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LAUNCH AND COMMISSIONING OF THE LUNAR RECONNAISSANCE ORBITER (LRO)

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The Lunar Reconnaissance Orbiter (LRO) launched on June 18, 2009 from the Cape Canaveral Air Force Station. LRO, designed, built, and operated by the National Aeronautics and Space Administration (NASA) Goddard Space Flight Center in Greenbelt, MD, is gathering crucial data on the lunar environment that will help astronauts prepare for long-duration lunar expeditions. To date, the Guidance, Navigation and Control (GN&C) subsystem has operated nominally and met all requirements. However, during the early phase of the mission, the GN&C Team encountered some anomalies. For example, during the Solar Array and High Gain Antenna deployments, one of the safing action points tripped, which was not expected. Also, the spacecraft transitioned to its safe hold mode, SunSafe, due to encountering an end of file for an ephemeris table. During the five-day lunar acquisition, one of the star trackers triggered the spacecraft to transition into a safe hold configuration, the cause of which was determined. These events offered invaluable insight to better understand the performance of the system they designed. An overview of the GN&C subsystem will be followed by a mission timeline. Then, interesting flight performance as well as anomalies encountered by the GN&C Team will be discussed in chronological order.

INTRODUCTION

The National Aeronautics and Space Administration (NASA) Lunar Reconnaissance Orbiter (LRO) launched on June 18, 2009 from the Cape Canaveral Air Force Station aboard an Atlas V launch vehicle and into a direct insertion trajectory to the Moon; Figure 1 is a drawing of LRO. LRO, designed, built, and operated by the NASA Goddard Space Flight Center (GSFC) in Greenbelt, MD, is gathering crucial data on the lunar environment that will help astronauts prepare for long-duration lunar expeditions. The mission, with a nominal one-year duration, utilizes a complement of seven scientific instruments to find safe landing sites, locate potential resources, characterize the radiation environment, and test new technologies[1]. From launch and early cruise operations through commissioning and into its one year mission, LRO has performed very well, with all subsystems operating well within the bounds of their respective design requirements, and has been collecting excellent science data. An early mission highlight was the release of images taken by the LRO Camera Narrow Angle Camera (LROC NAC) of the Apollo landing sites, ap-

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pearing on cable news networks, showing detailed images of landing hardware and faint but visible footprints left by Apollo astronauts[2].

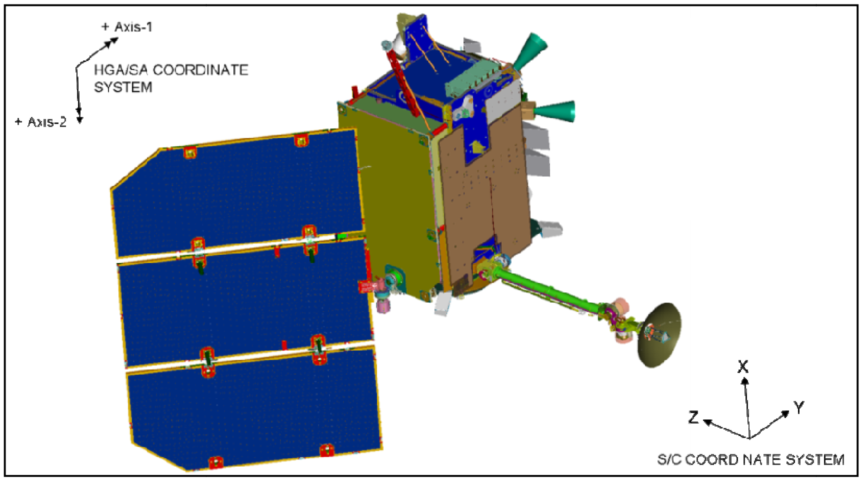
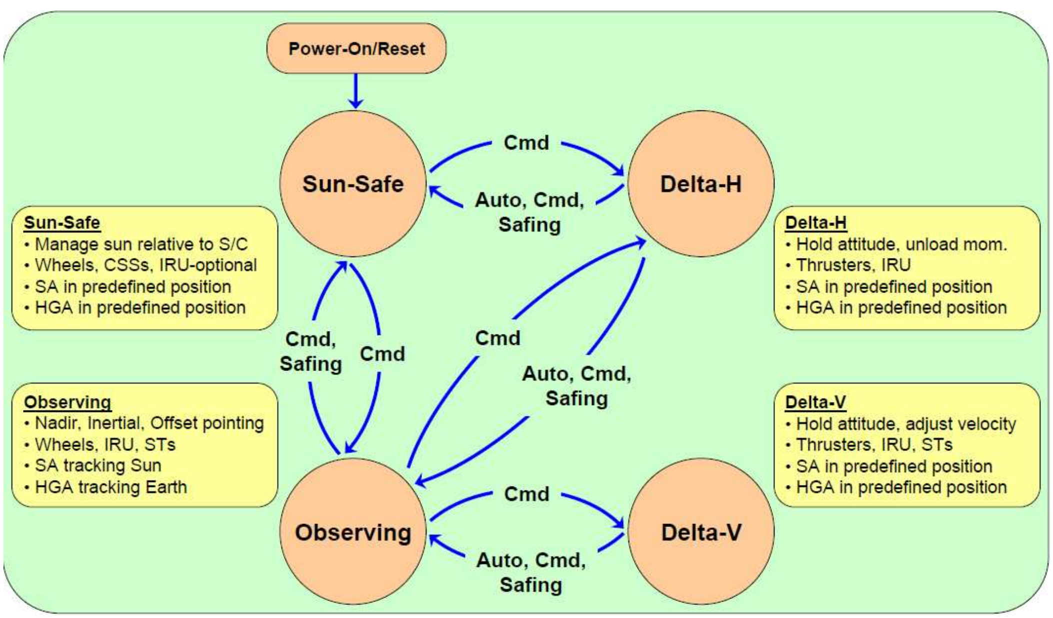


Figure 1. LRO mechanical drawing, with spacecraft body coordinate system (BCS) identified

This paper begins with an overview of the GN&C subsystem and a discussion of the LRO mission timeline. An early mission chronological narrative follows, describing GN&C subsystem performance and interesting GN&C events (including some unexpected safe mode entries) that occurred over a period from separation to cruise to lunar orbit insertion and the spacecraft commissioning phase.

LRO GUIDANCE, NAVIGATION AND CONTROL SUBSYSTEM

LRO's GN&C subsystem is made up of an onboard attitude control system (ACS), a collection of software algorithms based on high level and derived requirements, and a hardware suite of sensors and actuators. Figure 2 shows the LRO control modes, a brief description of their functions, hardware required and the transitions between each mode. Transitions are defined as automatic based upon defined criteria (Auto), commanded from ground or onboard algorithm (Cmd), or as the result of an anomaly (Safing). A discussion of each control mode and the hardware suite follows.



Primary Attitude Control Mode - Observing Mode

Observing mode is LRO's primary attitude control mode, which provides full-sky 3-axis spacecraft attitude slewing and fine pointing for lunar nadir and off-nadir as well as inertial targets[3]. This control mode is used for all nominal pointing and slewing operations including those for science data taking, instrument calibration, as well as initial attitude acquisition during Delta-V mode operation. The hardware complement used for this mode includes four reaction wheels (RW), two star trackers (ST), and one 3-axis inertial reference unit (IRU). Two quaternion output STs are operated continuously, providing a highly accurate attitude reference to ensure attitude performance during occasional single star tracker occultation. Attitude and rate sensor data is processed onboard using a six-state Kalman filter[4], which estimates IRU bias and spacecraft inertial attitude quaternions. The filtered quaternion data is used as the attitude source for onboard attitude control as well as the definitive spacecraft attitude for ground processing of science data. Control torque commands are generated using a standard PID type control law. This algorithm utilizes quaternion feedback proportionally-limited attitude error to ensure that all slews closely track the eigenaxis with a constant slew rate. An attitude limit is set to ensure ample torque margin at a slew rate of 0.1 deg/sec over the design system momentum range of 80 Nms.

