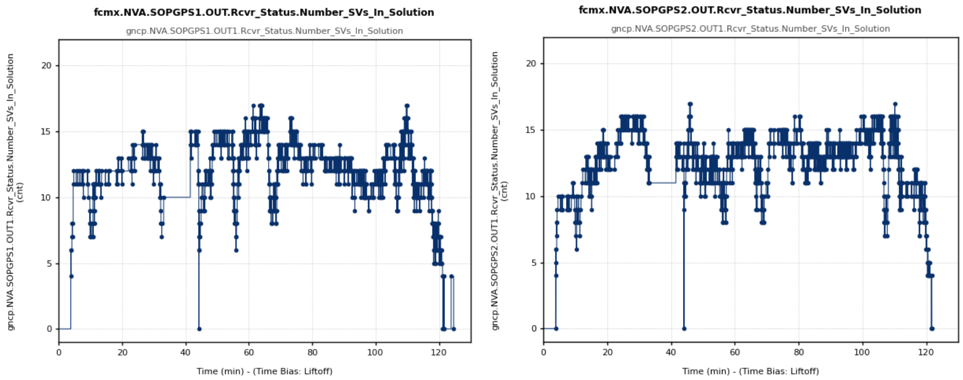

Figure 4. GPS satellite visibility on ascent

OUTBOUND LEG

The first challenge that was faced after orbit insertion was the debris that resulted from the Separation from the ICPS. This is shown in Figure 5. The preponderance of debris surrounding the vehicle significantly increased the Star Tracker mean background level readings as seen in Figure 6.

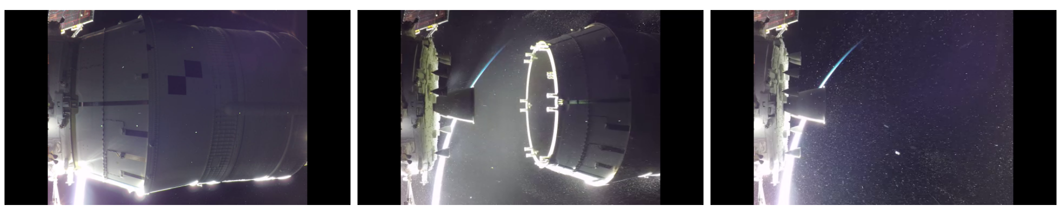

Figure 5. Separation debris seen during early mission

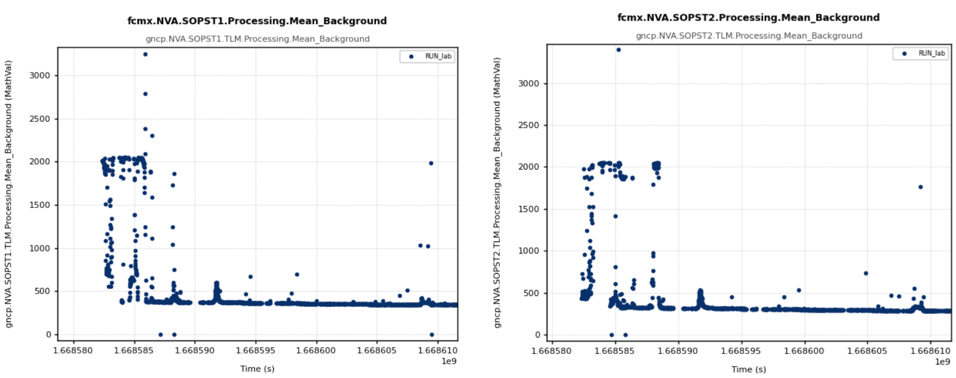

Figure 6. Star Tracker elevated mean background levels due to separation debris

This was not the only ST challenge that the operations team faced in the first days of the mission. It was observed that the STs would downmode from the nominal attitude tracking mode to an acquisition mode frequently and without warning. After a bit of detective work, it was observed that this phenomenon was correlated to the time of (certain) thruster firings, and that the illuminated plume would ”dazzle”, or temporarily blind, the star tracker. The RCS pluming shown in Figure 7. Figure 8 directly shows how well the star tracker dropouts correlated with RCS thruster firing and the resulting attitude rate . A formal Anomaly Resolution Team (ART) was convened and the findings unambiguously resolved that this ST downmoding occurred when Thruster 3 or Thruster 8 were firing as seen in Figure 9.

