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Lunar Reconnaissance Orbiter Mission and Spacecraft Design

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Abstract Launched June 18, 2009, with its primary mission scheduled to end September 2010, NASA's Lunar Reconnaissance Orbiter will be the first observatory ever to spend an entire year orbiting and observing the Moon at a low altitude of just 50 km. The spacecraft carries a wide variety of scientific instruments and will provide an extraordinary opportunity to study the lunar landscape at resolutions and over time scales never achieved before. This paper is intended as a companion to the series of papers released simultaneously in this journal detailing LRO's instruments and their planned measurements. The paper describes the design and key performance drivers of the LRO spacecraft and overall mission design. It presents a comprehensive description of the operation of the various systems that comprise the spacecraft and illustrates how these systems enable achievement of the mission requirements.

Keywords Lunar · Spacecraft · Mission design

1 Introduction and Overview

In January 2004 the President of the United States unveiled the Vision for Space Exploration which charted the path for the return of humans to the Moon and deep space. The first milestone in that plan was an unmanned lunar orbiter to be launched in late 2008. By late 2004 that first objective had become the Lunar Reconnaissance Orbiter (LRO) mission and as 2005 began development had started at NASA's Goddard Space Flight Center. As the precursor to the manned flights to the Moon the primary LRO mission objective was to

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collect the data necessary to support the design and development of the systems and missions which would return human beings to the Moon. The LRO mission requirements fulfill three broad objectives:

(1) Identify Safe Landing Sites This is accomplished via precision altimetry and high resolution imaging. The resulting global maps will be anchored with an improved global geodetic grid. The detailed topography and rock abundance information will enable the selection of safe landing sites.
(2) Locate and Characterize Potential Resources The primary focus is on Hydrogen/water at the lunar poles, regions of continuous solar illumination, and the overall mineralogical composition of the Moon.
(3) Characterize the Lunar Space Radiation Environment Principally the energy level and flux intensity of high energy particles, data which will be used to aid in the design of mitigation techniques to protect human explorers.

To meet the mission objectives NASA solicited proposals for instruments that could contribute to them. While the solicitation (NASA Announcment of Opportunity 2004) was fairly detailed in the types of data desired, the scope and top level requirements for the mission were not fully defined until after the selection of the instruments and their associated measurement proposals. The solicitation did not specify specific instrument types but did favor instrument designs with heritage from other planetary instruments in recognition of the aggressive LRO schedule. The suite of LRO instruments consists of six competitively selected instruments and a seventh directed technology demonstration instrument (Mini-RF). The instruments and key aspects of their design and data products are presented in Table 1. For more information on the LRO instruments and their planned measurement investigations, see Chin et al. (2007).

The Orbiter is illustrated in its operational configuration in Figs. 1 and 2 is a photograph of LRO in the open Atlas launch vehicle fairing atop the LCROSS spacecraft prior to final encapsulation. The LCROSS lunar mission (NASA: LCROSS Pages. www.nasa.gov\lcross) was LRO's sister lunar exploration mission and was launched with LRO on the Atlas rocket.

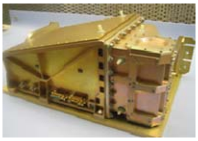
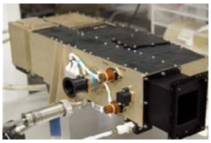
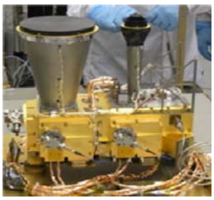
The fundamental mission and spacecraft performance requirements were defined by the requirement to accommodate the selected instruments and provide for the collection and communication of their measurement data products. Together with the programmatic constraints these resulted in a set of key driving requirements which led to the overall design of the mission and spacecraft. These are summarized and discussed below:

Discovery-Class Mission with an Accelerated Development Schedule LRO was a NASA directed, rather than competed, mission and it was stipulated when the project was initiated that a Discovery-class mission was envisioned and it was to be launched before the end of 2008 to support the NASA Exploration Program's requirements. This translated into an approximately $450M budget and four years to develop the mission from concept to launch. These realities influenced all aspects of the LRO design as it was clear that the success or failure of the mission would be judged not only on technical performance but also on meeting the schedule and budget. As NASA's first act in the Exploration Initiative it was imperative the mission was a complete success. Thus the use of available technologies and in-production hardware and an overall simplicity of design became central to the design effort to enable the rapid pace of development.

Orbit and Mission Duration To meet data resolution requirements and ensure data was collected across all seasonal lunar lighting conditions a 50 km polar orbit and a primary

Table 1 LRO instruments

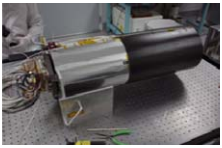
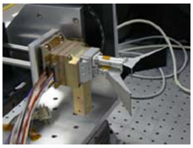
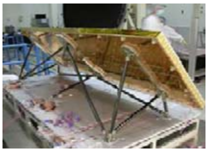
LRO Instrument Summary

Instrument		Primary Measurements	Key Characteristics
CRaTER—Cosmic Ray Telescope for the Effects of Radiation *Boston University/MIT*		• Lunar & Deep Space Radiation Environment	– Measures LET Spectra behind Tissue Equivalent Plastic (TEP) – Nadir FOV 70 deg. – Zenith FOV 35 deg. – Operates autonomously over entire orbit
DLRE—Diviner Lunar Radiometer Experiment *UCLA/JPL*		• 300–500 m resolution scale maps of surface temperature • Characterizes rock abundance and albedo • Silicate mineralogy	– 40–400 K 9 channel radiometer – 21 pixel push-broom line detector – 3.15 km swath – Near copy of MCS on MRO – Operates autonomously over full orbit
LAMP—Lyman-Alpha Mapping Project *Southwest Research Institute*		• Maps of frosts and landforms in permanently shadowed regions using Lyα albedo	– 465–1880 Å pass-band – 0.3 × 6.1 deg. Slit – 1.8 Å full slit spectral resolution – Near copy of ALICE on New Horizons – Operates primarily on night side of orbit
LEND—Lunar Exploration Neutron Detector *Russian Space Agency*		• Maps Hydrogen in upper 1 m of regolith at 10 km scale using neutron albedo	– Collimated neutron telescope – Measures and differentiates thermal, epithermal, and energetic neutrons – Derived from HEND on Mars Odyssey – Operates autonomously overfull orbit
LOLA—Lunar Orbiter Laser Altimeter *Goddard Space Flight Center*		• Global geodetic Topography with 10 cm vertical and 1 km (equator) to 25 m (poles) horizontal resolution • Characterize slopes, roughness, and brightness	– 1064 nm laser pulsed at 28 Hz – Beam split into 50 m/5 spot pattern – 4.4B measurements/year – Operates autonomously over full orbit

mission duration of 1 year was required. This in turn was a design driver for the spacecraft propulsion system design and reliability requirements.

Large On-board ∇V Capability In order to perform the necessary Lunar Orbit Insertion (LOI) and the subsequent regular orbit adjustments a large propulsion system was required.

Table 1 (*Continued*)

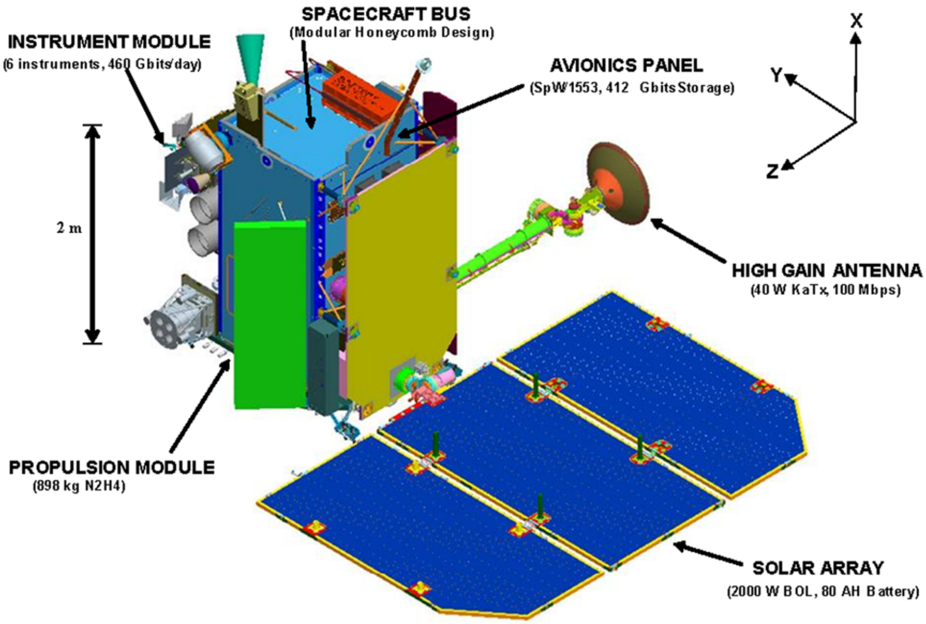
LRO Instrument Summary			
Instrument		Primary Measurements	Key Characteristics
LROC—Lunar Reconnaissance Orbiter Camera **NAC**—Narrow Angle Camera (2) **WAC**—Wide Angle Camera *ASU/MSSS*	 	• Target imagery at 0.5 m resolution at poles and selected sites. • Global imagery in visible and UV at 100 m resolution • Characterize lighting conditions • Multispectral imagery maps ilmenite and other minerals	<u>NACs:</u> – f3.59 Cassegrain (Ritchey-Chretien) – FL = 700 mm – FOV 2.86 deg per NAC – Push-broom line imager <u>WAC:</u> – f/5.1(vis)/f/5.5(UV), – FL = 6 mm (vis)/4.7 mm (UV) – FOV = 90 deg (vis)/60 deg (UV) – Push-broom frame imager – LROC is timeline driven and operates on day side of every orbit
Mini-RF—Technology Demo of Small Synthetic Aperture radar (SAR) *DOD/NAWC*		• Radar imagery and interferometry • Flown as demo • Can contribute to topography and resource identification	– X & S band radar – 15–150 m resolution (mode dependent) – Timeline driven when opportunities are available. Can operate over entire orbit

Almost half of LRO's mass at launch was fuel. Accommodation of the large fuel tanks was defining characteristic of the overall spacecraft physical configuration.

Near Continuous Nadir Pointed Operation To fulfill the science data product requirements LRO must be operated in a nadir pointed science operation mode at least 95% of the time during the year. This necessitated designing relatively complex continuously articulated antenna and solar array systems.