Thruster Maneuver Mode – Delta-V and Delta-H

Cruise trajectory, mid course correction, lunar orbit insertion, and orbit maintenance maneuvers are accomplished using a GSFC in-house developed propulsion system to adjust the spacecraft's velocity magnitude and vector direction. The system utilizes four 20-lbf thrusters (insertion thrusters or NT) and eight 5-lbf thrusters (attitude thrusters or AT) for orbit adjustments and attitude control during maneuvers. The NTs are reserved for providing orbit adjust for the lunar orbit insertion maneuvers following a trans-lunar cruise phase. The ATs are used for attitude control and/or orbit adjustment during all phases of the mission. Delta-V mode uses a PID controller to generate thruster torque commands for attitude control during orbit adjust operations. This mode uses an IRU propagated attitude for feedback control since the higher body rates and accelerations during propulsion may be too large to maintain reliable star tracker operation. Spacecraft angular velocity measurements from the IRU are also used for rate feedback in the Delta-V controller. Performance for this mode varies with the operational events during the mission lifetime. The performance requirement is averaged over the burn period because there are initial transient periods of high rate and attitude errors that settle out over time. To improve the transient response and settled attitude errors, a feed-forward torque, estimated based on thruster configuration and spacecraft mass properties, was added.

The ACS also provides for a thruster based system momentum management algorithm (known as Delta-H) to maintain the system and wheel momentum to within acceptable levels. Delta-H employs a similar algorithm as Delta-V, with the two differences being the method for computing the reaction wheel commanded torque and the removal of the integral term. Each reaction wheel is actively commanded in the controller to a commanded momentum level. A system momentum level command is also used to determine when the targeted momentum level has been reached, to within some acceptable tolerance. The ATs are used to maintain attitude pointing while the wheel and system momentum is 'dumped'. At the completion of the maneuver, the system reverts back to the mode from which Delta-H entered, typically Observing mode.

Spacecraft Safe Hold Control – SunSafe and SunSafe Gyroless

A coarse sun-pointing mode (SunSafe) is responsible for initial attitude acquisition, and in the event of a perceived spacecraft anomaly, providing a sun-pointing, power and thermally safe, attitude for an indefinite period of time. SunSafe mode makes use of ten coarse sun sensors (CSS) to determine a sun vector, and an IRU to determine the spacecraft angular velocity in the body frame. Through use of reaction wheels, SunSafe mode provides full-sky reorientation to a sun-pointing attitude along a ground supplied target sun vector in the body frame. To reduce the momentum accumulation due to solar radiation torques, a rate bias is commanded, resulting in a spacecraft roll about the sun target vector.

Nominally, SunSafe utilizes CSS and IRU data for feedback control, but in the event of an IRU failure, also includes a gyroless sub-mode. SunSafe Gyroless exists to compute rate information from the CSS data alone. Since the IRU is no longer in the feedback loop, the spacecraft rate about the sun line cannot be determined. Therefore, sun-pointing is maintained without control about the sunline. Another difference in these safe hold modes is that during eclipse; SunSafe Gyroless does not have attitude or rate feedback so the spacecraft is allowed to drift until sun presence returns (essentially open-loop control during eclipse).

Adcole Coarse Sun Sensors

The most basic sensors used on LRO to determine spacecraft attitude are ten Adcole coarse sun sensors. These small devices collect sunlight and convert the amount of sunlight into an electrical current. That current level is then translated into a digital signal, which is then processed by the onboard flight software. The amount of sunlight produced by each CSS is a measure of the angle between the line to the sun and the bore sight of the CSS. When combined with other CSS measurements, a three dimensional estimate of where the Sun is relative to the spacecraft body frame is computed.

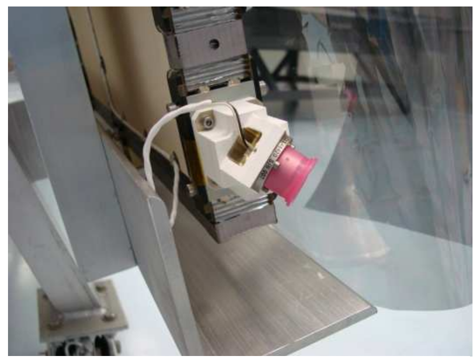


Figure 3. Adcole CSS mounted on LRO Solar Array

While relatively simple, these sensors are extremely important to the health and safety of the spacecraft. These devices are very small, relatively inexpensive, yet their input is the key to providing an attitude input to SunSafe. Without them working correctly, having the spacecraft thermally and electrically safe is questionable.

SELEX Galileo Star Trackers

Fine pointing attitude determination is accomplished by two SELEX Galileo Star Trackers. These STs have a 16° x 16° field of view and produce a quaternion attitude solution in the ST frame with respect the J2000 Earth Centered Inertial reference frame. Each tracker produces an attitude solution at 10 Hz but for the LRO application is read and processed at the ACS cycle of 5 Hz. Both trackers receive a one pulse per second (PPS) timing synchronization input from the spacecraft's onboard oscillators (clocks) and have their own 1553 Remote Terminal address for independent operation.

Star tracker processing includes a derived rate option that filters a differentiated quaternion output from the tracker in the event that the IRU onboard the spacecraft becomes unavailable. These derived rate measurements serve as a backup rate during critical Delta-V operations and can allow the Observing mode to function, albeit with degraded performance, to continue science instrument data collection and calibration.

Honeywell Miniature Inertial Measurement Unit

Honeywell's Miniature Inertial Measurement Unit (MIMU) serves as a single IRU. It contains 3 ring-laser gyros (one gyro per axis) that are used to produce spacecraft inertial body rates.

This unit outputs an accumulated angle, which is sampled by the onboard system at 10 Hz and differenced to produce the body rates. These 10 Hz rate estimates are averaged over two samples to provide a rate for use in the feedback loop at the ACS cycle time of 5 Hz. Due to accumulated angle roll-over ambiguity, the MIMU gyros provide only direction of spin but not the magnitude of the attitude rate for rates above a saturation level. For LRO the IRU saturation level is 18 deg/sec. The MIMU was turned on prior to launch and, has performed flawlessly in continuous operation during the LRO mission.

Figure 4. MIMU mounted on LRO

NASA GSFC Demiseable Integrated Reaction Wheels

Fine pointing control of the LRO spacecraft is accomplished through the use of four Reaction Wheels (RWs). These actuators were developed, built, tested, and delivered in-house by the NASA Goddard GN&C Hardware Branch. Each RW uses an 18-inch aluminum flywheel, providing inertia for momentum exchange. The RWs are mounted in a pyramid configuration with a reaction wheel assembly system momentum storage capacity of 130 Nms.

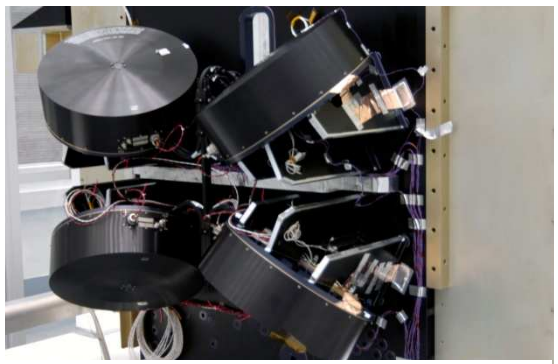


Figure 5. Reaction Wheel Assembly mounted on LRO in a tetrahedron configuration

The LRO RW contains several design features. Each wheel has software and firmware for controlling either wheel speed or wheel torque as well as the reporting of telemetry. Each wheel also has a separate 1553 Remote Terminal address. The internal software and firmware can be reprogrammed from the ground if desired during the mission. Although not used for this lunar mission, the wheel design has the added feature of nearly complete burn-up if it were to plummet into the Earth's atmosphere at the end of its life.

