NAVIGATION OF THE TWIN GRAIL SPACECRAFT INTO SCIENCE FORMATION AT THE MOON

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Abstract: *On February 29, 2012 the twin NASA Gravity Recovery And Interior Laboratory (GRAIL) spacecraft, Ebb and Flow, achieved precise synchronized formation for collecting highly sensitive lunar gravity data. This was accomplished after performing a total of 27 propulsive maneuvers between the two spacecraft (13 on Ebb, 14 on Flow) over six months. Each 300 kg GRAIL spacecraft independently flew a 3.8-month, low-energy trajectory to reach the Moon after separation from the launch vehicle on September 10, 2011. The spacecraft were captured into 11.5 hr co-planar polar orbits after performing Lunar Orbit Insertion (LOI) maneuvers on New Year's Eve (Dec 31, 2011) and New Year's Day (Jan 1, 2012), respectively, for Ebb, and Flow. Once captured, each spacecraft performed clusters of period reduction maneuvers to bring their orbit periods down to just less than 2 hrs. Finally, the orbiters were placed into science formation by performing five strategic maneuvers (2 on Ebb, 3 on Flow). These maneuvers ensured 3 months of orbit lifetime with mean altitudes of 55 km and separations of 82–217 km by targeting the orbits' eccentricity vectors to specific locations. This paper will discuss the navigation strategy and performance of the twin GRAIL spacecraft from the September 10, 2011 launch through the end of the Prime Mission Science Phase in June 2012.*

Keywords: *GRAIL, Lunar Gravity, Formation Flying, Low Altitude, Navigation, Low Energy Trajectory*

1. Introduction

The NASA Gravity Recovery And Interior Laboratory (GRAIL) mission is a Discovery-class mission proposed to map the lunar gravity field to high accuracy and spatial resolution. GRAIL (GR) is the lunar analog of the Gravity Recovery And Climate Experiment (GRACE) mission, which is currently mapping the Earth's gravity field to unprecedented resolution. The science payload consists of the Lunar Gravity Ranging System (LGRS), which was derived from GRACE. The LGRS measures the gravity field via the Ka-band ranging signals telecommunicated between the two nearly identical spacecraft GR-A (Ebb) and GR-B (Flow) separated by distances of 82–217 km in the same low altitude, near-circular, near-polar orbits.[1] The primary activity of the nine-month prime mission was the 89-day Science Phase consisting of slightly more than 3 gravity mapping cycles (or 3 lunar sidereal periods of 27.3 days each). The orbit mean altitude during this phase was approximately 55 km. The highly sensitive LGRS data will enable scientists to characterize the internal structure and thermal evolution of the Moon from crust to core. Each spacecraft also carries an education and public outreach imaging system called MoonKam, which consists of five camera heads. This system allowed US middle school students and teachers to participate in the mission by scheduling camera sequences to image specific areas on the lunar surface.

The GRAIL Project is managed by the Jet Propulsion Laboratory in Pasadena, CA, while the spacecraft were built and operated by the Lockheed Martin Space Systems Company (LM) in Denver, CO. Dr. Maria Zuber of the Massachusetts Institute of Technology is the Principal Investigator for this mission. This paper will discuss the navigation strategy and performance of the twin GRAIL spacecraft from the September 10, 2011 launch through the end of the Science Phase on May 29, 2012.

2. Mission Overview

2.1. Science Objectives

The two primary science objectives of the GRAIL mission are:

1. To determine the structure of the lunar interior, from crust to core.
2. To advance the understanding of the thermal evolution of the Moon.

[1]The twin orbiters were later named "Ebb" and "Flow" by students from Emily Dickinson Elementary School in Bozeman, Montana, following a NASA sponsored middle school contest.

A secondary objective is to extend knowledge gained from the Moon to other terrestrial planets.

These science objectives led to the following six different science investigations:

1. Map the structure of the crust and lithosphere
2. Understand the Moon's asymmetric thermal evolution
3. Determine the subsurface structure of impact basins and the origin of mascons
4. Ascertain the temporal evolution of the crustal brecciation and magmatism
5. Constrain deep interior structure from tides
6. Place limits on the size of a possible solid inner core

2.2. Navigation Requirements

These science objectives place requirements on the mission design and navigation of the twin orbiters to the Moon and into their synchronized low-altitude formation. One of the key and driving mission design/navigation requirements was to ensure the orbits of the two satellites during the Science Phase were co-planar, such that they have an inclination difference of less than 0.022° and a longitude of the ascending node difference of less than 0.02°. Other key and driving requirements for the Science Phase included: the maximum orbital inclination equaled 89.72° $\pm$0.152°; the periapsis altitude of the orbits was greater than 10 km, but did not exceed 55 km; the mean spacecraft separation distance during an orbit exceeded 50 km, but did not exceed 250 km and avoided separations where conditions give rise to orbiter-to-orbiter multipath off the lunar surface; the twin orbiters shall fly with GR-B in the lead position; the onboard ephemerides were updated, in order to maintain GR-A and GR-B attitudes and relative positions, with ephemeris pointing errors smaller than 0.073° (1σ). To ensure adequate power from the solar arrays which were aligned in the orbit plane, the solar beta angle (angle between orbit plane and the Sun direction) during the Science Phase was always greater than 49°. This last requirement was later ignored for the start of the Science Phase during operations when it was found that the solar panels were able to have adequate power starting with beta angle of 43°. Additional requirements ensured that the Science Phase would consist of at least 3 mapping cycles and it would be completed before the partial lunar eclipse on June 4th, 2012. Other requirements such as the orbiter-to-orbiter 10-km collision avoidance (COLA) requirement in the first six weeks of the lunar orbit phases were self-imposed by the Mission Design and Navigation Teams.

2.3. Mission Description & Operations Activity Timeline

Figure 1. GRAIL launch on September 10, 2011

The twin spacecraft were launched together, side-by-side aboard a Delta II 7920H 10C launch vehicle on Sep 10, 2011 (Figure 1). The launch period ranged from September 8, 2011 through October 3, 2011. Shortly after launch, the spacecraft separated from the second stage of the launch vehicle and independently flew 3.8-month low-energy trajectories to the Moon. Each spacecraft executed three propulsive Trajectory Correction Maneuvers (TCMs) during the Trans-Lunar Cruise (TLC) Phase to achieve the correct orbit conditions for orbit insertion at the Moon. These trajectories took the twin spacecraft toward the Sun and past the Earth-Sun Lagrange point 1 (L1) before heading back to the Moon. The spacecraft were captured into 11.5 hr polar orbits after performing Lunar Orbit Insertion (LOI) maneuvers on New Year's Eve (Dec 31, 2011) and New Year's Day (Jan 1, 2012), respectively, for GR-A, and GR-B. The dates of the LOI maneuvers were fixed regardless of when launch occurred within the launch period. A timeline of the GRAIL Prime Mission is illustrated in Figures 2 and 3. Figure 2 displays the operation activities during the Launch, TLC and LOI Phases while Figure 3 shows the operation activities in the lunar orbit phases from LOI, through the end of the Science Phases. These activities include maneuver development schedules, changes in attitude and spacecraft activities. The mission was designed to avoid a total lunar eclipse on December 10, 2011 and a partial lunar eclipse on June 4, 2012 in order to ensure adequate power for the orbiters during the mission. Once captured, the mission entered the Orbit Period Reduction (OPR) Phase where each spacecraft performed two clusters of Period Reduction Maneuvers (PRMs) to bring their orbit periods down to just less than 2 hrs within a relatively short amount of time. Finally, the orbiters were placed into formation in the Transition to Science Formation (TSF) Phase by performing five strategic Transfer to Science Formation Maneuvers (TSMs) (2 on GR-A, 3 on GR-B). These maneuvers ensured 3 months of orbit lifetime with mean altitudes of 55

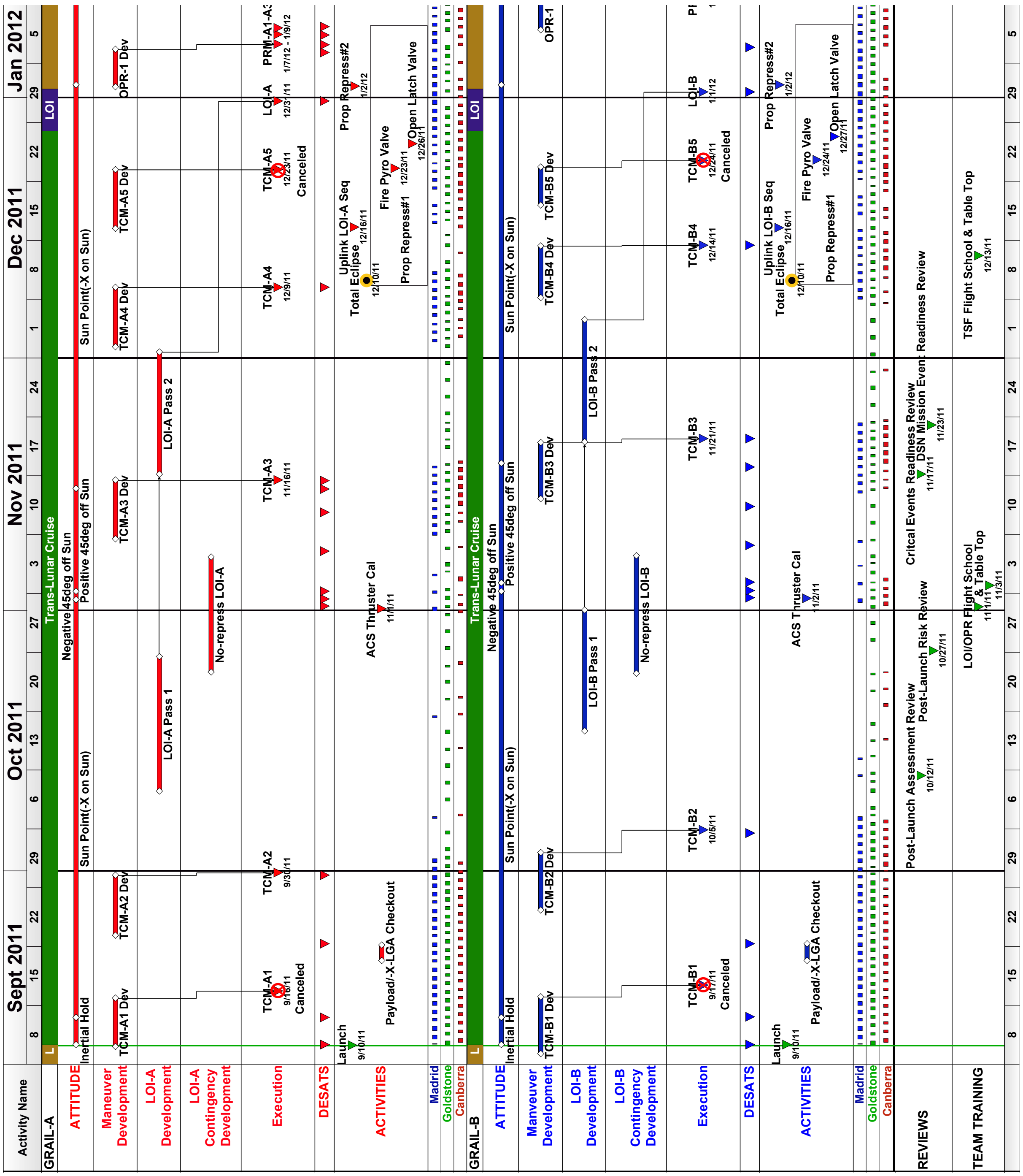


Figure 2. GRAIL TLC operations activity schedule

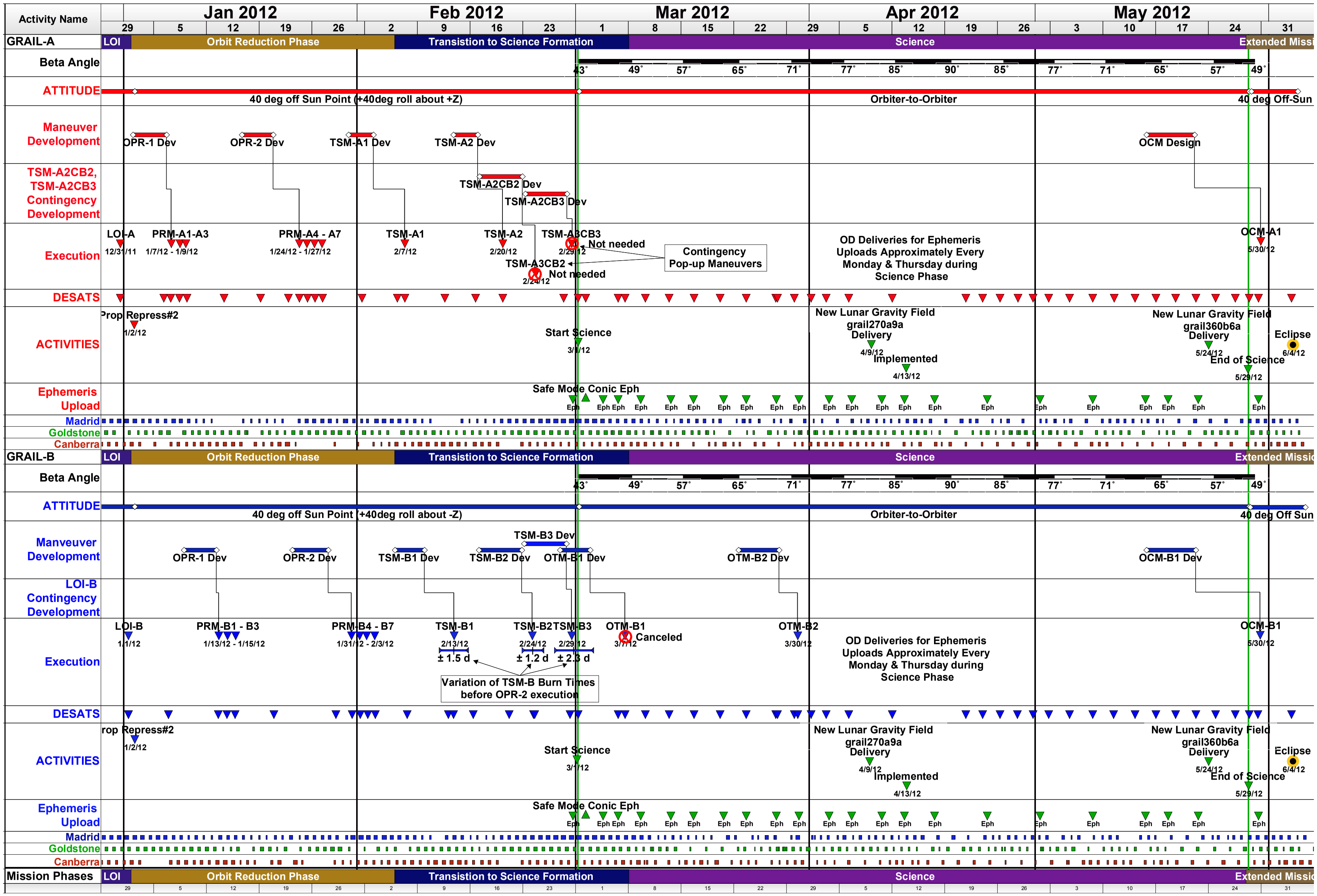


Figure 3. GRAIL OPR, TSF and Science Phase operations activity schedule

km during the Science Phase by targeting the orbits' eccentricity vectors to specific locations. The GR-A and GR-B orbiters achieved precise synchronized formation on February 29, 2012 after the last TSF maneuver, TSM-B3. This was accomplished after performing a total of 27 propulsive maneuvers between the two spacecraft (13 on GR-A, 14 on GR-B) over six months. This last TSM maneuver placed the separation of the two orbiters at the start of Science to 82 km with a slow drift rate to increase the separation at the end of the first mapping cycle. When the separation achieved 217 km on March 30th, 2012, the drift rate was reversed using an Orbit Trim Maneuver (OTM) on GR-B's Attitude Control System warm gas thrusters to bring the orbiters' separation to approximately 82 km at the end of the Science Phase on May 29, 2012.

The timeline of the nine-month GRAIL Prime Mission (PM) is divided into the following unique mission phases:

1.	Launch Phase (1 day), Launch Period:	Sep. 8, 2011	–	Oct. 3, 2011
2.	Trans-Lunar Cruise (TLC) Phase:	Sep. 11, 2011	–	Dec. 28, 2011
3.	Lunar Orbit Insertion (LOI) Phase:	Dec. 28, 2011	–	Jan. 2, 2012
4.	Orbit Period Reduction (OPR) Phase:	Jan. 2, 2012	–	Feb. 6, 2012
5.	Transition to Science Formation (TSF) Phase:	Feb. 6, 2012	–	Mar. 1, 2012
6.	Science Phase:	Mar. 1, 2012	–	May 29, 2012
7.	Lunar Eclipse (LEC) Phase:	May 29, 2012	–	Jun. 5, 2012

2.4. Extended Mission

After performance of the flight systems were evaluated from 7 months of flight experience, the Spacecraft Team (SCT) at LM determined that the vehicles could survive the partial eclipse on June 4th, so a proposal to extend the mission until December 2012 was made. The GRAIL extended mission (XM) was chosen to map the gravity at the even closer mean altitude of 25 km for three months beginning in September 2012[1]. In general, three maneuvers were required each week from mid-August 2012 to December 2012 to keep the orbiters from impacting the Moon at this close range and maintain orbiter-to-orbiter separation close to 60 km. The last phase in the PM summary above, the Lunar Eclipse (LEC) Phase, replaced a Decommissioning Phase where the PM mission was set to end when the orbiters would eventually impact the Moon on June 4th, 2012. In the LEC Phase, preparations to the orbiters' thermal and power management systems were made to give the orbiters the best conditions to survive the partial lunar eclipse. Essentially, the LEC Phase established the start of the XM when each orbiter simultaneously performed Orbit Correction Maneuvers (OCMs) on May 30, 2012 to raise the orbit periapsis and mean altitudes in order to avoid impact on June 4th. The XM extended from this date to the middle of December 2012. The navigation of the XM mission will be presented in a future paper.