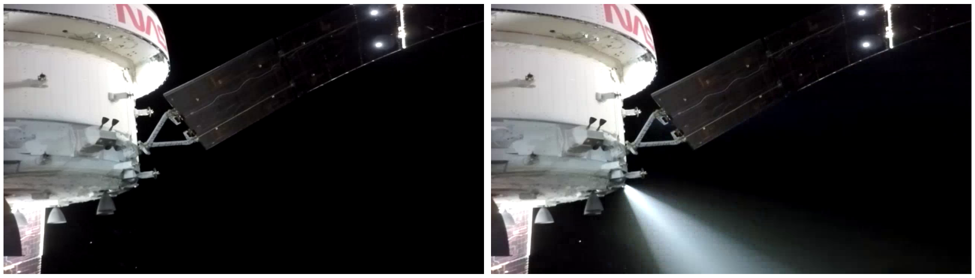

Figure 7. RCS plume illuminated by sunlight

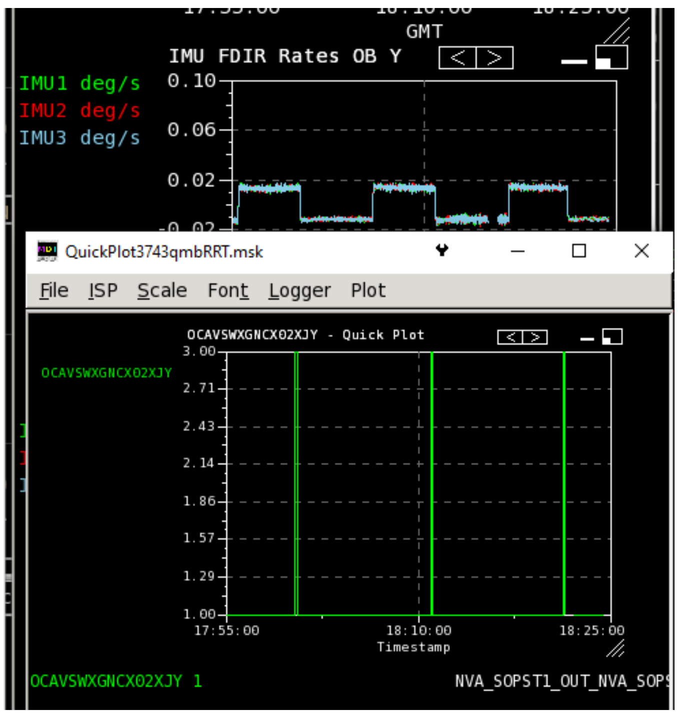

Figure 8. Star tracker dropouts correlated with RCS thruster firings (as evidenced in the Attitude rates)

The accelerations that were experienced during Outbound trajectory correction burns (in $\mathrm{ft/s^2}$) are depicted in Figure 10. Some of these correction burns were rather small ($\sim 0.1\,\mathrm{ft/s^2}$ (OTC2, OTC4)) and these used the RCS engines. The acceleration for OTC1 was large, because this burn had a deterministic component in order to test the Orion Main Engine (OME) was used to on OTC1. Some of the burns (OTC4, OTC5 and OTC6) used the AUX engines which provided accelerations on the order of 0.6 $\mathrm{ft/s^2}$.