Propulsion Deployment Electronics

The Propulsion Deployment Electronics (PDE) has a number of functions used to complete the LRO mission. First, the PDE commands the firing of the thrusters when desired as well as firing of the Propulsion NASA Standard Initiators (NSI) for opening of the propulsion pressurization tank. During the Solar Array (SA) and High Gain Antenna (HGA) deployment, the PDE is used to turn on the Non Explosive Actuators (NEA). Additionally, the PDE prevents the radio frequency transmitters from functioning until launch vehicle fairing separation, and provides extra switch capabilities for science payload heaters. This component was designed to allow the propulsion system to have redundant drive electronics to support the redundant set of thrusters. The PDE is designed

Figure 6. PDE

with four identical electronic boards and one module for the prevention of certain spacecraft functions at the desired times.

The design of the four PDE electronic boards allows for the critical Lunar Orbit Insertion (LOI) maneuvers to be completed, even if one of the four identical electronic boards is declared suspect. Each of the four electronic boards has their own 1553 remote terminal address. The thrusters are connected to electronic boards in a manner that allows LRO to complete its desired orbit adjustment despite a thruster failure.

LRO MISSION TIMELINE

The first twelve hours of the flight were packed with action (refer to Table 1). Immediately after separation, SunSafe, the default control mode, autonomously removed the tip-off rates and placed the spacecraft in a power-positive orientation. Afterwards, the deployables were extended to put the spacecraft in its flight configuration. The GN&C hardware as well as other subsystems were powered on, and the ACS mode was transitioned to Observing mode. Thruster one-shots were performed to ensure thrusters were working, and the communication was switched from omni-directional antenna to the HGA.

One day after launch, LRO conducted a mid course correction maneuver to adjust the inclination of its orbit. After the mid course correction, LRO cruised towards the moon with little excitement for three days. On the fifth day after launch, June 23, 2009 at 6:26:26am EDT, LRO completed a flawless LOI maneuver to place itself in an elliptical orbit about the Moon (Figure 7 shows the various phases of LRO's orbit during its mission: initial trajectory, LOI, commissioning orbit, and nominal orbit). Over the next five days, four more LOI maneuvers transferred LRO into its commissioning orbit.

Over the next ten weeks, the spacecraft and instrument teams commissioned their respective systems. Commissioning was when the GN&C Team had a chance to test out its primary attitude control mode as well as calibrate all of its systems. Once commissioning was completed, a series of thruster maneuvers, mission orbit insertion (MOI), lowered LRO into its nominal 50 km mean mission orbit where it began collecting meaningful science data.

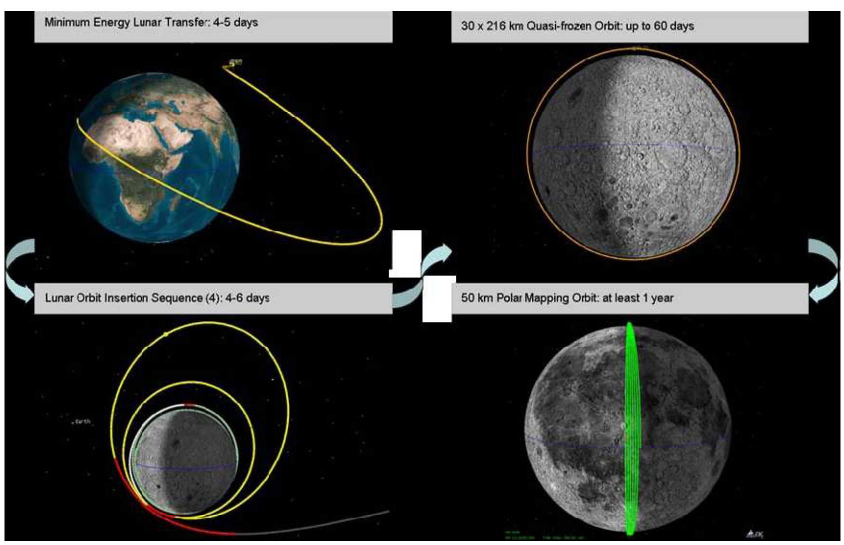


Figure 7. LRO Orbit Phases

Table 1. LRO Mission Timeline

Phase	Entry	Exit	Duration	Objectives
Pre-Launch	Start of LV Count-down Sequence	LV Lift-off	~1 Day	• Configure Orbiter into Launch Mode • Short Spacecraft Checkout
Launch	LV Lift-off	Payload Separation	~90 Minutes	• Achieve Trans-Lunar Trajectory
Early Cruise	Payload Separation	Observing Mode	~90 Minutes	• Sun Acquisition and Ground Acquisition • Deployments • Initial MCC Tracking
Mid Cruise	Observing Mode	Completion of MCC	~1 Day	• Propulsion Checks • Final MCC Planning • Execution of MCC Burn within L+24 hrs
Late Cruise	Completion of MCC	Start of LOI Sequence	3-4 Days	• LEND/CRaTER Early Turn-On Activities • Spacecraft Functional Checkout • LOI Planning
Lunar Orbit Acquisition	Start of LOI Sequence	Commissioning Orbit	4-6 Days	• Perform Lunar Orbit Capture Maneuver • Achieve 30x216 km Commissioning Orbit
Commissioning	Commissioning Orbit	Mission Orbit	Up to 60 Days	• Spacecraft Checkout and Calibrations • Instruments Checkout and Calibrations • Mission Orbit Adjustment
Nominal Mission	Mission Orbit	After 1-Year Nominal Operations	1-Year	• Routine Operations • Non-Routine Operations • Data Product Generation
Extended Mission	After 1-Year Nominal Operations	Impact	Up to 3 Additional Years	• Goals to be Determined • Impact Prediction/Activities
End-of-Mission	Impact	Completion of Closeout Activities	N/A	• Finalize Mission Operations/Activities

MISSION CHRONOLOGY

SUN ACQUISITION

LRO separated from the Atlas V launch vehicle with tip-off rates of approximately [-0.1, 0.2, -0.8] deg/sec in the x,y, and z axes, respectively. Figure 8 shows the autonomous acquisition of the Sun by the SunSafe control mode. Shown are the spacecraft rates and spacecraft body sun vector. During the separation, the spacecraft was in a sun-avoidance state, which seeks to keep the sun off of the spacecraft +Z-axis, to avoid instrument illumination. SunSafe simultaneously removed the tip-off rates, as well as began maneuvering the spacecraft so the Sun is on the -Z-axis. Once sun-avoidance was exited, SunSafe began maneuvering the spacecraft to point the folded SA towards the Sun (SA on the –Y-axis). The presented data shows that it took approximately 8.5 minutes to acquire the Sun after separation.

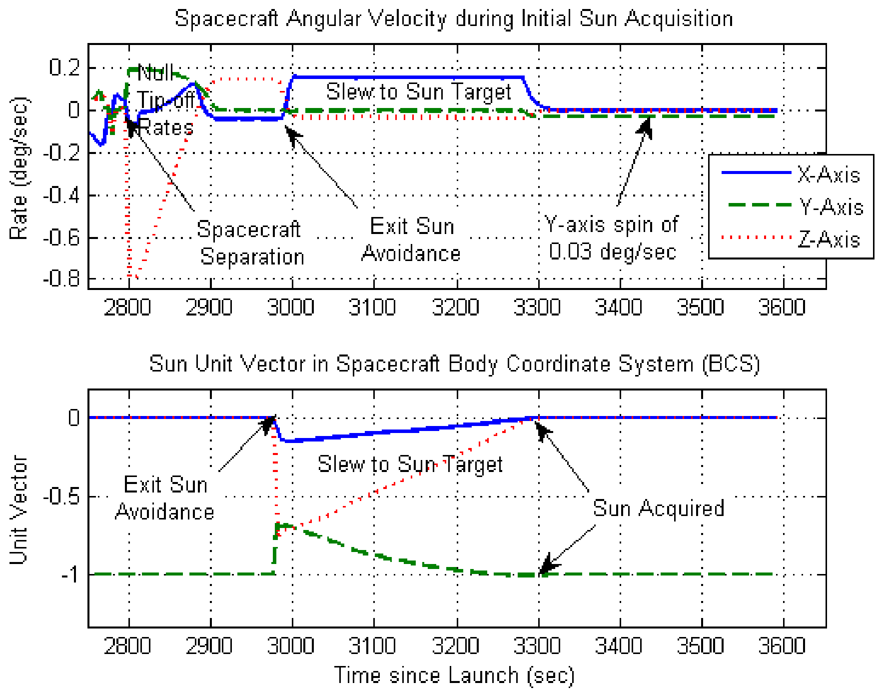


Figure 8. Spacecraft Angular Velocity and Sun Vector during Sun Acquisition. The Sun target is the –Y spacecraft axis – i.e. a target vector of [0,-1,0].

