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Mission Design Overview for the Gravity Recovery and Interior Laboratory (GRAIL) Mission

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The Gravity Recovery and Interior Laboratory (GRAIL) Mission – a NASA Discovery Program mission currently in development and scheduled to launch in September 2011 – will make a high resolution determination of the lunar gravity field in an effort to understand the internal structure and thermal evolution of the Moon "from crust to core" and will extend that knowledge to other terrestrial planets within the inner solar system. This will be accomplished by placing two nearly identical orbiters, flying in tandem, in a low-altitude, near-polar orbit around the Moon. Precise measurements of the relative velocity between the two orbiters combined with measurements of the absolute position of the orbiters about the Moon as determined via Earth-based tracking will allow the global lunar gravity field to be mapped to unprecedented accuracy and resolution. The development of the baseline strategy to achieve the objectives of this mission involves the integration of a variety of design elements into a coherent mission plan, including: trans-lunar trajectory and navigation design, in-flight calibration and checkout planning, lunar orbit insertion design, design of the maneuvers for orbit circularization and for the establishment of the formation necessary for gravity science data collection, and gravity science data collection design. These mission design elements are described in chronological order as the mission progresses through its seven mission phases: Launch, Trans-Lunar Cruise, Lunar Orbit Insertion, Orbit Period Reduction, Transition to Science Formation, Science, and Decommissioning.

I. Introduction

On December 10th, 2007 NASA announced the selection of the 11th Discovery Program mission. The winner of the 2-year competition was the Gravity Recovery and Interior Laboratory mission – or GRAIL for short. The objective of the GRAIL mission is to fly twin spacecraft in low-altitude orbits about the Moon to determine the structure of the lunar interior "from crust to core" and to advance the understanding of the thermal evolution of the Moon. To first order, the GRAIL mission can be thought of as the lunar analog of the highly successful NASA Earth System Science Pathfinder (ESSP) Gravity Recovery and Climate Experiment (GRACE) gravity-mapping mission. The measurement approach, instrument design, and science data analysis are all based upon the GRACE mission. Applying the techniques pioneered by the GRACE mission, the GRAIL mission will map the lunar gravity field to unprecedented accuracy and resolution – with expected improvements in the lunar farside gravity of greater than two orders of magnitude over existing lunar gravity fields.

The GRAIL mission is scheduled to launch in the fall of 2011 with the entire mission lasting only approximately nine months. The first six months are used to deliver the spacecraft to the Moon and maneuver them into their required orbits. The final three months are dedicated to the collection of gravity science data.

The Principal Investigator (PI) for the mission is Dr. Maria Zuber of the Massachusetts Institute of Technology. The mission is managed by the Jet Propulsion Laboratory (JPL) with the spacecraft developed by the Lockheed Martin Space Systems Company (LMSSC).

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II. Mission Overview

A. Science Measurement

The principle behind the science measurement is that the gravity field of the Moon influences the motion of the center-of-mass of each spacecraft in orbit about the Moon. Any surface feature (such as craters or mountains), subsurface mass anomalies (mass concentrations or mascons), or deep interior core motion perturbs the spacecraft orbits and introduces signatures in the relative motion between the spacecraft. By precisely measuring the change in distance between the spacecraft and combining that information with Earth-based tracking measurements of the absolute position of the spacecraft in orbit about the Moon, a high resolution gravity field can be developed.

The interspacecraft range rate is measured by a Ka-band Lunar Gravity Ranging System (LGRS) on board each spacecraft. The system provides continuous data measurement and is not dependent on line-of-sight visibility between the spacecraft and tracking stations on the Earth. Unlike the GRACE mission, which relied on the Global Positioning System (GPS) about the Earth for absolute orbit position and timing information, the GRAIL mission will replace these functions with X-band Doppler tracking from Earth-based tracking stations and an interspacecraft S-band Time Transfer System (TTS). An onboard Ultra-Stable Oscillator (USO) provides a stable reference frequency for the one-way X-band Doppler data. The X-band data are used for precision orbit determination and calibration of the payload clock. A separate S-band system is used for spacecraft telemetry, commanding, and navigation. The communications and measurement paths for the GRAIL mission are depicted in Fig. 1.

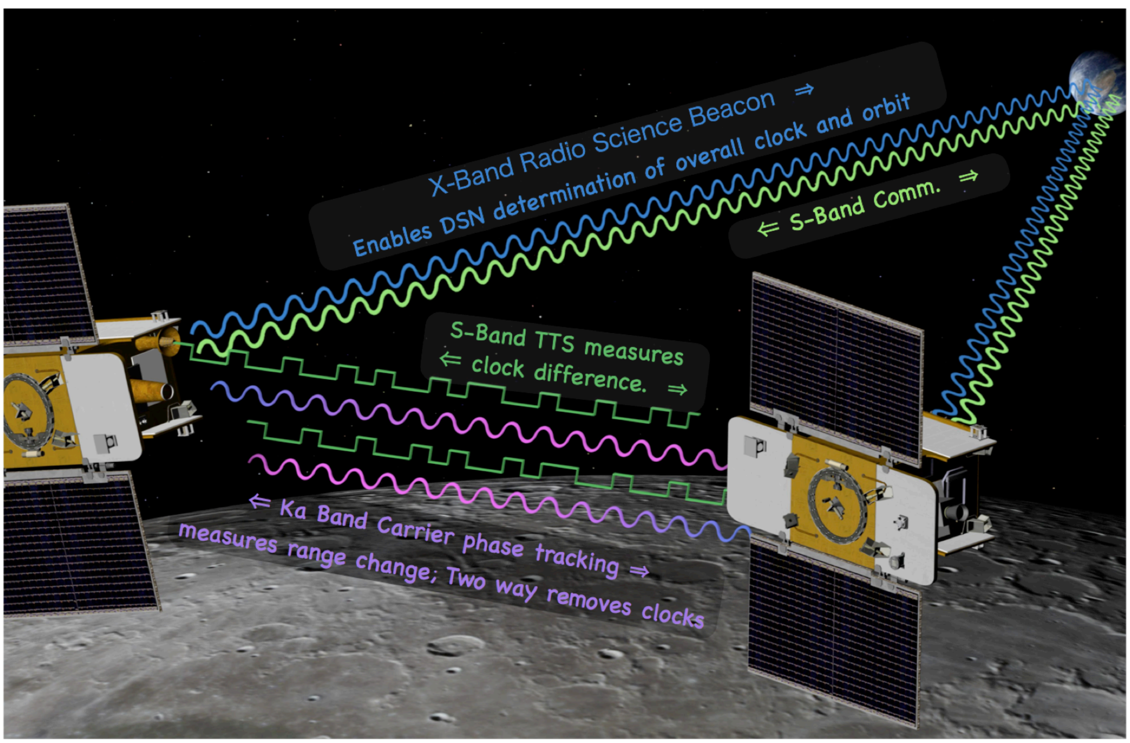

Figure 1. GRAIL Gravity Measurement

B. Science Objectives and Mission Success

The GRAIL mission has two primary science objectives:

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

A secondary science objective is

- To extend knowledge gained on the internal structure and thermal evolution of the Moon to other terrestrial planets.

The GRAIL science objectives lead to 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

The first four science investigations are defined by global, regional, and local scale gravity field measurement accuracies. Investigations five and six are defined in terms of the accuracy of gravity potential coefficients rather than physical scale.

Based on the six science investigations, NASA has levied a set of performance requirements on the GRAIL mission which constitute the requirements for achieving baseline and minimum mission success. These baseline and minimum success criteria are summarized in Table 1. Achievement of all six requirements will constitute baseline mission success, whereas achievement of the first four will constitute minimum mission success.

Table 1. Mission Success Criteria

Investigation	Baseline Mission Success	Minimum Mission Success
1	**Crust and Lithosphere** Determine the Moon's global gravity field with global average surface resolution of 30 km and with an accuracy of ±10 mGal.	Same as Baseline
2	**Thermal Evolution** Determine large regional gravity with global average surface resolution of 30 km and with an accuracy of ±2 mGal.	Same as Baseline
3	**Impact Basins** Determine small regional gravity that resolves basins and rings to a global average surface resolution of 30 km to a precision of ±0.5 mGal.	Same as Baseline
4	**Crustal Brecciation and Magmatism** Determine high-resolution local gravity fields with global average surface resolution of 30 km to a precision of ±0.1 mGal.	Same as Baseline
5	**Deep Interior** To constrain the deep interior from tides, determine the Love Number, k_2, to an accuracy of 6×10^{-4}.	N/A
6	**Core Detection** To place limits on the size of a possible solid inner core, determine the Love Number, k_2, to an accuracy of 2×10^{-4} and determine the second-degree and first-order gravity coefficients to an accuracy of 1×10^{-10}.	N/A

C. Mission Summary

The primary systems of the GRAIL project consist of twin spacecraft with nearly identical payloads, a single launch vehicle, and the ground systems to conduct mission operations. The two orbiters, referred to as GRAIL-A and GRAIL-B, will be launched on a Delta II 7920H launch vehicle from Space Launch Complex 17B (SLC-17B) at the Cape Canaveral Air Force Station in Florida. Launch will occur during a 26-day launch period extending from September 8 through October 3, 2011. Arrival at the Moon occurs on a fixed date for both orbiters and is independent of the launch date. The GRAIL-A orbiter arrives at the Moon on December 31, 2011 and the GRAIL-B orbiter arrives one day later, on New Year's Day, January 1, 2012.

The GRAIL mission is divided into seven mission phases as illustrated in Fig. 2. The figure shows that the mission was designed to avoid the lunar eclipses occurring on December 10, 2011 and June 4, 2012. Being in orbit about the Moon during a lunar eclipse would disrupt normal operations and might not be survivable (depending on the length or severity of the lunar eclipse).

The orbiters will leave the Earth following a low-energy trajectory headed towards the Sun and passing near the interior Sun–Earth Lagrange Point (EL1). The low-energy transfer to the Moon is beneficial to the GRAIL mission for a number of reasons: it reduces the Lunar Orbit Insertion (LOI) ΔV requirements, allows time for orbiter checkout and outgassing, permits the onboard Ultra-Stable Oscillator to be continuously powered for an extended period of time allowing it to reach a stable operating temperature, and permits the use of an extended launch period (as compared to a direct, Apollo-like trajectory to the Moon).

During the Trans-Lunar Cruise (TLC) Phase, up to five Trajectory Correction Maneuvers (TCMs) will be performed. The second and third TCMs are primarily deterministic maneuvers designed to separate the GRAIL-A and GRAIL-B arrival times at the Moon by approximately one day and insert them onto the desired low-energy trajectories. The remaining maneuvers are statistical in nature and are included to remove trajectory errors and target the final LOI conditions at the Moon.