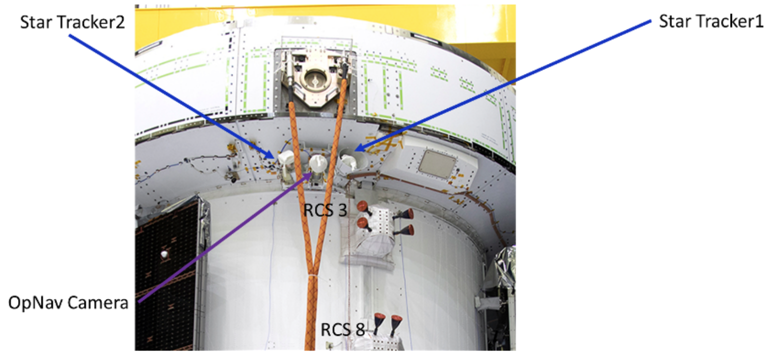

Figure 9. Star tracker, Optical Navigation Camera, and Thruster Layout

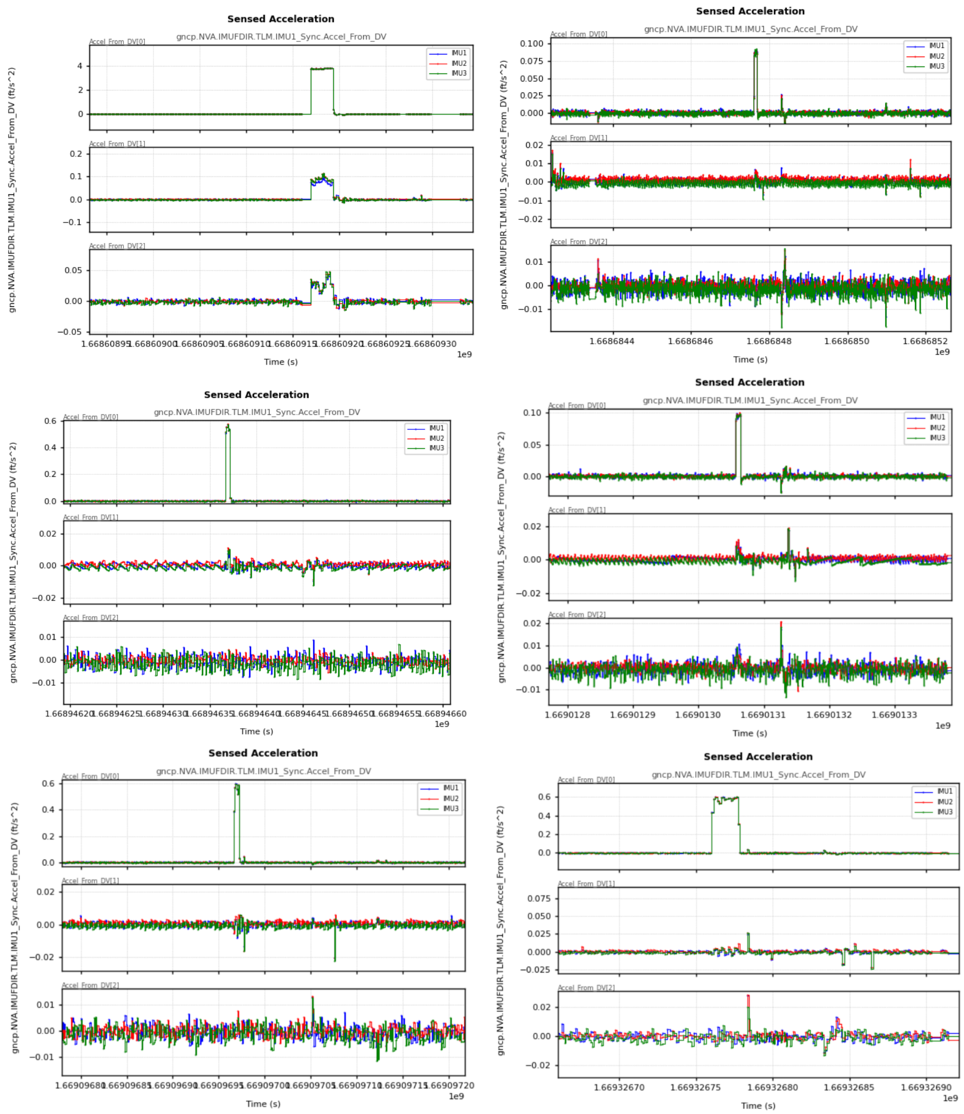

Figure 10. Outbound Trajectory Corrections 1-6

Optical Navigation

One of the primary FTOs of the Artemis I mission was to evaluate and certify the Optical Navigation system, which was designed to return the crew (on Artemis 2 and following) to Earth in the event of a Permanent Communication Loss (PCL) with the vehicle. To that end, a major focus during the outbound leg (particularly between TLI and OPF) was to certify the Optical Navigation (OpNav) system.

During cislunar flight the state is updated in one of two ways: a ground-based (DSN) update and an optical navigation update. The optical navigation update is done using onboard camera-based optical navigation measurements. These measurements are provided to the camera-controller computer which does the processing of the images obtained from the optical navigation camera. These images are time-tagged and correlated with the ST-based quaternion and are then used to generate a set of angles (actually tangents of the angle, derived from the centroid of the planet) and a range measurement (derived from the apparent diameter of the planet). These measurements were characterized pre-flight and a set of range-dependent noise and bias parameters from synthetic images of both Earth and Moon. They were anchored with a set of Earth-based Moon images and a few Earth images from missions OSIRIS-REx and Rosetta. It is not surprising that the Earth's atmosphere introduced a significant bias in the Earth measurements which was also range-dependent. For the Moon, noise due to the terrain was a larger factor than the bias. Due to