CRUISE

This section details GN&C activities after sun acquisition until lunar orbit acquisition. Fifty minutes after separation, the SA was deployed to increase power accumulation. Twenty-five minutes later, the HGA boom was deployed to put the spacecraft into its deployed configuration. Five minutes after deployments were completed, the star trackers were turned on and the ACS control mode was switched from SunSafe to Observing for the first time. Immediately afterwards, the GN&C Team ran a thruster one-shot test, where each of the eight ATs and each of the four NTs were pulsed one time while the spacecraft was holding an inertial attitude. These one-shot tests demonstrated that thruster performance was within specified tolerance of all pre-planned metrics.

Solar Array and High Gain Antenna Deployment

The Mechanical subsystem team requested that the GN&C team monitor the IRU rates in hopes that this data would help to verify proper deployment of the HGA and SA. In particular, they were interested in assessing proper functioning of the deployment latch mechanisms. While it was not possible to absolutely verify proper functioning of the latch mechanisms using IRU data alone, it was thought that careful inspection of the rate data would provide some corroborating evidence of proper deployment. Figure 9 presents the spacecraft angular velocity during the SA and HGA deployments. The deployment process began with the firing of a non-explosive actuator, which released the SA from the spacecraft. To allow the SA panels to unfold, the SA was mechanically driven to an extended position (identified as (90,-45) degrees). During this time, the two panels began to unfold with the assistance of spring mechanisms. The SA panels latched into place when they reached their deployed positions. Afterwards, the SA was driven back to its ‘index’ position, identified as (90, 45) degrees. A few minutes later, the HGA boom was released from the spacecraft body and extended to its deployed configuration with the aid of a non-explosive actuator and a spring-loaded mechanism. A latch engaged when the HGA boom arrived at its deployed position.

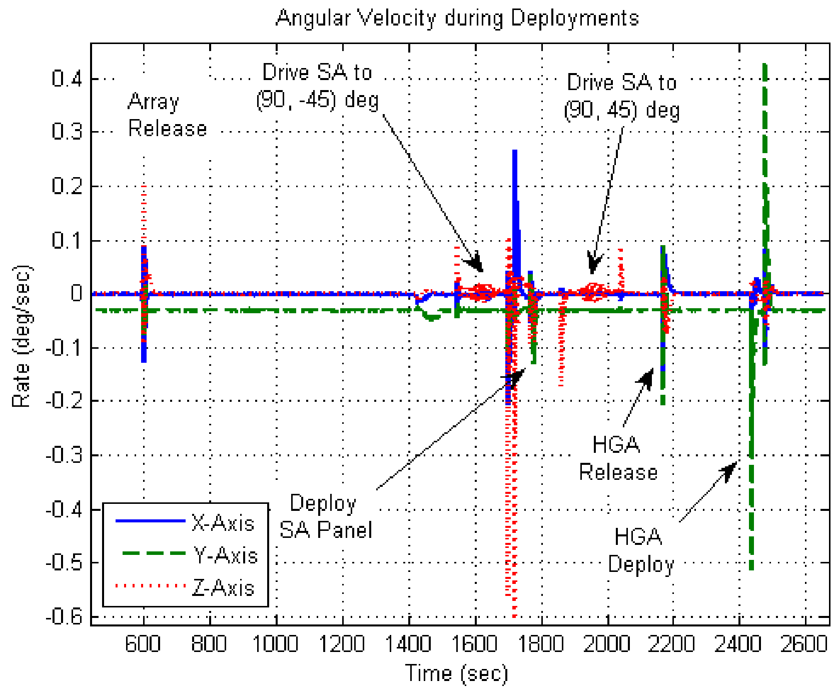


Figure 9. Spacecraft Angular Velocity during Deployments

During both deployments, a system momentum rate of change safing action point (AP) was tripped in the Fault Detection and Correction System (FDC). The system momentum rate of change limit was set to 3.3 Nm and the action point trips if the limit is exceeded for more than 3 seconds. As can be seen in Figure 10, the limit is exceeded for more than 3 times on a few occasions during the deployment sequence. The original purpose of this safing action point was to detect and prevent system momentum from growing too fast to handle with wheels, and to prevent fuel loss. When this AP tripped, the FDC commanded the spacecraft to isolate the propulsion system, transition to SunSafe, and stop any activities occurring on the spacecraft. In the deployment situation, there were no issues with the propulsion system because none of the isolation valves were open and the spacecraft was already in the SunSafe control mode.

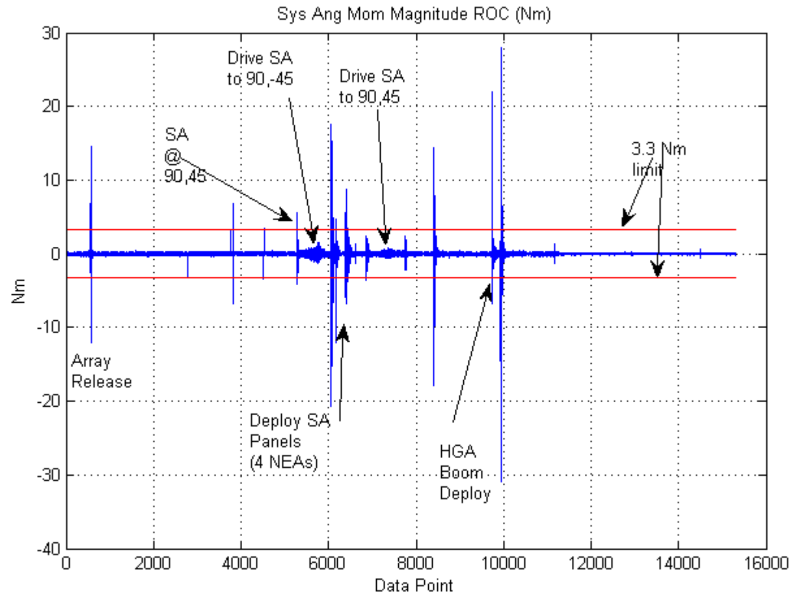


Figure 10. Spacecraft System Angular Momentum Rate of Change during Deployments

First Anomalous Entry into SunSafe

The spacecraft's first anomalous entry into SunSafe occurred soon after the thruster checkout and during the first transition into Observing mode. The transition into SunSafe was caused by not having ephemerides loaded to the spacecraft. During nominal operations, the spacecraft generates the target attitude using an on-board ephemeris uploaded from the ground. However, during early operations, the spacecraft was using a ground uploaded target attitude and therefore ephemerides were not needed. Unfortunately, the safing configuration did not check to see what the target source was and automatically transitioned the spacecraft back into SunSafe. There was very little impact to the spacecraft operations due to this event, and no further action was required.