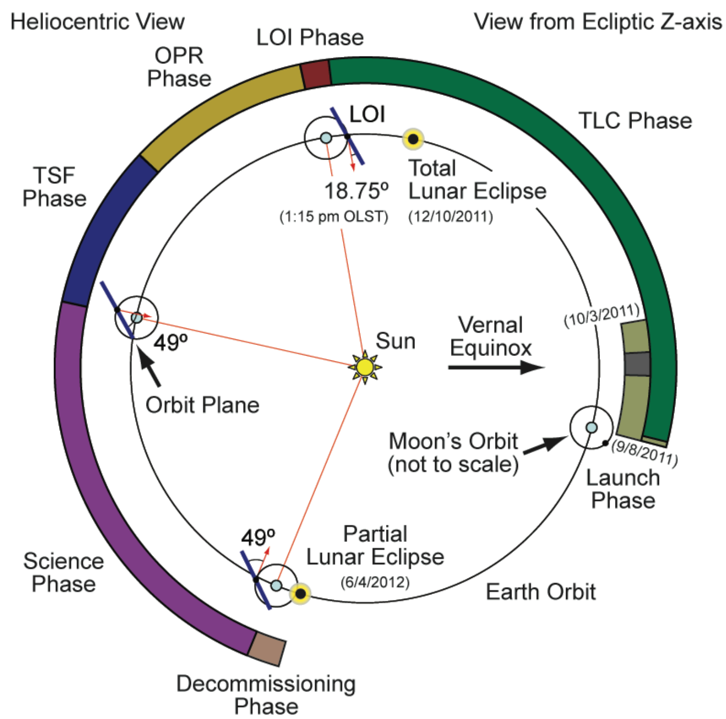

Figure 2. Heliocentric View of GRAIL Mission

At the conclusion of the TLC Phase, both orbiters will propulsively insert into lunar orbit. The LOI maneuvers will be separated by about 25 hours and will be simultaneously visible from DSN tracking stations in Madrid, Spain and Goldstone, California in order to provide dual complex coverage of these mission critical events. The orbiters approach the Moon over the lunar South Pole and are placed into elliptical orbits each with a targeted period of 11.5 hours.

Once in lunar orbit the mission will enter the Orbit Period Reduction (OPR) Phase. During this mission phase, a series of seven maneuvers will be performed by each orbiter to reduce the orbit period. These maneuvers are referred to as Period Reduction Maneuvers (PRMs) and are grouped into two clusters (three maneuvers in the first cluster, four maneuvers in the second cluster). Each of the maneuvers within a cluster is performed in the same inertial direction and with the same ΔV. This strategy increases mission robustness to a missed maneuver, simplifies operations by minimizing the number of separate maneuver designs, and reduces gravity losses. Each cluster is about a week long, with the clusters alternating between GRAIL-A and GRAIL-B. About five weeks after LOI, the orbit period on each orbiter will have been reduced to just under 2 hours.

From launch through the end of the OPR Phase, the two orbiters will be operated essentially independently. Activities on GRAIL-A and GRAIL-B will be separated in time in order to reduce operations conflicts and competition for ground resources, but no attempt will be made to fly the two orbiters in a coordinated manner. The purpose of the next mission phase, the Transition to Science Formation (TSF) Phase, is to establish the formation necessary for the collection of gravity science data and to check out the science payload prior to the start of the Science Phase.

During the TSF Phase, a series of rendezvous-like maneuvers will be performed in order to achieve the desired initial separation distance and ensure that the GRAIL-B orbiter is ahead of the GRAIL-A orbiter in the formation. Since the configurations of the two GRAIL orbiters are slightly different, the order in which they orbit the Moon is important. After a period of approximately three and a half weeks, with the execution of increasingly fine adjustments of the orbital conditions, the orbiters will be ready to test and calibrate the Ka-band science payload to ensure that the science payload on the two orbiters operate as a single instrument. Following a calibration period that lasts approximately a week, the collection of science data will begin.

At the start of the Science Phase, the GRAIL spacecraft will be in a near-polar, near-circular orbit with a mean altitude of 55 km. The initial conditions have been chosen to use the natural perturbations of the lunar gravity field to allow the orbit to evolve without requiring any orbit maintenance maneuvers. During the 82-day Science Phase, the Moon will rotate three times underneath the GRAIL orbit. The collection of gravity data over one complete rotation (27.3 days) is referred to as a Mapping Cycle. During Mapping Cycle 1, the mean separation distance is designed to increase from approximately 85 km to 225 km. A very small Orbit Trim Maneuver (OTM) executed near the end of Mapping Cycle 1 will then be used to change the separation drift rate. Following this OTM the mean separation distance will decrease from 225 km to approximately 65 km at the end of Mapping Cycle 3 – i.e. at the end of the Science Phase. The data collected at the lower separation distances help to determine the local gravity field, while the data collected at higher separation distances provide a greater sensitivity to large-scale features on the Moon.

The payload on each orbiter consists of the Lunar Gravity Ranging System and one Education and Public Outreach (E/PO) MoonKAM System. The LGRS will generate, transmit, and receive both Ka-band and S-band signals. These signals will be used to precisely measure the range rate between the two orbiters, as well as transfer timing information between the orbiters. The LGRS will also provide a one-way X-band link to the ground for precision orbit determination and to provide insight into performance of the USO. The MoonKAM consists of a single digital video recording unit with four camera heads – one slightly forward-pointed, two-nadir pointed, and one slightly backward pointed – to capture images and video of the lunar surface. Students at the University of California at San Diego will operate the cameras. During the Science Phase, operations will be conducted in a non-time-critical, ground-interactive mode. Data will be stored onboard and downlinked to Earth via S-band links as bandwidth becomes available. Each orbiter will receive an average of 12 hours of tracking per day by a 34m Deep Space Network (DSN) tracking station during the Science Phase. During a pass, real-time data will be transmitted along with recorded spacecraft telemetry and science data. The combined tracking passes for GRAIL-A and GRAIL-B will allow for the near-continuous absolute position determination of the orbiters.

At the conclusion of the Science Phase, both orbiters will transition to the Decommissioning Phase. During this short, 7-day mission phase, the orbiters will perform a final Ka-band science payload calibration, and will continue to acquire Science data as the power and thermal system allow prior to disposal on the lunar surface. No orbit maintenance or site impact targeting maneuvers are planned, nor are any orbiter passivation activities required. The orbiters will impact the lunar surface near the time of the partial lunar eclipse on June 4, 2012. A high-level overview timeline for the GRAIL mission is given in Fig. 3.

III. Flight System Description

The Flight System encompasses all of the components delivered by the GRAIL project that will fly into space. This includes both orbiters (where orbiter = spacecraft + payload) and a Launch Vehicle Adapter Assembly (LVAA). The two GRAIL orbiters share the same subsystem components and designs. Each orbiter includes a spacecraft derived from the Lockheed Martin Experimental Small Satellite (XSS-11) technology demonstration mission for the United States Air Force and the Mars Reconnaissance Orbiter (MRO) mission for NASA. Figure 4 shows the overall orbiter configuration.

The GRAIL orbiters are nearly identical, but there are a few important differences between them due to the need to point the spacecraft at one another during the Science Phase of the mission. These differences include the star tracker cant angle, the LGRS antenna cant angle, and the MoonKAM mounting. In addition, during the Science Phase, the orientation of the orbiters is different. In order to point the Ka-band antenna on one orbiter at the other, the GRAIL-A orbiter will fly with its -Y axis pointed towards the lunar surface, while the GRAIL-B orbiter will orient its +Y axis towards the surface.

The orbiters will be launched side-by-side on a Delta II 7920H launch vehicle housed in a 10 ft (3 m) composite payload fairing. Figure 5 shows the orbiters in the launch configuration. The LVAA includes a Flat Plate Adapter, and a SoftRide isolation system and Lightband separation system for each orbiter. The LVAA will remain with the second stage of the launch vehicle after separation of the orbiters.

The following sections provide a brief functional description of the major spacecraft and payload subsystems.