An overview of the GRAIL mission design is given in Reference [2]. Reference [3] describes the design of the low-energy TLC trajectories. The trajectory design work to bring the orbiters into the Science formation from LOI through OPR and TSF is discussed in Reference [4]. The complex GRAIL mission owes its success to the systems engineering of the Mission Operations System explained in Reference [5]. Reference [6] describes the mission operations. Reference [7] describes the Orbit Determination (OD) results. This reference gives more details of the navigation in the TLC, OPR and TSF Phases, such as the formulation of the pseudo B-plane used to track the consistency of OD solutions during lunar approach. This reference also explains the OD filter strategy used during each phase and methods for mitigating the effects of the gravity errors in the solutions.

3. Navigation Systems

3.1. Mission Design & Navigation Challenges

The uniqueness of the GRAIL mission and the Science formation requirements posed several challenges for the Mission Design and Navigation Teams. Some of these challenges include:

1. Determining the acceptable range of the incoming LOI targeting dispersions in order to determine the need of the TCM-5 statistical maneuvers. The Navigation (Nav) Team needed to understand how the achieved target errors on both orbiters affected the ability of the PRM and TSM maneuvers to achieve the Science formation. Non-linear navigation error analyses were needed to evaluate the complex error space of the independent navigation of the twin spacecraft.

The outcome of these analyses formulated the TCM-5 Go/No-Go criteria, which will be discussed later.

2. Like the first challenge above, the task of determining the range of acceptable PRM and TSM dispersions was also challenging.

3. The TLC orbits to the Moon after TCM-3 were not hyperbolic; the orbiters were actually captured within the Moon's influence by very long period elliptical orbits. This imposed challenges for the Nav Team to evaluate the consistency of the OD solutions and the progress of the orbiters at meeting the LOI target conditions. Typical hyperbolic flybys of celestial bodies or orbit insertions about them use the impact B-plane targeting plane (described in interplanetary navigation literature) to track navigation performance. The elliptical nature of the orbit at the LOI target interface required a special pseudo-B-plane coordinate system definition, which is described in Ref [7].

4. The success of the lunar orbit phase mission design was dependent on keeping on the timeline without missing maneuvers due to spacecraft safing events or command errors. The Nav Team had to be prepared for contingencies in the event of a missed maneuver. To be prepared, the Mission Design Team developed a "Contingency Playbook" prior to launch that evaluated the conditions in the event of a missed PRM or TSM maneuver on either GR-A or B and determined a contingency operations scenario involving one or more maneuvers to bring the orbiters into the correct Science formation. In all, over 330 contingency scenarios were investigated and made available in the online Playbook to the Nav Team as guidelines to achieve the Science formation. These scenarios had to be updated periodically with the latest orbit conditions as the mission progressed.

5. The gravity on the lunar far side was largely uncertain and posed challenges to the OD Team[7]. The best lunar gravity available at the start of the GRAIL mission was the spherical harmonic representation to degree and order 150, LP150Q field, developed using primarily 2-way S-band Doppler radio-metric tracking data from the Lunar Prospector (LP) mission[8]. Note, the Japan Aerospace Exploration Agency's (JAXA) Selene mission obtained backside lunar gravity information through "4-way Doppler" data obtained via a relay satellite. This gravity field was not available to the Nav Team. Because previous lunar gravity field determination efforts before GRAIL (and Selene) were performed using the 2-way measurements of LP or earlier lunar missions with Earth-based ground stations, no direct measurements of the gravity on the lunar far side could be made, and therefore this region of the lunar gravity field could only be inferred from long-term effects on the spacecraft orbits. These gravity errors affected the OD Team's predicted orbit accuracies. These errors also impacted the ability of the OD Team to accurately reconstruct the maneuvers in lunar orbit. Furthermore, the tracking data had to be deweighted significantly to mitigate the influence of gravity errors on the downtrack and crosstrack predictions. Lastly, it was computationally expensive to map the large gravity field covariance into downstream position uncertainties and therefore could not be performed within the operational timelines.

6. The mission design of the LOI, OPR and TSF Phases required accurate modeling of the thrust and mass flow rates in blowdown propulsion system (described below). This main engine finite burn model also included the contributions to the maneuver ΔV magnitude from the thrust vector control during the burn by the Attitude and Control System (ACS) reaction control warm-gas thrusters. The performance of the main engine degraded as the main engine fuel tank pressure decreased from launch and re-pressurization events. The gravity losses during the longer burn durations as the tank pressures were reduced needed to be accounted for. This model helped the Mission Design Team to estimate the fuel usage.

7. The burn-start epochs of the TSM-B maneuvers were subject to shift as much as ± 2.3 days from the nominal reference due to LOI, OPR and TSM errors (the timing uncertainties of TSM-B1, TSM-B2 and B3, were, respectively, ± 1.5 days, ± 1.2 days, and ± 2.3 days). These ranges in time were difficult to plan for in operations, especially in the earlier reference designs; the dispersion analyses showed that TSM-B2 and B3 could be required at anytime of the day. The Nav and SCT Teams would have to be ready to staff the third shift for the maneuver development with little advance warning. Later, prior to launch, a 4-day interval between maneuvers was imposed by the Mission Operations Team. With this rule, the TSF Phase could be redesigned and it was found that all maneuver development could be performed within the first and second work shifts.

8. The processing time to propagate the orbits using the full lunar gravity field posed issues to the timeliness of OD solution deliveries and maneuver design processes. The original plan was to truncate the 150x150 field to degree and

order 70, but pre-launch orbit simulation tests showed the full field was needed in the OD processing during the lunar orbit phases. The maneuver development timelines were adjusted to accommodate the longer processing times. Midway through the Science Phase, the Science Team delivered an improved gravity field (grail270a9a) of degree and order 270 on April 9th, 2012 using approximately 30 days of LGRS data. The Nav Team determined that truncation of this field to 200x200 produced significant improvements over the LP150Q field while keeping the processing time within reasonable limits. As more LGRS data were acquired, the Science Team found it necessary to produce larger gravity fields. Towards the end of the Science Phase (May 24, 2012), another lunar gravity field was delivered by the Science Team to Nav. This field now contained terms to degree and order 360 (grail360b6a) and was produced using LGRS data up to May 11th, 2012. Likewise, this field showed improvements over the previous field, but had to be truncated to 200x200 in order to produce reasonable OD delivery times.

3.2. Tracking

Navigation of the GRAIL spacecraft to the Moon was accomplished using the traditional navigation data types of 2-way S-band Doppler and range radio-metric data transmitted through the Deep Space Network (DSN) in Goldstone, CA, Canberra, Australia, and Madrid, Spain and transponded by 1 of 2 hemispherical Low Gain Antennas (LGAs) (mounted on the $\pm$ X-axis panels). Lunar orbit navigation was performed primarily using the 2-way Doppler data augmented occasionally with the 1-way X-band Doppler data which was transmitted by 1 of 2 Radio Science Beacon (RSB) antennas also mounted on the $\pm$ X panels. The S-band frequency spectrum of the tracking signal restricted the available tracking resources to the following S-band capable 34m tracking stations: Deep Space Station (DSS) 24, 27, 34, 45, 54, 65. DSS-24 (Goldstone), 34 (Canberra), and 54 (Madrid) are Beam Wave Guide (BWG) antennas. DSS-27 (Goldstone) is a High Speed Beam Wave Guide and DSS-45 (Canberra) and 65 (Madrid) are High Efficiency antennas. During the Science Phase, imbalances of the desaturation maneuvers were measured using the spacecraft-to-spacecraft Ka-band range data collected by the science LGRS payload. The TLC tracking schedule included continuous DSN coverage for both spacecraft for several days from launch, LOI and around maneuvers. Otherwise the data schedule was alternating 1 track per spacecraft every other day. From the OPR through the late TSF Phases, the DSN coverage consisted of two tracks per day per orbiter. The schedule changed to near continuous coverage from the late TSF Phase through the beginning of the Science Phase. From the start to the end of the Science Phase, the coverage consisted of at least one tracking pass per orbiter per day with no more than a 16 hour gap.

3.3. Navigation Team

The Nav Team was responsible for planning and executing the mission plan, to navigate the twin spacecraft from launch to the Moon, through LOI and into formation through the end of the Science Phase. The mission plan was formulated by the Mission Design Team. While in lunar orbit, the Nav Team was responsible for updating the predicted reference trajectories of both orbiters frequently for the background (BG) sequence development timelines, which usually required the team to update the reference trajectories which included downstream maneuvers 2-3 weeks ahead. It was important for the BG sequence to accurately account for the timing of the lunar occultations during the Science Phase to maximize the data return. To account for the occultation timing errors, the occultation entry and exit times were padded by up to 8 minutes to account for the errors in the 2-3 week old trajectory prediction. The Nav Team, which is led by the Team Chief and Deputy Team Chief, consisted of the following functional sub-teams: the Trajectory Design (Traj), the Orbit Determination (OD) and the Maneuver Design (MD) Teams. The OD and MD Teams were subdivided into separate GR-A and GR-B teams. Each GR-A and GR-B Team included 3 OD analysts and 2 Maneuver designers. The Traj Team consisted of 2 mission designers who were originally members of the Mission Design Team, one who doubled as a GR-A Maneuver designer. The Traj team was responsible for updating the reference trajectories of both spacecraft during all phases of the mission up to the start of Science to ensure the science requirements were met. The Traj Team also prepared the Project for contingencies in the case of missed maneuvers or the under-performance of LOI. The OD Team was responsible for predicting the trajectories of both orbiters based on the latest tracking data on a near-daily basis. They were also responsible for reconstructing all maneuver executions and delivering the latest predicted trajectories to the DSN in order to generate antenna pointing and frequency predicts, or to the SCT Team to upload to the orbiters for maintaining accurate pointing during the Science Phase. The MD Teams were responsible for designing all maneuvers. They were also responsible for performing statistical maneuver analyses. OD solution deliveries and final Maneuver designs were always verified by backup Nav Team analysts and peer reviewed by the 6–9 member Navigation Advisory Group (NAG) at JPL.

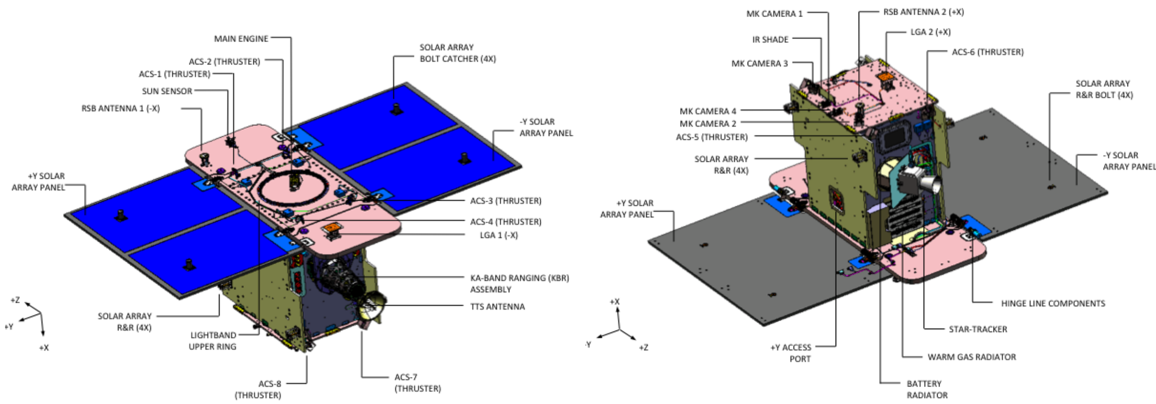


Figure 4. GRAIL Orbiter Configuration (GRAIL-B)

4. Flight System

The main science instrument carried onboard the twin GRAIL orbiters was the Ka-band Lunar Gravity Ranging System (LGRS), which measures inter-satellite range rate. The raw measurement of this system consisted of dual 1-way Ka-band range (KBR) or phase counts, which are post-processed on the ground to produce the range-rate observables with respect to the spacecraft center of mass with an accuracy of 4.5 μm/s averaged over 5 second sample times. Timing of the measurements between the orbiters was maintained through the use of the S-band Time Transfer System (TTS), which was based on the Ultra Stable Oscillators (USO) onboard both orbiters. These USOs were calibrated to the DSN clocks via the 1-way X-band Doppler signal transmitted through the RSB antennas.

The orbiters are powered by solar arrays (total cell area of 2.8 m^2), which were deployed shortly after separation from the launch vehicle. Rechargeable batteries maintained power in lunar orbit during the periods when the orbiters were in solar eclipse. The twin GRAIL orbiters are nearly identical. Figure 4 shows the configuration for the GR-B orbiter. The major differences between the orbiters include the canting of the star tracker and LGRS Ka-band antenna horn relative to the spacecraft Z-axis within the Y-Z-plane. The star tracker boresight is mainly pointed in the $+$Z-axis body direction for both orbiters, however to keep the lunar surface out of view, the boresight for GR-A was mounted with a negative 30° rotation about the $+$X-axis to point slightly in the $+$Y-axis direction and for GR-B it is rotated with a positive 30° angle to point in the $-$Y-axis direction. Similarly, the Ka-band antenna boresights for GR-A and GR-B are canted 2.1°, respectively, in the $-$Y and $+$Y-axis directions so that they can remain pointed toward each other for separation distances between approximately 65–225 km while the Y-body axes were aligned with the nadir direction in lunar orbit (see Figure 4).

Each orbiter's attitude was maintained by a three-axis stabilized Attitude Control System (ACS), which consisted of an Inertial Measurement Unit (IMU), Reaction Wheel Assembly (RWA), star tracker and Sun sensor. The IMU measures the spacecraft's linear accelerations and angular rates. The accelerometer in the IMU measures the Main Engine burn performance to cutoff the burn when the desired ΔV is reached. The RWA, which consists of a four-reaction-wheel pyramid design, has a 1.55 Nms momentum capacity and is capable of producing maximum torques of 0.006 Nm. The ACS maintains the attitude knowledge accuracy within 2.0 mrad (3σ) per axis over the mission using the IMU and star tracker. During the Science Phase, the accuracy to point the orbiter to the desired attitude needed to be controlled using the RWA with errors less than 1.0 mrad per axis (3σ) except during angular momentum desaturation (AMD) events. During the TLC Phase, the ACS Team discovered a star tracker misalignment problem. The GR-B star tracker alignment data received from the vendor was found to be wrong. This would have impacted the attitude determination during the LOI-B maneuver. In addition, the discrepancies between ACS & Navigation reconstruction of the TCM-B2 and B3 maneuvers led to the ACS Team finding the misalignment of the IMU accelerometer. The GR-B accelerometer misalignment was estimated by the ACS Team to be a 4.4 mrad rotation about the mechanical Z-axis. The root cause was found to be an error in the mounting of the IMU to the spacecraft mechanical frame; the wrong reference mirror was used for alignment to the mechanical frame. The same procedure error was found with the mounting of the IMU on the GR-A spacecraft, but was estimated to be smaller, 0.8 mrad. The Project later approved corrections to the IMU alignment matrices to be uploaded to both spacecraft. Because it was more critical to the LOI performance, the alignment matrix of the IMU to the mechanical frame was updated before TCM-B4. TCM-B4 served as the verification of this correction. The GR-A IMU alignment correction was less critical and was thus uploaded after LOI-A.

The propulsion system for each orbiter consists of a hydrazine propellant tank, a Main Engine (ME), a warm gas ACS thruster system, and a high pressure helium recharge tank. The 22 N ME was used for all maneuvers from the TCMs in the TLC Phase until the last maneuver in the TSF Phase (TSM-B3). The OTM in the Science Phase was performed using the warm gas ACS system. The ME nozzle, shown in Figure 4, is aligned along the spacecraft $-$X-axis producing thrust (ΔV) in the $+$X-axis direction. The ACS thrusters were designed for smaller maneuvers with ΔVs in the tenths of meters per second range (up to 0.3 m/sec) using 4 of the 8 ACS warm gas 0.9 N thrusters (ACS-1-8 in Figure 4). These 8-coupled thrusters are canted 15° from the X-axis in X-Z plane, and 35° from the X-axis in X-Y plane. ACS thrusters were also used during ME burns for thrust vector control.

The GRAIL propulsion system was designed to operate during the entire mission in a blowdown mode. To protect against cat bed damage, improve efficiency with higher thrust and maximize Isp over the entire mission, a recharge system was added to re-pressurize the propellant tank (for both orbiters) after TCM-5 and again after LOI (before the first PRM maneuver). Based on pre-launch TLC statistical $\Delta V99\%$ cost, the tank pressure was expected to reach below 300 psia following the execution of TCM-4. Instead, because of the lower TLC ΔV costs discussed later, the tank pressures, shown in Figure 5, only decreased to 358 psia for GR-A and 325 psia for GR-B. Prior to LOI, the SCT Team commanded the first re-pressurization. This brought the pressure back close to 400 psia before LOI. The LOI burn duration had been limited below 40 min to ensure that the tank pressure remained above 155 psia. The ME blowdown thrust performance through the mission is shown in Figure 6 as a function of accumulative ME on time. Telemetry of the ME tank pressures for the entire PM is displayed in Figure 5. The mission design of the Science formation required a high fidelity model of the blowdown thrust, which was a function of the tank pressure. The outcome of the targeted post-maneuver orbits were dependent on this model due to the long burn durations. The propulsion tank pressures, fuel mass remaining and other propulsion system parameters in this high fidelity finite burn model were periodically updated following each burn from telemetry and ground-based tools used by the LM Propulsion (Prop) Team.