High Rate Data Downlink and Large On-board Data Storage The high resolution imagery data products required a high data downlink rate (100 Mb/s) and high capacity on-board data storage to store data when the link is not available. Meeting this requirement led to the selection of a high bandwidth Ka-band science data communication system paired with a new dedicated ground station and a large capacity on-board solid state recorder integrated into the avionics.

Capability for Extended Mission The LRO primary mission requirement was 1 year, as noted above, but in addition LRO was required to have the capability of performing an extended mission of up to 3 additional years. This requirement had to be designed into the consumables budget. More significantly, this required LRO to be designed to survive the most severe lunar eclipses although they did not occur during the primary mission. These



LRO Orbiter Characteristics		
Mass (CBE)	1916 kg	Dry: 1018 kg, Fuel: 898 kg (1313 m/sec)
Orbit Average Bus Power	647 W @ Beta 0	
Data Volume, Max Downlink rate	461 Gb/day, 100 Mb/sec	
Pointing Accuracy, Knowledge	60, 30 arc-sec	

Fig. 1 Lunar Reconnaissance Orbiter—deployed configuration with key attributes given

eclipse periods became the worst-case design points for the spacecraft power and thermal systems.

Harsh Lunar Thermal Environment Designing a mission for a long-duration, low-altitude lunar mission with the added constraint that spacecraft must essentially point the instruments nadir continuously proved to be more challenging from a thermal control standpoint that initially anticipated. This reality manifested itself during the design and development in two areas: the overall configuration of the spacecraft which was dominated by the need to radiate almost all heat from a zenith face of the spacecraft (discussed in more detail in the thermal section that follows) and the extensive efforts required to adapt the heritage instrument designs for the environment.

This paper is intended as a companion to the series of papers published simultaneously in this journal which describe each of the LRO instruments and planned investigations in detail. It provides a detailed presentation of the LRO design with the goal, together with the instrument papers, of enabling a comprehensive understanding of how LRO functions and its capabilities. The remainder of this paper is divided into two main sections, the first presenting an overview of the mission design and operation, and the second presenting a description of the design and functional operation of the LRO spacecraft.

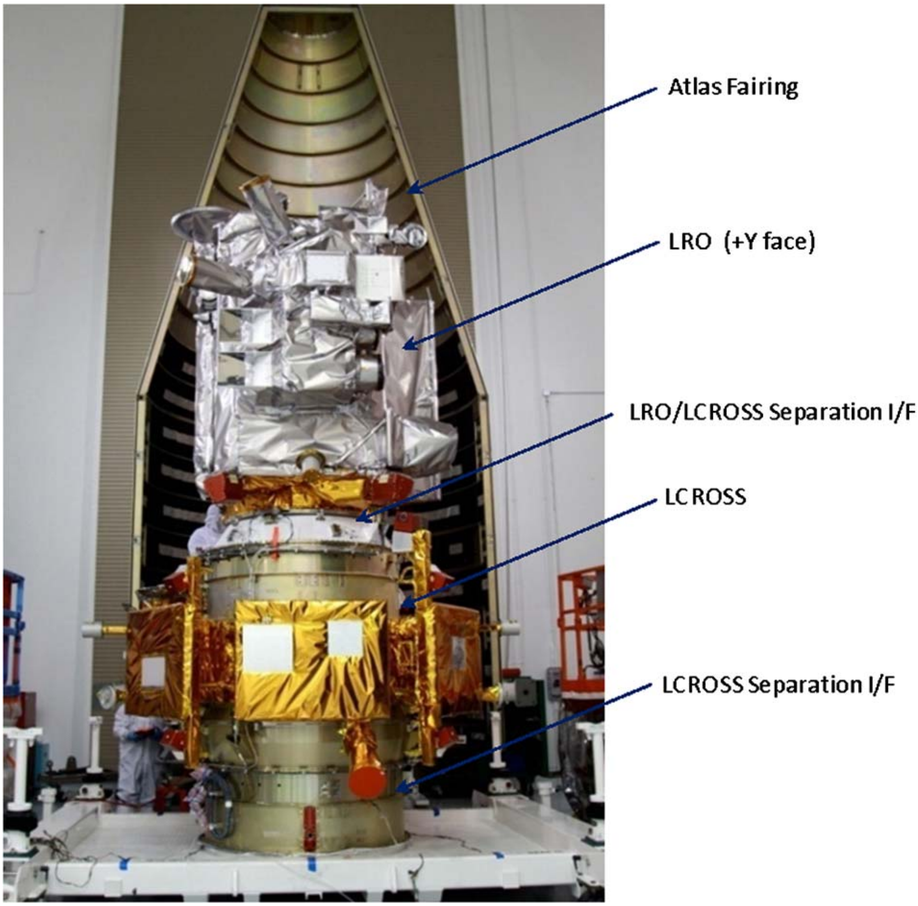


Fig. 2 LRO in Atlas Fairing with LCROSS Spacecraft

2 Mission and Operations Design

2.1 Baseline Mission

LRO was launched on an Atlas V 401 Evolved Expendable Launch Vehicle (EELV) from the Kennedy Space Center (KSC), located on the east coast of Florida, USA at 5:32 pm EDT June 18, 2009. It followed a minimum energy transfer trajectory to the Moon (see Fig. 3), taking four and a half days to complete the journey.

Once in the vicinity of the Moon, LRO began a sequence of Lunar Orbit Insertion (LOI) maneuvers, with the first maneuver (capture) nominally lasting roughly 40 minutes. Subsequent LOI maneuvers were performed over the next several days (see Fig. 4), culminating in a low maintenance, 30×216 km quasi-frozen orbit (Folta and Quinn 2006) that LRO used during its 60 day commissioning period (see Fig. 5). After completing its commissioning activities on September 15, 2009, LRO moved to its nominal 50 km polar mapping orbit (Fig. 6), where it will remain for a minimum of 1 year, collecting data over the entire lunar surface under all possible lighting conditions.

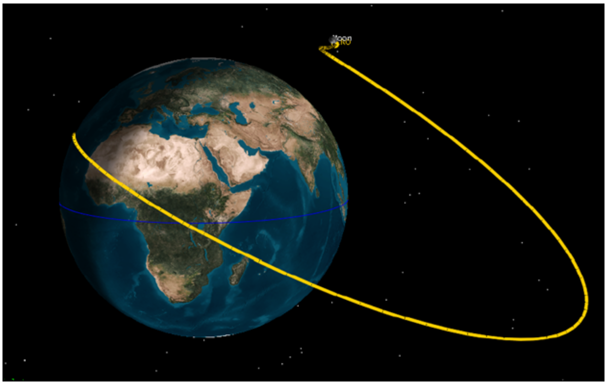


Fig. 3 Minimum energy transfer to the Moon (4–5 days)

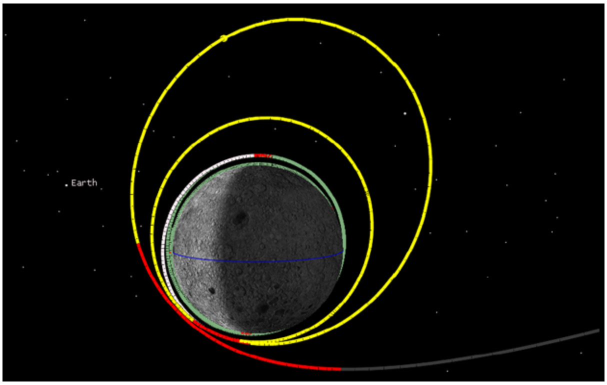


Fig. 4 Lunar orbit insertion sequence (4–6 days)

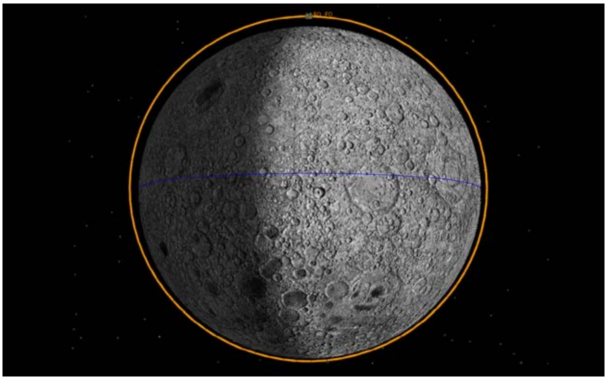


Fig. 5 30 × 216 km quasi-frozen orbit (up to 60 days)

2.2 Launch Windows

LRO launch window selection was primarily driven by the seasonal lighting conditions on the lunar surface and the desire to definitively identify any permanently lit or permanently shadowed areas near the lunar poles. To achieve these goals, LRO needed to be oriented in a particular orbit plane relative to the solar cycle, to maximize observation of the most extreme polar lighting conditions. Specifically, the LRO orbit plane must be oriented such

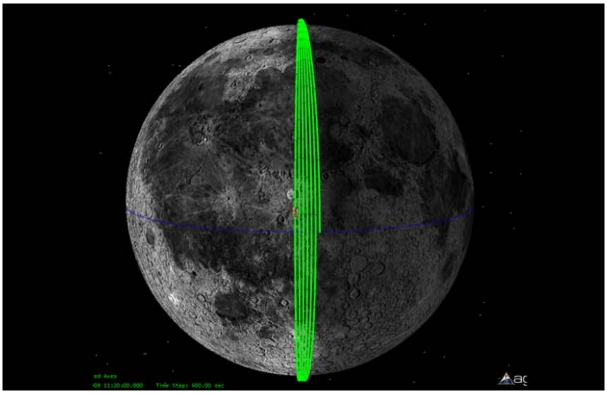


Fig. 6 50 km polar mapping orbit (at least 1 year)

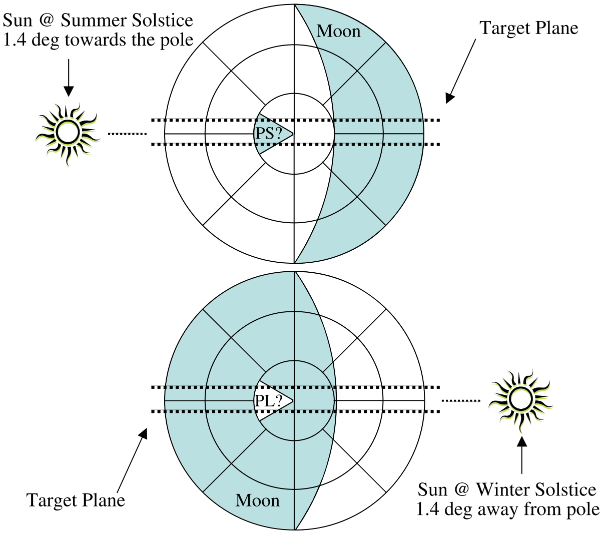


Fig. 7 The search for permanent light (PL) and shadow (PS) regions

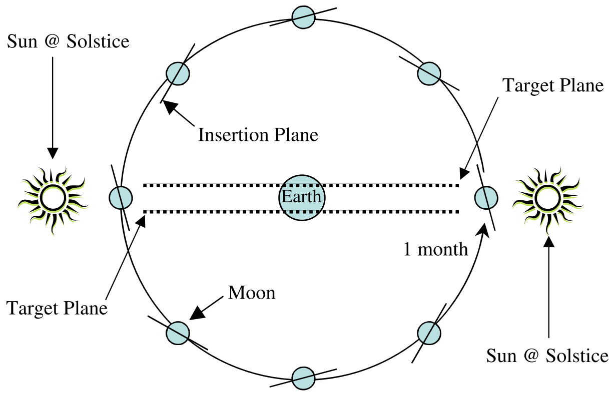
that it is near edge-on to the Sun (0 deg beta-Sun angle) during the lunar solstice periods (see Fig. 7).