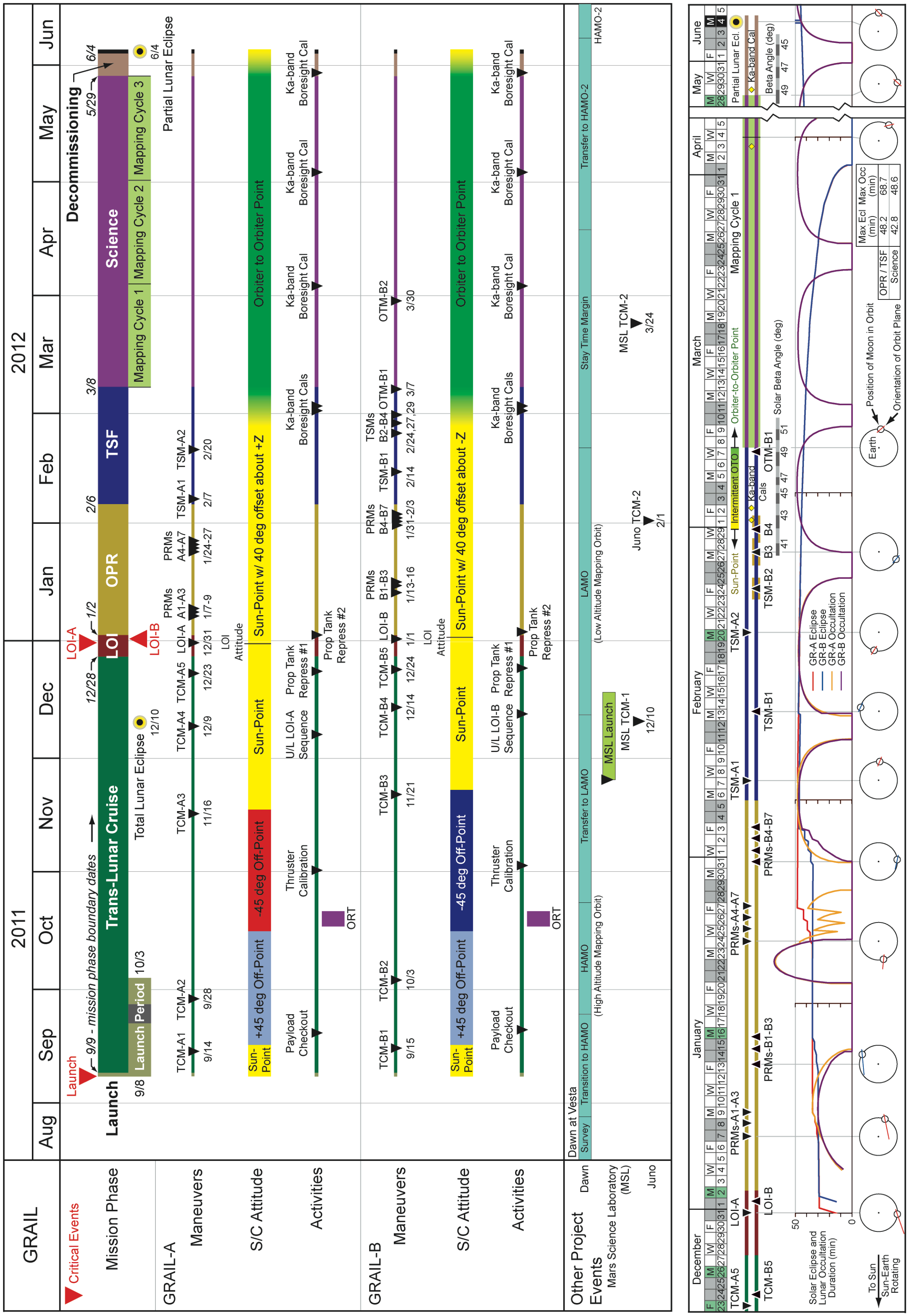

Figure 3. GRAIL Mission Timeline

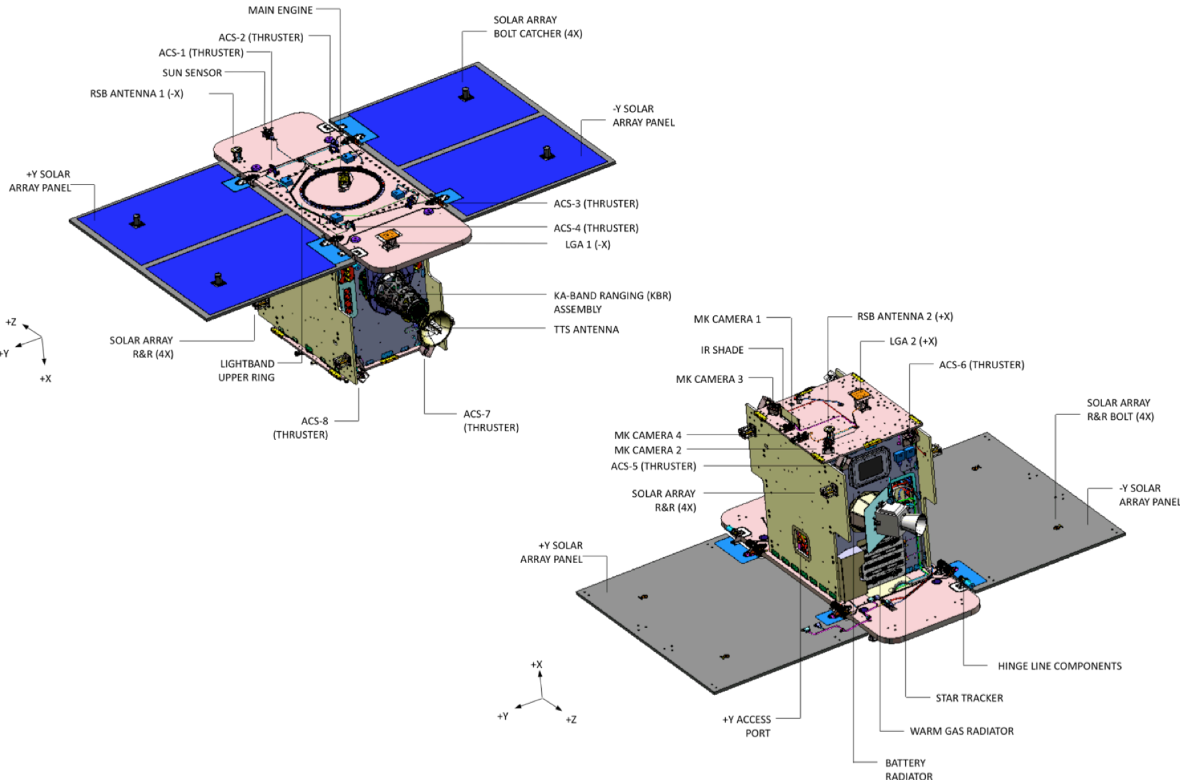

Figure 4. GRAIL Orbiter Configuration

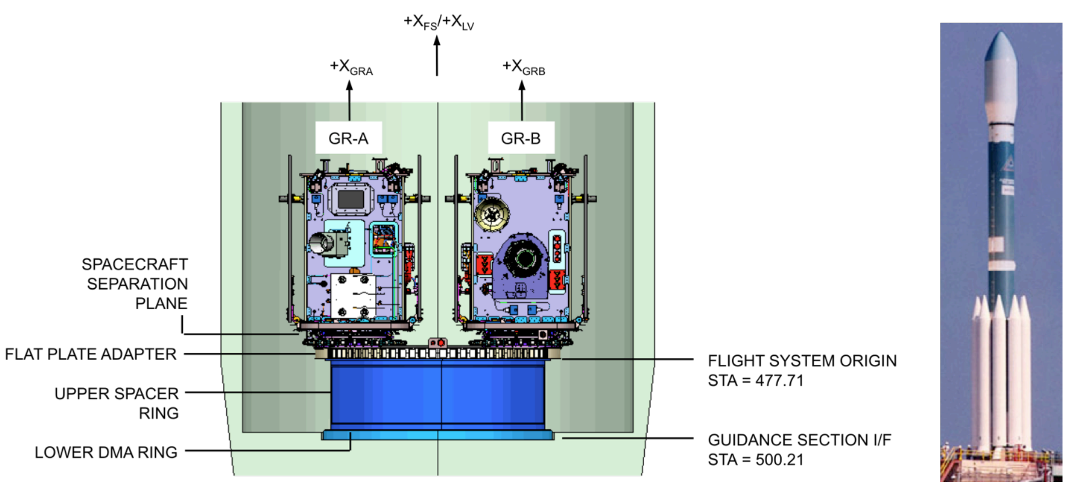

Figure 5. GRAIL Launch Configuration

A. Spacecraft

1. Attitude Control Subsystem (ACS)

The spacecraft Attitude Control Subsystem is used to provide three-axis stabilized control throughout the mission. The subsystem consists of a sun sensor, star tracker, reaction wheels, and inertial measurement unit. The onboard flight software can use the sun sensor data to determine the direction of the Sun and direct a slew maneuver needed to achieve Sun-point for the solar arrays, in the event that the star tracker fails to determine the spacecraft attitude. The star tracker, which is used to precisely determine the inertial attitude of the orbiter, relies on an onboard star catalog to match the patterns observed in flight. The star tracker is powered on immediately after separation from the launch vehicle and remains powered on for the duration of the mission. Each star tracker is canted up 30 degrees from the local horizontal to keep the Moon out of the Field-of-View (FOV) during the late TSF, Science, and Decommissioning Phases.

The Reaction Wheel Assembly (RWA) is the primary method of applying control torques to the orbiter and will be relied on to maintain the orbiter-to-orbiter pointing during the Science Phase. There are four reaction wheels each with a momentum capacity of 1.55 Nms and each can output a torque of at least 6 milliNewton-meters. The four wheels are mounted in a pyramid configuration to enhance the momentum storage capacity, increase the interval between reaction wheel desaturation events, and protect against wheel zero-speed crossings or the failure of any one wheel. The Inertial Measurement Unit (IMU), which measures spacecraft linear accelerations and angular rates, is powered on just prior to separation from the launch vehicle and remains powered on for the remainder of the mission. The IMU will be used for ΔV measurement and cutoff during all main engine burns.

2. Command and Data Handling (C&DH)

The Command and Data Handling subsystem is used for telemetry and command processing for the orbiter. The C&DH features the enhanced RAD-750 spacecraft computer with 128 Megabytes of SDRAM (Static Dynamic Random Access Memory) used to house the spacecraft flight software, fault protection, on-board file system, and data collection and data management functions. In addition, there are 512 Megabytes of storage within a Memory and Payload Interface Card (MPIC) for recorded data.

3. Electrical Power

The electrical power subsystem includes two solar arrays and a lithium ion battery. Each of the solar arrays consist of strings of second-generation Triple-Junction high-efficiency cells, and together are capable of producing 700 Watts at the end of the mission. The solar arrays are deployed shortly after separation from the launch vehicle and remain fixed throughout the mission. The Li-Ion battery has a capacity of 30 amp-hours, and is used to provide energy to the orbiter during periods when the solar arrays are not able to generate enough power for the orbiter systems.

4. Propulsion

The primary components of the propulsion system consist of a single, 22 N liquid hydrazine thruster (main engine), a propellant tank, a high-pressure helium tank, a warm gas attitude control system, and eight 0.9 N warm gas thrusters. The propellant tank operates in a blow-down mode, between approximately 400 psi and 150 psi, with the propellant and helium separated by a flexible diaphragm in a common tank. The propellant tank will be repressurized twice during the mission; once shortly before LOI and once again after LOI to improve engine performance. The current propellant load at launch is 106 kg which represents an 88.1% fill fraction. The warm gas ACS system provides thrust vector control during the main engine burns as well as the ΔV for small maneuvers in the TSF and Science Phases. The ACS thrusters are canted to provide coupled thrust during main engine maneuvers, reaction wheel desaturations, small ΔV maneuvers, and all safing slews.

5. Telecommunications

The telecom subsystem consists of an S-band transponder, two Low Gain Antennas (LGAs), and a single-pole double-throw coaxial switch used to alternate between the two antennas. The S-band transponder modulates signals sent to the ground and demodulates the commands sent from the ground. It is configured to receive commands at a rate of 1 kilobit/second (1 kb/s) and can transmit at selectable rates up to 128 kb/s. The transponder also provides two-way coherent ranging capability, which will be used during the Trans-Lunar Cruise Phase. The two LGAs are stacked patch antennas mounted on the +X and –X panels of the orbiter (i.e. pointed in opposite directions). This ensures a reliable communications link, even when the orbiter-to-Earth geometry is unfavorable. During the TLC Phase, the +X LGA will be used for communicating with the Earth. During the Science Phase, the orbiter will alternate between the two LGAs to minimize off-boresight angles.

6. Thermal

The thermal subsystem is responsible for regulating the temperatures of the spacecraft and payload. The subsystem relies on five items to effectively control the spacecraft environment: temperature sensors, heaters, thermostats, multi-layer insulation (MLI), and radiators. The temperature sensors are mounted throughout the spacecraft and are used to monitor component temperatures; the heaters, which are controlled by either the flight software or thermostats, are used to keep the orbiter components within their allowable thermal limits; the MLI covering around the spacecraft structure is used to maintain a constant temperature environment for the orbiters' systems; and the radiators, which are mounted on the +Y and –Y panels of the spacecraft, are used to dump heat from the spacecraft and payload.

B. Payload

There are two payload elements on each GRAIL orbiter – the Lunar Gravity Ranging System and the MoonKAM System. The LGRS is the primary science instrument and the MoonKAM System is used for Education and Public Outreach lunar imaging. The LGRS is based on the instrument flying as part of the GRACE mission.

1. Lunar Gravity Ranging System

The primary payload of the GRAIL spacecraft is the Lunar Gravity Ranging System. The LGRS is responsible for sending and receiving the signals needed to accurately and precisely measure the changes in range between the two orbiters. To accomplish this, the LGRS consists of an Ultra-Stable Oscillator, Microwave Assembly (MWA), a Time Transfer Assembly (TTA), and the Gravity Recovery Processor Assembly (GPA). The USO provides a steady reference signal that is used by all of the instrument subsystems. Within the LGRS, the USO provides the reference frequency for the MWA and the TTA. The MWA converts the USO reference signal to the Ka-band frequency, which is transmitted to the other orbiter. The function of the TTA is to provide a two-way time transfer link between the spacecraft to both synchronize and measure the clock offset between the two LGRS clocks. The TTA generates an S-band signal from the USO reference frequency and sends a GPS-like ranging code to the other spacecraft. The GPA combines all the inputs received from the MWA and TTA to produce the radiometric data that is downlinked to the ground. In addition to acquiring the inter-spacecraft measurements, the LGRS also provides a one-way signal to the ground based on the USO, and is transmitted via the X-band Radio Science Beacon (RSB). The steady state drift of the USO is measured via the one-way Doppler data provided by the RSB. The LGRS is being developed by the Jet Propulsion Laboratory. A block diagram of the LGRS is illustrated in Fig. 6.