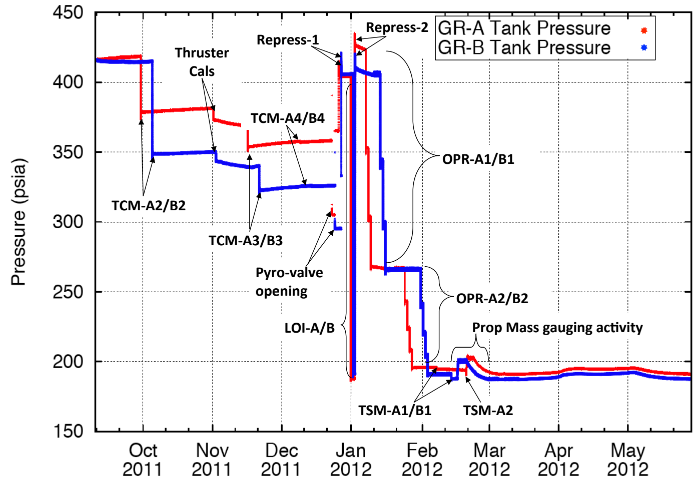


Figure 5. ME tank pressures of GR-A and GR-B

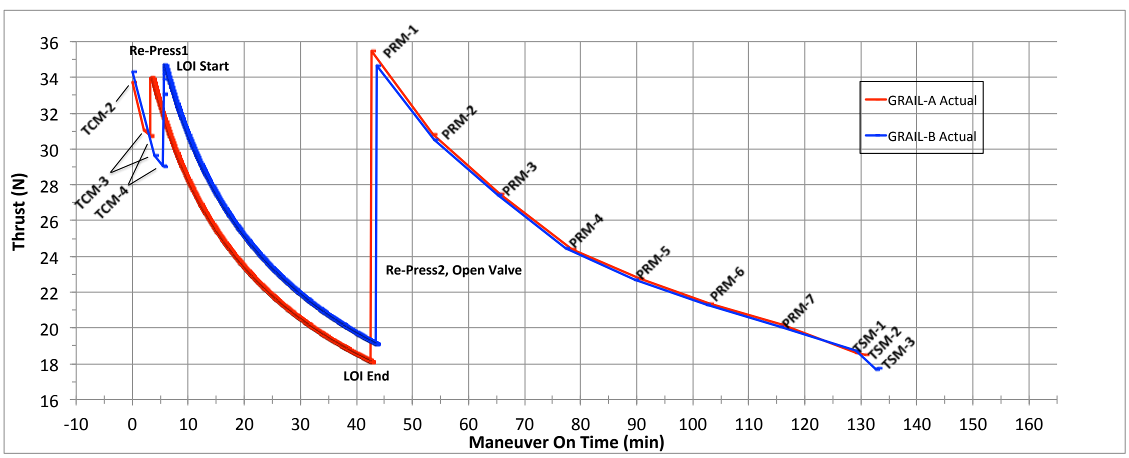


Figure 6. Blowdown main engine thrust performance of GR-A and GR-B

The RWA Angular Momentum Desaturations (AMDs or desats) manage the wheel speeds within acceptable levels by firing the coupled ACS thrusters. The ACS thruster couples are designed to impart zero ΔV. The RWA wheel speeds increase to maintain attitudes by compensating for solar torques and gravity gradient torques in lunar orbit. Only a few AMDs on both spacecraft occurred during TLC because of the mainly Sun-pointed attitude flown through this Phase. During the TLC, and OPR mission phases, the AMDs executed autonomously. There were three commanded AMDs between the PRMs and first maneuvers in the TSF Phase. In the Science Phase, the AMDs were commanded in the background sequence to occur over the lunar poles (above latitudes of 45° or below altitudes of -45°)when the RWA reached approximately 70% wheel capacity. This was to reduce science data outages due to the fact that redundant data are collected in the polar regions. The frequency of the AMDs in the Science Phase was one every 2 to 4 days. Generally, an AMD was commanded before every maneuver and attitude change. All AMD events are shown in Figures 2 and 3.

4.1. Attitude Profile

Shortly after launch, both spacecraft slewed to an inertial hold Sun-point attitude for approximately 80 hours, where the solar panels were faced directly at the Sun. Afterward both spacecraft attitudes were actively positioned to keep the solar panels directed towards the Sun through Nov 2nd, 2011 on GR-A and Nov 3rd on GR-B. The attitudes of both S/C were adjusted for calibrating the solar pressure model parameters on each vehicle by rotating the solar panels 45° off the Sun with a negative rotation about the S/C Y-axes for approximately one day then with a positive rotation about the S/C Y-axes from Nov 3rd to Nov 15th for GR-A and from Nov 4th to Nov 18 for GR-B. Each spacecraft autonomously slewed by 180° when they passed the Sun-Earth line twice during TLC to keep the star tracker boresight pointed away from Earth. The attitudes were nominally positioned back to the Sun-point attitude prior to the TCM-3 maneuvers through January 2nd, 2012. From this date through the end of the TSF Phase on March 1, 2012, the spacecraft were positioned with a 40° biased off Sun-point attitude as shown in Figure 8a for GR-A (40° roll about the +Z-axis for GR-A and 40° roll about the -Z-axis for GR-B). This attitude reduced the thermal irradiation from the lunar surface on the orbiter and positions the Y-panel radiators away from the Moon. During this time, both orbiters autonomously slew 180° about the X-axis when they cross the Earth-Sun line to keep the star tracker positioned away from the Earth; this occurs approximately every 13.7 days as the Moon orbits the Earth. With the exception of the post-launch inertial-hold attitude, all attitudes described above were implemented onboard using Sun-Earth, Sun-S/C and Sun-Moon ephemerides based on the September 9th launch TLC trajectories which were uploaded prior to launch. Differences in the attitude pointing profiles from the September 10th launch trajectories were negligible.

The attitude strategy during the Science Phase is shown in Figure 8b. The orbiter-to-orbiter vector aligned with the Ka-band antenna horn boresights was maintained by the latest predicted dual spacecraft ephemeris of GR-A and GR-B onboard both orbiters. The orbiter -X axis points in the orbit normal direction such that the solar arrays remained parallel to the orbit plane. The Nav Team provided the predicted spacecraft ephemerides of both orbiters to the ACS Team. The ACS Team then processed these predicted trajectories to form a dual spacecraft ephemeris for each spacecraft, which contained the Chebyshev polynomial representations of each orbiter ephemeris. Finally, these dual ephemerides were uploaded to each orbiter. The frequency of ephemeris uploads depended on how long the accuracy of the orbit prediction remains less than the navigation error allocation of the 0.073 (1σ) orbiter-to-orbiter pointing requirement. Nominally, during the Science Phase these deliveries were made every Monday and Thursday. This frequency was dependent on AMD or desat frequency and performance, and accuracy of lunar gravity models. It was anticipated

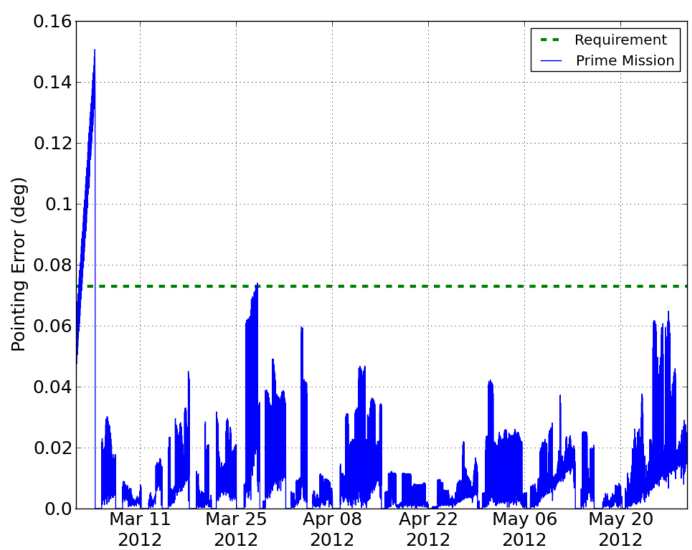
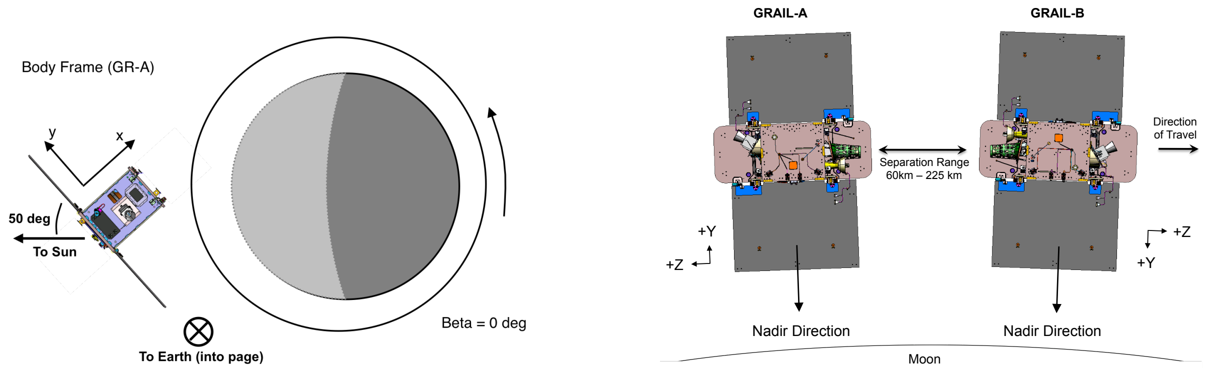


Figure 7. **Orbiter-to-orbiter pointing errors of onboard ephemeris during Science Phase**



(a) 40° off Sun-point OPR & TSF Phases

(b) Orbiter-to-orbiter point in Science Phase

Figure 8. GRAIL Attitude configuration through OPR and TSF in (a) and through Science Phase in (b)

that once the desat frequency was reduced during the higher solar beta angles, where the solar torque was reduced on the RWAs, the upload frequency could be reduced due to better downtrack predictions. However, the Nav Team determined using the LGRS range measurement that the ΔV from the desat thruster imbalance was sufficiently small (< 0.2 mm/s) to not induce significant downtrack errors. The ACS Team noted that the reduced desat frequency during much of this higher beta angle period was not realized due to higher than expected gravity torques in the low orbit. The pre-launch lunar gravity model was the primary source of both the downtrack and crosstrack errors in the predicted ephemerides. These errors necessitated the twice per week uploads. Once a better gravity model using the LGRS data was delivered by the Science Team in early April, the frequency of deliveries generally reduced to once per week. Figure 7 shows the orbiter-to-orbiter pointing error of the onboard ephemeris during the entire Science Phase. The first week's onboard ephemeris exceeded the Nav requirement. This was due to the fact that the onboard ephemeris had to be created ahead of the TSM-B3 execution, which saw a significant 2.6σ magnitude error. The thicker error bands or larger oscillations in Figure 7 are due to the larger crosstrack errors in either the onboard ephemeris or later OD solutions.

4.1.1. Safe Mode Conic Ephemeris

A circular-orbit-pointing strategy was developed to minimize disruptions in the Science Phase in case either orbiter entered safe mode. This would reduce the time and complexity of returning to orbiter-to-orbiter pointing for science gathering, which would result from the typical Sun or Earth-pointing safe-mode algorithms. Upon entering safe mode, this conic orbit pointing strategy consists of the orbiter aligning the Z-axis (-Z for GR-A) along the velocity direction of a mean conic orbit, which was fit over the entire Science Phase orbits. This mean conic orbit, referred to as the safe mode conic ephemeris, was updated with the latest predicted mapping orbit and uploaded to each orbiter just prior to the start of science. In order to keep thermally sensitive instruments from pointing at the lunar surface, the requirements of this safe mode ephemeris were to maintain attitudes within 7° (3σ) in-plane and 1.5° out-of-plane (3σ) of the predicted orbiter point attitude during the Science Phase. This safe mode conic ephemeris was routinely compared to the latest OD solutions to ensure it met the requirements. Figure 9 shows how the safe mode ephemeris maintained acceptable downtrack and crosstrack angular errors relative to the reconstructed orbits during the entire Science Phase.

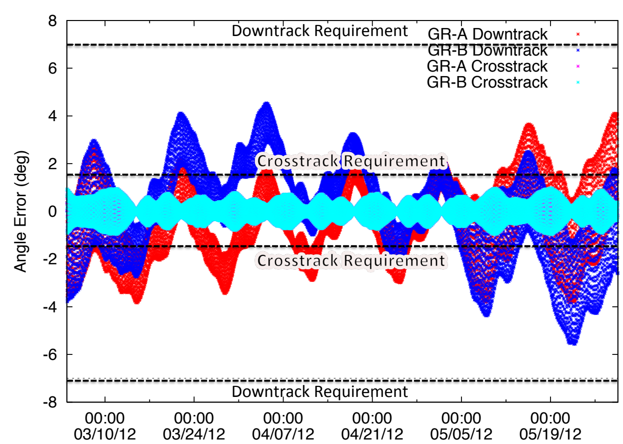


Figure 9. **Safe Mode Ephemeris of GR-A and GR-B compared to reconstructed OD solutions.**

4.2. Maneuver Development

All propulsive maneuvers were created using a five-shift maneuver development schedule. This schedule included the delivery of the orbit determination solution and the maneuver design by the Nav Team, maneuver implementation by the ACS and Prop Teams, verification of the implemented maneuver by the Nav Team, sequence generation and validation by the Mission Planning & Sequencing and Spacecraft Teams (MPST and SCT), testing on LM's Spacecraft Test Lab (STL) and review of test results by the SCT, command approval by Mission Manager and uplink by the SCT. For most maneuvers, this schedule allowed the five shifts to be performed over five workdays, usually with approximately one day of margin before execution. The maneuver development schedules were preceded by a kick-off meeting using preliminary maneuver designs to inform the Project of the maneuver and post-maneuver trajectory characterizations. After this meeting the Nav Team produced a preliminary Maneuver Profile File (MPF) to give to the ACS and Prop Teams so that they could prepare a Maneuver Performance Data File (MPDF). The MPF file contains the burn time, ΔV magnitude and burn direction. The MPDF file would then allow the Nav Team to correctly model the thruster performance of the upcoming maneuver. After the orbits were determined by the OD Team, the Maneuver Team would design the upcoming maneuver and deliver a final MPF to the ACS and Prop Teams, who would then implement the burn into configuration files. These configuration files were used to set parameters in the onboard maneuver block sequence; these parameters included the burn time, slews to and from burn attitude, the ΔV and minimum and maximum burn timers. ME maneuvers used the accelerometers in the onboard IMU to determine the burn duration. Maneuvers on the ACS thruster system used timed cutoffs, since the accelerometers were not sensitive enough to adequately measure the thrust performance of these small thrusters. The final maneuver design was presented to the Project and approved during the MPF teleconference, usually the morning following the maneuver design. Once implemented by the ACS, Prop and SCT Teams, a Maneuver Performance Implementation File (MPIF) was delivered to the Nav Team along with the maneuver configuration files. These were then verified by the Nav Team and their comparisons to the MPF designs were presented to the Project during the MPIF teleconference, usually following the next morning after the implementation. Some maneuver designs required compressed timelines. These were achieved using multiple shifts per day. TCM-A1/B1 were planned to be developed over a 4-day schedule, but were cancelled on the second day after launch. Due to the short duration between LOI-A, LOI-B and the first cluster of maneuvers on GR-A (OPR-A1), the PRM-A1-3 maneuvers were developed over a 4-day schedule. The TSM-B2, B3 and OTM-B1 maneuvers required a 3-day development timeline.

A two-pass LOI maneuver development schedule was planned. The second pass (Pass-2) allowed for an update in case of larger than expected TCM-3 maneuver errors occurred. The first pass (Pass-1), which assumed nominal performance of TCM-3, began approximately 1.5 weeks after the TCM-2 maneuvers had executed. Tracking data were collected up to Oct 10 for GR-A and Oct 18 for GR-B. The first pass extended over approximately 2 weeks to ensure extra margin because of the large set of LOI risk reduction tests to be run on the STL. Plans to develop the LOI-A and B contingency maneuvers were also scheduled concurrently over a two-week period beginning Oct 24. These were to be redesigns of the LOI given the contingency that the re-pressurization of the fuel tank prior to LOI did not happen. It was determined that with the lower tank pressure, the LOI contingency burns didn't need to be redesigned. However, the minimum and maximum burn timers that limit the burn duration in case the IMU's accelerometer failed had to be increased due to the lower engine performance. The second pass was ultimately not needed due to nominal performance of the TCM-3 maneuvers.

5. Operations

5.1. Launch Phase

The GRAIL twin spacecraft were launched aboard a Delta II 7920H-10C vehicle from Space Launch Complex 17B (SLC-17B) at the Cape Canaveral Air Force Station on September 10, 2011 at 13:08:52 UTC. The launch occurred on the second opportunity of the third day of a 42-day launch period.[2] The launch vehicle departed on the long coast trajectory with a flight path azimuth of 99° from true north (Figure 10). There were two instantaneous launch opportunities per day with launch azimuths of 93° and 99° (separated by 39–44 minutes). The first day launch opportunities were aborted due to high altitude crosswinds at the last minutes of the countdown. The second day opportunities were scrubbed

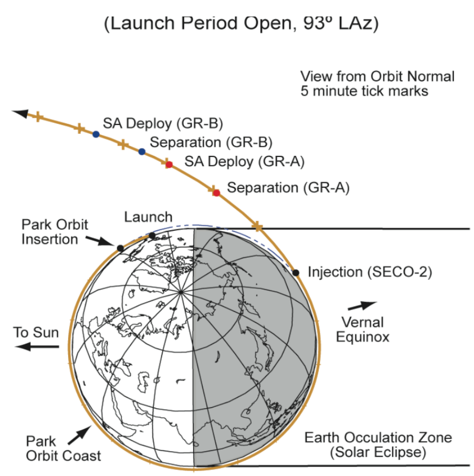


Figure 10. Earth departure trajectory

[2]The launch period was effectively reduced to 32 days after the loading of the Stage 2 propellant on September 2.

due to additional time needed to analyze a test code violation of the launch vehicle engine section heater. The first opportunity on Sep 10 was skipped again due to high winds aloft.