Observing Mode Checkout

Shortly after the thruster one-shot tests and spacecraft appendage deployment were successfully completed, a series of attitude slews were performed using the Observing mode controller to provide an initial set of data for calibration of the IRU bias, alignment, and scale factor. This "mini-cal" of the IRU was followed up later with a more precise IRU calibration involving a series of large angle slews over several hours. At this time, a relative ST calibration was also performed using ST quaternion data to compute corrections to ST alignments, removing any residuals between the ST quaternion outputs. This corrected a known error in the ST alignment, loaded at launch, which was on the order of a few tenths of a degree. Calibration parameters were uploaded to the spacecraft, and the onboard attitude Kalman Filter (KF) was initialized for the first time using measured data from both STs. During the cruise phase, both STs were operating with nominal performance and the KF remained converged with a steady-state attitude estimation error covariance of approximately 2 to 3 arc-sec (1 sigma). Since fine pointing performance was not needed during the cruise phase, the KF output was not used for attitude feedback or for IRU bias correction. Instead, a ST was selected as the attitude source and a constant bias correction was applied to the IRU rates used for feedback control. This provided the team with the opportunity to trend the KF performance during the transit to the moon before utilizing KF output in the feedback, as required for fine pointing performance during the rest of the mission. Figure 11 shows a plot of typical attitude errors during cruise, when the attitude feedback was set to the ST output. Note the boresight of this ST is aligned within about 30 deg from the spacecraft Z-axis and perpendicular to the X-axis of the spacecraft. Thus, the larger ST errors about the boresight result in relatively larger attitude errors about the Z and Y axes.

Occasionally, the attitude source was set to IRU propagation. This was primarily done in preparation for Delta-V operations and during ST occultations, which occurred when the spacecraft maintained inertial attitude during initial lunar orbit operations. This gave the team a chance to observe the best controller performance that could be achieved when targeting inertial attitudes with minimal disturbances present. Figure 12 shows typical 1 arc-sec level attitude error performance attained using IRU propagation as attitude feedback, when the SA was fixed, no spacecraft instruments were moving, and the reaction wheels were at low momentum levels. The data from Figure 12 was captured on Day of Year (DOY) 174. Attitude error when the IRU propagation was used as the attitude source, during inertial attitude targeting, demonstrated the best controller performance that was to be expected when the KF was used as the attitude source.

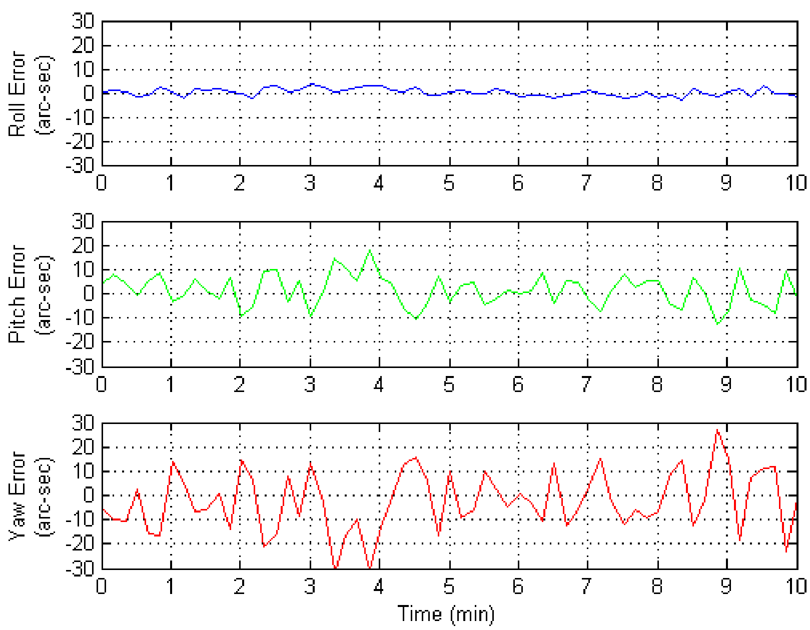


Figure 11. Typical Attitude Error when ST selected for Attitude Feedback during Cruise

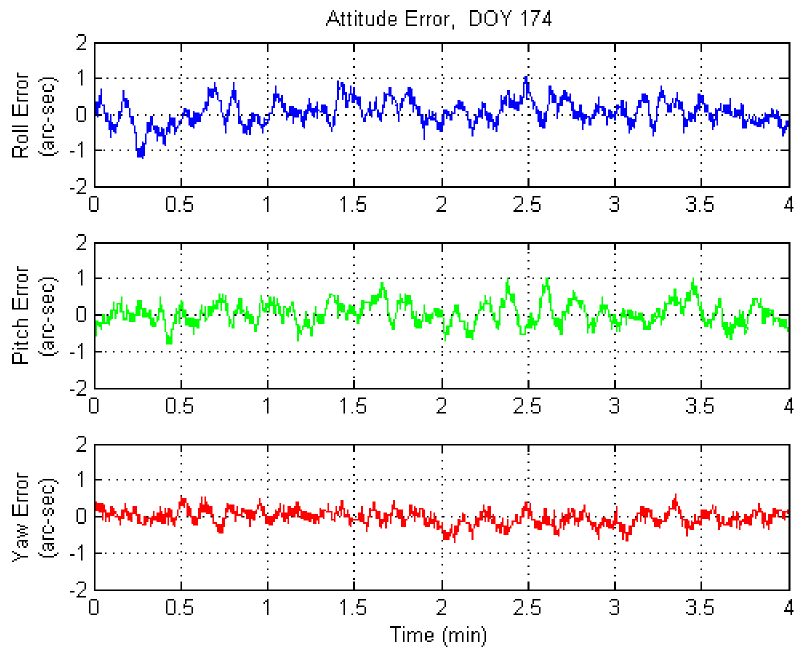


Figure 12. Best Attitude Performance Using IRU Propagation During Cruise Phase (DOY 174)

LUNAR ORBIT ACQUISITION

The LRO spacecraft performed a series of seven thruster maneuvers to reach the spacecraft's commissioning orbit with the first and third maneuvers being critical and the third maneuver the most critical. This section discusses the first maneuver, Mid Course Correction 1, and the third maneuver, Lunar Orbit Insertion 1.

Mid Course Correction 1

Mid Course Correction 1 (MCC-1) was performed on June 19, 2009, approximately 24 hours after launch and lasted about 48 seconds. MCC-1 used 4 out of the 12 thrusters on-board the spacecraft in an off-pulsing configuration. The maneuver executed flawlessly and phase plane plots of the maneuver are shown in Figure 13. The horizontal axis of each subplot represents the attitude error and the vertical axis of each plot represents the rate error seen by the Delta-V controller. The maximum absolute attitude error was 0.57 deg and the maximum absolute rate error was 0.13 deg/sec.

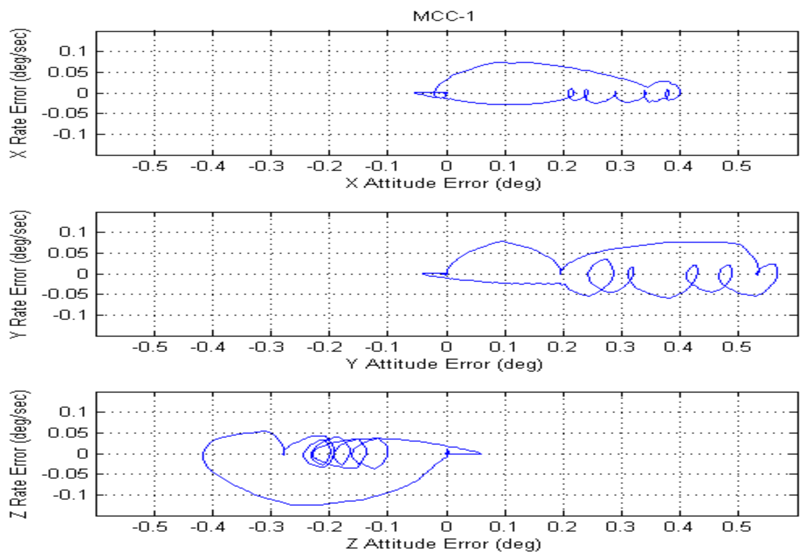


Figure 13. Mid Course Correction Maneuver Phase Plane Attitude and Rate Error

Lunar Orbit Insertion 1

The third Delta-V maneuver performed by the spacecraft was Lunar Orbit Insertion 1 (LOI-1). LOI-1 was the longest thruster maneuver executed by the spacecraft, 40 minutes, and used all 12 of the thrusters on the spacecraft. The 4 NTs were full on, and the 8 ATs were controlled in an On-Pulsed Mode. LOI-1 was used to slow down the spacecraft in order to be captured by the Moon's gravitational field. Any issues with this critical maneuver could have resulted in a degraded science mission or loss of mission. This maneuver executed flawlessly with small attitude and rates errors as shown by the phase plane plots in Figure 14. The maximum absolute attitude error was 3.2 deg and the maximum absolute rate error was 0.36 deg/sec.