At insertion (LOI), LRO's orbit plane is fixed with respect to the Earth (∼85 deg beta-Earth angle), regardless of the relative geometry between the Earth, Moon, and Sun (see Fig. 8). Therefore, since the lighting constraint drives the target plane to a *particular inertial* orientation (low beta-Sun angle at the solstices), it constrains the Earth departure (launch) to be within a few days of the point at which the natural insertion plane coincides with the inertial target plane. Forcing the difference between the two to be below 20 deg constrains the launch window to 2–3 day periods every two weeks (see Fig. 9 for examples of launch windows).

2.3 Lunar Orbit Insertion

LRO's launch enabled it to reach the Moon in roughly four and a half days, at which time a successful Lunar Orbit Insertion maneuver (LOI-1) was required to capture into a stable orbit around the Moon. The LOI-1 burn and the subsequent LOI maneuvers were mission

Fig. 8 LRO insertion plane relative to Earth and target plane



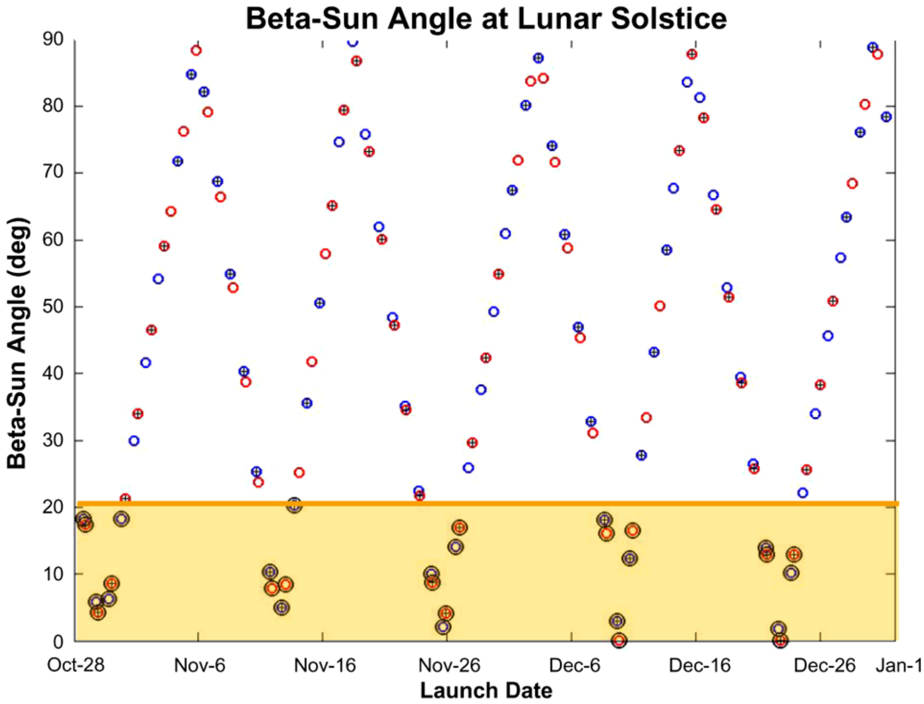


Fig. 9 LRO launch windows; 2–3 days every 2 weeks meet the beta-sun angle requirement

critical, and many steps were taken to ensure their flexibility and probability of success. In addition, these maneuvers determined the fuel budget for the entire LRO mission.

The first of these maneuvers (LOI-1) provided the necessary change in velocity (∇V) to allow LRO to be captured by the Moon's gravitational field. The magnitude of this maneuver is a function of the exact geometry of the transfer trajectory and the allocation of an adequate amount of fuel to the LOI-1 maneuver in turn establishes the fuel budget for the mission (see Table 2) (Beckman 2007). Although LRO's launch vehicle imposed a 2000 kg mass limit on LRO, the orbiter carried a ∇V requirement of 1270 m/s and had a maximum fuel load of

Table 2 LRO ∇V and fuel budget pre-launch

Mission plan	ΔV (m/s)	Fuel (kg)	Source
Mid-Course Correction	30	28.6	3σ LV errors
LOI-1 (with Checkout)	591	454.5	Deterministic
LOI-2 thru LOI-4	362	226.3	Deterministic
Mission Orbit Insertion	56	31.9	Deterministic
Orbit Station-Keeping	162	91.5	Deterministic
Ext. Mission/Margin	69	33.1	3 yrs Frozen
Momentum Unloading	–	17.0	4 years Total
Other (Residuals)	–	15.4	Conservative
Total	1270	898	

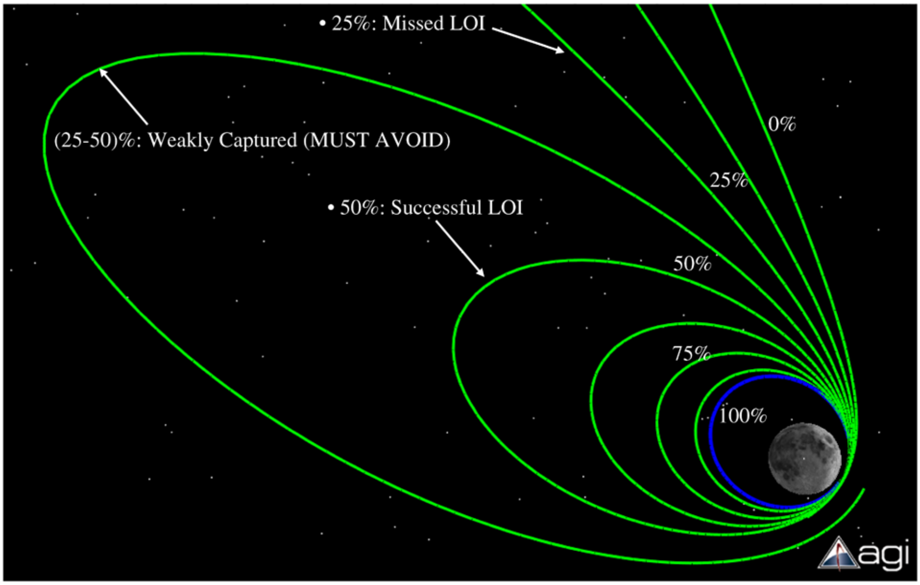


Fig. 10 Lunar Orbit Insertion (LOI) orbits shown as a function of percent completion of LOI maneuver

895 kg (hydrazine). The maximum fuel load was set by the capability of the available tanks LRO was designing with. The required ∇V, in combination with the expected effective specific impulse (Isp) of LRO's propulsion system (212.2 s), set the maximum allowable total launch mass for LRO at 1965 kg. LRO's final measured mass was 1915 kg, with the difference contributing to a significant increase in reserve ∇V.

Many measures were implemented to ensure the success of this critical maneuver. Firstly, the burn (lasting approximately 40 minutes), was designed such that only 50% of the ∇V would be required to capture into orbit (see Fig. 10). In addition, it was designed such that capture could be achieved either through thrust or duration, defending against possible failures in the propulsion system, as well as possible interruptions caused by software and/or processor faults.

The propulsion system, described in a following section, is fully redundant with two banks of two 88 N insertion thrusters and two banks of four 20 N attitude control thrusters. The insertion thrusters deliver a total force of ∼350 N, twice the required thrust needed for lunar capture. In the event of any thruster failure, either of the insertion thruster banks could have been used with either of the attitude thruster banks in order to successfully execute an insertion maneuver. With half the thrust, the burn duration would slightly more than double (due to increased finite burn losses), but the resulting orbit would be stable, and the mission could continue.

Similarly, at full thrust, a 40 minute burn duration is twice what is needed for a successful lunar capture. If the burn was interrupted for less than 20 minutes, for any reason, the resulting orbit would still be stable, and LRO would be able to continue with its mission. Restart procedures were created for all conceivable scenarios, and the LRO operations team was thoroughly trained in executing these procedures.

In the event of a catastrophic failure or interruption (one that results in less than 25% of the required ∇V), LRO would not be captured by the Moon's gravity (see Fig. 10), but would have one final chance of getting into lunar orbit. It would have required a deep space maneuver within 10 days of the first lunar encounter, used to target a second lunar encounter about 90 days later (see Fig. 11). This would have depleted significant amounts of fuel, but LRO would still have been able to get into lunar orbit, albeit with limited options in terms of orbits and durations. It would have been able to fly in a higher (∼215 km), circular orbit for up to a year, or go into and stay in the low maintenance, 30 × 216 km, quasi-frozen orbit originally meant for commissioning for up to 3 years.

Finally, if, as a result of failures and/or interruptions, the delivered ∇V was in the 25–50% range, LRO would have been weakly captured by the Moon's gravity. This would have resulted in chaotic behavior and have had an irrecoverable impact on LRO's orbital inclination. If this happened, LRO would not have been able to achieve any sort of polar lunar orbit, and its primary objectives could not be met. This catastrophic threat factored into all contingency planning, and was a prime focus of early mission rehearsals and training.