The Delta II launch vehicle injected the twin spacecraft into a low energy trajectory to the Moon with a launch energy, C3, of -0.69 km^2/s^2, the declination of the injection orbit apoapsis vector (DAV) equal to -6.16°, and the right ascension of the injection orbit apoapsis vector (RAV), equal to 190.55° defined at the Target Interface Point (TIP) (8 minutes after Second Engine Cutoff-2 (SECO2)). The total injected mass, which included the two spacecraft and the launch vehicle adapter assembly (LVAA), was 758.3 kg. The fully loaded GR-A and GR-B spacecraft, respectively, weighed 306.2 kg, 307.5 kg and each included 106.0 kg of propellant. Table 1 compares the actual weighed-in mass of the flight system against the capability of the launch vehicle. These weights and propellant load effectively gave the total mission ΔV capability of 938 m/s and 934 m/s, respectively, for GR-A and GR-B. Shortly after launch the Delta II second stage was inserted into a low altitude (90 nmi), circular parking orbit. After the long coast period of approximately 1 hour, the second stage was used again to inject the combined Stage II/GR-A/GR-B stack onto the departure trajectory. Following 9.5 minutes after SECO2, the GR-A spacecraft separated from the LVAA, then GR-B separated 8.25 min later. Shortly after separation each spacecraft quickly deployed their solar array, turned to an inertial Sun Point attitude and turned their transmitters on.

Table 1. Total Flight System Mass

Item	GRAIL-A	GRAIL-B	Capability
Spacecraft	182.3 kg	183.6 kg	–
Payload	17.7 kg	17.7 kg	–
Orbiter Dry Mass	200.0 kg	201.3 kg	226.0 kg
Propellant (fuel)	106.0 kg	106.0 kg	106.0 kg
He	0.2 kg	0.2 kg	0.25 kg
Orbiter Wet Mass	306.2 kg	307.5 kg	332.2 kg
LVAA Mass	144.7 kg		153.5 kg
Flight System Mass	758.3 kg		818.0 kg

The US Department of Defense's Strategic Command (STRATCOM) tracked the two spacecraft and the launch vehicle 2nd stage for the first 2 hours via the Space Surveillance Network to provide alternative tracking should a contingency arise. After approximately 78 minutes from launch, the DSN's Goldstone antennas first received signals from GR-A and GR-B. It took the DSN approximately 28 and 33 seconds, respectively, to lock up on the GR-A and GR-B signals.

The achieved performance of the Delta II vehicle was well within its expected dispersions. The difference between the launch vehicle target and the achieved parameters are tabulated in Table 2[9]. The differences are also compared against the expected dispersion statistics in Table 2. The errors in the target parameters were less than 0.35-σ. Continuous 2-way Doppler and range tracking data from approximately 1 hour after TIP through September 15 were used for this determination.

5.2. Trans-Lunar Cruise Phase

Each GRAIL spacecraft followed a low-energy trajectory to the Moon over 3.8 months. These trajectories reduced the Lunar Orbit Insertion (LOI) requirements by over 100 m/s in ΔV cost compared to typical 3-6 day direct lunar transfers. The longer transfer time of the these trajectories allowed more time for various spacecraft system checkouts, Ultra Stable Oscillator (USO) stabilization, outgassing, solar pressure characterization and ACS thruster system characterization. The pre-launch low-energy TLC trajectory designs required two deterministic TCMs to achieve the orbit insertion target at the Moon[3]. The first of these deterministic maneuvers would separate the lunar arrival conditions by 25 hours so that the operations teams could monitor each spacecraft's LOI on different days. The second TCM would then insert the spacecraft into the lunar approach trajectory. For operations, three additional statistical maneuvers were planned to clean up errors after launch and reduce trajectory errors during final lunar approach. The LOI targets of GR-A and GR-B were designed together through a combined optimization process described in Ref[3].

Table 2. Launch Vehicle Performance, Achieved vs. Target Parameters at TIP‡

Parameter	L.V. Target	Achieved	Error (A-T)	Uncertainty(1σ)	Error (No. of σ)
C3 (km^2/s^2)	-0.6956	-0.6928	0.0028	0.0119	0.24
DAV (deg)	-6.1595	-6.1638	-0.0043	0.0136	0.32
RAV (deg)	190.5432	190.5484	0.0052	0.0187	0.28

‡ September 10, 2011 14:27:12.669 UTC

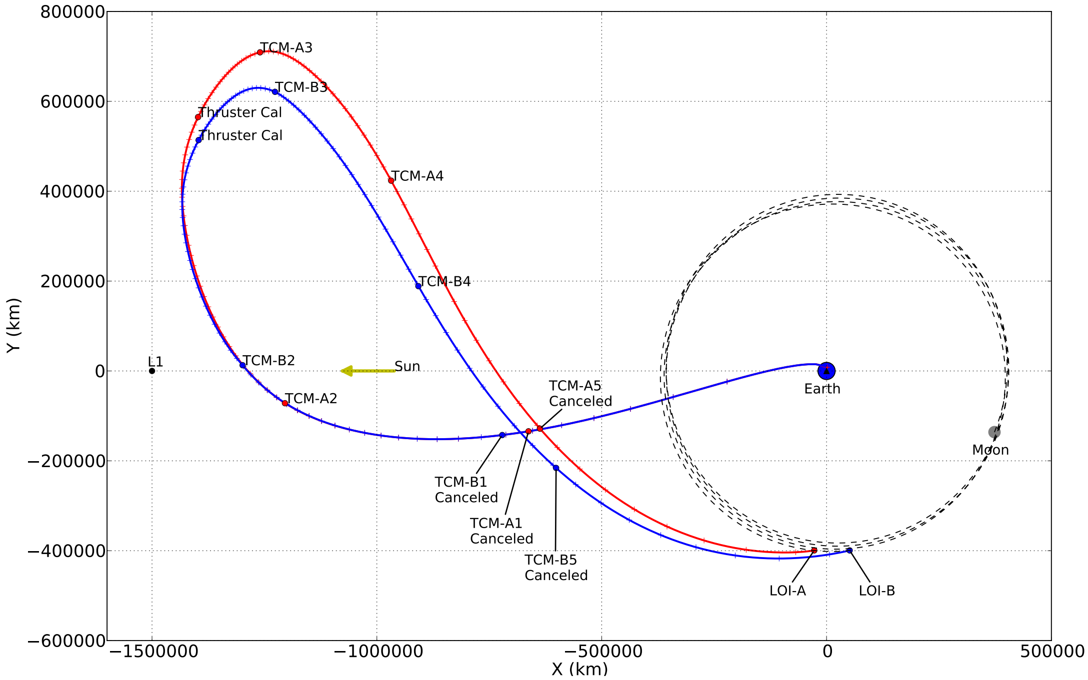


Figure 11. Trans-lunar Cruise Orbit in Sun-Earth Rotating Frame

To clean up possible launch errors, TCMs-A1, B1 were planned to take place six and seven days, respectively, after launch. At 20 and 25 days after launch, the TCM-A2, and B2 maneuvers separated the dates of the LOI maneuvers as mentioned above. The third maneuvers (TCM-A3, B3) executed on November 16 and 21, 2011 to insert the spacecraft onto their lunar approach trajectories. The statistical maneuvers TCM-A4, and B4 at three weeks (December 9, 14, 2011) prior to LOI-A and B were planned to clean up TCM-A3 and B3 errors. However, depending on the post-launch injection, two to four of these maneuvers were planned to be optimized together to save ΔV cost or improve LOI targeting. The final statistical LOI targeting maneuvers TCM-A5 and B5 were planned eight days before LOI-A and B to ensure precise LOI conditions were obtained.

5.3. TCM-A1/B1 and LGRS Checkout

Accurate performance of the Delta-II 7920H 10C launch vehicle led to the cancellation of the first of the five planned TCMs on each spacecraft to target the required lunar orbit insertion conditions. When included in the four maneuver optimization, TCM-A1 was close to zero, while TCM-B1 was close to 1 m/s. TCM-A1 and B1 could be canceled with negligible cost. Each GRAIL TLC trajectory shown in Figure 11 was optimized using 3 TCMs (TCMs 2-4). The last maneuvers TCM-A5, B5, which were planned to clean up trajectory errors 8 days from lunar orbit insertion, were canceled due to good performance of the earlier TCMs. The LOI maneuvers (LOI-A and LOI-B) executed on New Year's Eve (Dec 31, 2011) and New Year's Day (Jan 1, 2012), respectively, for GR-A, and GR-B. Each LOI burn lasted approximately 38 min long and inserted the spacecraft into an 11.5-hour period capture orbit.

Once TCM-A1, B1 were cancelled, the spacecraft-to-spacecraft range was found to be sufficiently small to perform a check out of the LGRS system. This activity was not planned pre-launch because of the high probability of the TCM-1 maneuvers, which would have rapidly increased the separation distance beyond the LGRS capability. Since no development was necessary for TCM-1 maneuvers, the SCT, MPST and Payload Teams were able to quickly plan a LGRS check-out sequence and execute it. Each spacecraft slewed on the reaction wheels to point the Ka-horns towards each other. This checkout was successfully executed on September 22nd at a separation distance of approximately 500 km. During the slews to the orbiter-to-orbiter pointing attitude described beforehand, both spacecraft experienced an 'outgassing' acceleration event as the new attitudes exposed different surfaces to the Sun. These events were evident as sudden increases in the 1-way X-band and 2-way S-band Doppler tracking data over a 40–50 min period. The accelerations on the vehicles were estimated at $1\text{–}3\times10^{-10}$ km/s^2[7].

Once the TCM-1 burns were canceled, a post-launch statistical maneuver analysis was performed. The results of this analysis are compared to the pre-launch analysis in Table 3. These results narrowed the range of the TLC ΔV costs for GR-A to be within 20.5 to 23.2 m/s and for GR-B to be within 33.9 to 35.1 m/s. These would represent the 11 to 39 percentile of the pre-launch ΔV cost for GR-A and the 21 to 35 percentile for GR-B.

Table 3. Comparison of post-launch TLC maneuver ΔV cost analysis to pre-launch results

Maneuver	Pre-launch results			Post-launch predictions		
	1%	50%	99%	1%	50%	99%
	GRAIL-A					
TCM-A1	0.00 m/s	3.81 m/s	20.33 m/s	0.00 m/s	0.00 m/s	0.00 m/s
TCM-A2	11.37 m/s	14.23 m/s	15.05 m/s	13.93 m/s	13.97 m/s	14.00 m/s
TCM-A3	4.97 m/s	6.07 m/s	9.32 m/s	6.26 m/s	6.53 m/s	7.74 m/s
TCM-A4	0.02 m/s	0.12 m/s	0.40 m/s	0.06 m/s	0.44 m/s	1.51 m/s
TCM-A5	0.00 m/s	0.00 m/s	0.01 m/s	0.00 m/s	0.02 m/s	0.06 m/s
Cumulative *Percentile*	19.95 m/s	24.51 m/s	40.39 m/s	20.45 m/s *10.9%*	20.94 m/s *17.2%*	23.23 m/s *39.1%*
	GRAIL-B					
TCM-B1	0.00 m/s	4.70 m/s	22.48 m/s	0.00 m/s	0.00 m/s	0.00 m/s
TCM-B2	21.26 m/s	23.42 m/s	24.09 m/s	25.04 m/s	25.10 m/s	25.15 m/s
TCM-B3	5.21 m/s	9.22 m/s	11.67 m/s	8.51m/s	8.78 m/s	9.37 m/s
TCM-B4	0.04 m/s	0.21 m/s	0.66 m/s	0.05 m/s	0.24 m/s	0.70 m/s
TCM-B5	0.00 m/s	0.00 m/s	0.00 m/s	0.00 m/s	0.01 m/s	0.04 m/s
Cumulative *Percentile*	32.45 m/s	37.12 m/s	53.55 m/s	33.89 m/s *20.5%*	34.11 m/s *22.5%*	35.08 m/s *32.1%*

5.4. TCM-A2/B2

TCMs-2–4 on each spacecraft were designed as part of a multi-TCM optimization strategy which targeted the spacecraft 6-coordinate Cartesian state at LOI using pre-launch LOI targets. In order to achieve the 6-state targets at LOI, the TCM-2 and 3 maneuvers targeted the 6-coordinate state at the TCM-4 burn location, while the final maneuver, TCM-4, targeted time to periapsis, inclination, and radius of periapsis at the time of LOI. The data cutoff (DCO) for the final TCM-A2 and B2 designs were on Sep 23rd and 29th, respectively (6-7 days ahead of execution). TCM-A2 was 14 m/s, while TCM-B2 was 25 m/s. Table 6 lists the errors in the burn magnitude and pointing. The designed TCM-A2 burn fell within the 38% of the pre-launch ΔV cost analysis (Table 4). The TCM-B2 design fell outside the ΔV99 percentile due to the canceled TCM-B1 burn, which was optimized to be ΔV of 1 m/s (see Table 4).

Table 4. GRAIL-A Maneuver Summary

Phase	Maneuver	Epoch (UTC)	Traj Design ΔV (m/s)	ΔV99 Percentile (m/s)	Median ΔV (m/s)	Design ΔV (m/s)	Percentile (%)
GRAIL-A							
TLC	TCM-A1	16-Sep-11 18:00:00	–	20.33	3.81	Canceled	0.0%
	TCM-A2	30-Sep-11 18:00:00	12.54	15.05	14.23	13.97	37.5%
	TCM-A3	16-Nov-11 18:00:00	8.15	9.32	6.07	6.45	57.8%
	TCM-A4	9-Dec-11 18:00:00	–	0.40	0.12	0.23	86.8%
	TCM-A5	23-Dec-11 18:00	–	0.01	0.00	Canceled	–
		TLC Total	20.69	45.10	24.24	20.66	13.5%
LOI	LOI-A	31-Dec-11 21:21:20	189.80	189.80	–	190.78	–
OPR-1	PRM-A1	7-Jan-12 13:36:58	79.60	79.60	–	79.29	–
	PRM-A2	8-Jan-12 17:21:14	79.60	79.60	–	79.29	–
	PRM-A3	9-Jan-12 17:30:42	79.60	79.60	–	79.29	–
OPR-2	PRM-A4	24-Jan-12 15:29:07	69.50	69.50	–	69.54	–
	PRM-A5	25-Jan-12 18:20:36	69.50	69.50	–	69.54	–
	PRM-A6	26-Jan-12 16:58:03	69.50	69.50	–	69.54	–
	PRM-A7	27-Jan-12 16:48:47	69.50	69.50	–	69.54	–
Statistical ΔV associated with LOI/OPR ΔV99			–	1.30	–	–	–
TSF	TSM-A1	7-Feb-12 16:32:41	15.30	15.30	–	8.23	–
	TSM-A2	20-Feb-12 15:05:14	14.00	14.00	–	19.28	–
		Lunar Orbit Total	735.90	737.20	–	734.32	–
		Cumulative Total Prime Mission ΔV	756.59	782.30	–	754.98	–
		ΔV Capability	–	938.00	–	938.00	–
		Margin	–	155.70 16.75 kg	–	183.02 19.68 kg	–

Table 5. GRAIL-B Maneuver Summary

Phase	Maneuver	Epoch (UTC)	Traj Design ΔV (m/s)	ΔV99 Percentile (m/s)	Median ΔV (m/s)	Design ΔV (m/s)	Percentile (%)
GRAIL-B							
TLC	TCM-B1	17-Sep-11 18:00:00	–	22.48	4.70	Canceled	0.0%
	TCM-B2	5-Oct-11 18:00:00	25.86	24.09	23.42	25.10	100.0%
	TCM-B3	21-Nov-11 18:00:00	7.48	11.67	9.22	8.85	44.4%
	TCM-B4	14-Dec-11 18:00:00	–	0.66	0.21	0.26	63.6%
	TCM-B5	24-Dec-11 18:00:00	–	0.01	0.00	Canceled	–
		TLC Total	33.34	58.90	37.56	34.20	23.6%
LOI	LOI-B	1-Jan-12 22:05:16	191.40	191.40	–	192.32	–
OPR-1	PRM-B1	13-Jan-12 21:00:49	74.50	74.50	–	74.11	–
	PRM-B2	15-Jan-12 1:35:34	74.50	74.50	–	74.11	–
	PRM-B3	16-Jan-12 2:51:04	74.50	74.50	–	74.11	–
OPR-2	PRM-B4	31-Jan-12 16:29:06	70.70	70.70	–	70.63	–
	PRM-B5	1-Feb-12 17:23:39	70.70	70.70	–	70.63	–
	PRM-B6	2-Feb-12 16:48:20	70.70	70.70	–	70.63	–
	PRM-B7	3-Feb-12 17:20:09	70.70	70.70	–	70.63	–
Statistical ΔV99 associated with LOI/OPR ΔV99			–	1.30	–	–	–
TSF	TSM-B1	14-Feb-12 3:15:25	17.50	17.50	–	17.50	–
	TSM-B2	24-Feb-12 17:24:33	1.90	1.90	–	2.07	–
	TSM-B3	29-Feb-12 13:46:37	2.80	2.80	–	0.51	–
Science	OTM-B1	7-Mar-12 20:43:31	0.01	0.01	–	Canceled	–
	OTM-B2	30-Mar-12 18:41:40	0.03	0.03	–	0.03	–
		Lunar Orbit Total	719.94	721.24	–	717.28	
		Cumulative Total Prime Mission ΔV	753.28	780.14	–	751.48	
		ΔV Capability	–	934.20	–	934.20	
		Margin	–	154.06	–	182.72	
		Fuel left after PM		16.50 kg		19.93 kg	

5.5. ACS Thruster Calibrations

The thruster calibration of the ACS thrusters were performed on Nov 1 and 2, respectively, for GR-A and GR-B. These calibrations involved measuring the translational ΔVs from the warm-gas ACS thrusters by thrusting at 3 orthogonal orientations with respect to the Earth-line direction to calibrate the thrust levels and alignments.