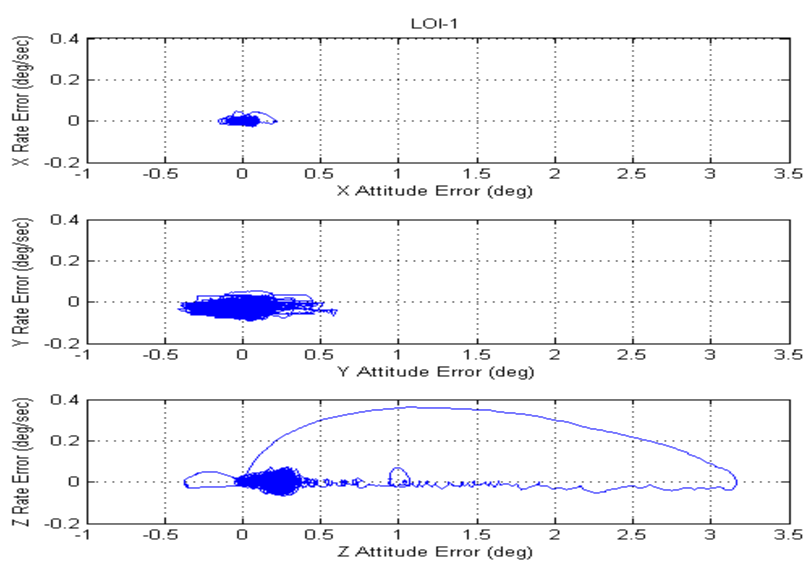


Figure 14. Lunar Orbit Insertion Maneuver Phase Plane

A time history of the attitude error during the LOI-1 maneuver is shown in Figure 15. As shown in the figure, an initial transient was seen in the controller and these errors were damped out within a couple of minutes in the x and y axes and within 10 minutes in the z axis.

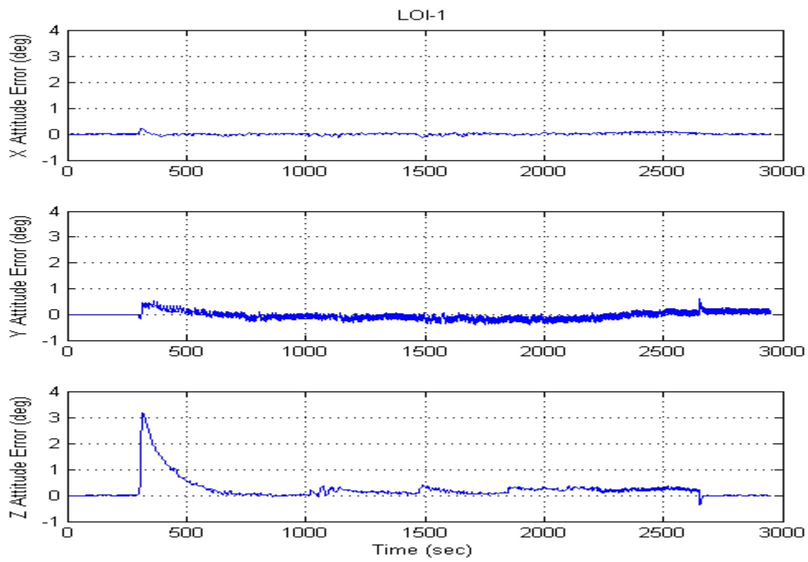


Figure 15. Lunar Orbit Insertion Maneuver Time-History Attitude Error

Second Anomalous Entry into SunSafe

Another unexpected entry into SunSafe mode occurred during the first lunar orbit after LOI-1. Both STs were occulted by the Moon for a duration of 1 hour. This was expected since the LRO had not yet started Nadir targeting operations and the onboard target was inertial. Both STs entered standby mode when the ST data was marked as invalid during the lunar occultation. The attitude source was set to ST1 at this time, so the onboard attitude was IRU propagated from the last valid ST1 quaternion (just prior to Moon occultation). The FDC system includes monitors for each ST operational mode. If either ST's operational mode is "standby", then it is commanded back into the autonomous attitude determination (AAD) mode at 1 minute intervals, causing the ST to attempt to re-acquire the attitude from a "lost in space" condition. During one of these attempts to promote ST1 from standby mode to AAD mode during the occultation, ST1 computed a "valid" quaternion, with also a "valid" rate, for a single ACS cycle. This "valid" quaternion had a very large error in excess of 100 deg. But, since it was marked as valid, it was processed by the onboard system for use. An additional delta quaternion check, which compares the ST quaternion to a valid one from the previous cycle, failed to catch the error since the previous cycle was marked as invalid. As a result, this single piece of inaccurate attitude data was used as the attitude source for one ACS cycle. After this cycle, the ST data immediately became invalid again, and the ST dropped back into standby mode. So, the onboard attitude was again the IRU propagated attitude, but unfortunately propagated from this erroneous ST quaternion (which had been marked as "valid"). Thus, the onboard attitude solution remained inaccurate, thereafter and the onboard FDC sent LRO into SunSafe due to the persistent large attitude error.

LOI 2 through LOI 5

The remaining LOI maneuvers were accomplished using the same two NTs and all eight of the ATs. With each successive maneuver, the GN&C Team estimated a new feed forward torque to apply to the controller in order to reduce the initial attitude transient. Table 2 shows the maximum attitude error and rate error the spacecraft saw during each maneuver. Notice that the maximum attitude error decreased with each successive maneuver.

Table 2. LOI-2 to LOI-5 Performance

	Max Abs Attitude Error (deg)			Max Abs Rate Error (deg/sec)		
Maneuver	X	Y	Z	X	Y	Z
LOI-2	0.65	1.46	1.16	0.11	0.12	0.28
LOI-3	0.06	0.92	0.27	0.02	0.20	0.07
LOI-4	0.07	0.83	0.20	0.02	0.21	0.07
LOI-5	0.06	0.78	0.21	0.04	0.20	0.08

COMMISSIONING

During the Commissioning phase of the mission, LRO engineers and scientists checked-out and calibrated their respective systems. The operations team planned and executed a number of off-point maneuvers to accommodate the science and engineering teams. During this time, the SA unexpectedly hit a gimbal lock situation. Also, the Observing mode control mode was tested in great detail and disturbances causing attitude transients were studied and understood.

Solar Array Gimbal Lock Event

The SA encountered a gimbal lock situation on July 1, 2009 when the sun target vector approached [0,-1,0], i.e., when the sun is directly on the –Y-axis of the spacecraft based on uploaded ephemeris. This happens because there are an infinite number of possible solutions for that particular array orientation. Observed performance showed the commanded SA Gimbal 1 angle swerve, stop, turn around, and continued to swerve for an orbit; refer to Figure 16. Since the spacecraft was flying with a 30 degree offset (for thermal reasons), the SA really was not at the gimbal lock orientation. But, the SA targeting algorithm uses a target sun vector (computed from on-board ephemeris) instead of the measured sun vector. Therefore, when the target sun vector became [0,-1,0], the algorithm detected gimbal lock. However, the observed flight behavior was not seen during testing, which led to a deeper study of the event.