LRO successfully completed LOI maneuvers, without the necessity of exercising any contingency procedures, in late June 2009 and achieved the planned commissioning orbit. Commissioning took roughly 60 days, and LRO was then transitioned into the primary mission 50 km polar mapping orbit on September 15, 2009.

2.4 On-Orbit Operations

Once in orbit around the Moon, LRO's universe became essentially Moon-centered (see Fig. 12). From this perspective, the Earth circumnavigates the Moon once a month, and the Sun circumnavigates the Moon once a year. During normal operations, the $+Z$ axis of the Orbiter is pointed toward the Moon, with the Sun remaining on the $-Y$ side of the X–Z plane. The solar arrays track the Sun in two axes, and the high-gain antenna points toward Earth whenever it is in view. LRO's orbit has a mean period of 113 minutes and a maximum eclipse time of 48 minutes, occurring at 0° Sun-beta angle. As previously discussed, LRO's orbit is targeted so that the lunar solstices occur near these 0° Sun-beta angle periods. These also mark the points at which LRO executes a 180° yaw flip to keep the Sun on the solar-array-side ($-Y$) of the spacecraft.

LRO's orbit will be in full view of the Earth for ∼2 days twice per month, and in full view of the Sun for ∼1 month twice per year. LRO's momentum management and station-keeping maneuvers are executed while the orbit is in full view of the Earth.

LRO will make use of a global network of S-band ground stations for nominal spacecraft tracking (at least 30 minutes per orbit) and one Ka-band station for downlink of all the stored

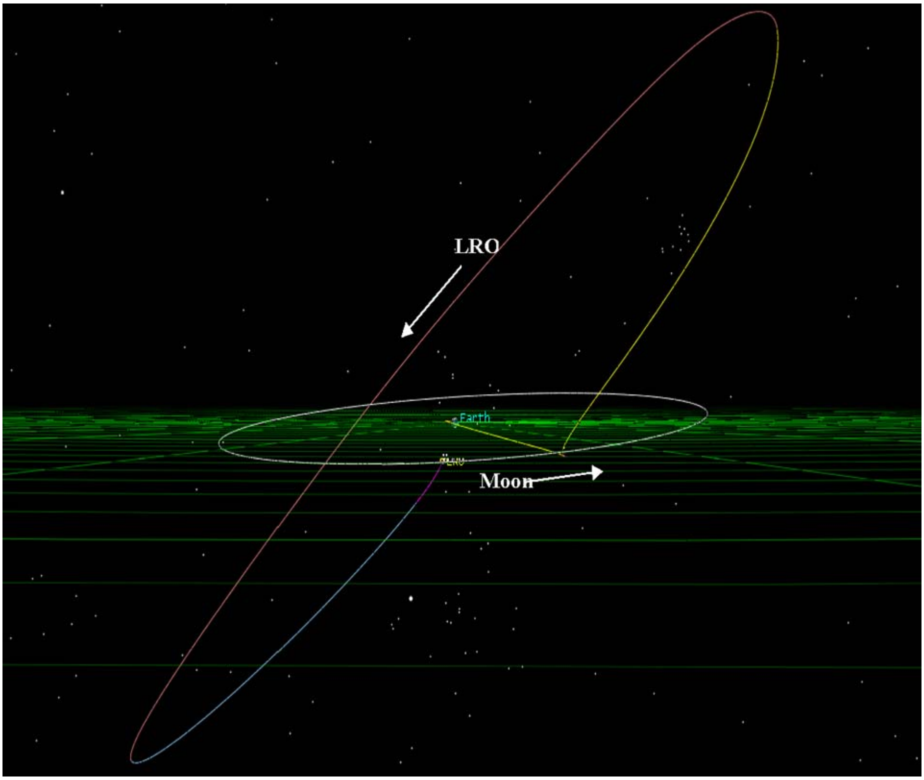


Fig. 11 Deep space recovery trajectory in the event of a missed LOI or less than 25% complete LOI propulsion burn

instrument and spacecraft data. Nominally, LRO will never be out of contact with the ground for more than one hour at a time. A depiction of the LRO ground system is given in Fig. 13.

A snapshot of the nominal on-orbit operations over 3 different time scales is shown in 14. Station-keeping maneuvers and instrument calibrations occur once a month. Momentum management maneuvers occur every 2 weeks. There is an S-band pass every orbit (12 per day), and 4 (on average) Ka-band passes every day. Most of the instruments operate autonomously over the course of a single orbit, while two (LROC and Mini-RF) require daily tailored command timelines. Nominally, LRO will receive a new command timeline from the ground once per day.

2.5 Data Downlink

On a given day, about 460 Gbits of data are generated on-board the LRO spacecraft (see Table 3). Data is downlinked at 100 Mbps through a single Ka-band ground station at White Sands, New Mexico, USA (designated WS1). On average, there are four passes between LRO and WS1 every day, each lasting 45 minutes, but the actual number fluctuates between 2 and 6, as the Moon moves through its entire declination range each month (as seen from the Earth). Table 4 shows the effect that this has on the Ka-band link utilization. Even in the worst case (2 passes), there is sufficient time to downlink the entire day's data volume.

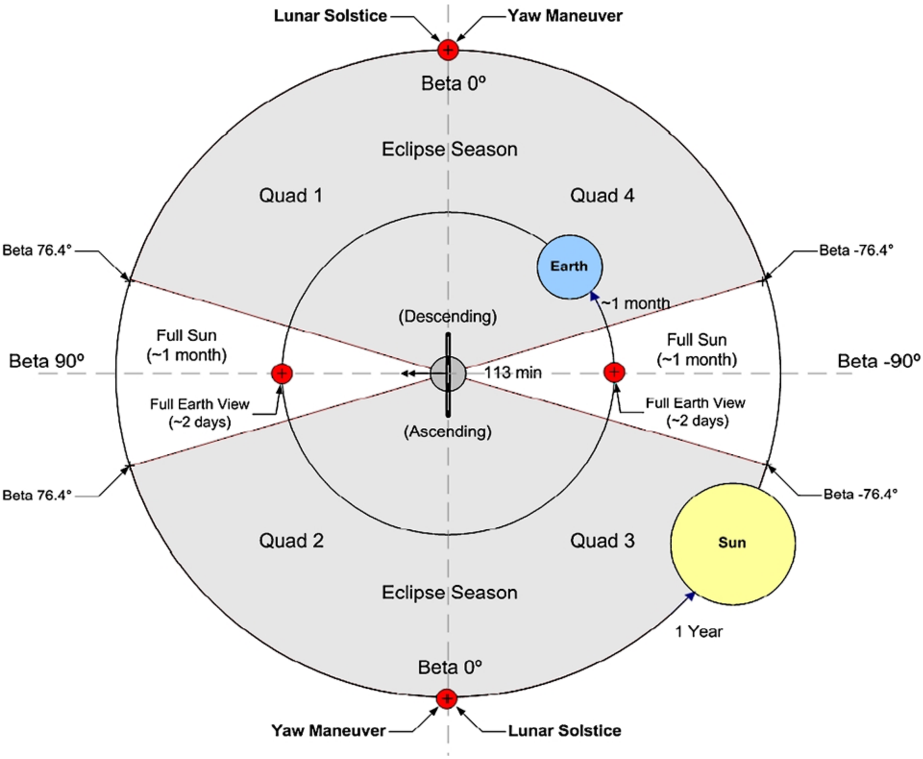


Fig. 12 "Moon Centered Universe" provides useful diagram for depicting the relationship of the LRO orbit to the Earth and Sun

Table 3 Breakdown of LRO's daily data volume

LRO Data Volume Budget				
Type	Data per Orbit (Mbits)	Data per Day (Gbits)	Files per Orbit	Files per Day
CRaTER	610.20	7.78	2.00	25.49
Diviner	180.52	2.30	22.60	288.00
LAMP	168.22	2.14	2.00	25.49
LEND	20.52	0.26	2.00	25.49
LOLA	226.29	2.88	2.00	25.49
LROC	34,659.16	441.67	28.00	356.81
Spacecraft Housekeeping	379.65	4.84	2.00	25.49
Total (Gbits):	36.24	461.87	61	772

2.6 Lunar Orbit Station Keeping

Lunar orbits can be characterized by the evolution of their eccentricity and argument of periapsis over time. The Moon's non-uniform gravitational field causes significant perturbations to these two parameters (Beckman 2007). Figure 15 illustrates the evolution of these parameters, over time, for LRO's two main lunar orbits (the 30×216 km quasi-frozen com-

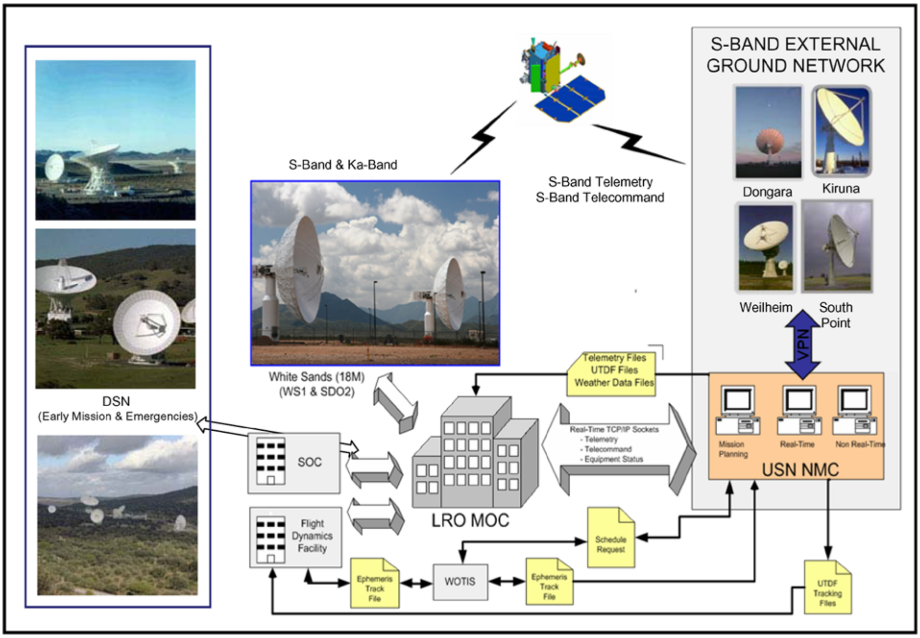


Fig. 13 Representation of the LRO Ground System showing relationship between Mission Operations Center (MOC) and Flight Dynamics Facility at NASA GSFC, the ground stations, and the Instrument Science Operations Centers (SOC). Note only one of the seven SOCs is shown

Table 4 Ka-band downlink utilization

Ka-band Dowlink Utilization*					
Passes	2	3	4	5	6
No.	Pass Utilization minutes)				
1	45.0	45.0	45.0	45.0	45.0
2	40.1	33.4	26.7	20.0	13.3
3	–	6.7	6.7	6.7	6.7
4	–	–	6.7	6.7	6.7
5	–	–	–	6.7	6.7
6	–	–	–	–	6.7
Used	94.6%	63.0%	47.3%	37.8%	31.5%
Margin	5.4%	37.0%	42.7%	62.2%	68.5%

*Based on D/L rate of 100 Mbps—10% overhead

missioning orbit and the 50 km polar mapping mission orbit). The quasi-frozen orbit shows virtually no secular growth in eccentricity or argument of periapsis over time. The altitude is bounded, and the periselene remains over the South Pole. This orbit requires very little station-keeping fuel (about 5 m/s per year).