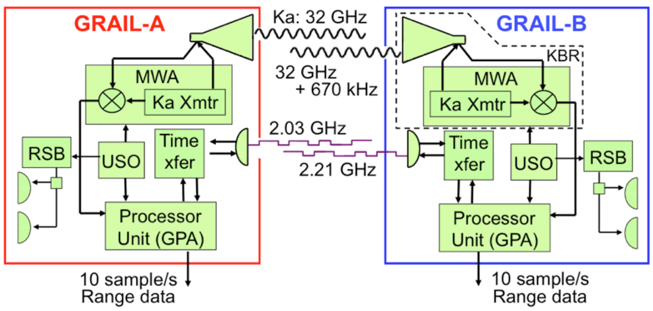

Figure 6. Payload Block Diagram

2. MoonKAM

MoonKAM is a digital video imaging system that is used as part of the Education and Public Outreach (E/PO) activities for GRAIL. Each MoonKAM system consists of a digital video controller and four camera heads. The digital video controller serves as the main interface to the spacecraft and provides storage for images acquired by the camera heads. The four cameras are mounted in such a way that they provide different views of the lunar surface during the Science Phase. The cameras are pointed 60 degrees away from each other in the Y-Z plane, with one pointed slightly forward, two pointed nadir, and one pointed slightly backward. One of the nadir-pointed cameras has a half-angle FOV of 2.7 degrees; all of the other cameras have a half-angle FOV of 22 degrees. The MoonKAM system is provided by Ecliptic Enterprises Corporation and will be operated by undergraduate students at the University of California at San Diego under the supervision of faculty and in coordination with Sally Ride Science.

C. Mass Summary

Table 2 provides a mass breakdown for the GRAIL flight system. Since the development of the flight system is not yet complete (approximately 40% of the flight system components have been delivered and weighed), the masses listed in the middle column are representative of the Current Best Estimate (CBE) plus growth uncertainty of the flight system dry mass components. The indicated orbiter dry mass can be applied to both GRAIL-A and GRAIL-B. The flight system mass, which is the total mass of two orbiters and the LVAA, is well within the launch vehicle allocation of 818.0 kg.

Table 2. Flight System Mass Breakdown

Item	CBE+Growth (kg)	Capability (kg)
Spacecraft	~ 190.0	—
Payload	19.6	—
Orbiter Dry Mass	~ 209.6	226.0
Propellant	106.0	106.0
Helium	0.2	0.2
Orbiter Wet Mass	~ 315.8	332.2
LVAA	~ 151.5	—
Flight System Mass	783.1	818.0

IV. Mission Phases

A. Launch Phase

The Launch Phase begins when the orbiters transfer to internal power at five minutes before liftoff and extends through the initial 24 hours after liftoff. During this phase, the orbiters will undergo launch, separation from the launch vehicle, initial DSN acquisition, solar array deployment and initial commanding.

The GRAIL launch period was designed to satisfy a number of important mission requirements while minimizing the orbiter ΔV requirements during the Trans-Lunar Cruise (TLC) and Lunar Orbit Insertion (LOI) Phases. The mission requirements included ensuring visibility of the critical LOI maneuvers from Earth and selecting an arrival geometry at the Moon that provides an adequate amount of time from the LOI maneuver to the start of the science data collection. In addition, other design goals, such as employing a constant arrival date in order to simplify operations, were key factors in the development of the overall launch period strategy.

Figure 7 illustrates the deterministic (impulsive) ΔV cost for both GRAIL orbiters during the TLC and LOI Phases (i.e. TCM-2 + TCM-3 + LOI ΔV) for each day within a 30-day interval. The non-shaded portions of Fig. 7 represent the 21 launch days currently selected for the GRAIL mission. The noncontinguous 21 launch days are divided into two groups within a 26-day launch period beginning on September 8 and extending through October 3, 2011. The two groups are separated by a 5-day stand-down period. The stand-down period was initially established in order to avoid perturbations from the Moon on the Earth departure trajectory that increase the mission ΔV and injection energy requirements, however, use of the stand-down days can be revisited once launch vehicle and orbiter performance margins are known more precisely closer to launch.

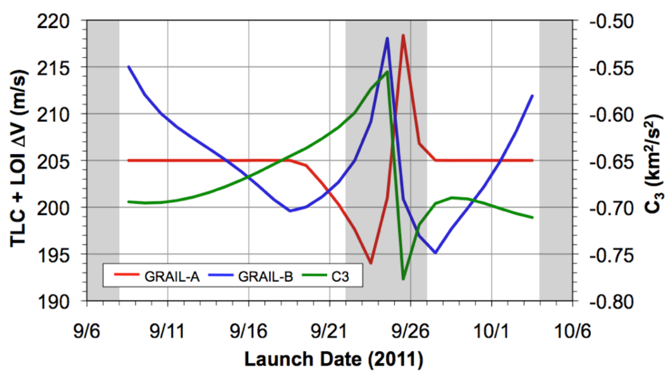

Figure 7. TLC + LOI ΔV Cost Throughout LP

To further ensure the likelihood of launching on any given day of the launch period, two different launch azimuths will be used. The two different launch azimuths provide two instantaneous launch opportunities per day in order to more easily accommodate and deal with minor hardware problems, short-term weather violations at the launch pad, and launch holds caused by potential collisions with objects in Earth orbit. The two azimuths are 93 degrees and 99 degrees and the daily launch opportunities are separated in time by approximately 39 minutes.

In general, for any given launch azimuth, two distinct launch opportunities exist every day for a rocket to launch and inject its payload onto the proper Earth departure trajectory. The primary difference between these opportunities, usually referred to as the short-coast and long-coast opportunities, is the length of time that the launch vehicle must coast in the parking orbit before reaching the proper location to perform the Earth departure injection burn. From a spacecraft power perspective, the short-coast option is generally preferred since it minimizes the time that the spacecraft is operating on internal power. For GRAIL, however, given the current launch period the short-coast opportunity resulted in much higher TLC Phase ΔV requirements (~ 40 m/s). Therefore, the long-coast opportunity was selected for both launch azimuths.

The Earth relative target conditions that must be achieved by the Delta II launch vehicle in order to place the spacecraft onto a low-energy trajectory to the Moon are specified by defining twice the injection energy per unit mass (C_3), the declination of the injection orbit apoapsis vector (DAV), and the right ascension of the injection orbit apoapsis vector (RAV) at a specific time after the second Stage II burn. The specific time is referred to as the Targeting Interface Point (TIP) and is defined to occur 8 minutes after the second Stage II engine cutoff (SECO-2). The launch vehicle targets represent the conditions on the osculating departure trajectory at the TIP. Since the GRAIL departure trajectory is along an ellipse as opposed to a hyperbola with respect to the Earth, the traditional set of targets used for nearly all planetary missions, involving C_3 and the declination and right ascension of the departure asymptote, cannot be used. Table 3 indicates the launch vehicle targets for four days spanning the 26-day GRAIL launch period.

Table 4 indicates the timing of the key events during the Launch Phase. Figure 8 shows the Earth departure trajectory for the open and close launch dates and Figure 9 provides a illustration of some of the key spacecraft events following injection, including spacecraft separation and solar array deployment.

Table 3. Launch Vehicle Targets

Launch Day	Launch Date	Launch Time (hh:mm:ss) (UTC)	Launch Vehicle Targets C_3 (km^2/s^2)	DAV (deg) EME2000	RAV (deg) EME2000
93˚ launch azimuth					
1	08 Sep 2011	12:35:51	–0.694	–6.223	190.390
14	21 Sep 2011	11:38:13	–0.615	–6.988	190.213
20	27 Sep 2011	11:12:27	–0.696	–5.846	187.493
26	03 Oct 2011	10:47:14	–0.711	–5.676	186.764

Table 4. Key Launch Phase Events

	GRAIL LP Open	GRAIL LP Close
Launch Date	08 Sep 2011	03 Oct 2011
Coast Opportunity	Long	
Launch Azimuth	93˚	
Launch Time (hh:mm:ss, UTC)	12:35:51	10:47:14
Event	**Mission Elapsed Time (sec)**	
Liftoff	0.0	0.0
MECO	263.2	263.2
Stage II Ignition	276.7	276.7
Payload Fairing Jet.	281.0	281.0
SECO-1	426.5	426.5
Enter Solar Eclipse	2680.3	3067.6
Restart Stage II	4082.7	4065.6
SECO-2 (Injection)	4344.2	4327.1
Exit Solar Eclipse	4634.5	N/A
Goldstone Rise (7º)	4728.8	4721.6
TIP	4824.2	4807.1
Exit Solar Eclipse	N/A	4860.0
GR-A – Stage II Sep.	4914.2	4897.1
GR-A Solar Array Dep.	5223.2	5206.1
GR-B – Stage II Sep.	5409.2	5392.1
GR-B Solar Array Dep.	5718.2	5701.1
Madrid Rise (7º)	7096.1	7161.8
Madrid Set (7º)	15857.6	15642.6