The in-flight results would be used to update the thruster properties onboard and in ground-based tools. Better knowledge of the thrusters allows for more precise maneuvering and reconstruction. The thruster calibration was primarily the responsibility of the ACS Team to build and execute. However, the Nav Team had insight into the execution performance through the receipt of real-time 1-way X-band and 2-way S-band Doppler data. Therefore, the OD Team had two main responsibilities. First, the OD Team was responsible for monitoring the real-time Doppler residuals during the maneuvers, processing the Doppler data into estimated velocity changes, and then delivering them to the ACS Team to use in their analysis. The second task was to perform a complimentary analysis to estimate the thruster characteristics based solely on Doppler data. The Navigation analysis of the thruster calibration provides ACS with a secondary estimate of the thruster magnitude and directions using a Doppler-only data structure.

Each GR-A and GR-B thruster calibration activity was a three-hour sequence that performed a specified set of firings at three orthogonal attitudes. There were three different calibration tests at each attitude, an eight thruster calibration, a single thruster calibration, and a four thruster calibration. The eight thruster calibration fired all eight thrusters simultaneously for 80 seconds. Ideally, if all thrusters are balanced, the residual ΔV observed from this maneuver should be zero. Otherwise, observation of the ΔV from this maneuver can tell us in which direction the thrusters have more magnitude. The single thruster calibration fired each thruster individually in a specified order to minimize overall ΔV. Each thruster fires 160 pulses, 60 milliseconds each, and 1.5 seconds between thrusters. Each single thruster calibration imparts 7.2 mm/s of ΔV per attitude. This calibration was the primary measurement used to characterize the thrusters. The four thruster calibration fires four thrusters at a time. The first set of four induces a +Z rotation, while second set of four induces a -Z rotation. The thrusters fire 250 pulses of 50 ms for 50 sec at a 50% duty cycle. This set-up was chosen to mimic a desaturation calibration maneuver, used in the Science Phase to unload momentum from the reaction wheels. Each four thruster calibration firing contributes 10 mm/s of ΔV per attitude. The thruster calibration on GRAIL-A was

Table 6. Maneuver Performance Summary

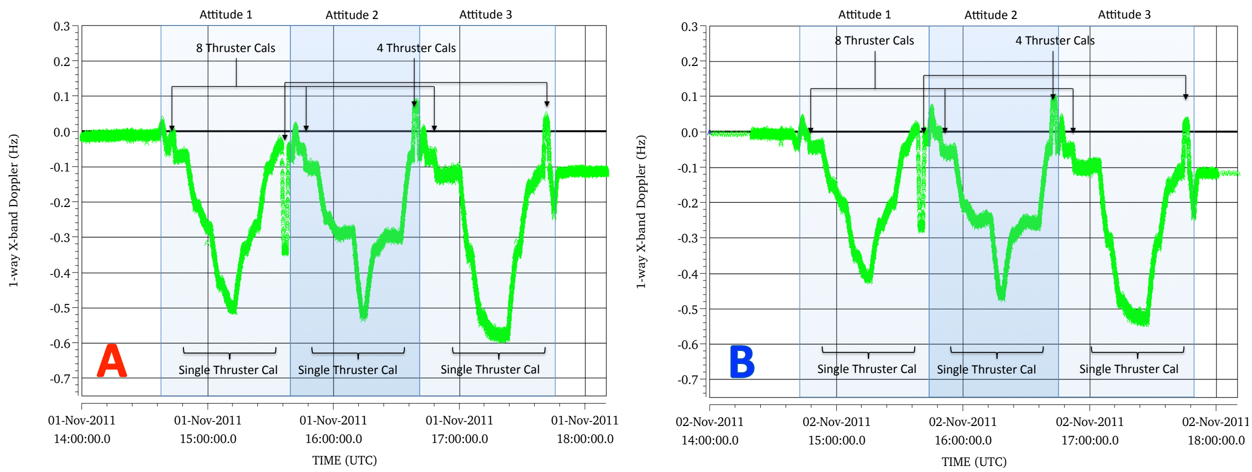
Phase	Maneuver	Epoch (UTC)	Design ΔV (m/s)	Reconstructed ΔV (m/s)	Direction RA (deg)	Direction Dec (deg)	Duration (sec)	ΔV Mag Error (mm/s)	Pointing Error (mrad)	Timing Update (sec)	OD Reconstruction Solution	Lunar Phase Post-Mnvr Period (hrs)	Period Errors (sec)
GRAIL-A													
TLC	TCM-A1	16-Sep-11 18:00:00	Canceled	Canceled	–	–	–	–	–	–	–	–	–
	TCM-A2	30-Sep-11 18:00:00	13.97	13.97	82.62	-28.46	128.0	0.11 (0.01σ)	4.32 (1.5σ)	–	od023v1	–	–
	TCM-A3	16-Nov-11 18:00:00	6.45	6.44	118.71	-2.51	62.6	-2.58 (0.4σ)	1.19 (0.8σ)	–	od031v1	–	–
	TCM-A4	c9-Dec-11 18:00:00	0.23	0.23	0.73	-5.15	14.8	-5.2 (3.9σ)	1.70 (0.8σ)	–	od043v1	–	–
	TCM-A5	23-Dec-11 18:00:00	Canceled	Canceled	–	–	–	–	–	–	–	–	–
LOI	LOI-A†	31-Dec-11 21:21:20	190.78	191.07	330.22	-68.45	2295.7	290.00 (1.8σ)	3.21 (1.6σ)	None	od051v5	11.44	-234 (-164)
OPR-1	PRM-A1	7-Jan-12 13:36:58	79.29	79.25	86.30	-69.38	650.9	-32.30 (0.5σ)	5.41 (1.9σ)	–	od059v3	7.00	5.25
	PRM-A2	8-Jan-12 17:21:14	79.29	79.27	86.30	-69.38	712.8	-18.13 (0.3σ)	7.49 (2.6σ)	33.0	od060v3	4.83	8.70
	PRM-A3‡	9-Jan-12 17:30:42	79.29	79.26	86.30	-69.38	764.7	-22.06 (0.3σ)	6.46 (2.2σ)	75.2	od061v3	3.65	2.67 (8.1)
OPR-2	PRM-A4	24-Jan-12 15:29:07	69.54	69.54	86.73	-69.82	716.3	-8.38 (0.1σ)	3.88 (1.3σ)	–	od072v3	2.98	0.46
	PRM-A5	25-Jan-12 18:20:36	69.54	69.54	86.73	-69.82	743.4	-6.87 (0.1σ)	2.92 (1.0σ)	None	od073v3	2.51	0.79
	PRM-A6	26-Jan-12 16:58:03	69.54	69.53	86.73	-69.82	765.9	-14.00 (0.2σ)	4.44 (1.5σ)	None	od074v3	2.17	1.02
	PRM-A7‡	27-Jan-12 16:48:47	69.54	69.54	86.73	-69.82	784.3	-4.90 (0.08σ)	3.27 (1.1σ)	20.7	od075v3	1.90	0.49 (1.3)
TSF	TSM-A1	7-Feb-12 16:32:41	8.23	8.23	114.58	-11.24	93.6	-3.92 (0.5σ)	8.02 (5.5σ)	–	od086v3	1.90	-0.61
	TSM-A2	20-Feb-12 15:05:14	19.28	19.28	103.91	-50.44	223.6	2.25 (0.1σ)	0.97 (0.7σ)	–	od098v3	1.89	0.10
GRAIL-B													
TLC	TCM-B1	17-Sep-11 18:00:00	Canceled	Canceled	–	–	–	–	–	–	–	–	–
	TCM-B2	5-Oct-11 18:00:00	25.10	25.09	315.97	-36.61	235.7	-7.23 (0.3σ)	5.32 (1.8σ)	–	od025v1	–	–
	TCM-B3	21-Nov-11 18:00:00	8.85	8.84	284.43	16.76	90.3	-2.68 (0.4σ)	4.49 (1.5σ)	–	od031v1	–	–
	TCM-B4	14-Dec-11 18:00:00	0.26	0.26	109.38	-25.89	20.9	-0.50 (0.3σ)	0.77 (0.4σ)	–	od045v1	–	–
	TCM-B5	24-Dec-11 18:00:00	Canceled	Canceled	–	–	–	–	–	–	–	–	–
LOI	LOI-B†	1-Jan-12 22:05:16	192.32	192.32	329.52	-69.33	2256.4	194.00 (1.2σ)	1.88 (0.9σ)	None	od053v5	11.46	-159 (-64)
OPR-1	PRM-B1	13-Jan-12 21:00:49	74.11	74.12	89.02	-68.01	621.6	15.56 (0.3σ)	2.15 (1.5σ)	–	od065v3	7.15	3.18
	PRM-B2	15-Jan-12 1:35:34	74.11	74.11	89.02	-68.01	674.5	3.66 (0.06σ)	1.67 (1.2σ)	11.0	od066v3	5.05	4.83
	PRM-B3‡	16-Jan-12 2:51:04	74.11	74.10	89.02	-68.01	718.4	-8.69 (0.1σ)	0.95 (0.7σ)	34.1	od067v3	3.85	2.1(5.50)
OPR-2	PRM-B4	31-Jan-12 16:29:06	70.63	70.66	85.83	-69.14	731.2	28.00 (0.5σ)	2.07 (1.4σ)	–	od079v3	3.11	-1.08
	PRM-B5	1-Feb-12 17:23:39	70.63	70.64	85.83	-69.14	759.8	3.98 (0.07σ)	2.70 (1.9σ)	None	od080v3	2.60	-0.15
	PRM-B6	2-Feb-12 16:48:20	70.63	70.63	85.83	-69.14	783.7	-3.48 (0.06σ)	2.08 (1.4σ)	-18.7	od081v4	2.23	0.32
	PRM-B7‡	3-Feb-12 17:20:09	70.63	70.63	85.83	-69.14	803.2	0.55 (0.01σ)	2.25 (1.6σ)	-24.8	od082v3	1.95	0.28(0.13)
TSF	TSM-B1	14-Feb-12 3:15:25	17.50	17.49	84.07	-69.57	202.9	-1.31 (0.1σ)	1.66 (1.1σ)	–	od093v3	1.90	0.15
	TSM-B2	24-Feb-12 17:24:33	2.07	2.06	269.30	-46.59	24.1	-4.61 (1.6σ)	7.41 (5.1σ)	–	od102v3	1.89	0.15
	TSM-B3	29-Feb-12 13:46:37	0.51	0.50	117.73	7.21	6.2	-6.07 (2.6σ)	1.44 (0.9σ)	–	od111v4	1.89	0.06
Science	OTM-B1	7-Mar-12 20:43:31	Canceled	Canceled	–	–	–	–	–	–	–	–	–
	OTM-B2	30-Mar-12 18:41:40	0.03	0.03	306.82	-45.29	83.3	-0.98 (0.4σ)	62.96 (5.2σ)	–	od141v4	1.89	0.0059

† Period errors for LOI are given with respect to the Pass-1 reference, refod017v1, and with respect to the last pre-LOI reference, refod048v1 in parentheses.

‡ Period errors for PRM-3 and PRM-7 maneuvers are with respect to the individual burn and to the total cluster in parentheses.

performed on November 1, 2011. The thruster calibration on GRAIL-B was performed on November 2, 2011. Both tests began around 14:00 UTC and lasted until 17:53 UTC. Figure 12 shows the thruster calibration activities during each of the three attitudes on both vehicles in the 1-way X-band Doppler pass-through residuals. The ΔVs of each thruster firing were not modeled in predicted trajectory, resulting in the structure observed in Figure 12.

The thruster calibrations on GR-A and GR-B performed in-flight executed very close to the nominal plan. The results from the Navigation analyses of both S/C showed that the differences between the estimated values and the modeled values were very small. These errors were small enough, given the uncertainties, that it was deemed not necessary to update the values for the thrusters of both GR-A and GR-B in flight or ground tools. The navigation and ACS analyses were complimentary. The results were close enough with independent methods that there was confidence in the results and decision.



(a) GR-A (b) GR-B

Figure 12. GRAIL-A/B 1-way X-band Doppler pass-through residuals during thruster calibration

5.6. TCM-A3/B3

The DCOs for the TCM-A3 and B3 designs were on Nov 10th and 15th, respectively (6 days ahead of execution). These maneuvers were re-optimized with TCM-4 and were designed to insert the GRAIL spacecraft into the lunar-approach manifold. TCM-A3 was designed to execute on Nov 16, 2012 18:00 UTC with a ΔV magnitude of 6.5 m/s. TCM-B3 executed five days later on Nov 21, 18:00 UTC with a magnitude of 8.9 m/s. As shown in Table 6, the TCM-A3 and B3 maneuver magnitude errors were less than 1σ of the nominal values. The pointing errors were approximately 1 mrad for TCM-A3 and 4 mrad for TCM-B3.

5.7. TCM-A4/B4

The DCOs for the TCM-A4 and B4 burn designs were on Dec 4th and 8th, respectively (5–6 days ahead of execution). The designed ΔV magnitude for TCM-A4 was 0.23 m/s and 0.26 m/s for TCM-B4. In order to achieve better accuracy using the ME for these small maneuvers, the duty cycle needed to be reduced as determined by the LM Prop Team to 15%. Since this was a first-time event on either orbiter, the duration of the ME at 15% duty cycle was called into question. Re-evaluation of the thrust performance under the 15% duty cycle by the Prop Team showed that the duration of the burn could be longer than the nominal 6% maximum burn timer used for the majority of the GRAIL maneuvers. Therefore, the maximum burn timers were increased to 10%. However, the TCM-A4 maneuver had terminated due to achieving the maximum burn timer. This meant that the 15% duty cycle main engine performed less than expected. This resulted in a larger than expected under-burn magnitude error close to 4σ less than the nominal. Despite this error, the post-burn orbit results mapped to the LOI aim point were found to lie close to the targeted values. As a corrective action to this issue, the TCM-B4's maximum burn timer was increased to 20% higher duration. The TCM-B4 burn performed within 1σ of the expected performance.

5.8. TCM-5 Go/No Go Criteria

Due to the complexity of achieving the Science Phase orbit, it was a difficult task to determine how LOI insertion errors mapped into ΔV cost for the lunar phase. For instance, the Traj Team designed the orbits to be co-planar by targeting each orbiter's inclination and node at LOI to be offset from each other. The evolution of the osculating orbits, as seen in the inclination and node plots in Figure 13, from LOI through OPR and the TSF Phases brought the orbits into the same osculating values. It was hoped that the TCM-4 maneuvers could be performed accurately enough so the TCM-5 maneuvers were not needed. The Nav Team had to determine what orbit errors could be tolerated from both orbiters yet still achieve Science formation with high confidence with the available propellant. Several analyses (Monte Carlo and worst-case) were performed using the expected post-TCM-A4 and B4 3σ dispersions based on post-Launch statistical maneuver analysis (from OD solution od017v1) in radius of periapsis, inclination, node, time of periapsis and argument of periapsis. To evaluate how these errors affected the ability to obtain the desired Science orbit formation within the Science requirements, each of these dispersed cases for GR-A and GR-B were propagated from LOI to Science formation. It was determined that the post-TCM-4 dispersions in the node, inclination and argument of periapsis could be corrected fairly easily within the TSF Phase with manageable ΔV cost. The dispersions in the radius and time of periapsis were found to be more important to control as they affected the post-LOI orbits and could violate the self-imposed 10 km collision avoidance requirement in the OPR through the beginning of the TSF Phase. These analyses indicated regions in the ballistic LOI target radius of periapsis and time to periapsis space where COLA between the two orbiters was more severe. These regions indicating possible COLA violations, however, were subject to the Project's discretion, meaning that if the solutions were clearly in these regions, the Project could overrule the 'Go' criteria and cancel TCM-5. Other regions outside of the 3-σ errors of both of these parameters were considered errors needing corrections. If not corrected with TCM-A5 or B5, one or two additional maneuvers may have been needed to achieve Science formation. With this, criteria for needing to perform TCM-5 were created. Figures 15a and b show the TCM-A5 and B5 Go/No-go criteria relative to the incoming LOI target orbit radius and time to periapsis errors. The expected dispersion (3σ) of the TCM-4 burn performance is indicated by the scatter of red dots. The expected 3-σ dispersions of the TCM-5 burn are represented by the scatter of blue dots. The diagonal-vertical lines indicating COLA violations in the TCM-A5 and B5 corridors move depending on the location in the corridor the other orbiter achieved. The upper left side of the TCM-B5 corridor is bounded by cases where the altitude after TSM-B1 would be too low. The lower right side of this corridor was bounded by cases where the TSM-A2 maneuver would grow too large. The timeline from TCM-4 to the TCM-5 Go/No-go meeting is shown in Figure 14. The scheduled tracking passes used in the Go/No-Go evaluations for each orbiter are shown in this timeline. The covariance studies of the execution errors TCM-A4 and B4 on the orbit dispersions showed that knowledge of the orbit parameters would converge on Dec 17 for GR-A and Dec 18 for GR-B. Given these criteria, it can be seen from the final OD solutions in Figures 15a and b that both TCM-A5 and B5 were not needed and thus cancelled.