In contrast, the 50 km mission orbit shows significant evolution in eccentricity and argument of periapsis from month to month. In Fig. 15 this is most clearly seen in the increasing eccentricity as the orbit evolves. If left uncorrected, these perturbations would cause LRO

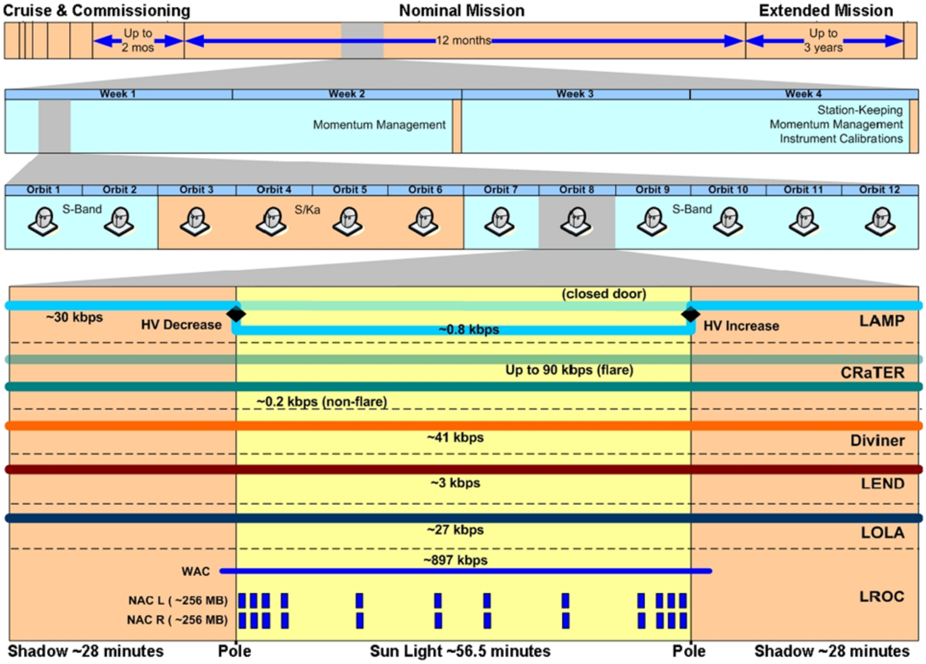
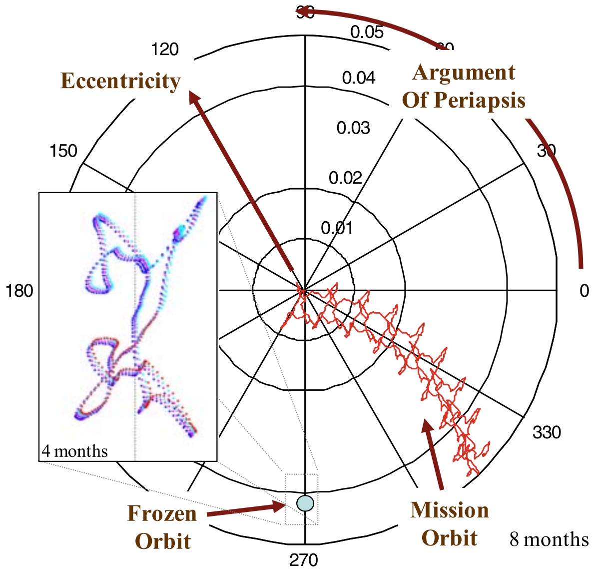


Fig. 14 Typical LRO on-orbit operations

Fig. 15 Evolution of eccentricity and argument of periapsis



to hit the lunar surface within roughly 68 days. The variation in altitude with and without station-keeping is shown in Fig. 16. The LRO station-keeping strategy makes use of the repeating pattern that can be seen in the eccentricity and argument of periapsis parameters (see Fig. 17). The goal is to precisely reset the pattern at the end of each month, so that the

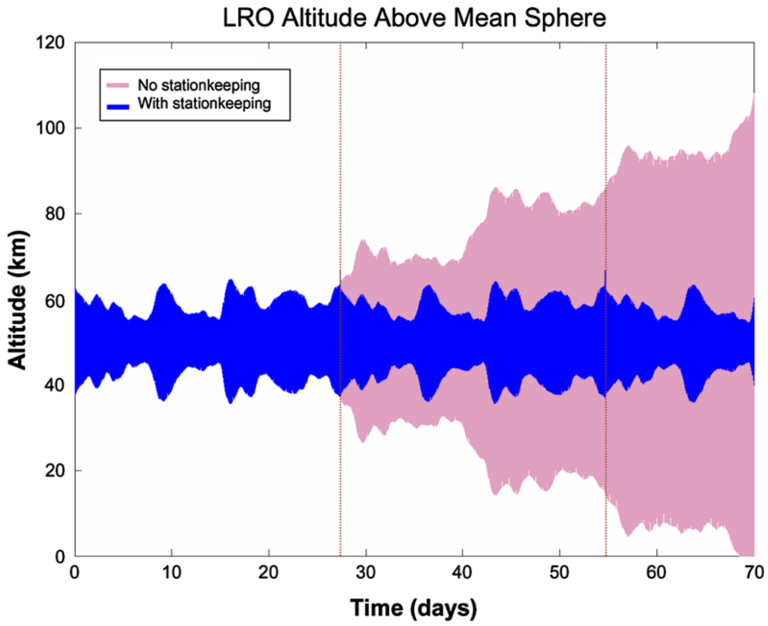


Fig. 16 Simulated LRO mission orbit altitude variation with and without station-keeping, thus showing the necessity of the monthly station-keeping maneuvers

evolution is bounded. This is accomplished with a 2-burn sequence that first circularizes the orbit and then de-circularizes it in the proper direction so as to center the repeating pattern around zero eccentricity. These maneuvers keep LRO's altitude within 15 km of the target 50 km orbital altitude and a safe distance from the lunar surface. This is an unavoidably fuel intensive procedure, consuming approximately 150 m/s of fuel per year.

2.7 Lunar Eclipses

Looking back at Fig. 12, it is clear that the Earth will pass between the Sun and the Moon every month. Although Earth will usually pass far enough above or below the Sun-Moon line that its shadow will not fall on the Moon, twice a year (on average) it will pass close enough to the line that it will cast a significant shadow on the Moon. These lunar eclipses vary in severity over a cycle of roughly four years; with a peak occurring in the middle of 2011 (see Table 5). For descriptive purposes, these eclipses have been labeled as Type 1 (Moon partly in the penumbra only), Type 2 (Moon fully in the penumbra), Type 3 (Moon partly in the umbra), and Type 4 (Moon fully in the umbra), based on existing data like that shown in Fig. 18 (NASA: Eclipse Pages. sunearth.gsfc.nasa.gov/eclipse).

In order to assess the effects of these lunar eclipses on the LRO spacecraft, geometric models were developed to estimate the amount of solar input that would be received during each event (see Fig. 19). These models show LRO coiling through space as it orbits the Moon, while it (with the Moon) passes through the Earth's shadow. These results were used to size the spacecraft battery and design the power management schemes to ensure LRO

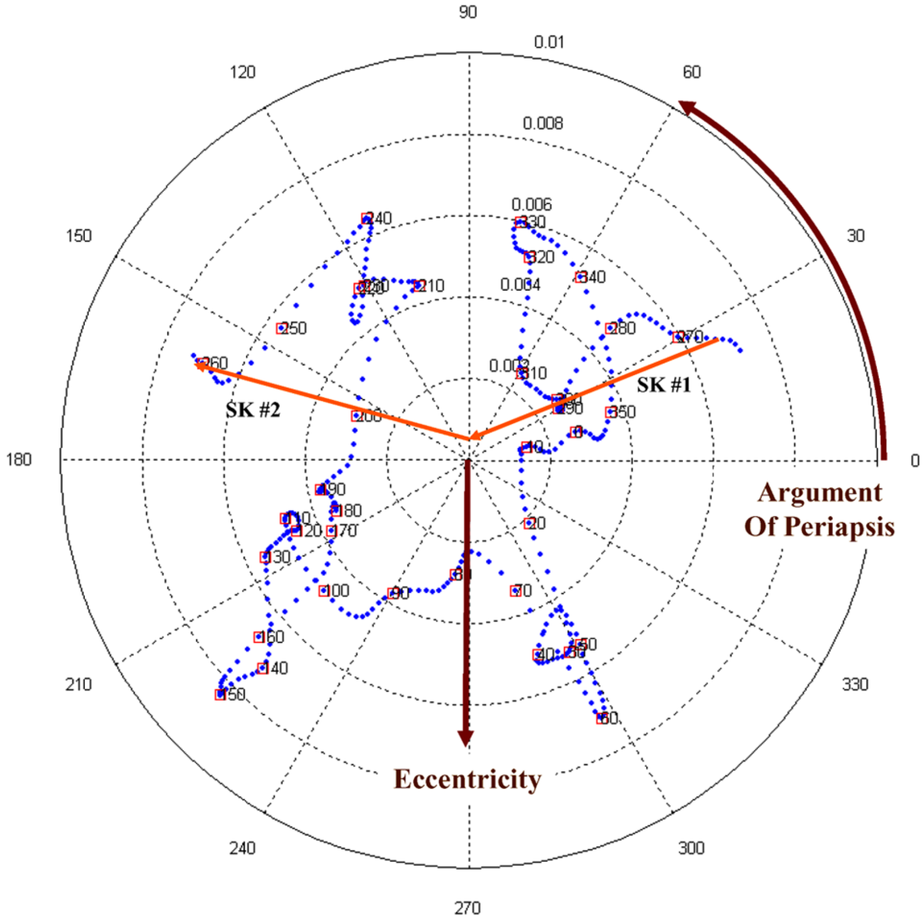


Fig. 17 Mission orbit station-keeping strategy—each month two station-keeping (SK1 & SK2) maneuvers reset the perturbation driven evolution of the orbit eccentricity and argument of periapsis to maintain the orbit at 50 km ±15 km

can survive the eclipses. During its nominal mission, LRO will encounter Type 1, 2, and 3 lunar eclipses. The worst of these will occur in June 2010 resulting in a worst case battery depth of discharge of ∼30%, which is within the normal operating range for the spacecraft. During the potential extended mission timeframe, lunar eclipses become increasingly severe culminating with the worst of the cycle in June 2011. This event will result in a worst case depth of discharge of almost 60% and in addition requires the spacecraft to undergo a preheat operation prior to entering the eclipse in order to survive. The spacecraft was designed and qualified to survive this worst case event.