software limitations, the same constant noise parameters were chosen for Earth and Moon (they were not range-dependent); likewise the same bias parameters were chosen for Earth and Moon. This resulted in a selection of conservative tuning parameters and thereby the performance was appropriately impacted.[1]

Prior to each OpNav pass, a camera calibration pass, consisting of rotating the vehicle to image stars, is performed. This was done to ensure that the proper camera parameters (and interlock angles between the STs and the OpNav Camera) are obtained and used. Immediately following this calibration pass, the vehicle was rotated to image the target body (either the Earth or the Moon, as appropriate) and then a OpNav pass commenced. During the Earth and Moon certification passes, a 2 hour navigation pass was performed. The measurements were processed (in Channel 3) in the CLEKF and the residuals and covariances of the position and velocity were evaluated. If they passed the pre-selected (before launch) criterion, that target body pass was declared to be certified. On the outbound leg, there was one Earth pass (with a backup) and one Moon pass which was used for the certification of the OpNav system for a autonomous return in a PCL scenario. During the remainder of the mission (after certification), there were OpNav passes but these were just image-gathering passes (for further analysis post-mission) and the duration of these passes was only 15 minutes each. The images associated with the Earth passes are shown in Figure 11.

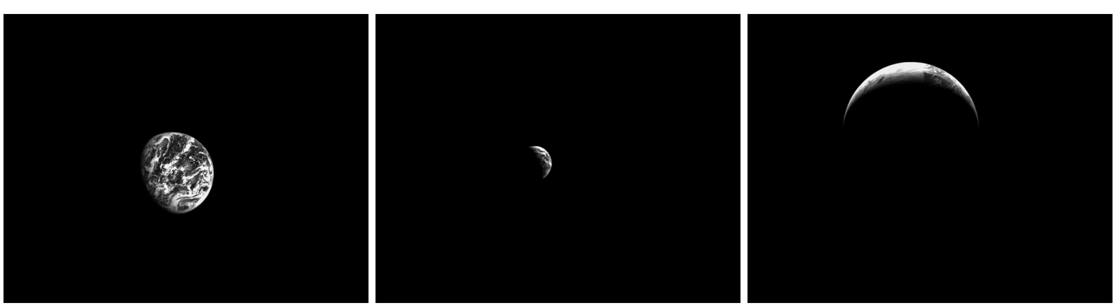

Figure 11. Earth images captured by OpNav Camera - close, far, thin

The OpNav Moon images shown in Figure 12. During the OpNav passes, careful attention was