5.9. Pre-LOI Tank Re-pressurization

Because of the low ΔV cost expended during TLC, the tank pressures in both spacecraft were sufficiently high enough to perform nominal LOI maneuvers without the re-pressurizations. The pre-LOI pressure on GR-A was approximately 358 psia and 325 psia on GR-B. The contingency no-repress LOI design had kept the design of the nominal LOI burns, but changed the maximum burn timers to allow for the longer burns. With the no-repress LOI burns, the pressures at the end of LOI would reach down to approximately 153 psia on GR-A and 146 psia on GR-B. These lower limits were found acceptable by the Prop Team. With this information, the Project briefly entertained the notion of skipping the pre-LOI re-pressurizations; however, this scheme violated the JPL "test-as-you-fly, fly-as-you-test" principle. Therefore, in preparation for LOI, the ME tanks were planned to be re-pressurized on December 23, but GR-A showed a faulty pressure transducer reading when a pyro valve was fired. "The helium supply is separated from the propellant tank by a high-pressure latch valve and a pyro valve. The latch valve is upstream of the pyro valve."[6] Instead of increasing the pressure reading a small amount when the pyro valve was fired before opening the latch valve, the propellant tank unexpectedly decreased by 60 psi.[6] This caused a delay for the GR-B tank re-pressurization and the completion of the GR-A re-pressurization opening of the GR-A latch valve until this behavior was better understood. It was determined that this behavior had been observed on previous LM-built missions, such as Mars Odyssey, and understood as a zero-bias shift in the pressure transducer caused by the pyro shock.[6] The pyro valve was fired on GR-B on Dec 24, 2012, then the latch valves were cycled for both GR-A and GR-B, respectively, on Dec 26 and 27. After the pyro event on GR-B, the pressure transducer showed a similar decrease in the tank pressure (-37 psi). Overall, the tank

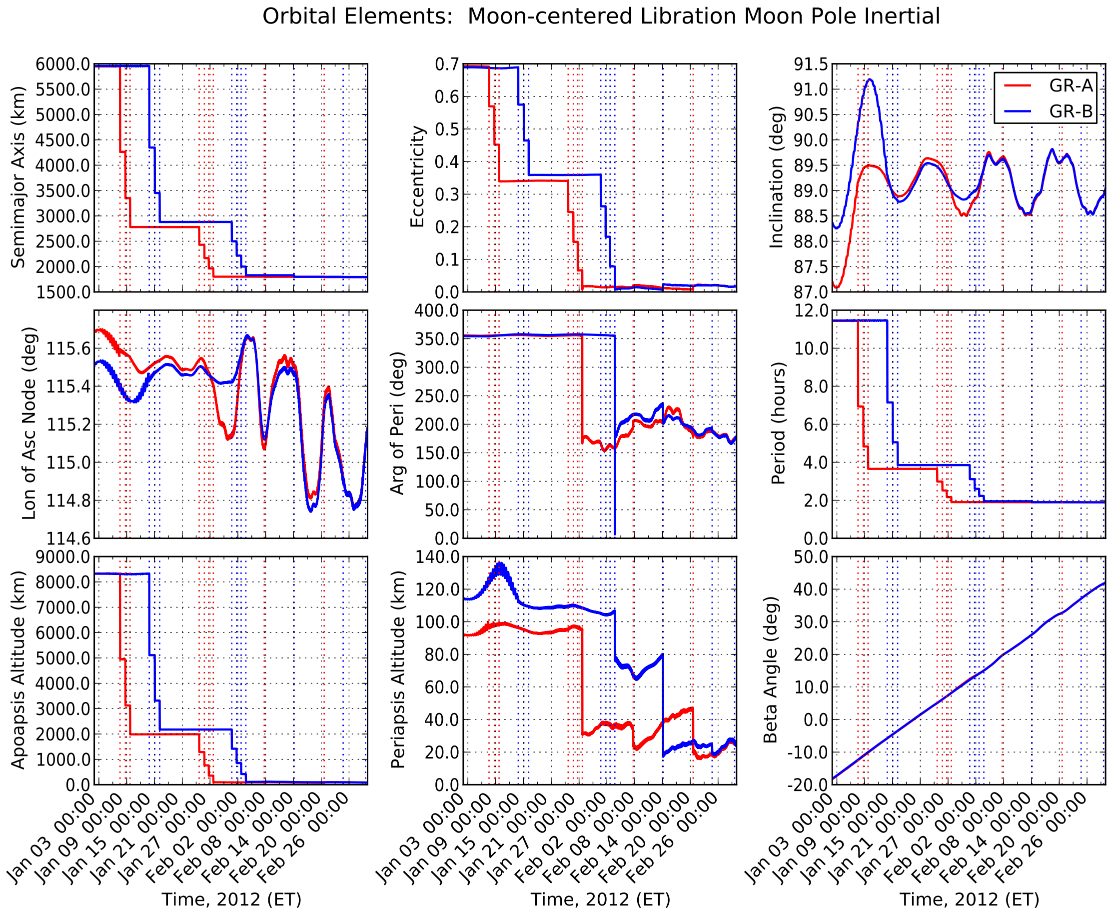


Figure 13. GR-A (red) and GR-B (blue) orbital elements during the OPR and TSF Phases

re-pressurizations achieved approximately 404 psia on GR-A and 406 psia on GR-B before the LOI burns took place. Figure 5 shows the telemetry of the ME tank pressure levels during these events.

5.10. Lunar Orbit Insertion

The LOI burns were designed early in the TLC phase during the LOI Pass-1 development timeline. The Pass-1 timelines took place a couple weeks after the TCM-A2 and B2 maneuvers. The LOI burn designs were targeted to match the post-LOI orbit state at the apoapsis following the LOI burn from the post-Launch reference, refod017v1. The LOI-A and B burn magnitudes were 190.9 m/s and 192.3 m/s and required approximately 24 kg of fuel each. The nominal durations were 39.9 min and 38.7 min, respectively, for LOI-A and LOI-B. The burn timers for LOI were set at $\pm 10\%$ from the nominal duration. Due to the significant time associated with running and evaluating the LOI risk reduction tests, the SCT recommended only allowing changes from the nominal LOI designs in the LOI ΔV burn magnitude, burn start time, or minimum and maximum burn timers for the contingency no-repress LOI and Pass-2 redesigns. The Nav Team found that it was not necessary to perform a redesign of the LOI burns for the no-repress cases. If the re-pressurization failed to take place, the LOI burns were expected to execute approximately 5 and 8 minutes longer than the nominal durations, respectively, for GR-A and GR-B (45 min and 47 min). The maximum burn timers in the contingency case would have to be updated to allow for the longer burn duration due to the degraded ME performance at the lower pressures. Just prior to LOI, the Prop Team found that the thrust levels for LOI were predicted to be slightly higher than the levels used in the Pass-1 design. The updated durations would be approximately 1–1.5 minutes shorter than the nominal design.

The LOI maneuvers executed on New Year's Eve (Dec 31, 2011, 21:21 UTC) and New Year's Day (Jan 1, 2012, 22:05 UTC), respectively, for GR-A, and GR-B. The LOI burns were controlled by optimized in-plane, constant pitch rate

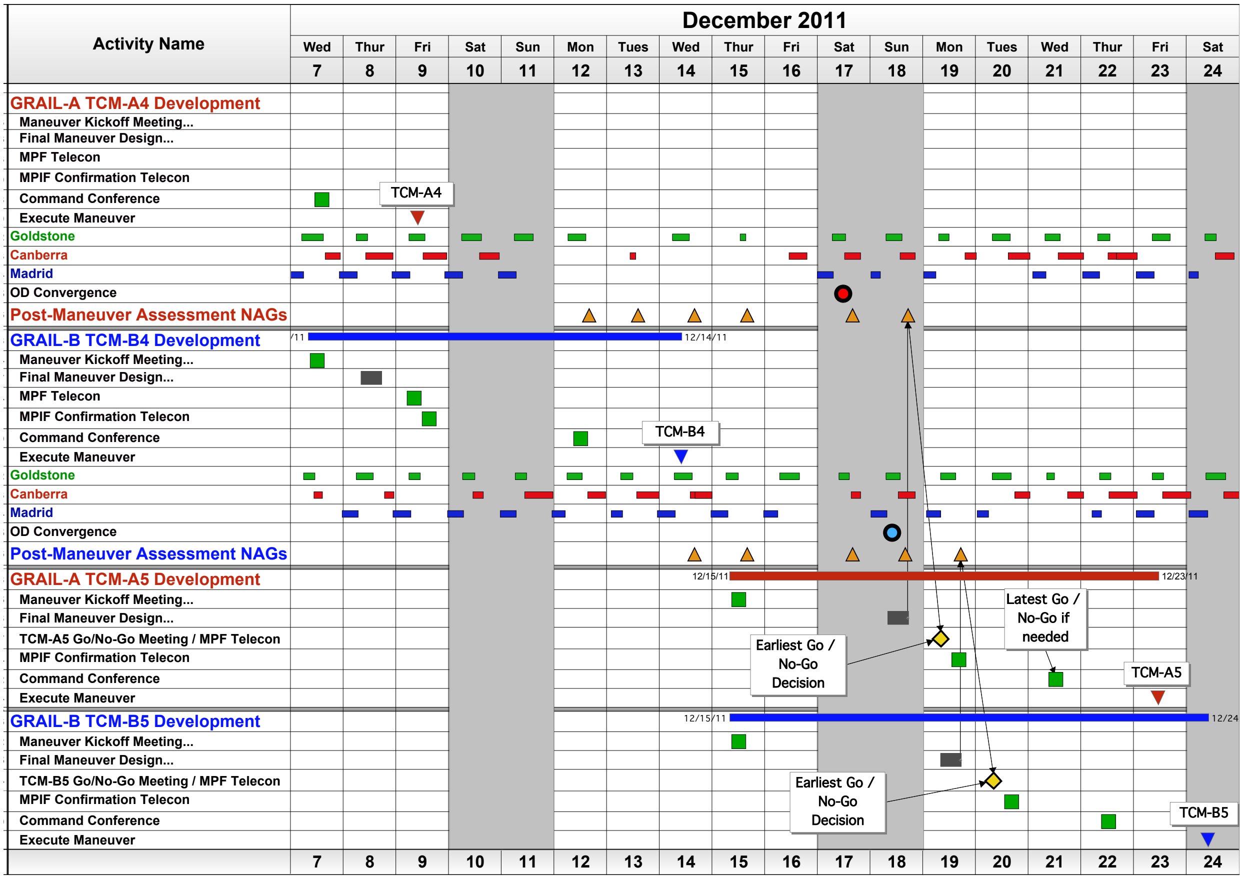
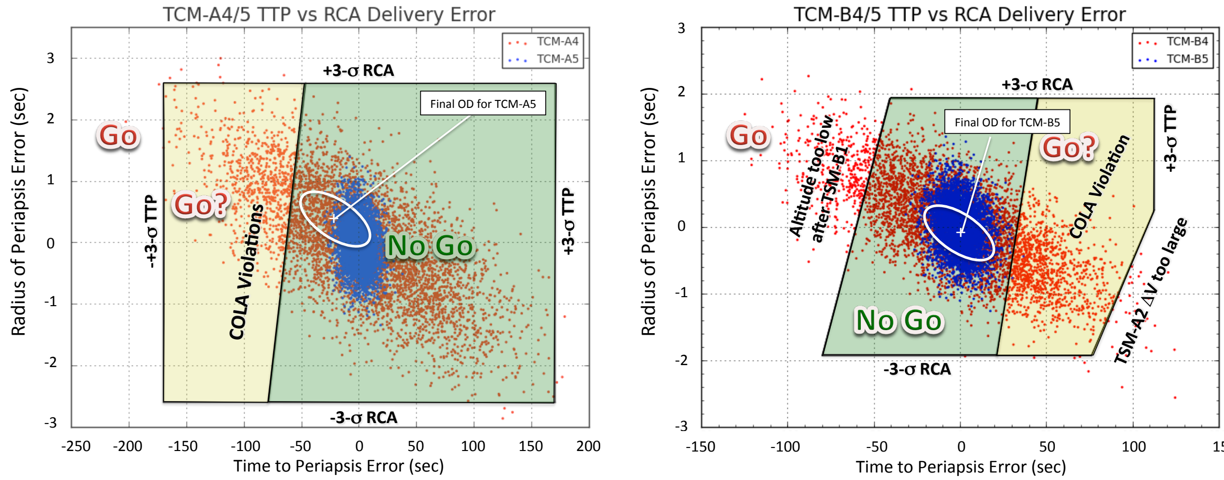


Figure 14. Timeline for TCM-5 Go/No-Go

thrust vector steering. Each LOI burn lasted approximately 38 min and inserted the spacecraft into an approximate 11.5-hour period capture orbit. At approximately 11 minutes before LOI, each orbiter slewed to the initial burn attitude. A 30° roll about the -X-axis was also performed to keep the Moon out of the star tracker field of view. During the burn, the orbiters pitched over about the orbit normal at a pre-defined rate to keep the ME thrust vector nearly aligned with the lunar-relative velocity vector. This strategy reduced the gravity losses during the burn and optimized the ΔV and fuel use. Different levels of fault protection were disabled beginning approximately 2 days before to 15 minutes before LOI for both orbiters to ensure the LOI commands would execute. The telecom system was configured two hours before LOI to turn ranging off, lower the data rate to 1000 bps and turn the 2-way coherency off in order to increase the signal-to-noise margin during the LOI slews and burn execution. After the burn, the fault protection responses were re-enabled, the orbiters slewed back to Sun-pointed attitude and the telecom system was reconfigured to increase the data rate, and turn coherency on. The ranging signal, however, was not turned back on as planned. The LOI burns were designed to occur on consecutive days during the Madrid-Goldstone tracking pass overlaps to provide redundant coverage of these critical events in the event of a tracking station failure. Telecommunications were maintained throughout the burn via the LGA mounted on the orbiters' +X-panel. Due to power restrictions during LOI, the LOI burn performance could only be monitored by the 1-way X-band Doppler signal from the RSB antennas. Although the low antenna gain pattern drops off significantly at approximately 90-120°, the signal was adequate to monitor the burn performance during each maneuver.

Thrust levels of LOI-A and LOI-B, respectively, ranged from 34 N and 34.5 N at the start to approximately 18 N and 19 N at the end as shown in Figure 6. The corresponding tank pressures on GR-A were reduced from 404 to 177 psia, while GR-B's tank pressures were reduced from 406 to 187 psia after completion of the LOI burns (Figure 5). Figures 16a, and b show the 1-way X-band Doppler residuals of the LOI-A and B maneuvers relative to predicted trajectories without the burns modeled to observe the LOI maneuvers. Figures 16c and d compare these Doppler signals against the predicted trajectories with the LOI maneuvers modeled. These figures indicate how the burns performed relative to the



(a) TCM-A5 Go/No-Go Corridor

(b) TCM-B5 Go/No-Go Corridor

Figure 15. TCM-5 Go/No-Go Corridor

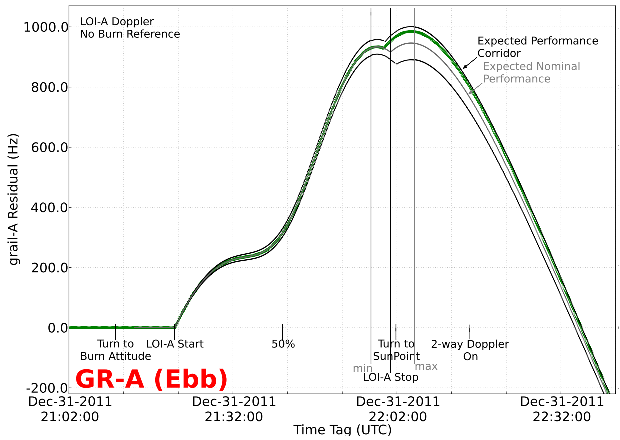
Table 7. **The post-LOI-A & B orbits compared to the LOI targets and last pre-LOI orbit reference, refod048v1**

Target Parameter	LOI Targets Pass-1 Reference refOD017v1	Post-Repress#1 Reference Update refOD048v1	Post-LOI Reconstruction	Uncertainty (1σ)	Error from LOI Targets refOD017v1	Error from Reference Update refOD048v1	Predicted Error (1σ)
GRAIL-A							
Semi-major Axis (km)	5970.745	5964.000	5948.200	0.012	-22.545	-15.800	16.000
Eccentricity	0.694	0.693	0.692	4.7E-07	-0.001800	-0.001160	0.000610
Inclination (deg)	87.435	87.464	87.463	6.9E-05	0.028	-0.001	0.044
Long. of Asc. Node (deg)	115.679	115.690	115.680	2.6E-05	0.001	-0.010	0.005
Arg. o f Periapsis (deg)	355.714	355.780	356.020	2.4E-04	0.306	0.240	0.180
Period (sec)	41400.021	41330.000	41166.000	0.126	-234.021	-164.000	180.000
Periapsis Range (km)	1827.108	1828.900	1830.900	0.006	3.792	2.000	2.500
GRAIL-B							
Semi-major Axis (km)	5970.745	5961.600	5955.500	0.021	-15.245	-6.100	16.000
Eccentricity	0.690	0.690	0.689	9.6E-07	-0.000940	-0.000440	0.000750
Inclination (deg)	88.331	88.328	88.326	3.4E-05	-0.005	-0.002	0.052
Long. of Asc. Node (deg)	115.552	115.520	115.520	2.0E-05	-0.032	0.000	0.010
Arg. of Periapsis (deg)	354.132	354.430	354.500	3.6E-04	0.368	0.070	0.170
Period (sec)	41400.021	41305.000	41241.000	0.219	-159.021	-64.000	180.000
Periapsis Range (km)	1850.931	1851.100	1851.800	0.011	0.869	0.700	3.600

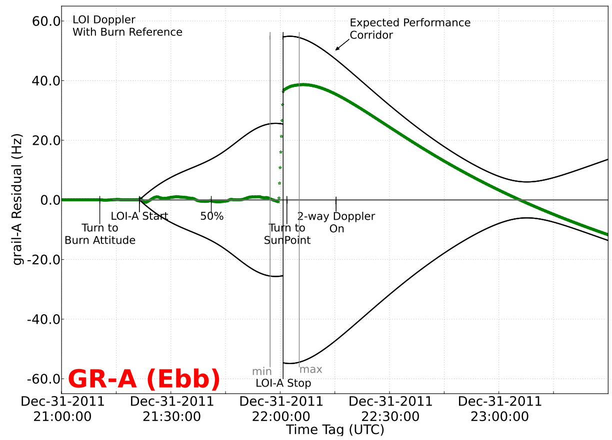
expected 3-σ dispersions. Due to increased thrust levels from the nominal, the LOI-A and LOI-B burns completed, respectively, 1.6 minutes and 1 minute less than the nominal burn durations. The LOI-A burn achieved a 11.44 hr orbit with 87.4° inclination[3] and periapsis altitude of 88 km. The orbit period was 4 minutes less than the Pass-1 design and 3 minutes less than the nominal design with the updated thrust data. The LOI-B burn achieved a 11.46 hr orbit with an inclination of 88.3°, and the periapsis altitude of 108 km. This orbit period was approximately 3 minutes less the nominal design and approximately 1 minute less than the nominal design with the updated thrust data. Table 6 lists the LOI-A and LOI-B burn errors. The ΔV magnitude errors on these burns were 290 mm/s for LOI-A and 194 mm/s for LOI-B greater than the nominal magnitudes. The achieved orbit was within 2σ of the expected dispersions. The post-LOI osculating orbit elements are compared to the targeted values from the Pass-1 design reference in Table 7. The post-LOI orbit inclination values were in error from the targets by 0.03° and -0.005°, respectively, for GR-A and GR-B, while the node errors were, respectively, off by 0.001° and -0.032°. The periapsis radius error was 3.8 km for GR-A and 0.9 km for GR-B. When the reference trajectory was updated after the first re-pressurization prior to LOI using refod048v1 and the latest LOI thrust information and incoming LOI state conditions, the post-LOI errors were significantly smaller. These errors give a better indication how LOI performed. These errors, also compared in Table 7, show the period error to be approximately 1 minute less in both the GR-A and GR-B orbits.