2.8 Extended Mission

Once its one year nominal mission is complete, LRO will, by design, have a significant amount of fuel remaining. Fuel to provide 65 m/s was allocated in the fuel budget for an extended mission and an additional 165 m/s worth of fuel was gained largely due to the

Table 5 Tabulation of lunar eclipses during LRO mission timeframe

Lunar Eclipses: 2009–2013				
Date	Type	Penum.	Partial	Total
2009 Feb 09	(2)	4:03	–	–
2009 Jul 07	(1)	2:12	–	–
2009 Aug 06	(1)	3:16	–	–
2009 Dec 31	(2)	4:15	1:02	–
2010 Jun 26	(3)	5:26	2:44	–
2010 Dec 21	(4)	5:38	3:29	1:13
2011 Jun 15	(4)	5:39	3:40	1:41
2011 Dec 10	(4)	6:00	3:33	0:52
2012 Jun 04	(3)	4:33	2:08	–
2012 Nov 28	(2)	4:41	–	–
2013 Apr 25	(2)	4:12	0:32	–
2013 May 25	(1)	0:54	–	–
2013 Oct 18	(2)	4:04	–	–

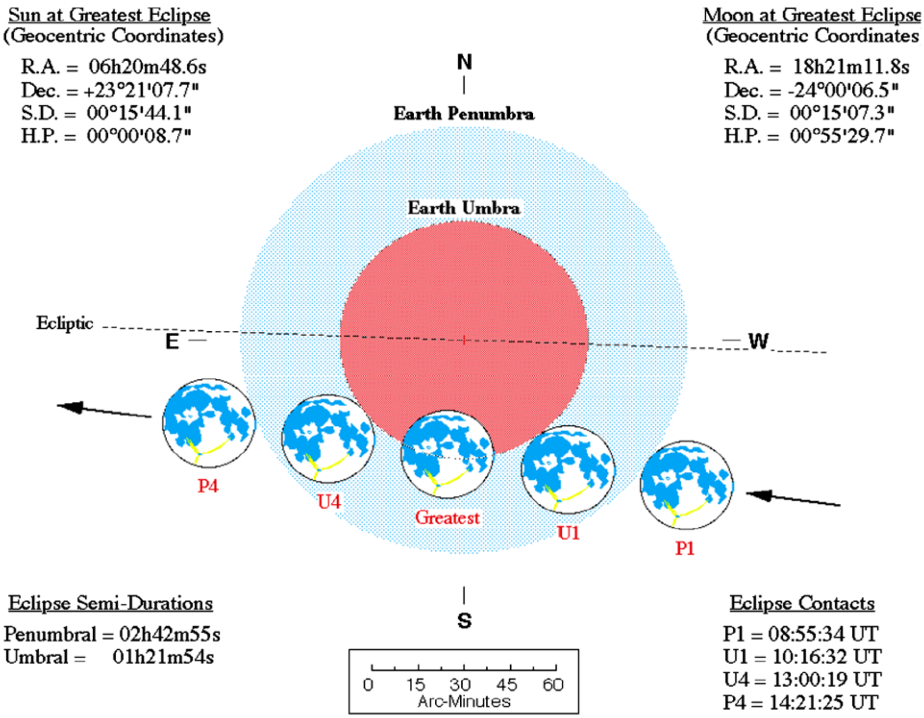


Fig. 18 Example eclipse data—June 26, 2010 Lunar Eclipse

positive mass margin at launch, better-than-budgeted thruster performance, and the very accurate trans-lunar injection by the launch vehicle (thus requiring very little correction). The extended mission is likely take on one of two forms. With the currently projected fuel reserve LRO could remain in the nominal 50 km mission orbit for an additional 18 months.

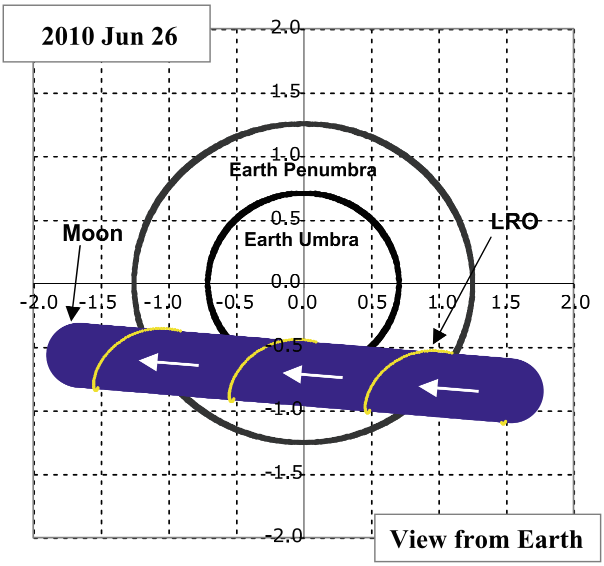


Fig. 19 Geometric model of 26-Jun-2010 Lunar Eclipse showing LRO's path (*spiral path*) about the Moon as it passes through the eclipse. This model was used to design the LRO power and thermal systems to survive the eclipses

Alternatively, that fuel could be used to transition back to the 30 × 216 km quasi-frozen commissioning orbit (∼50 m/s) and remain there for well over 10 years (5 m/s per year), although this is well beyond the design life of the spacecraft. At present NASA has committed to an extended LRO science mission of at least 2 additional years. The desired orbit has not yet been determined.

Regardless of the extended mission selected, LRO will eventually impact the lunar surface. Once its fuel is depleted, it will no longer be able to maintain its orbit, and the perturbations caused by the non-uniform gravitational field (even in the frozen orbit) will eventually result in an impact.

3 Spacecraft Design

LRO's spacecraft design was driven by the very aggressive schedule, diverse suite of seven instruments selected for the mission, and unique challenges of a long duration, low altitude lunar mission. These key drivers manifested themselves in LRO's highly modular physical design. Key aspects of the modularity are listed below:

- Propulsion module includes all plumbing and thermal control, enabling independent assembly and proof testing.
- Propulsion module includes two available from the NASA X-38 program, enabling early assembly of propulsion module.
- Spacecraft avionics attached to a single panel, containing embedded heat pipes, enabling a table-top integration and thermal coupling of most heat sources.
- Optical instruments mounted on a separate instrument module, composed of aluminum honeycomb panels with graphite-composite face skin, decoupling high-precision pointing instruments from the rest of the structure.
- Single, zenith-facing radiator, decoupled from large swings in lunar heat load and coupled to most of the structure to provide a large thermal mass to damp variations induced by the environment.

The Orbiter is shown in Figs. 1 and 2. Figure 20 shows an expanded representation of the Orbiter which shows the major components and illustrates the modularity of the spacecraft. Figure 21 shows multiple views of the actual Orbiter to enable key components to be seen. In response to both schedule and cost constraints, GSFC drew upon highly successful in-house experience with rapidly executed single-string design missions and designed LRO as a high-reliability largely single-string spacecraft (Lee 2007). The LRO system block diagram is shown in Fig. 22 which serves as a roadmap for the descriptions of the major subsystems that follow.

3.1 Command and Data Handling Subsystem

The LRO Command and Data Handling (C&DH) subsystem is the centerpiece of the LRO control and data flow architecture. The C&DH subsystem is housed in a single enclosure, see Fig. 23, and includes a Single Board Computer (SBC), a data storage system, a SpaceWire interface (SpW) (ESA Standard 2008) which serves as a high speed connection for some of the instruments and C&DH dataflows, a MIL-STD-1553B low-speed bus controller (Department of Defense Interface Standard for Digital Time Division Command/Response Multiplex Data Bus 1978), interface card to the communication subsystem, analog telemetry conditioning cards and two ultra-stable oscillators (USO). Each of these has heritage from previous NASA and/or DOD missions (Nguyen et al. 2008).

The primary functions of the C&DH system are:

- Hosting the Guidance, Navigation and Control (GN&C) and C&DH Flight Software (FSW)
- Command and data handling functions (command acceptance and distribution, telemetry collection, science data storage) for the instruments
- Telemetry, command and control interface for spacecraft subsystems
- Interface to the spacecraft communications transmitter and transponder for high speed telemetry
- Science data formatting

A block diagram of the C&DH system and its relationships to adjacent systems and instruments is shown in Fig. 23.

The LRO C&DH consists of seven major the elements which are briefly described below.