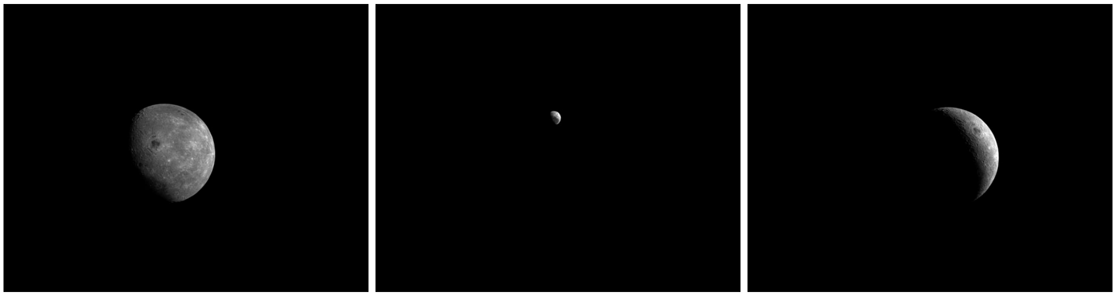

Figure 12. Moon images captured by OpNav Camera - close, far, thin

paid to the temperature of the OpNav camera. A typical temperature profile is found in Figure 13.

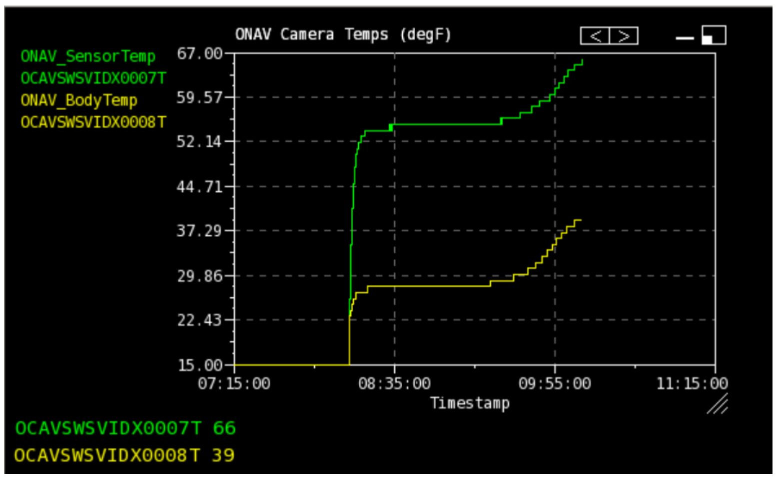

Figure 13. OpNav camera temperatures during a pass

The Optical Navigation camera captured spectacular lunar surface imagery during the return flyby encounter, shown in Figure 14. These images were part of a developmental flight test objective experiment to collect test imagery for lunar feature terrain-relative navigation studies. Apollo landing sites were captured during the Artemis I return flyby as seen in Figure 15. The Apollo 14 approach navigation craters Fra Mauro Y and D are clearly visible. The Apollo 15 Rima Hadley valley exploration target is also clearly visible.

DISTANT RETROGRADE ORBIT (DRO)

During the DRO phase, the navigation system performed as expected. There were several thermal and propulsion system DFTOs that were inserted. As well, additional OpNav images were obtained and downlinked to further add to the library of images to better characterize the optical navigation system. As in the other phases, these images were gathered during daily 15 minute OpNav Image gathering passes. The Navigation system performed nominally during this phase with (thankfully) no surprises. There were three small correction burns to maintain the DRO orbit, but these were small and not particularly noteworthy.