The SA targeting algorithm determines gimbal lock by comparing the target vector to a tolerance value. When that tolerance value is set properly, gimbal lock is identified early and the SA is sent to a pre-defined angle of + 90 deg for gimbal 1. Flight results show that gimbal lock was not detected early. Gimbal 1 was commanded to +90 deg, but then exited gimbal lock and proceeded to follow a swerving command. This means that the tolerance value was not set properly (it's a table value – current value is $1x10^{\wedge -16}$ rad and it should be $1x10^{\wedge -10}$ rad). Figure 16 shows the expected behavior with the correct tolerance setting, "HiFi Predicted Ang".

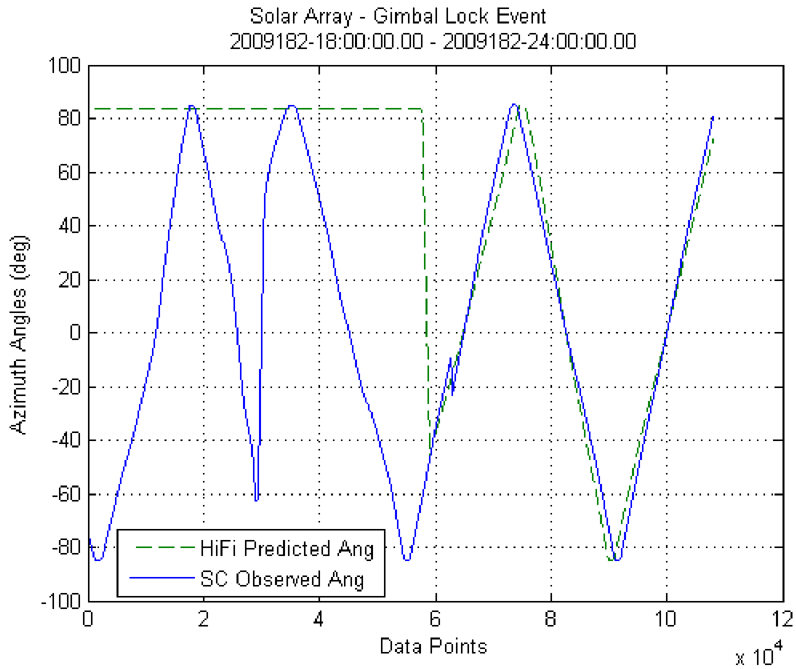


Figure 16. Array Gimbal Lock Event plotted against High Fidelity Simulation Results

The easiest solution to prevent the observed performance when gimbal lock is encountered again is to change the tolerance value to $1x10^{\wedge-10}$ rad. But this requires the creation, testing and uploading of a new table. Gimbal lock only occurs when the spacecraft is at a high solar beta angle, and since the nominal operational plan is to park the SA at high solar beta angles, this situation should never occur again. The reason it was encountered is because the nominal SA operational plan was not being followed.

Observing Mode Performance

LRO demonstrated excellent pointing performance during Observing mode nadir and inertial attitude target operations during the entire Commissioning phase. LRO pointing requirements are given as pointing accuracy and knowledge relative to the prime ST reference frame. Since there is no direct measure of pointing accuracy relative to this frame, the derived attitude control error requirement of 15 arc-sec (3 sigma), consistent with the LRO knowledge and attitude accuracy requirements, was used to access pointing performance.

Transient LRO attitude errors observed during commissioning, shown in Figure 17 and Figure 18, resulted primarily from three sources: Diviner instrument calibrations, RW zero crossings, and SA articulation. A description of each of these disturbances is documented in the subsections below.

An additional error source seen in Figure 17, which will not be described in detail, was due to a non-constant nadir target rate resulting from the low-lunar-elliptic commissioning orbit. A small amplitude pitch error variation at the orbit period of 2 hours exists. This sinusoidal variation of attitude error of about +/- 2 arc-sec, was eliminated after LRO was maneuvered into the nearly circular mission orbit.

Even during times of considerable disturbance, primarily due to RW zero crossings and SA articulation as shown in Figure 18, the attitude errors were maintained below the statistical attitude error requirement level of 15 arc-sec (3 sigma) as documented in Table 3.

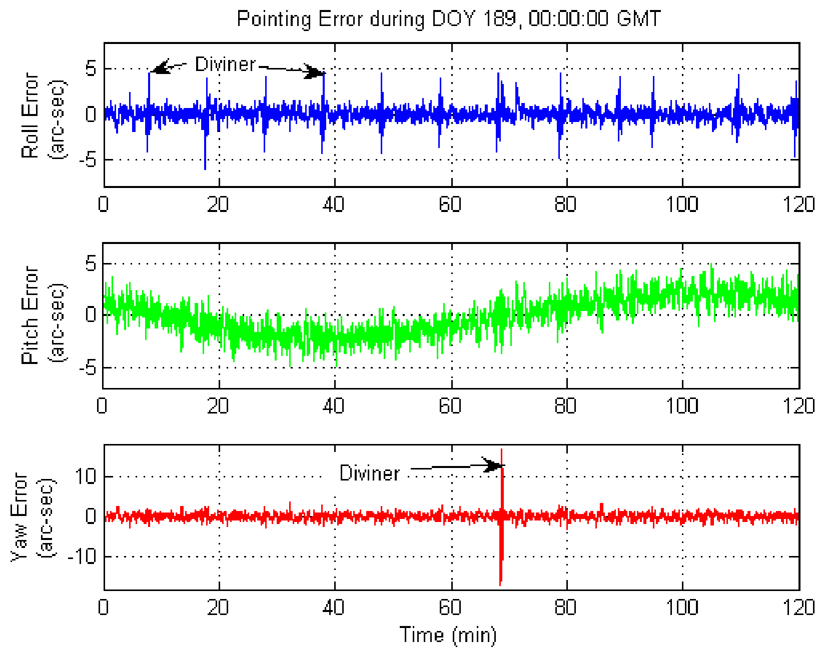


Figure 17. Attitude Error during DOY 189, 00:00 GMT

Diviner Instrument Calibration Disturbance. Figure 17 shows attitude error for a typical orbit during nadir target operations on DOY 189 during times when the SA is not moving and RWs do not cross zero speed. Attitude errors remain below 2 arc-sec, with the exception of transients caused by Diviner instrument motion. There are two types of Diviner motion that caused spacecraft pointing error. The smaller disturbance was caused by a "barrel roll" or elevation-only gimbal motion for pointing the instrument to deep space for calibration. These occurred at 10 minute intervals and resulted in less than 5 arc-sec pointing error, primarily about the spacecraft roll axis. The other Diviner disturbance is a result of two axis motion of the Diviner azimuth and elevation gimbals occurring once per orbit. The effective inertia load of the azimuth gimbal is much larger than that of the elevation gimbal. Thus, the corresponding spacecraft attitude disturbance, primarily about the spacecraft yaw axis, is larger respectively, as shown in Figure 17 at approximately 67 minutes. The Diviner azimuth disturbance is clearly present at time = 470 minutes and has an amplitude of approximately 23 arc-sec, refer to Figure 18.

Reaction Wheel Zero Crossing Disturbance. Data taken during DOY 220 (Figure 18) shows somewhat increased pointing errors during the times when the SA is tracking the Sun and the RWs are crossing zero speeds. The disturbances at time = 430, 442, 464, 470, 493, 497 minutes, and during period from time = 520 to 530 minutes are all caused by reaction wheel zero crossings. Worst case transients from wheel #3 cause pointing error transients of ~50 arc-sec as shown in Figure 18 at time=465 min. Wheel speed zero crossings continued to occur each orbit as X-Z momentum increased at higher rates than pitch axis momentum. During subsequent Delta-H wheel momentum unloading, pitch momentum biasing was used to minimize low speed wheel operation while maintaining system momentum within 80 Nms over a required two week period. Although using this strategy, it was not always possible to entirely avoid zero crossings within the system momentum constraints.