3.1.1 Single Board Computer (SBC)

LRO's Single Board Computer (SBC) is the processing platform for the Flight Software (FSW). It utilizes the BAE RAD750 133 MegaHertz (MHz) processor and carries 36 Megabytes (MB) of Static Random Access Memory (SRAM) storage to store the executable code, and provide backup data storage. In addition, it provides 64 KB of Start-Up Read Only Memory (SUROM) storage for Boot Code and 4 MB of Electrically Erasable Programmable Read Only Memory (EEPROM) of storage for Application Code. The spacecraft's MIL-STD-1553B Low Speed Bus (LSB) used to communicate with the majority of subsystems and instruments is integrated into the SBC. The Single Board Computer supports SpaceWire interfaces to communicate with LROC, Mini-RF, the HK/IO card, and the Communication cards. It communicates with the Data Storage Board (DSB) cards via a Compact Peripheral Component Interface (cPCI) interface on the backplane

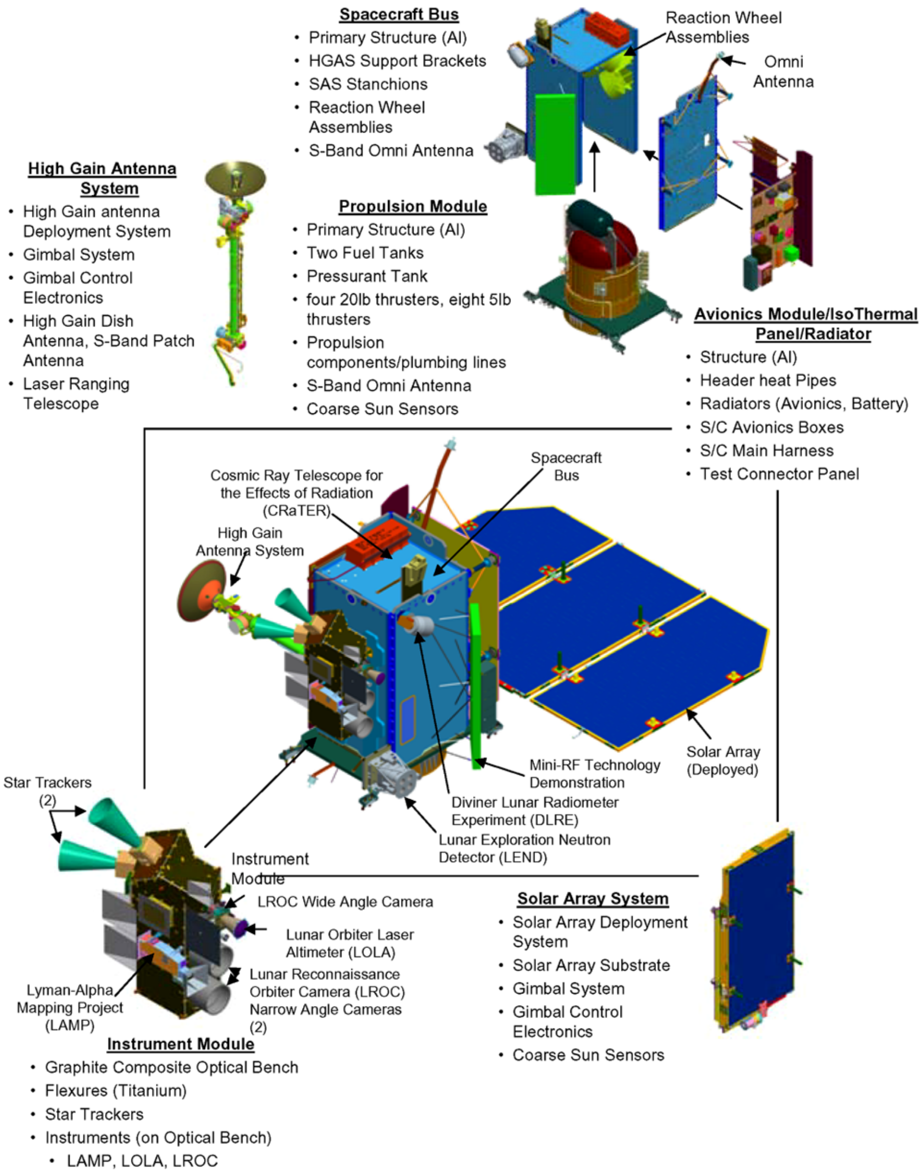


Fig. 20 LRO design overview showing modular construction

3.1.2 Communication Cards (Ka-Comm & S-Comm)

LRO's C&DH subsystem includes two communication cards, the Ka-Comm and S-Comm, which enable spacecraft communications with the ground.

The Ka-Comm card is designed to accommodate a high-speed telemetry interface to Earth using the Ka-Band frequencies. It is connected to the SBC via a SpaceWire link.

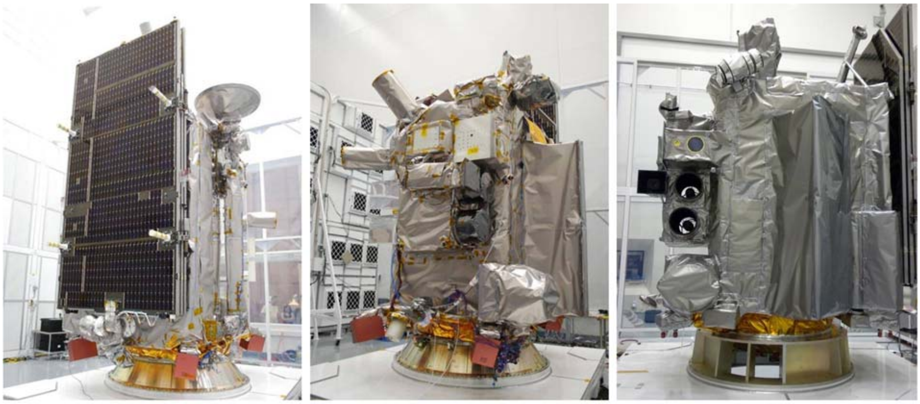


Fig. 21 LRO in launch configuration

During a Ka-Band pass, the data from the SBC flows directly into the Ka-Comm at the SpaceWire link rate of 132 Mega bits per second (Mbps), where the link rate equals the data rate plus the SpaceWire overhead. The data rate is a maximum 100 Mbps. The data is encoded for transmission per Consultative Committee for Space Data Systems (CCSDS) (Consultative Committee for Space Data Systems (CCSDS). http://public.ccsds.org/) recommendations for telemetry using concatenated encoding. Prior to sending on to the Ka-band system for modulation onto the RF carrier the data stream is also split into two streams for Offset Quadrature Phase Shift Keying (OQPSK) modulation.

The S-Comm card is designed to accommodate a telemetry interface to Earth using S-Band frequencies. The S-Comm card is connected directly to the SBC via a 10 Mbps SpaceWire link. During a ground station pass the data from the SBC flows directly into the S-Comm card and is encoded for transmission per CCSDS recommendations for telemetry encoding using concatenated encoding. The S-Comm provides one telemetry stream at up to 1.093 Mbps to the transmitter for Bi-Phase Shift Keying (BPSK) modulation onto the RF carrier.

The S-Comm accepts uplinked CCSDS telecommands from the S-Band transponder at 4 Kbps. Normally commands are forwarded to the SBC for further processing via the SpaceWire interface unless they are tagged as hardware-decoded commands. Hardware-decoded commands are performed directly by the S-Comm card. The S-Comm card can support up to 8 hardware-decoded commands, 4 of which are RS-422 outputs. Hardware-decoded commands are only used in contingency situations.

Both the S-comm and Ka-comm card provide control of the respective Communication Subsystem transmitters via asynchronous low rate serial interfaces.

3.1.3 HK/IO Board

The Housekeeping Input/Output (HK/IO) board maintains the spacecraft mission elapsed timer (MET), maintains the time of received uplink commands, provides timing synchronization to the instruments through a once per second pulse (1 PPS), and provides a unique high speed science data and a low speed command/telemetry interface to the LAMP instrument (all other instruments communicate via SpaceWire or MIL-STD-1553). The HK/IO board interfaces with the C&DH system via SpaceWire thought the Ka-comm card.