MECO = Main Engine Cutoff
SECO = Second Stage Engine Cutoff

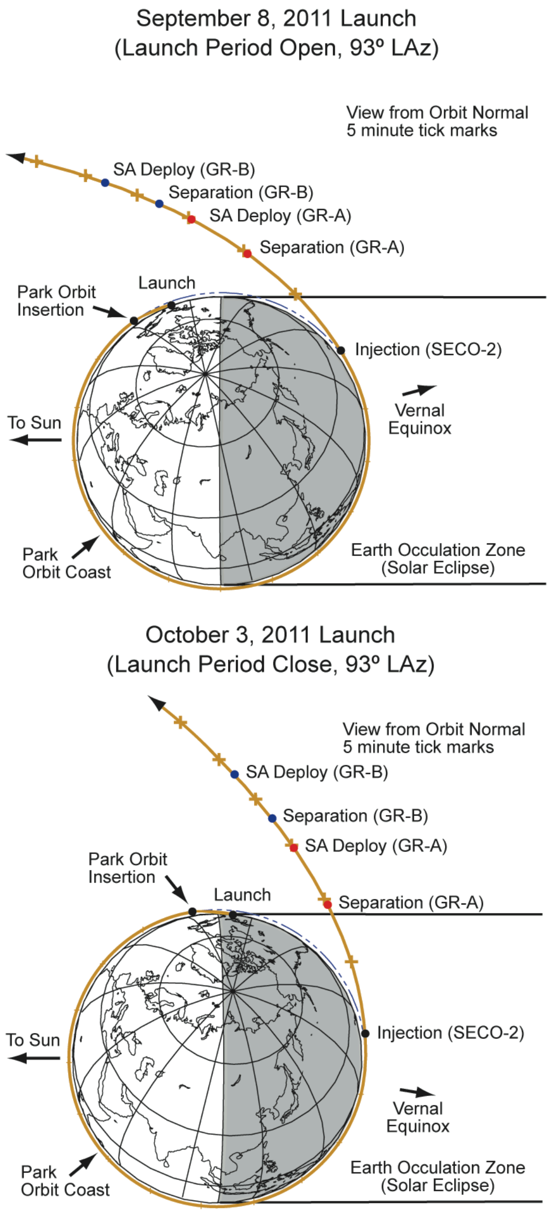

Figure 8. Earth Departure Trajectory

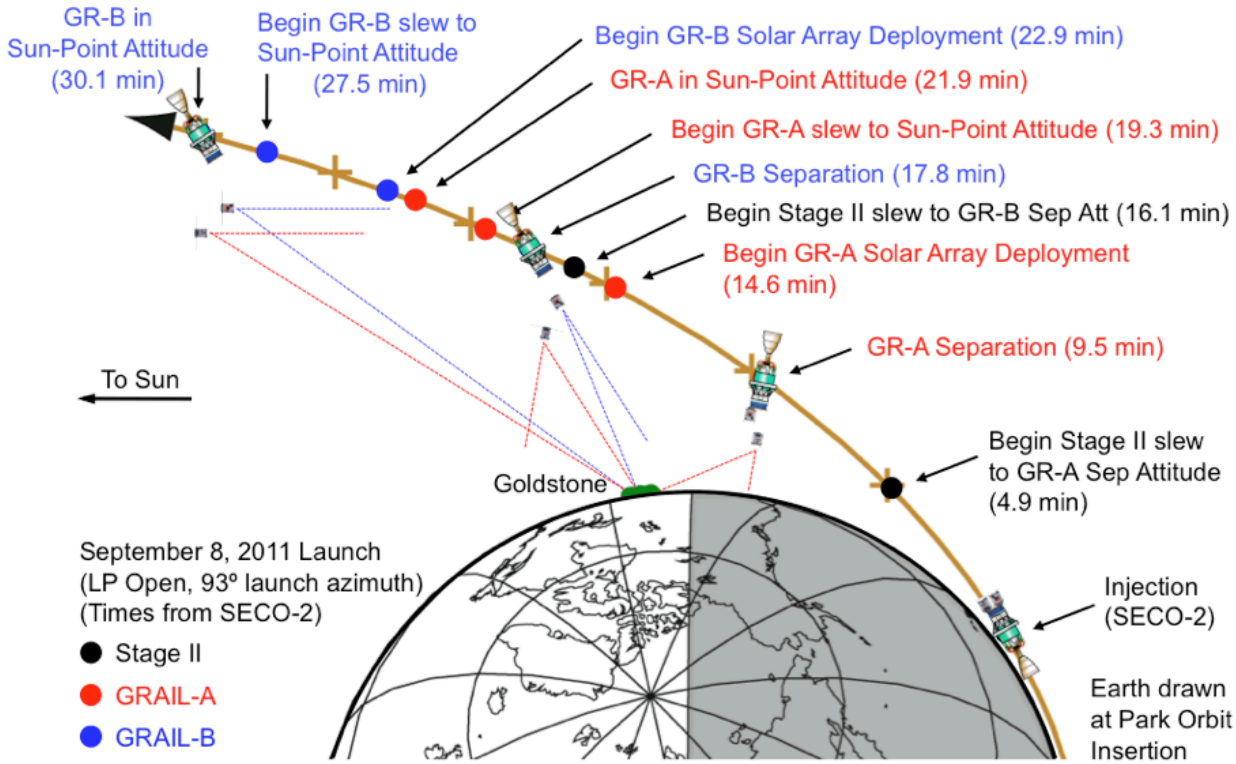

Figure 9. Spacecraft Event Timeline After Injection

B. Trans-Lunar Cruise

The Trans-Lunar Cruise Phase is the period of time when both orbiters will be en route to the Moon via a low-energy trajectory. The major activities planned during this mission phase include a series of checkouts and calibrations to characterize the performance of the spacecraft and payload subsystems, navigation activities required for determining and correcting the vehicle's flight path, and activities to prepare for Lunar Orbit Insertion (LOI) including repressurization of the propulsion system. Prior to arrival at the Moon, spacecraft fault protection will be reconfigured to prepare for the mission critical LOI maneuvers used to capture the spacecraft into orbit about the Moon.

The GRAIL orbiters will follow a low-energy transfer trajectory to the Moon. The trajectory type is referred to as "low-energy" since it reduces the LOI ΔV requirements at the Moon by over 100 m/s compared to the more familiar 3- to 6-day transfer trajectories used by most other lunar missions. The lower ΔV cost, however, is traded for flight time. The GRAIL low-energy trajectory takes approximately 3 to 4 months to get to the Moon. The longer flight time is beneficial to the GRAIL mission for a number of reasons: it allows more time for checkout of the orbiter systems, allows more time for outgassing to occur, permits the on-board Ultra-Stable Oscillator to be continuously powered for several months allowing it to reach a stable operating temperature before beginning the collection of science measurements that rely on its use in lunar orbit, and permits the use of an extended launch period (i.e., 21 launch days within a 26-day launch period) similar to those used for planetary missions.

Another useful characteristic of the low-energy trajectory is the fact that the arrival time at the Moon can be held constant for all launch days. This greatly simplifies the overall mission design effort by decoupling the pre- and post-LOI analyses. With a fixed arrival date, the flight time to the Moon varies depending on the launch date. The flight time ranges from 115 days for a launch at the open of the launch period to 90 days at the close of the launch period.

The GRAIL TLC trajectory is illustrated in Fig. 10. The figure shows the GRAIL-A and GRAIL-B trajectories for a launch at the open and close of the launch period. The low-energy trajectories leave the Earth following a path towards the Sun and passing near the interior Sun-Earth Lagrange Point 1 (EL1) before heading back towards the Earth-Moon system. A detailed discussion of the trades and analyses involved in designing and optimizing the TLC Phase trajectories is provided in Ref. 1.

Navigation during the TLC Phase is performed relying only on two-way S-band Doppler and range data. Up to five Trajectory Correction Maneuvers (TCMs) will be performed in order to deliver the orbiters to the desired conditions at LOI. Table 5 lists the TCMs and their primary purpose. The second and third TCMs are primarily deterministic maneuvers designed to separate the GRAIL-A and GRAIL-B arrival times at the Moon by approximately one day and insert them onto the desired lunar approach trajectories. The other TCMs are primarily statistical in nature and are used to correct launch vehicle injection errors and target the LOI conditions.

Table 5. TCM Schedule

Mnvr	Schedule	Mnvr	Schedule	Purpose
TCM-A1	L + 6 days	TCM-B1	L + 7 days	Correct LV injection errors
TCM-A2	L + 20 days	TCM-B2	L + 25 days	LOI separation (deterministic)
TCM-A3	16-Nov-2011	TCM-B3	21-Nov-2011	Manifold insertion (deterministic)
TCM-A4	09-Dec-2011 (LOI-A – 22 days)	TCM-B4	14-Dec-2011 (LOI-B – 18 days)	Correct TCM-3 errors
TCM-A5	23-Dec-2011 (LOI-A – 8 days)	TCM-B5	24-Dec-2011 (LOI-B – 8 days)	LOI targeting

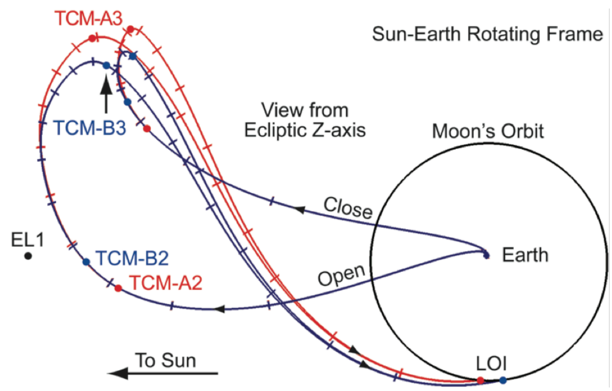

Figure 10. Trans-Lunar Cruise Phase Trajectories

During the TLC Phase, the orbiters will spend time in three different attitudes. For the first week following launch, the orbiters will be in a Sun-point attitude, which has the –X axis of the spacecraft pointed directly towards the Sun, with the –Z axis pointed in the Sun-Earth plane. For the next two months, the orbiters will be off-pointed from the Sun by 45 degrees. The first half of this period will be spent in an attitude with a plus 45-degree rotation about the orbiter +Y axis (a plus 45 degree off-point) and the second half will be spent in an attitude with a negative 45 degree rotation about the orbiter +Y axis (a negative 45 degree off-point). These two off-pointed attitudes will illuminate areas on the orbiters that will be exposed to the Sun during the Science Phase. This will facilitate outgassing from the spacecraft structure which will reduce the amount of non-gravitational perturbations experienced by the orbiters during the Science Phase. Spending time in these different attitudes will also help determine the induced translational ΔVs that are imparted during reaction wheel desaturation maneuvers, as well as help estimate the effects of solar radiation pressure on the orbiters. Following TCM-3, each orbiter will return to the Sun-point attitude for the remainder of the TLC Phase. In this attitude, the solar torque on the spacecraft is minimized, reducing the need to perform RWA desats. This will reduce disturbances that may affect the navigation of the orbiters leading up to LOI.

C. Lunar Orbit Insertion Phase

At the end of the TLC Phase, both spacecraft will enter the Lunar Orbit Insertion Phase. The primary focus for this phase is the critical LOI maneuver. This phase begins three days prior to GRAIL-A arrival at the Moon and ends one day past the LOI of GRAIL-B. As part of the transition to this mission phase, both orbiters will reconfigure their on-board fault protection settings to increase the likelihood of completing the maneuver. Once the burn is completed, the fault protection settings will be reset to the original configuration.

The LOI maneuver for each orbiter is approximately 41 minutes in duration and will change the orbiter velocity by about 193 m/s. A constant pitch rate thrust vector steering strategy will be used in order to reduce gravity losses during the burn. The LOI maneuver will put each orbiter into a near-polar, elliptical orbit with a period of 11.5 hours.

Both orbiters approach the Moon from the south, flying nearly directly over the South Pole. The orbiters are inserted into nearly the same orbital plane with an Orbit Local Solar Time (OLST) of the descending node of approximately 1:15 PM (or a solar beta angle of about –19 degrees). The lunar approach trajectory and capture orbit are illustrated in Fig. 11.

During the LOI maneuver, each spacecraft will transmit real-time engineering telemetry to the ground using its +X axis LGA. The LOI maneuvers were designed to occur during the Madrid-Goldstone DSN tracking station overlap on consecutive days in order to provide continuous coverage of the mission critical events from two different DSN complexes.

Once an orbiter has completed its LOI maneuver, it will return to a Sun-pointed attitude. Approximately a day later, however, it will slew to a biased Sun-pointed attitude – different from the attitude that was used during the TLC Phase. Due to additional heat that is reflected off the lunar surface, particularly at low beta angles, the spacecraft must orient the Y-panel radiators away from the Moon. To achieve this, the spacecraft will be off-pointed by 40 degrees about the Z-axis. For GRAIL-A, this will be a 40 degree roll about +Z axis and for GRAIL-B, it will be a 40 degree roll about the –Z axis.