The propellant tank for each orbiter was again re-pressurized on Jan 2, 2012 following the LOI burns. These re-pressurizations involved opening the high pressure latch valve then closing it once the pressure reached predicted levels. Finally, the valves were then left open once the telemetry indicated it was safe to do so. The tank pressures, as shown in Figure 5, reached 430 psia for GR-A and 408 psia for GR-B prior to the start of the OPR Phase.

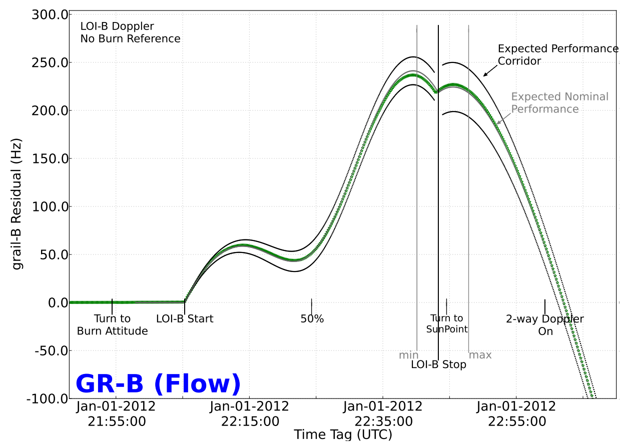
[3]relative to the Moon's equator in the Libration Moon Pole Inertial frame, which is defined by the Moon's principal axes at a fixed epoch



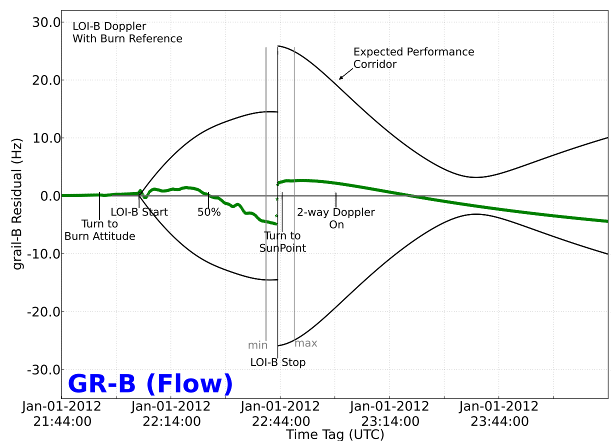
(a) LOI-A residuals with respect to the no-burn predicts



(b) LOI-A residuals with respect to the with-burn predicts



(c) LOI-B residuals with respect to the no-burn predicts



(d) LOI-B residuals with respect to the with-burn predicts

Figure 16. Real-time Doppler residuals during the LOI burns plotted relative to the 3-σ performance corridor

5.11. Orbit Period Reduction

Shortly following the LOI burns, the spacecraft entered the OPR Phase, where the orbit periods for both spacecraft were reduced to just below 2 hours using a concept similar to the aerobraking strategies developed for the Mars missions (MGS, Odyssey, and MRO). Instead of atmospheric drag, however, maneuvers are repeatedly performed at periapsis to reduce the orbit period. This was accomplished within approximately one month through the execution of 7 Period Reduction Maneuvers (PRMs) per orbiter grouped into two clusters (three in OPR-1 and four in OPR-2). Figure 17 illustrates how the orbit period is reduced following the PRM maneuvers. To reduce operation activities, the same maneuver design (ΔV magnitude and inertial direction) was used for each burn in a cluster; the burns were executed on consecutive days in January and beginning of February 2012. The clusters of PRMs were staggered between the two orbiters by 1-week intervals, beginning January 7, 2012 for GR-A, followed by January 13, 2012 for GR-B. During this time a collision avoidance strategy was implemented whereby the orbits between GR-A and GR-B were designed not to intersect. The self-imposed COLA requirement was set at 10 km. The reference trajectory updates for this phase implemented this strategy. The PRM burns were optimized to reduce gravity losses over three days in cluster 1 and four days in cluster 2. The 10-13 min long

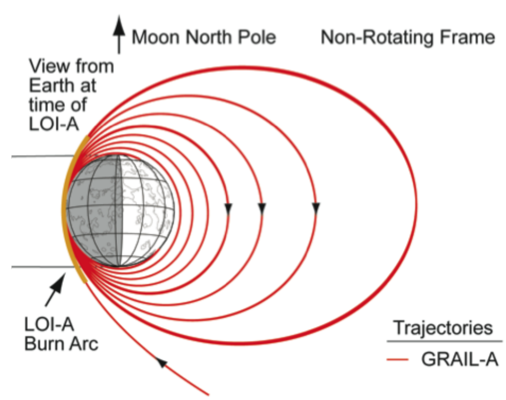


Figure 17. **The reduction of the orbit period during the OPR Phase**

Table 8. Comparison of Post-OPR-1 orbits to target†

Target Parameter	Target†	Reconstruction	Uncertainty (1σ)	Error	Predicted Error‡ (1σ)
GRAIL-A: Post-OPR-A1					
Inclination (deg)	89.490	89.498	0.0006	0.0086	0.013
Longitude of Ascending Node (deg)	115.548	115.548	0.0002	-0.0004	0.002
Arg. of Periapsis (deg)	356.326	356.113	0.0001	-0.2120	0.828
Period (sec)	13143.192	13151.280	0.0065	8.0900	11.200
Time to Periapsis (sec)	-6538.128	-6464.490	0.0009	73.6000	NA
Periapsis Range (km)	1836.191	1836.389	0.0018	0.1980	0.125
GRAIL-B: Post-OPR-B1					
Inclination (deg)	88.785	88.781	4.46E-06	-0.0043	0.006
Longitude of Ascending Node (deg)	115.496	115.496	2.10E-06	-0.0006	0.002
Arg. of Periapsis (deg)	357.465	357.373	0.0001	-0.0918	0.699
Period (sec)	13861.293	13866.823	0.0058	5.5303	10.100
Time to Periapsis (sec)	-6810.611	-6775.830	0.0017	34.7803	NA
Periapsis Range (km)	1846.794	1846.797	0.0009	0.0022	0.019

† Not all orbit elements were targeted; the OPR-1 cluster targets were semi-major axis or average period, argument of periapsis and inclination. The non-targeted parameters were the corresponding values on latest reference trajectories.
‡ Predicted errors only included the mapped dispersions of the maneuver errors; OD errors were not included due to the difficulty of including the gravity field covariance.

burns executed while inertially pointed primarily in the anti-velocity direction at periapsis on consecutive days. The ΔV of the first cluster burns were, respectively, 79.6 m/s and 74.5 m/s for GR-A and GR-B. The 2nd cluster included four burns of magnitudes equal to 69.5 m/s for GR-A and 70.7 m/s for GR-B. Total ΔVs to reduce the orbiters' periods from 11.5 hr down to just below 2 hr totaled 516 m/s for GR-A and 506 m/s for GR-B. To improve the efficiency of these burns, timing updates were planned to recenter the centroid of the consecutive maneuvers in a cluster after the orbit periods were redetermined from 6 hours of post-maneuver tracking. The minimum criteria for requiring a timing update was set to 10 sec. Timing updates for PRM-A2 and PRM-A3 were 33 sec and 75 sec, while timing updates for PRM-B2 and PRM-B3 were 11 sec and 34 sec. In the 2nd cluster, only PRM-A7, PRM-B6 and PRM-B7 required timing updates; these were, respectively, 21 sec, -19 sec and -25 sec.

The achieved post-OPR cluster-1 osculating orbit element parameters are compared to the targeted values in Table 8. The period error after the 3 PRM burns was 8 sec for GR-A and 5.5 sec for GR-B. Table 9 lists the post-OPR cluster-2 orbit element comparisons to the targeted values. After the four PRM burns, GR-A achieved a period error of 0.5 sec while GR-B achieved a period error of 0.1 sec.

Table 9. Comparison of Post-OPR-2 orbits to target†

Target Parameter	Target†	Reconstruction	Uncertainty (1σ)	Error	Predicted Error‡ (1σ)
GRAIL-A: Post-OPR-A2					
Inclination (deg)	88.938	88.924	5.4E-04	-0.0136	0.004
Longitude of Ascending Node (deg)	115.402	115.404	9.9E-05	0.0021	3.9E-04
Arg. of Periapsis (deg)	176.310	177.264	8.9E-04	0.9532	0.760
Period (sec)	6856.910	6857.398	6.6E-04	0.4886	0.832
Time to Periapsis (sec)	-3360.998	-3342.615	0.0022	18.3826	NA
Periapsis Range (km)	1768.750	1768.885	1.4E-04	0.1354	0.287
GRAIL-B: Post-OPR-B2					
Inclination (deg)	89.484	89.469	4.33E-03	-0.0158	0.007
Longitude of Ascending Node (deg)	115.659	115.661	1.81E-04	0.0015	4.7E-04
Arg. of Periapsis (deg)	177.653	179.811	0.0030	2.1572	1.980
Period (sec)	7024.455	7024.568	0.0020	0.1128	2.070
Time to Periapsis (sec)	-3381.698	-3359.576	0.0105	22.1215	NA
Periapsis Range (km)	1815.544	1815.580	0.0004	0.0363	0.728

† Not all orbit elements were targeted; the OPR-2 cluster targets were semi-major axis or average period, argument of periapsis and inclination. The non-targeted parameters were the corresponding values on latest reference trajectories.
‡ Predicted errors only included the mapped dispersions of the maneuver errors; OD errors were not included due to the difficulty of including the gravity field covariance.

5.12. Transition to Science Formation

Once the orbit periods for both orbiters had been successfully reduced to less than 2 hours, the mission entered the TSF Phase. During this nearly month-long phase, a series of maneuvers, referred to as Transition-to Science-Formation Maneuvers or TSMs, were performed strategically to achieve the Science formation. The TSF Phase strategy involved performing two deterministic TSMs on GR-A to establish the science orbit and three deterministic TSMs on GR-B to place it 85 km ahead of GR-A in the same orbit. The maneuver design strategy during the TSF Phase was to match the reference trajectories, which were updated (based on the final OD deliveries) by the Traj Team in the maneuver design process. Post-maneuver orbit parameter targets from these reference trajectories were then used by the MD Team in the maneuver search process. The GR-A TSM's were essentially independent of GR-B's orbit, whereas the GR-B maneuvers relied on the latest OD solutions of both orbiters.

The TSF strategy shown in Figure 18 included the following:

1. Each TSM targets a period so that the period difference between the two spacecraft orbits controls the rate of change of the phase difference, i.e., the approach rate.
2. TSM-A maneuvers establish the science orbit
   (a) TSM-A1 fixes the orbit inclination and sets the orbit period to be 29 sec greater than the designed science orbit period.
   (b) TSM-A2 initializes the semi-major axis, eccentricity, and argument of periapsis for the science orbit
3. TSM-B maneuver times are adjusted to control the final relative phase of each spacecraft, i.e., relative time and locations of periapse, measured after TSM-B1 by the amount of time GR-B is ahead of GR-A
4. Uncertainties in initial phasing of the spacecraft and in time-ahead after the TSM-A1, B1 maneuvers are dealt with by adjusting the number of orbits between maneuvers
   (a) Between TSM-A1 and TSM-B1, the orbit period of GR-B is 3 min greater so that GR-A laps GR-B in three days; execute TSM-B1 to match periods when GR-B is 30-33 min ahead
   (b) TSM-A2 reduces period of GRL-A by 29sec, so execute TSM-B2 when GRL-B is 6.6–7.1 min ahead, leaving GR-B in 5.6sec to 6.1sec longer orbit
   (c) Execute TSM-B3 when GRL-B is 41–49sec (69 to 81km) ahead
   (d) Target TSM-B3 to a separation of 100.5km at March 8 18:00 UTC

5.12.1. TSM-A1

The TSM-A1 maneuver was designed to ensure the COLA requirement was met by keeping GR-A's orbit eccentricity vector (eccentricity and argument of periapsis) close to that of GR-B's orbit. It also was designed to reduce GR-A's orbit period, so that GR-A would lap GR-B every 3 days. The targets were inclination, average period, argument of periapsis and eccentricity. The burn executed on Feb 7, 2012 16:32 UTC with a ΔV magnitude of 8.2 m/s. The maneuver under performed by 4 mm/s (0.5σ) with a fairly large pointing error of 8 mrad ($> 3\sigma$). The pointing error could be attributed to the shift of the center of gravity (CG) due to fuel movement during the burn. The second integrator feed-forward torque gains in the thrust vector control system onboard and the ME alignment vector which were updated for the burn were suspect, but it was found that the pointing would have been even worse without these updates.

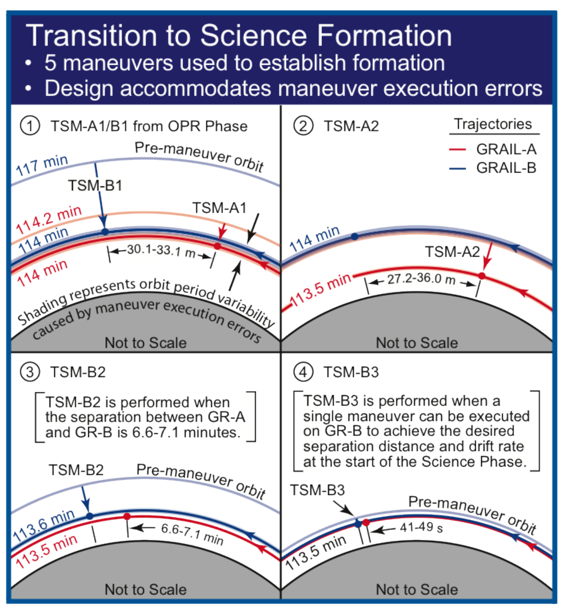


Figure 18. The TSF strategy

5.12.2. TSM-B1

The DCO for TSM-B1 was six hours after the TSM-A1 maneuver on Feb 7th. The TSM-B1 maneuver was designed to begin the process for getting into formation by matching the orbit period of GR-A and placing GR-B approximately 33 min ahead of GR-A. It also targeted the eccentricity vector such that the downstream TSM-B3 would be less than 5 m/s. It was desired to keep the TSM-B3 maneuver small so that there was a good chance of not needing a statistical maneuver afterwards to clean up the orbit errors. The TSM-B1 targets included matching the post-TSM-A1 reference semi-major axis, eccentricity, inclination, and argument of periapsis. In addition to the large downtrack component, the burn also had a significant radial component. The burn executed nominally on Feb 14, 2012 03:15 UTC with a magnitude of 17.5 m/s.

5.12.3. TSM-A2

The TSM-A2 maneuver was designed to finalize the Science orbit by targeting the average period, periapsis altitude, and argument of periapsis to achieve 82-day Science Phase without a maneuver. The burn took place on Feb 20, 2012 15:05 UTC with ΔV magnitude of 19.3 m/s. The maneuver performed nominally. TSM-A2 was a mostly radial burn. It used the TSM-A2 reference trajectory targets of the orbit inclination, average period, eccentricity, and argument of periapsis. This burn placed the eccentricity vector closer to that of GR-B's and it shortened GR-A's period by approximately 30 sec, so now GR-A was approaching GR-B. After the burn, the apoapsis and periapsis altitudes of the orbit were 90 km and 20 km, respectively.

5.12.4. TSM-B2

The TSM-B2 maneuver was performed four days after TSM-A2 on Feb 24 at 17:24 UTC when GR-B was ahead of GR-A by approximately 7 minutes. The maneuver was designed to reduce the drift rate between two orbiters and match GR-B's orbit plane to GR-A's. In order to achieve this, the maneuver targeted to the inclination, average period, argument of periapsis, and longitude of ascending node from the TSM-B2 reference trajectory. The burn magnitude was 2.1 m/s and consisted mostly of downtrack and crosstrack components. Like the TSM-A1 maneuver, this burn had a larger than expected pointing error. Both GR-A and GR-B's OD solutions were needed in the reference update process. In order to account for the execution errors of TSM-A2, the GR-A OD solution used a DCO approximately 6 hrs after the TSM-A2 burn executed.

In the event of a missed TSM-B2 maneuver, the GR-A Team designed a contingency orbit pop-up maneuver, TSM-A3CB2, concurrently with the TSM-B2 design. This 2.2 m/s maneuver was designed to arrest GR-A's drift toward GR-B. The burn time was scheduled approximately 5 hrs later than the TSM-B2 time, so that GR-A would not lap GR-B. If this scenario played out, the Contingency Playbook had a plan to bring the two orbiters back into formation. This plan would require two more maneuvers, one on GR-A and one on GR-B, to complete the Science formation.