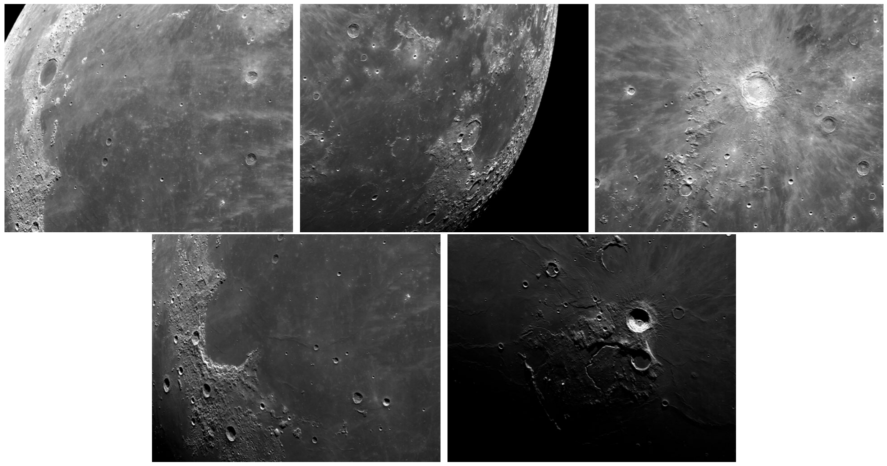

Figure 14. Lunary Flyby Imges

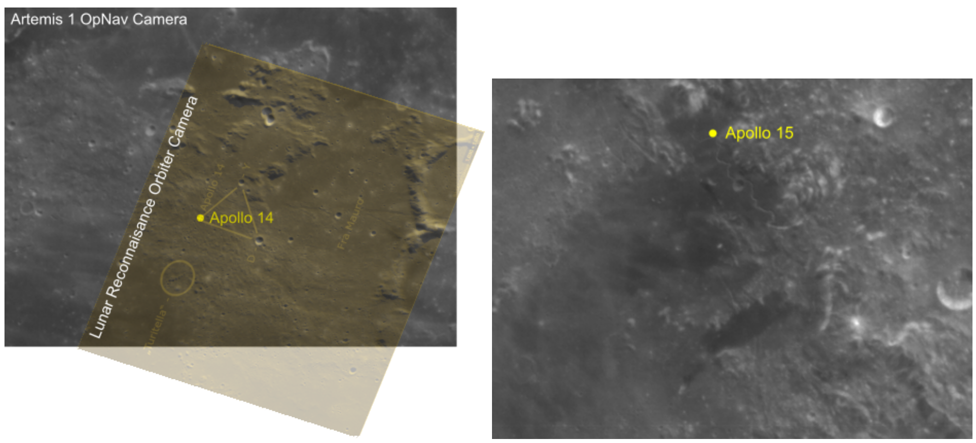

Figure 15. Apollo landing sites imaged by Artemis I OpNav camera

RETURN LEG

During the return leg, which occurred after the departure from the DRO, through the Return Powered Flyby (RPF) maneuver, there were 6 correction burns (3 before RPF and 3 after) performed. The final three (after RPF) are depicted in Figure 16. RTC4 and RTC6 were performed with the RCS thruster system; RTC5, which was larger (because there were several days between RTC4 and RTC5) was performed with the AUX engines. It should be noted that RTC4 occurred 18 hours after RPF, RTC5 occurred 22 hours before EI, and RTC6 occurred 5 hours prior to EI.

Figure 16. Return Trajectory Correction Burns

ENTRY, DESCENT, AND LANDING

In contrast to the Return leg, the Entry, Descent, and Landing (EDL) phase of the Artemis I mission was rather more eventful. As one might expect during the run-up to entry interface, all eyes were on the GPS to see when it would acquire. All the entry sensors agreed within expectation as shown in Figure 17. The backup barometric altimeters were able to sense the atmosphere and provide measurements at around 80kft.