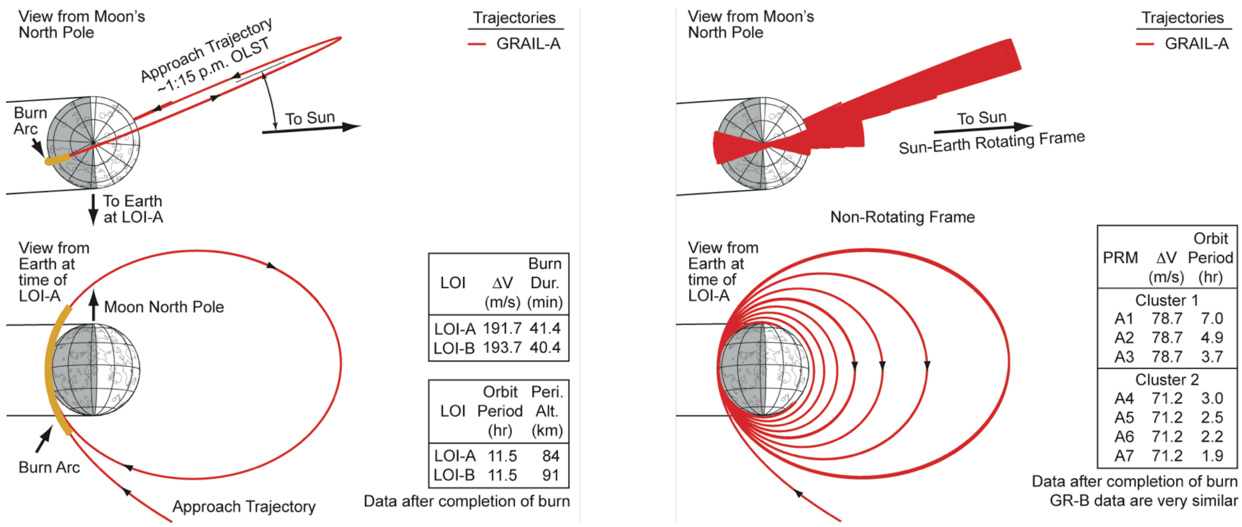

Figure 11. LOI and OPR Phase Trajectories

D. Orbit Period Reduction Phase

The Orbit Period Reduction Phase starts one day after the LOI maneuver for GRAIL-B and continues for five weeks. During this mission phase a series of maneuvers, grouped into two different clusters for each orbiter, are performed in order to reduce the orbit period of the elliptical capture orbit from 11.5 hours down to just under two hours.

Once safely in lunar orbit following the critical LOI maneuvers, the next mission objective is to maneuver the GRAIL orbiters into low-altitude, near-circular orbits just slightly larger than the final orbit desired for science. The ΔV required to perform this transition is approximately 520 m/s. Given the magnitude of the required ΔV and the limited thrust levels provided by the main engine of the GRAIL propulsion system, a large number of separate burns, performed on different orbits, must be executed. A strategy involving seven separate burns has been devised to impart the desired ΔV during the OPR Phase.

The maneuvers, referred to as Period Reduction Maneuvers (PRMs), are grouped into two different clusters for each orbiter. Within each cluster, a daily maneuver is executed over successive days using a single maneuver design. In other words, maneuvers within a cluster are repeatedly performed in the same inertial direction with the same ΔV. Each cluster is about one week long and the clusters alternate between GRAIL-A and GRAIL-B. The first cluster contains three PRMs, and the second contains four. The additional days in the weeklong clusters can be used as backup or contingency maneuver opportunities if a PRM is missed or delayed.

Figure 11 illustrates the reduction in orbit size during the OPR Phase for GRAIL-A. The figure also includes the ΔV and orbit period following each maneuver. The timing of the PRMs in the OPR Phase is illustrated in Fig. 3.

Due to the fact that the PRMs are performed over a period of one month, the orbit geometry as viewed from Earth changes considerably. In general, this means that the maneuvers in only one of the two clusters for each orbiter will be visible from Earth. The orbit geometry with respect to the Earth and the Sun is also illustrated in Fig. 3.

Throughout the OPR Phase, the orbiters will continue to employ the biased, Sun-pointed attitude strategy that was adopted upon completion of the LOI maneuver in order to reduce the thermal input reflected off of the lunar surface.

E. Transition to Science Formation Phase

The Transition to Science Formation Phase has two primary objectives: 1) to establish the orbit formation and initial conditions necessary for the collection of gravity science data, and 2) to test and calibrate the payload prior to the beginning of the Science Phase. Until this point in the mission, the two GRAIL orbiters will have been essentially operated independently. This is the first mission phase where the position of one orbiter relative to the other becomes relevant. This phase begins when both orbiters have been placed in low-altitude, near-circular orbits with periods just slightly larger than the final, targeted orbit period for science operations and ends when the orbit formation and initial conditions have been established, the science payload has been tested and calibrated, and the spacecraft power can support continuous operation of the payload in the attitude required for science operations.

The only orbit parameter actively controlled during the OPR Phase is the orbit period. No attempt is made to control the orbit plane, the orbit eccentricity, or the location of periapsis (i.e. argument of periapsis). In order to avoid the need for orbit maintenance maneuvers during the Science Phase, eccentricity and argument of periapsis must actively be controlled during the TSF Phase. This means that all of the maneuvers in the TSF Phase must accommodate and take advantage of the natural evolution of the orbital elements caused by the lunar gravity field. The maneuvers in this mission phase are referred to as TSMs for Transition to Science formation phase Maneuvers.

At the end of the OPR Phase, the orbit period for GRAIL-B is targeted to be approximately 3 minutes larger than the period of GRAIL-A. This difference in period means that the GRAIL-A orbiter, flying below the GRAIL-B orbiter, will lap the GRAIL-B orbiter once every three days. This creates the situation where subsequent maneuvers can be timed such that the distance between the two orbiters can be managed and reduced in a controlled manner.

At the start of the TSF Phase, the relative phasing between the GRAIL-A and GRAIL-B orbiters is assumed to be random. This is due to the build up of orbit period errors caused by execution errors from all of the maneuvers performed since LOI. In order to maneuver the orbiters into the proper formation with the GRAIL-B orbiter in the lead position and a desired separation distance between the two orbiters, a maneuver strategy similar to an orbit rendezvous was developed.

The first maneuvers in the TSF Phase are used primarily to adjust the orbit eccentricity and argument of periapsis (i.e. the eccentricity vector). TSM-A1 is performed first and, in terms of period, slightly reduces the GRAIL-A orbit period by 0.2 minutes to 114 minutes. Thus, after this maneuver GRAIL-A will still lap GRAIL-B every three days. The timing of TSM-B1 is selected such that it is performed when GRAIL-B is between about 26 and 29 minutes ahead of GRAIL-A. In conjunction with changes in eccentricity and argument of periapsis, TSM-B1 reduces the orbit period of GRAIL-B by 3 minutes to match that of GRAIL-A. Thus, after the completion of TSM-A1 and TSM-B1, both orbiters will generally be in the correct orbit, have essentially the same orbit period, and GRAIL-B will be ahead of GRAIL-A. In other words, the basic orbit formation will have been established. The remaining TSMs are used to carefully reduce the separation distance, match orbit planes, and continue to control the eccentricity vector. The maneuvers are precisely timed to produce the desired initial conditions at the start of the Science Phase.

During the Science Phase, the orbit period stays relatively constant. The lunar gravity field, however, causes the location of the periapsis to move from the descending node side of the orbit to the ascending node side. With the proper initial conditions, the periapsis altitude starts low at approximately 20 km altitude above a reference lunar radius, naturally increases to just over 50 km, then decreases again to approximately 16 km. In order to avoid targeting a very low lunar altitude during the TSF Phase, the next maneuver, TSM-A2, will not be performed before February 20, 2012.

The purpose of TSM-A2 is to reduce the period of the GRAIL-A orbit to the final, desired orbit period of approximately 113.5 minutes. Given a difference in orbital periods, the GRAIL-A orbiter will once again start to approach the GRAIL-B orbiter from behind. The next maneuver, TSM-B2 will be performed when GRAIL-A is only 3.7 to 4.2 minutes behind GRAIL-B. TSM-B2 reduces the orbit period of GRAIL-B to 113.6 minutes.

At this point, GRAIL-A will be approaching GRAIL-B very slowly. When a separation distance of approximately 75 km is reached, the final deterministic TSF Phase maneuver, TSM-B3, will be performed. Due to maneuver execution errors, a statistical clean up maneuver to fine tune the separation rate between the two orbiters may be needed. That maneuver is referred to as TSM-B4. Figure 12 illustrates the TSF Phase maneuver strategy. The reference ΔVs for the TSF Phase maneuvers are listed in the Appendix along with the entire mission ΔV budget.

Since the GRAIL-A maneuvers in this mission phase establish the science orbit, they can be performed at a predetermined date and time. The GRAIL-B maneuvers, however, have to be performed relative to the position of GRAIL-A and thus cannot be precisely specified ahead of time. They are dependent on the maneuver execution errors associated with the preceding maneuvers. Prior to the start of the TSF Phase, the variability of TSMs B2-B4 is on the order of ±1 day around a reference or baseline value. As each maneuver in the TSF Phase is executed the timing uncertainty of the GRAIL-B maneuvers is decreased.

Throughout the TSF Phase, the orbiters will continue to employ the biased, Sun-pointed attitude strategy. Once the orbiters are properly positioned within their orbits, they may be commanded to point their Ka-band antennas at each other if power resources allow. In the orbiter-pointed attitude, GRAIL-A will have its –Y-axis pointed towards the Moon, and GRAIL-B will have its +Y-axis pointed towards the Moon.

The first opportunity for the orbiters to go to the orbiter-point attitude occurs late in the TSF Phase with TSM-B4, after the orbiters are properly positioned in their orbits. This statistical clean-up maneuver will be performed on GRAIL-B using only the ACS thrusters while in the orbiter-point attitude. GRAIL-A will also be commanded to orbiter-point during the GRAIL-B maneuver and the LGRS will be powered on both orbiters so that the Science Team can preview the LGRS instrument performance and the Navigation Team can assess the spacecraft-to-spacecraft tracking data during the maneuver.

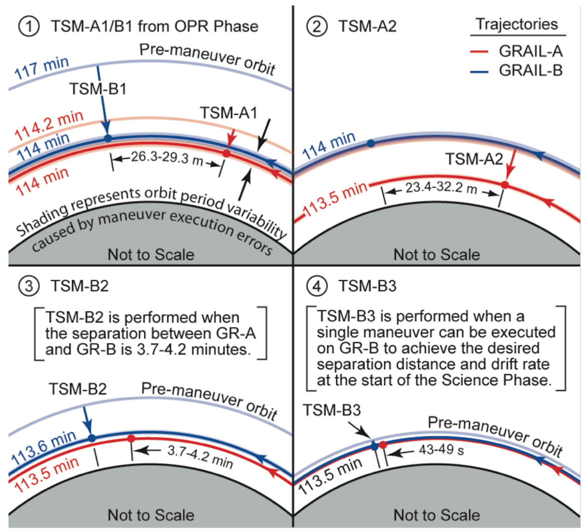

Figure 12. TSF Phase Maneuver Strategy

During the two days following TSM-B4, two Ka-band boresight calibrations will be performed to measure the Ka-band antenna phase center with respect to the orbiter center of mass. Each orbiter will perform a series of small slews while in orbiter-point as LGRS and attitude data are acquired so that the ideal pointing of the Ka-band antennas can be determined. Once the boresight vectors have been determined by the Operations Team, they will be uplinked to the orbiters to optimize the orbiter pointing.

Thermal and power analyses suggest that the orbiters will not be able to remain in a continuous orbiter-pointed attitude with the LGRS powered on prior to the start of the Science Phase, when the solar beta angle is less than 49 degrees. Since the late TSF Phase activities will be performed in the week before the start of the Science Phase, it is expected that the orbiters will only be able to stay in orbiter-point with the LGRS powered on for a limited time before needing to return to the biased, Sun-pointed attitude with the RSB, MWA, and GPA components of the LGRS powered off.