5.12.5. TSM-B3

Finally on Feb 29, 2012 13:46 UTC, the TSM-B3 maneuver executed with a ΔV of 0.5 m/s to bring the orbiters into the final Science formation. The burn was designed to reverse the drift rate between the orbiters by a small amount, and involved a slight change in strategy from the mission design strategy shown in Figure 18. The TSM-B3 maneuver was performed when GR-B was ahead of GR-A by approximately 75 km (47 sec) just as originally planned, but was sized to adjust the drift rate between orbiters such that the separation at the time of the OTM-B2 maneuver on March 30, 18:00 UTC was 225 km. This revised strategy offered the advantage of nominally eliminating the need of the OTM-B1 maneuver planned on March 7 to adjust the drift rate for the Science Phase. The TSM-B3 maneuver was also designed to match the eccentricity vectors between GR-A and GR-B so they also would survive the 82-day Science Phase without a maneuver. The maneuver targeted the

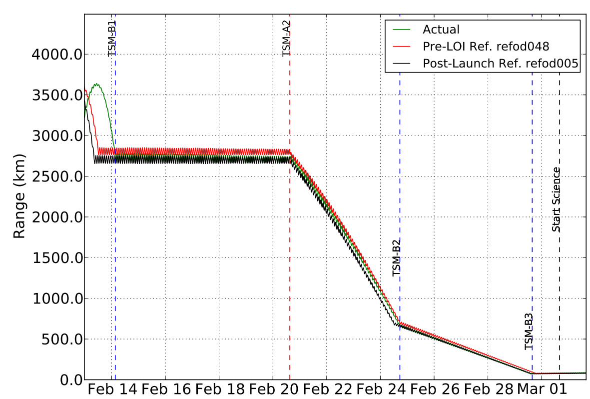


Figure 19. **Comparison of orbiter-to-orbiter separation during the TSM-B1 through TSM-B3 maneuvers**

Table 10. Comparison of Post-TSF orbits to target†

Target Parameter	Target†	Reconstruction	Uncertainty (1σ)	Error	Predicted Error‡ (1σ)
GRAIL-A: Post-TSM-A2					
Inclination (deg)	89.625	89.618	2.1E-03	-0.0073	2.6E-04
Longitude of Ascending Node (deg)	115.349	115.351	2.1E-04	0.0015	6.0E-04
Arg. of Periapsis (deg)	184.436	184.446	6.1E-04	0.0107	0.097
Period (sec)	6813.695	6813.796	1.2E-03	0.1003	0.414
Time to Periapsis (sec)	-3277.005	-3277.053	3.5E-03	-0.0480	NA
Periapsis Range (km)	1758.349	1758.372	2.5E-04	0.0228	0.092
GRAIL-B: Post-TSM-B3					
Inclination (deg)	88.948	88.944	4.5E-04	-0.0037	1.8E-06
Longitude of Ascending Node (deg)	115.142	115.142	8.1E-06	0.0003	2.8E-05
Arg. of Periapsis (deg)	176.750	176.685	6.8E-04	-0.0650	0.010
Period (sec)	6814.806	6814.871	4.7E-04	0.0648	0.029
Time to Periapsis (sec)	-3283.069	-3285.018	6.8E-03	-1.9496	NA
Periapsis Range (km)	1763.304	1763.347	2.1E-04	0.0433	0.005

† Not all orbit elements were targeted with these last TSF maneuvers; the non-targeted parameters were the corresponding values on latest reference trajectories.
‡ Predicted errors only included the mapped dispersions of the maneuver errors; OD errors were not included due to the difficulty of including the gravity field errors.

semi-major axis, eccentricity, argument of periapsis, and longitude of ascending node from the TSM-B3 reference trajectory. The maneuver performed with a fairly large under-burn magnitude error of 2.6 σ. Despite this error, the Science formation was successfully achieved. Table 10 compares the post-TSM-A2 and TSM-B3 orbit elements against the target values. These errors were evaluated at the apoapsis following the burn. The TSM-B1, TSM-A2 through TSM-B3 maneuver sequence was considered to be the most challenging to the navigation processes, but the actual results compared very well to the pre-launch and post-launch references in Figure 19.

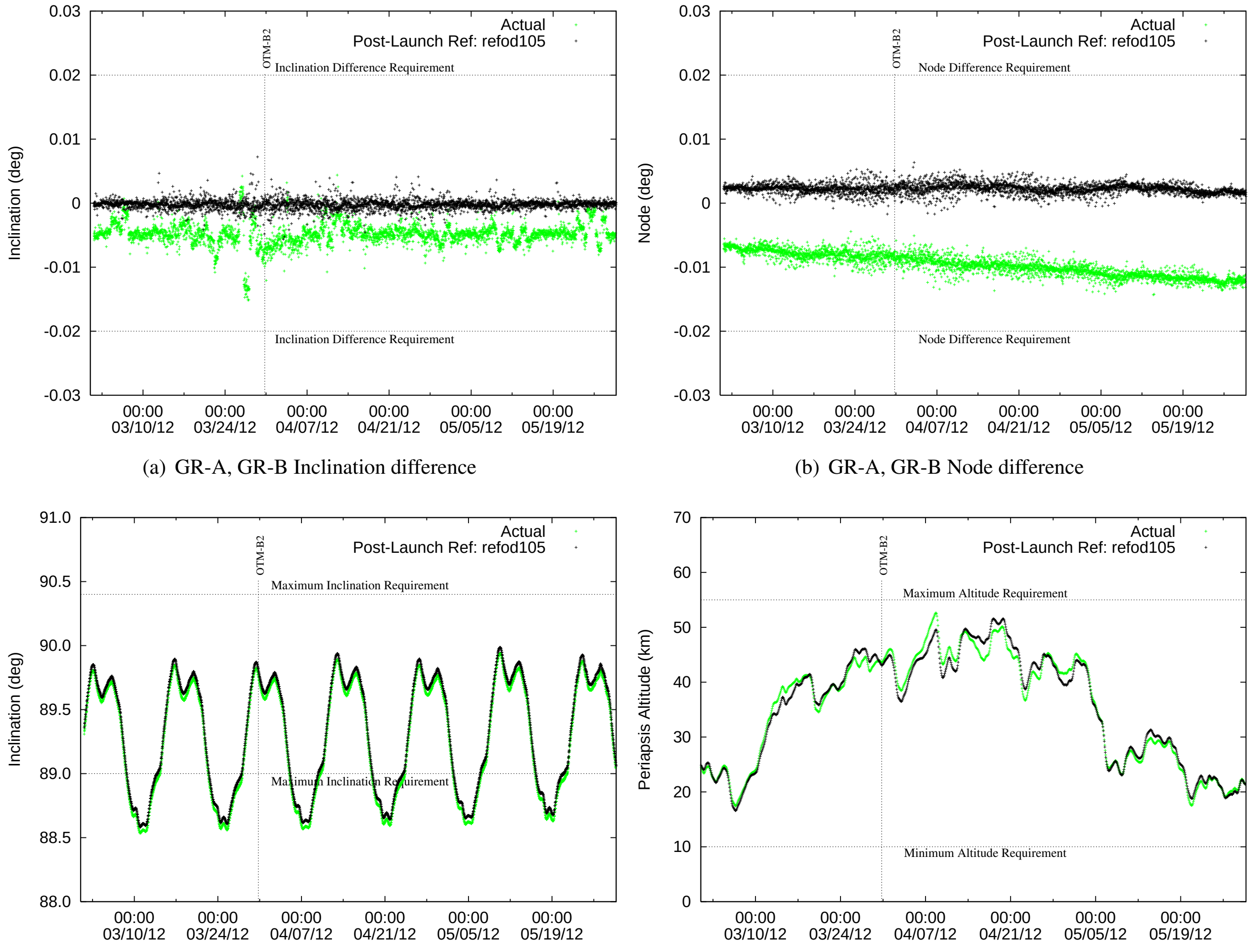
In case of a TSM-B3 failure, the GR-B orbit would now be the Science orbit and GR-A's orbit would need to match it. The contingency maneuver, TSM-A3CB3, was designed to set GR-A's drift relative to GR-B's orbit period such that it would reach 225 km at the time of OTM-B2. This maneuver was designed concurrently to the TSM-B3 design. It was designed to match GR-A's eccentricity vector to that of GR-B's. It was scheduled approximately 5.7 hours later than TSM-B3. In this scenario, no additional deterministic maneuvers would be required to form the Science formation.

5.13. Science Phase

On March 1, the orbiters were oriented into the orbiter-to-orbiter active pointing attitude to align the Ka-band antenna horns towards each other to collect data with the solar arrays aligned within the orbit plane. The original plan was to periodically place the orbiters into this orientation for a few hours at a time through March 8 to check out the LGRS system, perform Ka-boresight calibrations, then return to the nominal 40° off-Sun point attitude. However, early in operations, the SCT determined that the energy collected from the solar arrays at the solar beta angle of 43° would be adequate with enough margin to remain in this configuration through the original start of Science on March 8th.

A statistical ACS maneuver, OTM-B1, was planned before the beginning of the Science Phase on March 7th, 2012 to adjust the separation drift rate errors from the post-TSM-B3 orbits. The separation profile following OTM-B1 would reach 225 km on March 30, 2012 where another ACS maneuver, OTM-B2 reversed the separation rate. The design of the OTM-B1 maneuver began on March 1, one day following the TSM-B3 burn. The orbiter-to-orbiter separation determined from the post-TSM-B3 orbits showed it to be close to the distance between the reference trajectories, and thus the OTM-B1 maneuver was canceled.

The OTM-B2 maneuver was designed using the DCO of the GR-A and GR-B orbits on March 26, 2012. The process of designing the OTM was different than the other maneuvers, since the orbiter remained in the orbiter-to-orbiter pointing attitude at the start of the burn. This maneuver had to use this starting attitude with either the + or − Z-axis ACS thrusters in the design. OTM-B2 was designed as a pitch-over maneuver with the pitch rate about the X-axis equal to the mean motion of the orbit. This would keep the Ka-horns of the two orbiters pointed at each other during the burn,



(a) GR-A, GR-B Inclination difference

(b) GR-A, GR-B Node difference

(c) Maximum Inclination

(d) Periapsis Altitude

Figure 20. **Comparing actual to post-launch reference, inclination, node differences, maximum inclination and periapsis altitudes against science requirements**

so that the LGRS link could be maintained. The maneuver was scheduled over the North lunar pole where redundant LGRS data had been collected, so that it would not disrupt the Science gathering. The maneuver adjusted the period of GR-B by a small amount to target the separation at the end of the Science Phase to 65 km on May 29th. The designed burn had a ΔV magnitude of 28.2 mm/s and would be performed on the $+Z$-axis ACS thrusters. The burn error was about 1 mm/s under the design. The pointing error for this burn was rather large at 63 mrad, but this error did not contribute to significant orbit errors.

Figures 20a–b compare the achieved GR-A, GR-B Science orbit inclination and node differences to the reference and the Science co-planar orbit requirements. which include the orbit co-planar requirements of 0.02° in inclination and node. Figure 20c compares the achieved GR-A, GR-B Science orbit inclination to the reference and the Science maximum inclination requirements. The achieved GR-A and B Science orbit periapsis altitudes are compared to the reference and the Science requirements in Figure 20d. The Science requirements for these parameters were met. Figure 21 shows how the separation profile achieved during the Science Phase compared to the reference design. It also shows that the separation profile met the Science requirements. Notice the offset from the reference was acceptable to the Science Team. The Nav Team was prepared to correct the separation profile with DeltaV Correction Maneuvers (DCMs), which would be designed like an OTM over the North or South lunar pole to adjust the period of GR-B relative to GR-A, but these were never needed. Figure 22 compares the evolution of the eccentricity vector over the Science Phase to the LOI Pass-1 reference orbits (refod017v1). Finally, the osculating Science Phase orbits of both orbiters are displayed in Figure 23.

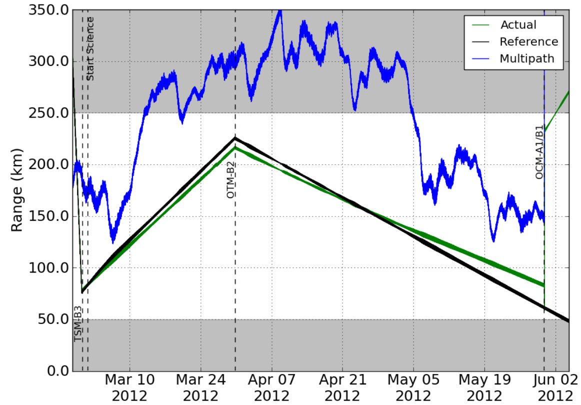


Figure 21. **Comparison of actual separation (green) to the reference design (black) during Science Phase**

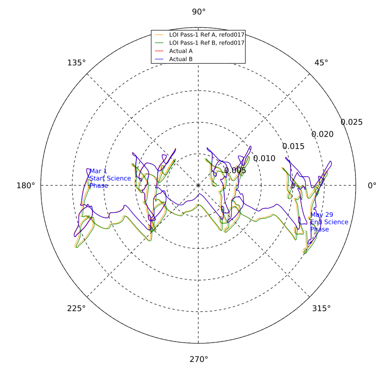


Figure 22. **Comparing GR-A and GR-B eccentricity vector (e Vs ω) progression during Science Phase to LOI Pass-1 reference**

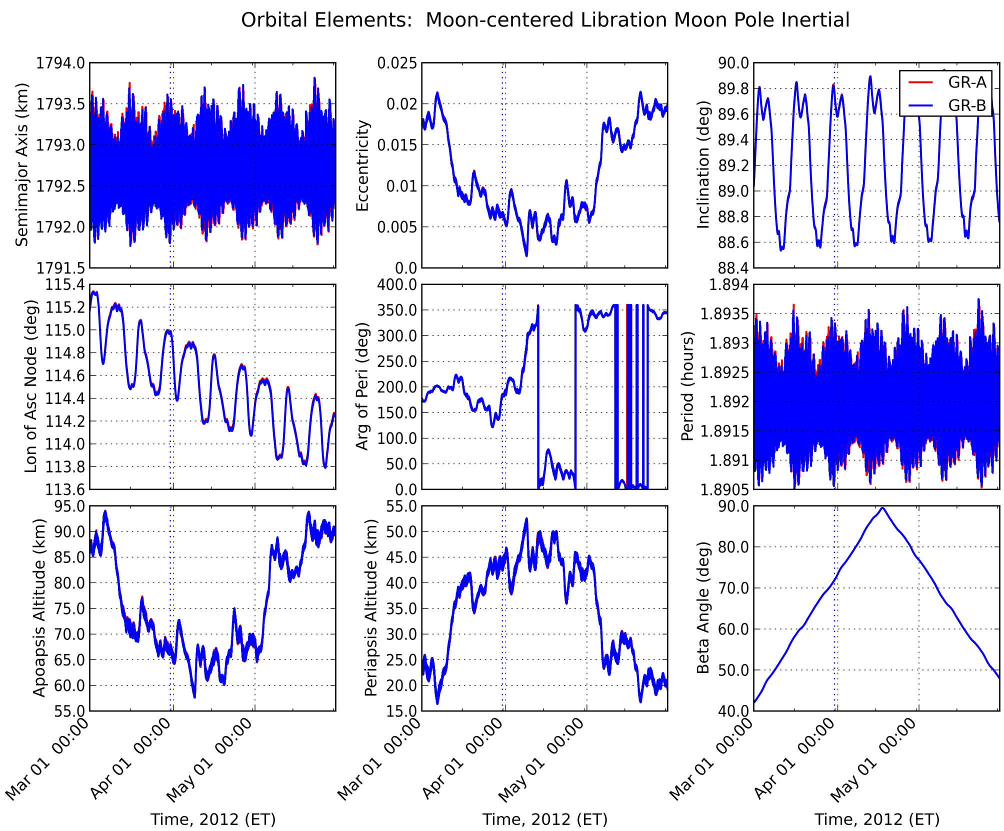


Figure 23. GR-A (red) and GR-B (blue) orbital elements during the Science Phase

5.14. Navigation Performance

Tables 4 and 5 compare the actual maneuver ΔV designs against the pre-launch expected TLC $\Delta V99\%$ and reference lunar orbit ΔV costs, respectively, for GR-A and GR-B. The actual TLC ΔV costs were within the 14th percentile of the expected performance for GR-A and 23rd percentile for GR-B. The maneuver errors of the TCM maneuvers are summarized in Table 6. The TCM maneuvers consumed 2.9 kg of propellant from GR-A and 4.6 kg from GR-B. The LOI maneuvers burned approximately 24 kg each. The first OPR clusters expended 28 kg of fuel for GR-A and 26 kg for GR-B. The second OPR clusters expended approximately 29 kg for both orbiters. The TSF maneuvers consumed 2.7 kg on GR-A and 1.9 kg on GR-B.

The total GR-A ΔV cost for the lunar phase was 734.2 m/s, which was 1.6 m/s less than the pre-launch reference. The GR-B lunar phase ΔV cost of 717.3 m/s was also less than the pre-launch design (-2.6 m/s). The total mission cost for GR-A and GR-B left a ΔV reserve of 183 m/s onboard both orbiters for the XM (approximately 19.9 kg for GR-A and 19.7 kg for GR-B) .

6. Conclusions

The twin GRAIL spacecraft were successfully flown from launch to the Moon using only 3 out of the 5 planned TCMs. Then after successful LOI maneuvers, the orbiters were navigated through the OPR and TSF phases to bring the orbiters into precise Science formation. This configuration met all Science requirements. The orbiter-to-orbiter pointing errors were kept small with frequent ephemeris uploads. This frequency was reduced from twice per week to once per week after a new gravity field using the LGRS data was delivered by the Science Team. After that initial gravity field delivery, the Science Team has produced subsequent fields, each with increasing fidelity. Navigation of the GRAIL spacecraft, now in the XM, continues to be successful and should enable the collection of high-quality science data and completion of the science objectives.

Acknowledgments

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