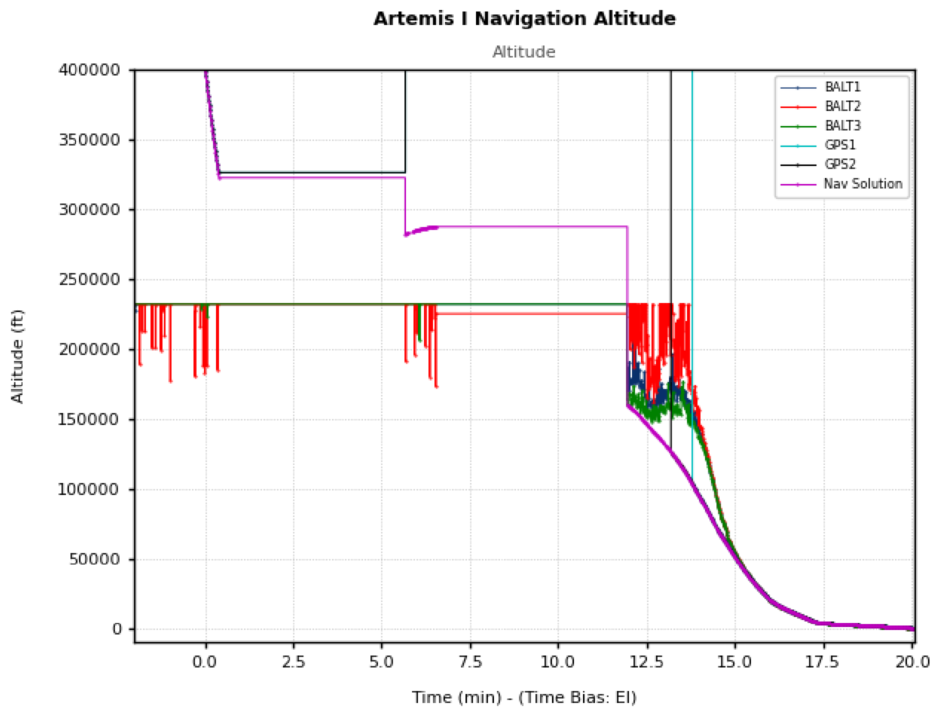

Figure 17. Altitude readings from nav sources during entry

The IMUs tracked the entry acceleration profile well as shown in Figure 18. There are notable data dropouts during communication loss, but the profile shows between 3-4 g's acceleration when the vehicle emerges from blackout at around 160 kft.

GPS had good processing both before and after the plasma blackout as seen in Figure 19. As was predicted by a number of pre-flight simulations, GPS did not reacquire during the brief exo-atmospheric skip. The GPS processing residuals from the navigation filter are shown in Figure 20, showing rapid convergence and low noise levels.