Solar Array Elevation Gimbal Step Disturbance. On DOY 220 attitude transients apparent at time = 427, 437, 480, and 500 min were due to SA elevation angle command step transients, Figure 18. Generally, the SA elevation gimbal is stopped or moving somewhat slowly during nadir target operations. Integrator wind-up in the SA gimbal controller resulted in instantaneous SA motion of about two cardinal steps (0.015 deg). These disturbances resulted in rate transients of about 30-40 arc-sec/sec about roll axis and 20-30 arc-sec/sec about yaw axis, as shown in Figure 19. These rates were somewhat large, but still within the LRO stability requirements. Options for elimination of this disturbance are to zero out the integrator gain or to reduce the integral limits in the SA elevation controller, neither of which was implemented during the commissioning phase.

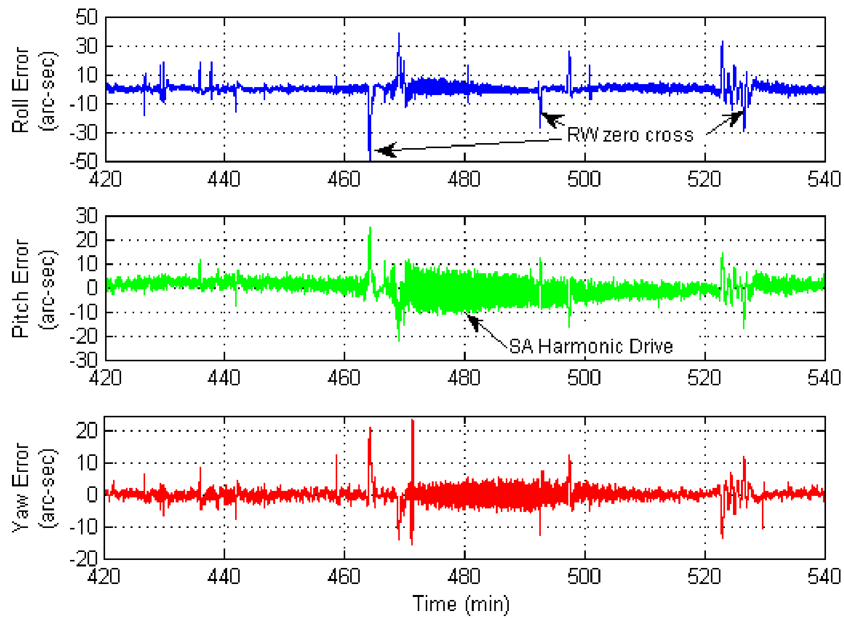


Figure 18. Attitude Error during DOY 220, GMT 07:00

Solar Array Harmonic Drive Gear Transmission Error Disturbance. Figure 18 also shows typical attitude error response due to SA harmonic drive gear transmission error, primarily about the pitch axis. This error is caused by a sustained SA oscillation driven at harmonics of the SA gimbal drive speed. The dominant disturbance occurred at a 2x harmonic of the SA rate, with a period of approximately 15 to 20 sec due to the rather high 200:1 gear ratio of the drive system. This effect resulted in a maximum attitude error of ~11 arc-sec over a typical orbit, shown in Figure 18 on DOY 220. This maximum error coincides with the SA rewind when the SA azimuth gimbal reverses direction at time = 470 min. While this error was predicted prior to flight, it was somewhat surprising that the magnitude of the error seemed to be dependent on the direction of gimbal rotation and perhaps the loading of the gears during the reversal. Note that the error decreases as the SA continues to rewind after time = 480 min. This gear transmission error has been trending somewhat lower as the solar beta angle decreases, as a result of the decreased inertia loading about the +Y gimbal axis.

Solar Array Rewind Disturbance. Since LRO is Nadir pointing in a polar orbit, the SA +Y gimbal reverses direction twice an orbit over the lunar poles. The attitude disturbance resulting from this SA "rewind" is minimized by using acceleration profiling. As the SA approaches the rewind condition, the gimbal rate is slowly reduced to zero over a period of about two minutes. After a few minutes in a stopped condition the SA rotation is reversed and the rate is slowly increased, again over a two minute period. The effect of this rewind disturbance can be seen in the body rate plot for DOY 220, Figure 19, at time = 465 and 525 min. As shown in Figure 19, maximum rates from this disturbance were about 20 arc-sec/sec. These rate spikes occur at the end of deceleration and beginning of acceleration from a stopped condition due to the drive having a non-zero minimum rate. Figure 18 shows that the attitude transients due to SA rewind was typically less than 10 arc-sec.

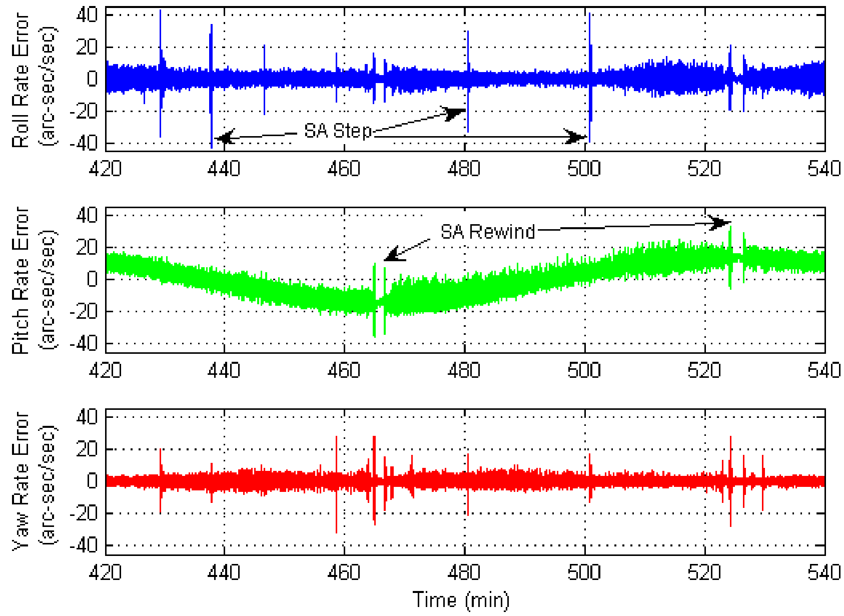


Figure 19. Rate Error during DOY 220, GMT 07:00

Table 3. Observing Mode Controller Error Statistics

	Roll (arc-sec, 3 sigma)	Pitch (arc-sec, 3 sigma)	Yaw (arc-sec, 3 sigma)
Nadir with SA fixed and no RW zero crossings (DOY 189)	2.2	5.3	2.8
Nadir with SA tracking and RW zero crossings (DOY 220)	13.8	10.9	7.1
Requirement	15.0	15.0	15.0

CONCLUSION

This paper detailed experiences from the Guidance, Navigation and Control (GN&C) Team during launch through commissioning of the Lunar Reconnaissance Orbiter (LRO). An overview of the GN&C subsystem was followed with a brief discussion of the mission timeline. An early mission chronological narrative described the GN&C subsystem performance and interesting GN&C events. Spacecraft angular velocity data showed that the initial attitude acquisition, following launch vehicle separation, was successful. Flight telemetry received both in real time and stored for later downlink demonstrated excellent GN&C system performance. With the exception of a few anomalous entries into SunSafe mode, the system operation was flawless. These safe hold transitions, while unplanned, provided the team a unique opportunity to demonstrate the reliability of SunSafe mode and recovery operations.

Assisting in the early LRO mission and spacecraft commissioning provided the GN&C team invaluable experience operating a spacecraft being inserted into Lunar orbit. We are all very proud and grateful to have had the privilege to participate in this historic NASA return to the Moon. Currently, LRO is conducting its nominal science mission collecting much needed data for NASA's eventual return to the Lunar surface.

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