Following TSM-B4, through the final days of the TSF Phase, including the Ka-band boresight calibration activities, the orbiters will be commanded to the orbiter-point attitude for increasingly longer durations with the LGRS powered on as the solar beta angle increases. Once the power and thermal systems allow, the orbiters will be left in orbiter-point with all of the LGRS components powered on. This is expected to occur by the time the solar beta angle reaches 49 degrees.

One day before the start of the Science Phase, a small orbit trim maneuver, OTM-B1, will be performed to change the drift rate between the orbiters. The mean separation distance between the orbiters will slowly increase from 85 km when the OTM is performed to 225 km late in the first Mapping Cycle of the Science Phase, when OTM-B2 will be performed to once again change the drift rate. These small magnitude maneuvers (< ~0.03 m/s) will be executed along the GRAIL-B ±Z-axis using four ACS thrusters without having to disrupt the orbiter-to-orbiter link.

The timing of the maneuvers and activities in the TSF Phase is illustrated in the timeline at the bottom of Fig. 3. The trajectory design strategies employed in the TSF Phase – and throughout all of the lunar orbit mission phases – are described in more detail in Ref. 2, while Ref. 3 provides a mathematical and geometric background for some of the techniques used to design the TSF Phase maneuvers.

F. Science Phase

The Science Phase is defined as the period of time during which the orbiter can support continuous operation of the payload. During this period, precise measurements of the range rate between the two orbiters are made. The Science Phase begins after the orbiters have been maneuvered into the desired orbit, the proper formation has been established (GRAIL-B leading GRAIL-A), orbiter-to-orbiter pointing has been achieved, and the payload has been tested and calibrated. It is designed to start on March 8, 2012, when the orbit plane relative to the Sun has exceeded a solar beta angle of 49 degrees and end on May 29, 2012 when the solar beta angle once again decreases below 49 degrees.

The science orbit is designed to satisfy the basic science requirements of the GRAIL mission, which are for a low-altitude, near-circular, near-polar orbit that does not require any maneuvers to maintain the orbit during the Science Phase. In addition, requirements also place limits on the separation distance between the orbiters and the degree to which the GRAIL-A and GRAIL-B orbits are coplanar.

In order to meet the science requirements, nearly all of the orbit parameters are tightly bounded. The primary design parameter from a science perspective is the orbit altitude, since sensitivity to the lunar gravity field is driven by orbit altitude (i.e. the lower the altitude, the more sensitive the science measurements). Limits on the minimum orbit altitude are driven primarily by orbit lifetime considerations. In order to maximize the orbit lifetime while still meeting the science requirements, the initial eccentricity and argument of periapsis must be selected such that the periapsis altitude starts low and naturally increases due to the perturbations from the lunar gravity field before falling again. This behavior, along with the solar beta angle constraints (i.e., the power constraints on orbiter/payload operation), limits the duration of the Science Phase. Finally, an orbit inclination near 90 degrees is selected to provide near global coverage of the lunar surface.

The GRAIL science orbit has a mean semi-major axis of approximately 1792.4 km, or when expressed as an altitude, a mean orbit altitude of approximately 55 km. Figure 13 shows the variation in periapsis altitude during the Science Phase for both GRAIL-A and GRAIL-B. Two different types of altitudes are shown in the figure. One altitude is calculated relative to a reference lunar sphere with a radius of 1737.4 km (depicted by the bold, thicker red and blue lines in the figure), while the second altitude is calculated relative to the actual lunar topography (shown by the thin red line). The data for the actual lunar topography were developed by the Lunar Reconnaissance Orbiter mission. The figure shows that during the Science Phase the periapsis altitude ranges from approximately 16 km to 51 km above a reference lunar sphere with a minimum periapsis altitude of approximately 12 km above the actual lunar topography occurring near the end of the Science Phase.

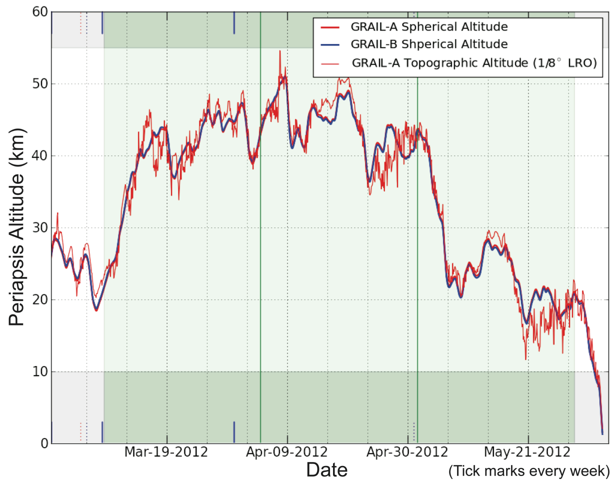

Figure 13. Periapsis Altitude During the Science Phase

In addition to the limits imposed by orbit altitude, the duration of the Science Phase is also limited by the 49 degree solar beta angle constraint imposed by the capability of the spacecraft electrical power system. When the solar beta angle is less than 49 degrees, the spacecraft is not expected to be able to generate sufficient power from the solar arrays to continuously operate the science payload as well as the other orbiter systems. Figure 14 illustrates the time history of the solar beta angle during the Science Phase. The figure also illustrates the relationship between beta angle and the duration of solar eclipse during the Science Phase. Eclipse durations are a maximum at the beginning and end of the Science Phase. When the solar beta angle is near 90 degrees near the middle of the Science Phase, there are no solar eclipses. Figure 15 shows the orientation of the science orbit as viewed from the Sun at the beginning, middle, and end of the Science Phase.

The influence of the lunar gravity field, and to a lesser degree the third-body perturbations from the Earth, are evident in the time histories of all of the science orbit parameters. Figure 16 shows the variation in the osculating orbital inclination during the Science Phase. The figure shows a cyclical pattern that repeats every two weeks with a maximum value of each cycle targeted to be within a 0.7 degree band centered on 89.7 degrees. The slight offset from 90 degrees provides slightly improved gravity recovery solutions without losing too much in terms of global coverage.

Over the course of the Science Phase, the mean separation distance is designed to slowly increase and then slowly decrease. The mean separation distance will slowly increase from 85 km at the start of Mapping Cycle 1 to 225 km late in Mapping Cycle 1. A small Orbit Trim Maneuver (OTM) will then be performed on GRAIL-B late in Mapping Cycle 1 that will cause the separation distance between the orbiters to slowly decrease from 225 km to approximately 65 km at the end of Mapping Cycle 3. Figure 17 illustrates the orbiter separation strategy during the Science Phase.

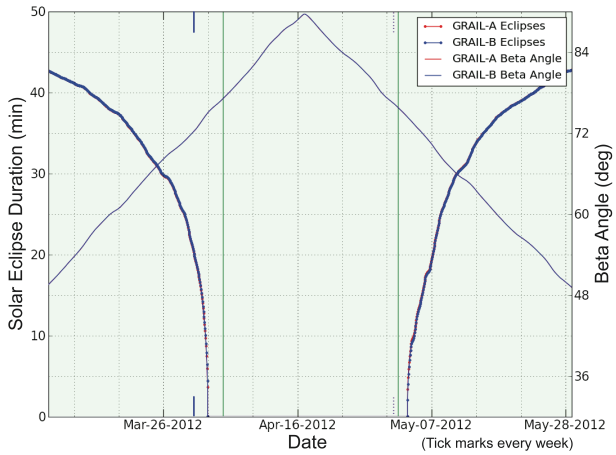

Figure 14. Solar Beta Angle and Solar Eclipse Duration During the Science Phase

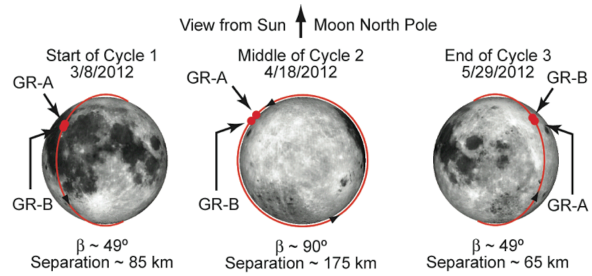

Figure 15. Orbit Orientation as Viewed from the Sun During the Science Phase

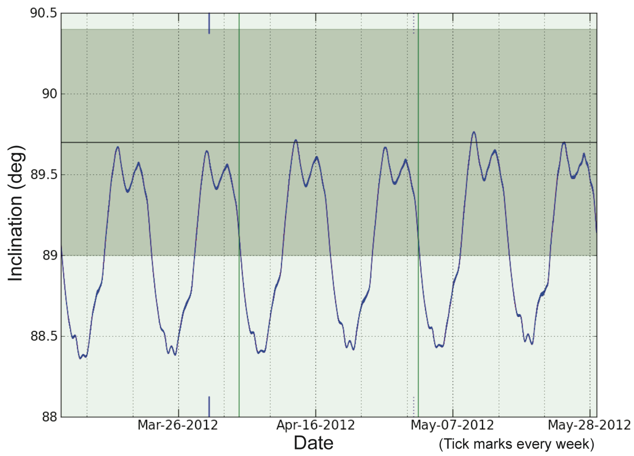

Figure 16. Orbit Inclination During the Science Phase

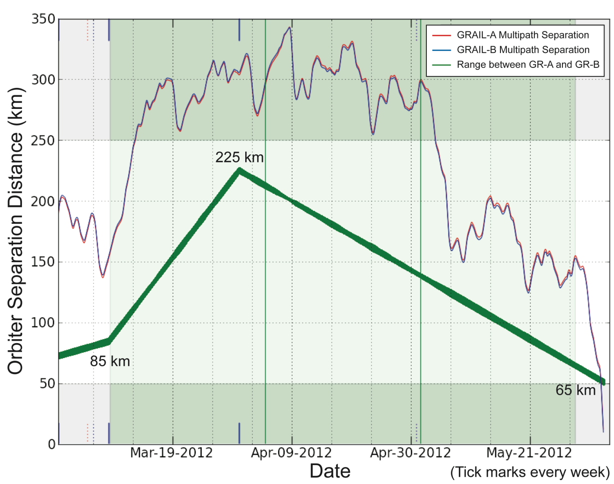

Figure 17. Oribiter-to-Orbiter Separation Distance During the Science Phase

The change in separation distance is required to meet the GRAIL science objectives. The data collected when the orbiters are closer together helps to determine the local gravity field, while the data collected when the separation distances are larger, are more useful in satisfying the science objectives related to detection of a lunar core. In addition, the separation distances are designed to ensure that there is no degradation or corruption of the Ka-band signal, due to multipath off the lunar surface.

In order to continuously collect science data, each orbiter must ensure that the Ka-band antenna boresight is pointed to within 0.3 degrees (3σ) of the line-of-sight to the other orbiter. In addition, to maintain adequate power, the solar arrays on each orbiter are pointed in the direction of the orbit normal. The orbiter-point attitude is maintained using reaction wheels and is based upon ephemeris files that are uplinked to each orbiter twice a week. The ephemeris files for both spacecraft are loaded onto each orbiter. To allow for possible updates in orbiter pointing, two additional Ka-band boresight calibrations are planned during the Science Phase – one near the beginning of Mapping Cycle 2 and another near the beginning of Mapping Cycle 3. The boresight calibrations may be more effective after a more refined gravity field has been developed or after the orbiters have operated under more thermally stable conditions (i.e. during the solar eclipse free period in the middle of the Science Phase). Figure 18 illustrates the orbiter attitude during the Science Phase.