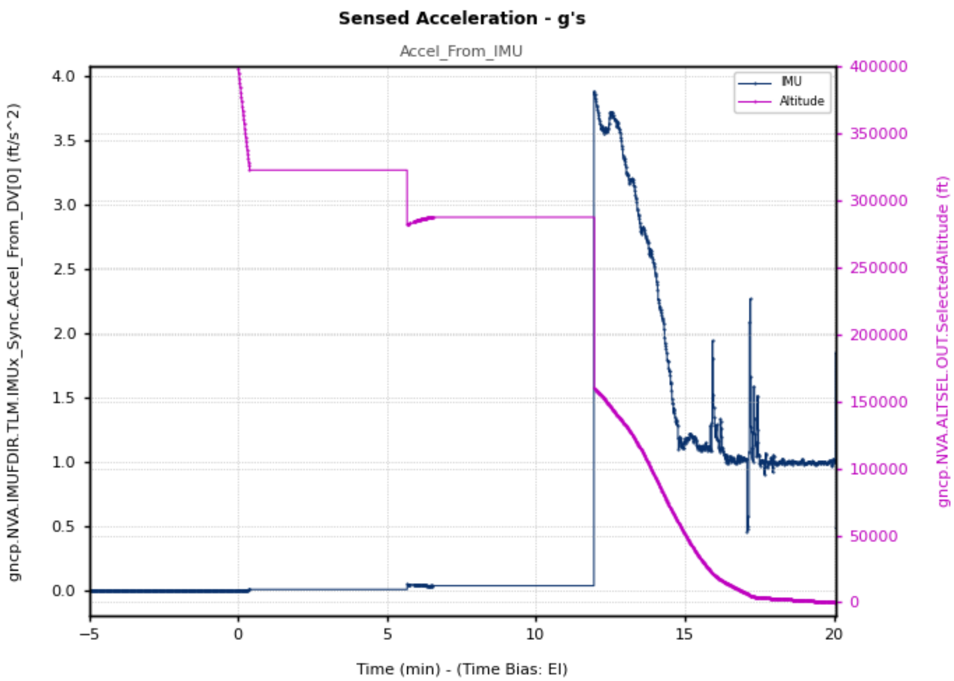

Figure 18. Entry sensed acceleration

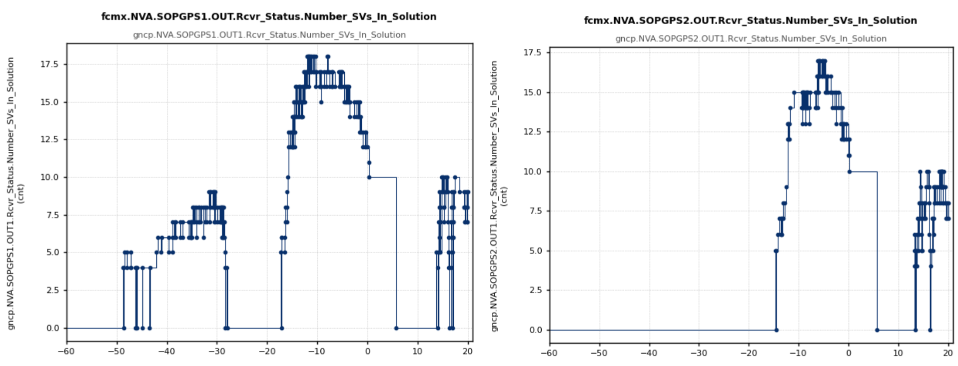

Figure 19. GPS satellite visibility on entry

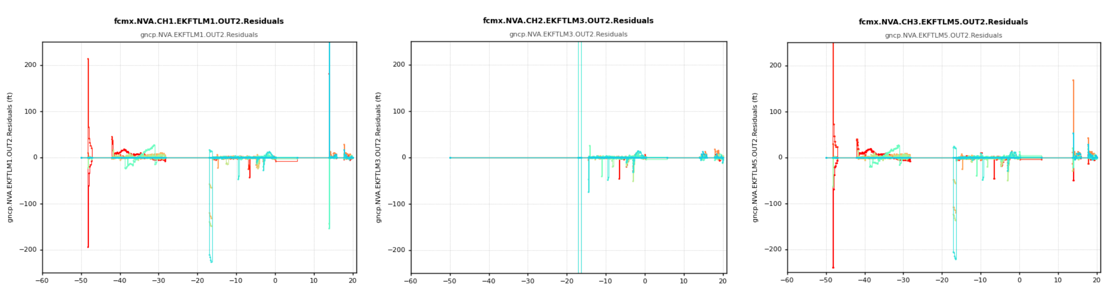

Figure 20. GPS residuals on entry

CONCLUSIONS

The Artemis I mission was a resounding success by any and all standards. The Navigation System, apart from the few initial growing pains involving the Star Trackers, performed flawlessly. The navigation filters, all of them, exceeded expectations. In fact, none of them rejected any measurements. The Optical Navigation system performed better than expectations, especially considering most of the images used in the development and testing were synthetic images. Whereas there is not an insignificant amount of forward work involving the analysis of the data (and the images), and there will be minor changes to the navigation software for Artemis 2 and following, the Artemis I mission validated the design choices made during the development. The Artemis navigation system is well positioned to support the crewed missions for future Artemis flights.

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