As the Moon orbits the Earth, the angle between the Earth and the boresight of the two Low Gain Antennas changes. Since the LGAs only provide hemispherical coverage, there will be a need to switch antennas when the Earth is beyond 90 degrees for either antenna – approximately every two weeks. The LGA-to-Earth angle also affects the telecom link performance. Depending on the angle, the Operations

Team will choose a data rate that is achievable from a list of predefined rates (from 1 kb/s to 128 kb/s). This selection, in turn, affects the amount of data that is sent down to the ground. With this variability, there are days when data volumes greatly exceed the needs of the engineering telemetry and science. There are also days when the data volume is insufficient to return even all the recorded engineering telemetry. Figure 19 shows the mission data volume capability during the Science Phase for each orbiter (the GRAIL-A data volume is stacked on top of the GRAIL-B data volume) and reflects the variability in the data return volume arising from the impact of the LGA-to-Earth angle on the telecom link performance. The only impact this variable downlink bandwidth has on GRAIL is in the allocation to the MoonKAM activity.

On days where the data rate exceeds 16 kb/s, there is adequate downlink volume to accommodate all the recorded engineering and science data accumulated since the last DSN pass. Any extra bandwidth is available for use by MoonKAM. On the days where the data rate is less than 16 kb/s, there is insufficient bandwidth for even just the recorded engineering telemetry. This results in the data being stored onboard, within the Memory and Payload Interface Card of the spacecraft C&DH subsystem. Though there is storage for up to seven days worth of engineering and science data, the maximum amount of data accumulating in the onboard storage never exceeds five days for engineering telemetry, and six days for science data.

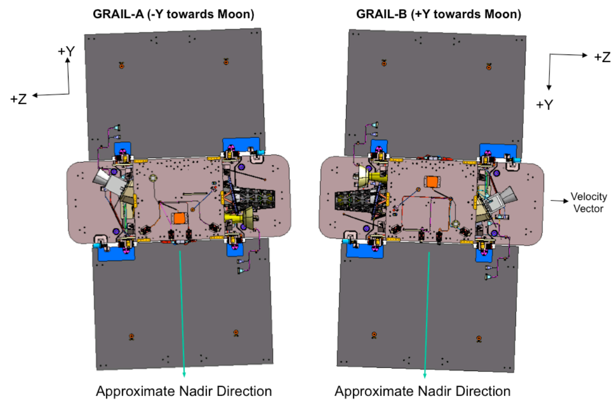

Figure 18. Orbiter-Point Attitude During the Science Phase

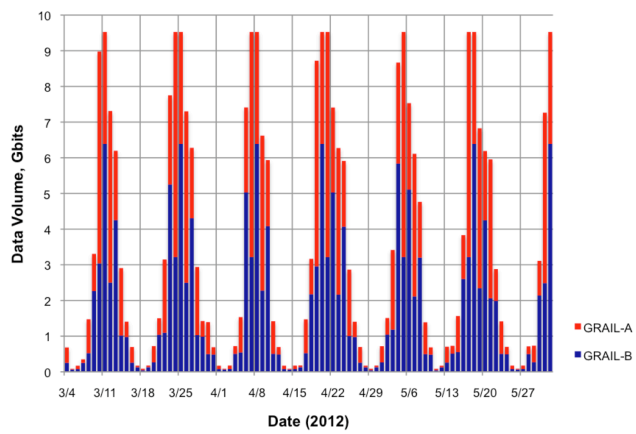

Figure 19. Mission Data Volume During the Science Phase

In addition to the Science objectives of the mission, there is a separate activity for Education and Public Outreach (E/PO) that is operated by Sally Ride Science and the University of California at San Diego (UCSD). This activity utilizes the MoonKAM system, which acquires images of the lunar surface during the Science Phase.

The MoonKAM program is designed to engage middle school students in the areas of math and science. These students will submit imaging requests to the MoonKAM Operations Center (MOC) at UCSD and undergraduates there will build commands to acquire the images. The commands will be delivered to JPL for automated processing into a command file that will then be uplinked to the spacecraft. As the image data are taken, telemetry will be sent to the ground indicating which observations are available for downlink. The MOC will then specify the order in which the image data, stored onboard the MoonKAM, are transferred to the spacecraft for downlink. After the image data are downlinked and processed, image files will be distributed to the middle school students.

G. Decommissioning Phase

The final phase of the GRAIL mission is the Decommissioning Phase. This mission phase lasts only 7 days and has very few required activities. During this phase, the orbiters will perform a final Ka-band boresight calibration and will continue to acquire science data as power and thermal resources allow. The mission phase ends at the time of a partial lunar eclipse on June 4, 2012.

The design of the GRAIL science orbit is such that, at the end of the Science Phase, the orbit periapsis altitudes are approximately 15-20 km above the lunar surface and decreasing. Without the execution of any maneuvers, both orbiters will impact the lunar surface during, or shortly after, this mission phase. There will be no attempt to target the impact points on the lunar surface.

As the solar beta angle decreases throughout the Decommissioning Phase, the orbiters will not be able to maintain a power-positive state in the orbiter-pointed attitude. Since no passivation measures on the orbiters are required prior to impacting the lunar surface, the orbiters will continue to collect science data as the power and thermal systems allow – although as the orbiters descend to lower orbit altitudes, the science data may become degraded due to Ka-band multipath off the lunar surface. Finally, power and thermal analyses indicate that during the partial lunar eclipse the orbiters will likely experience total battery depletion and will experience temperatures below acceptable flight levels. Thus, for any number of reasons, the GRAIL mission is expected to end within a few days of the partial lunar eclipse in June 2012.

V. Mission Status

The GRAIL project is currently a little over one year from launch. At this stage of the project, the flight system is preparing to start the integration and test of flight hardware and software while the mission system is beginning tests to prepare for operations. In the coming year, the GRAIL Mission Design Team will finalize the concepts and strategies described in this paper and will develop contingency plans in response to mission anomalies, such as a missed or delayed maneuver in lunar orbit.

Appendix

The GRAIL mission ΔV budget listed in Table A is the result of the combination of a number of different types of analyses and assumptions where conservative estimates have been applied worst-case on top of worst-case. The components of the ΔV budget include:

Analyses

- deterministic analyses – ΔVs associated with the baseline mission design
- statistical analyses – ΔVs estimated using Monte Carlo techniques and expected performance dispersions

Spacecraft performance assumptions

- propulsion system assumptions – analyses assumed a 3σ low specific impulse (Isp) for the main engine and a maximum propellant load of 106 kg
- ACS propellant assumptions – propellant budget includes 3σ mass expenditures for non-ΔV events
- spacecraft mass assumptions – analyses assumed the maximum orbiter dry mass (226 kg)

For the TLC Phase, a statistical maneuver analysis was performed in which the TCMs were used to correct launch vehicle injection errors, orbit determination errors, and maneuver execution errors. The overall ΔV was minimized by employing a multi-TCM maneuver optimization strategy. The TLC Phase ΔVs in Table A represent the 99% confidence level ΔVs (or ΔV_{99}) for the opening date of the launch period.

For the lunar orbit mission phases, a detailed simulation was developed modeling the blowdown performance of the GRAIL propulsion system, including the performance changes associated with the multiple repressurization events. All maneuvers were modeled as finite burns and the LOI maneuvers were based on an optimized in-plane, constant pitch rate thrust vector steering strategy.

Considering the factors described above, the estimated ΔV margin at the end of the mission for both GRAIL-A and GRAIL-B for a launch on the opening launch date, is approximately 65-70 m/s. However, this ΔV margin is considered "encumbered", since there are additional factors that need to be accounted for in order to develop a robust mission ΔV budget. These include:

Analyses

- contingency analyses – estimating the ΔV necessary to recover from various mission contingencies, most notably a missed or delayed maneuver

Analysis assumptions

- design and modeling errors – changes (or variations) in the ΔV associated with design assumptions, such as the choice of lunar gravity field or the assumed timing of maneuvers
- variation across the launch period – for example, the ΔV requirements during the TLC Phase are not the same for every launch day

After these types of factors have been included, the amount of ΔV remaining is considered "unencumbered" or "unallocated". For GRAIL, the unencumbered ΔV margin is approximately 25 m/s for both orbiters.

At this stage in the development of the project, however, the thing that could have the most impact on the ΔV margin is for the orbiter dry mass, once its completely built, to come in less than the maximum capability (see Table 2). If the orbiter dry mass stays at its present level and the project still decides to load 106 kg of propellant, the (encumbered) mission ΔV margin would grow to well over 100 m/s.

Table A. GRAIL Mission ΔV Budget

Msn Phs	GRAIL-A		GRAIL-B		Maneuver Description
	Maneuver	ΔV (m/s)	Maneuver	ΔV (m/s)	
TLC	TCM-A1	21.4	TCM-B1	22.7	Correct launch vehicle injection errors
TLC	TCM-A2	12.4	TCM-B2	26.8	LOI separation (primarily deterministic)
TLC	TCM-A3	11.1	TCM-B3	6.3	Manifold insertion (primarily deterministic)
TLC	TCM-A4	0.38	TCM-B4	0.82	Correct TCM-3 errors
TLC	TCM-A5	0.06	TCM-B5	0.04	LOI targeting
TLC					TCM ΔVs scaled such that the sum = ΔV(99)
TLC	Total ΔV(99) = 45.4		Total ΔV(99) = 56.6		99% ΔV (deterministic + statisical)
LOI	LOI-A	191.7	LOI-B	193.7	Lunar orbit insertion (period 11.5 hours)
OPR	PRMs A1-A3	3 x 78.7	PRMs B1-B3	3 x 76.1	Period reduction (post ΔV period ~ 3.7 hours)
OPR	PRMs A4-A7	4 x 71.2	PRMs B4-B7	4 x 70.0	Period reduction (post ΔV period ~ 1.9 hours)
LOI OPR		1.3		1.3	Statistical ΔV associated with LOI/OPR ΔV(99)
TSF	TSM-A1	11.7	TSM-B1	24.3	Orbit targeting (a, e, ω) + minor period reduction
TSF	TSM-A2	11.6	TSM-B2	3.4	Establish orbit formation (GR-B leads GR-A)
TSF			TSM-B3	0.4	Establish separation distance/rate for start of Science
TSF			TSM-B4		Refine separation drift rate (with ACS)
SCI			OTM-B1	0.02	Change separation drift rate (with ACS)
SCI			OTM-B2	0.03	Change separation drift rate (with ACS)
DCM					No Decommissioning ΔV requirements
Sub-Total ΔV		**782.4**		**787.8**	**Translational ΔV (without margin)**
Margin		71.4	Margin	65.7	ΔV margin intended to cover: • Statistical ΔV in TSF and Science Phases • Design and modeling errors for TSMs • Variation across launch period • Mission contingencies
Total ΔV		**853.8**		**853.5**	**Translational ΔV Capability (with margin)**
					ΔV Liens and Encumbrances
TLC		12.0		3.0	Worst case launch dates: GR-A 9/25, GR-B 9/24
TSF		10.0		10.0	Estimate of ΔV variation during TSF Phase
All		25.0		25.0	Contingencies (ΔV for delayed/recovery maneuvers)
Total		**24.4**		**27.7**	**Unallocated Margin**

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- Maneuver Analysis: Troy Goodson, Ram Bhat, Chris Potts
- Mission Planning: Peter Xaypraseuth

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