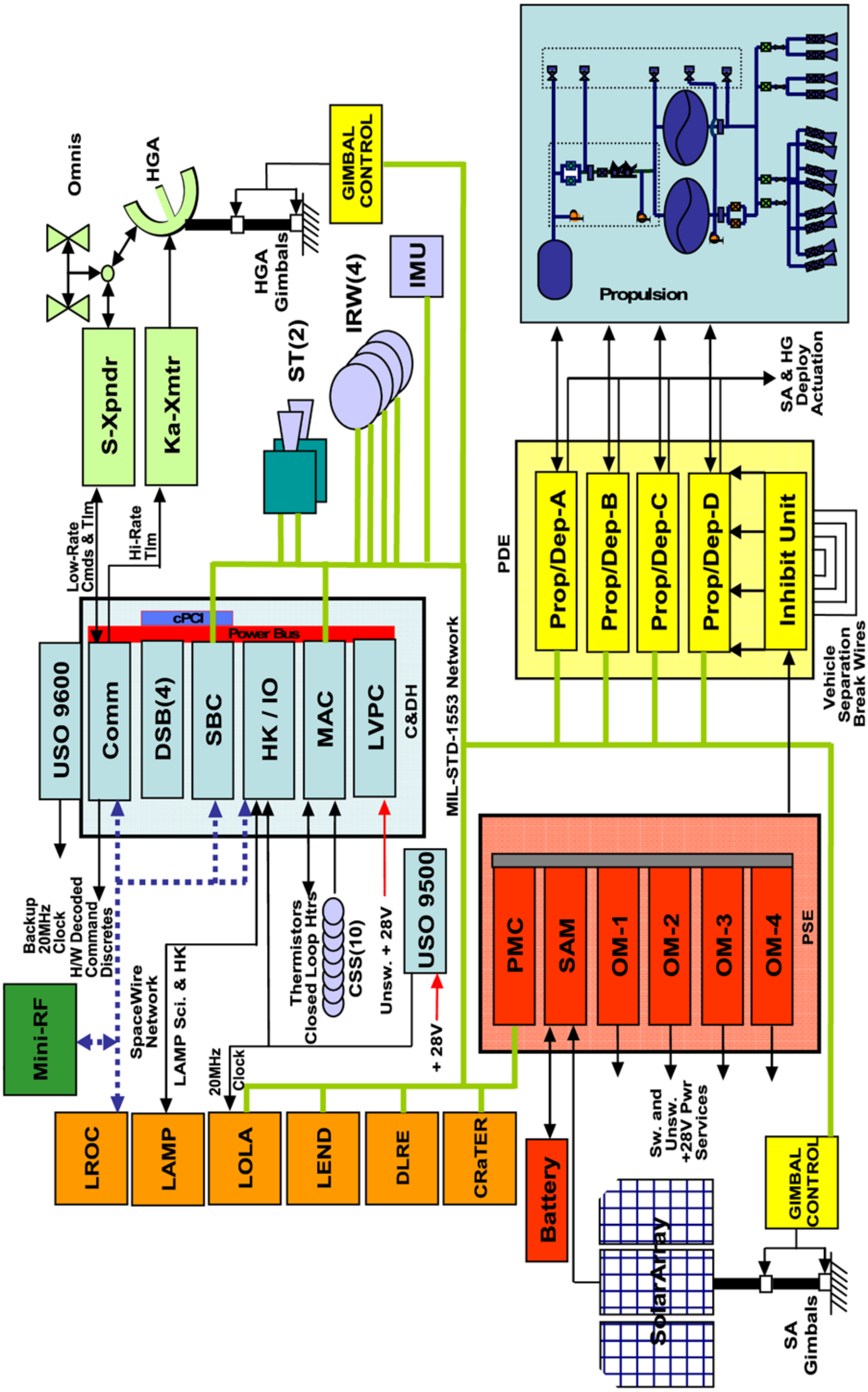


Fig. 22 LRO simplified system block diagram

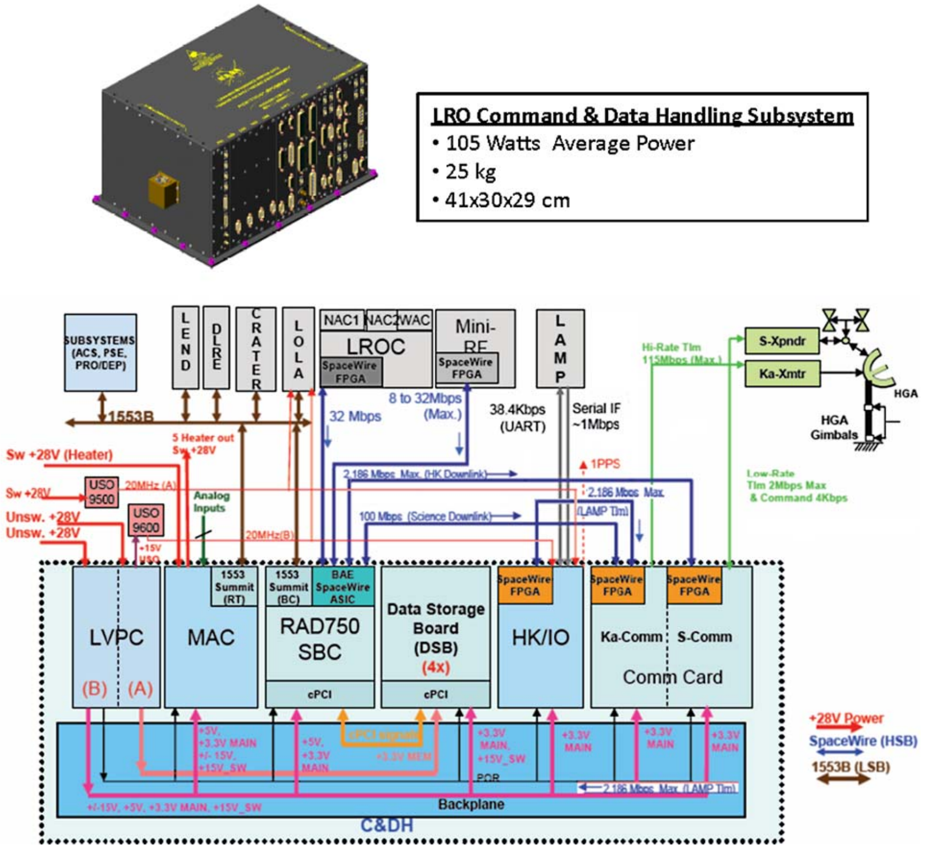


Fig. 23 LRO C&DH system

3.1.4 Multifunction Analog Card

The Multifunction Analog Card (MAC) is a set of interface electronics that samples and conditions a variety of spacecraft housekeeping inputs, including thermistors, hinge potentiometers, coarse sun sensors (CSS), analog voltage inputs from the Ka-band Travelling Wave Tube Amplifier (TWTA), Analog to Digital (A/D) converter scale reference voltages, and C&DH subsystem LVPC outputs. The MAC also controls the transmission of +31VDC power to switched heater services. All MAC data transfer to/from the SBC is over the MIL-STD-1553 bus.

3.1.5 Data Storage Board (DSB)

The data recorder on LRO is a solid state design consisting of four DSB cards designed as a file system to handle the storage and retrieval of files. The DSBs provide 384 Gigabits (Gbits) of memory capacity for incoming data files for a minimum of 17.5 hours of science data and HK data collection. Data transfer to and from the SBC is via a cPCI interface on the backplane. A robust error detection and correction scheme will correct up to 2 nibbles of 4 bits each in the 32-bit words.

3.1.6 Low Voltage Power Converter (LVPC)

The primary task of the LVPC is to provide power to the subsystems within the C&DH enclosure, including the four DSB boards, SBC, MAC, HK/IO Board, and Communication Cards. The LVPC converts an unswitched primary input voltage with an operational range of +21 Volts Direct Current (VDC) to +35VDC into secondary power outputs. The LVPC also houses the circuitry that drives the magnetic relays to provide power for the RF systems.

3.1.7 Ultra Stable Oscillators (USO)

The C&DH subsystem includes both a primary and redundant USOs to provide precision timekeeping onboard the spacecraft. Each oscillator provides two 20 MHz signals. One signal goes to the C&DH HK/IO card and the second goes directly to the LOLA instrument.

Only one of the oscillators will be powered at any given time. The primary USO has a frequency stability factor of 10 parts per billion (ppb) over one millisecond (ms). It was chosen to provide sufficient stability to meet the needs of the laser ranging system. It receives a +31VDC switched service from the Power Subsystem Electronics (PSE).

The redundant USO has a frequency stability factor of 0.3 parts per million (ppm) over one millisecond (ms). It was chosen to provide sufficient stability to meet the needs for LOLA reconstruction of the orbital ephemeris. It receives a +15VDC switched service from the LVPC.

3.2 Attitude Control Subsystem

The Attitude Control System (ACS) controls the pointing of the LRO spacecraft. ACS control algorithms are implemented in flight software hosted by the LRO C&DH and control the distributed sensors and actuators via the MIL-STD 1553 bus. Through the use of Star Trackers and Coarse Sun Sensors, the ACS determines where the spacecraft is pointing in fine and course accuracies, respectively. A three axis, ring laser gyro (called an Inertial Measurement Unit or IMU) measures the rate at which the spacecraft is rotating. The use of Reaction Wheels allows the spacecraft to smoothly point in any desired direction, as well as compensate for disturbance torques (such as High Gain Antenna movement and Solar Array movement). Eight small thrusters (20 N) are used to provide steering (attitude control) during large thruster (80 N) firings for Lunar Orbit Insertion (LOI). Additionally, the Propulsion System 20 N thrusters are used every two weeks to zero out the momentum buildup in the Reaction Wheels (primarily due to gravity gradient effects) and perform station keeping maneuvers while in the mission orbit. The ACS subsystem components are described in Table 6 and the Reaction Wheels and Star Trackers are shown in Fig. 24.

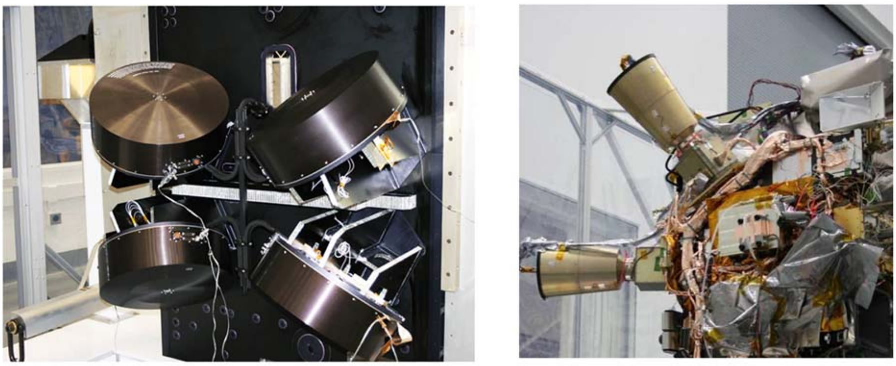
The Attitude ACS also determines spacecraft attitude, provides guidance to reach the desired inertial pointing vector, and uses actuators to achieve the desired pointing vector. Additionally, the ACS provides pointing support for High Gain Antenna and Solar Array. The nominal ACS control flow is illustrated in Fig. 25.

There are four primary LRO ACS control modes: Sun Safe, Observing, Delta-V, and Delta-H which are illustrated in Fig. 26. Each is described in detail below.

Sun Safe is the mode in which the Orbiter is starts in and was used to acquire the Sun after separation from the launch vehicle. It is the mode that Orbiter will eventually return to in the event of anomalies detected on-board. This mode moves the solar array to the "index" position (with the panel against the Orbiter facing $-Y$), damps any body rates, and points the

Table 6 LRO ACS components

ACS Component	Power (W)	Mass (kg)	Qty.	Performance
Star Tracker (ST)	11.2	4.2	2	20 arc-sec, 90 arc-sec about boresight, 16 deg field of view
Inertial Measurement Unit (IMU)	25	4.5	1	0.2 arc-sec resolution up to 375 deg/s, 100 ppm linearity, 5 ppm stability
Coarse Sun Sensors (CSS)	0	0.011	10	170 deg field of view
Reaction Wheels	16.6	11.9	4	$\sim$90 N m s at 24 V, $\sim$120 N m s at 31 V, $>$0.16 N m torque
Propulsion-Deploy Electronics (PDE)	13.2	9.4	1	FPGA based electronics that control the thrusters, valves, and deployment initiators. Provides safety inhibits



LRO Reaction Wheels mounted to spacecraft structure   LRO Star Trackers mounted on Instrument Module

Fig. 24 LRO reaction wheels & star trackers

$-Y$ axis toward the Sun. Sun Safe uses the coarse Sun sensors and the IMU to sense attitude and rates, and it uses the Reaction Wheels to lower body rates and point the Orbiter. This mode is capable of operation without the IMU in the event it is off or producing bad data by using the Sun sensors to derive rates. Sun Safe mode can also be used to accommodate a stuck solar array.

Observing mode is the primary mission control mode. It uses the star trackers and the IMU, combined with spacecraft ephemeris to determine the Sun, Earth, and Moon vectors in inertial space. The ground operators select solar inertial or lunar nadir pointing, and the Reaction Wheels provide the control. In Observing mode the Orbiter can point to and track any commanded inertial target. This also mode controls the 2-axis gimbals on both the solar array and the high-gain antenna, for tracking of the Sun and Earth respectively.

Delta-V mode provides changes in the Orbiter velocity for orbit control. This mode holds the commanded attitude by propagating rates measured by the IMU. It uses the thrusters for both the velocity change and for attitude control. The 80 N insertion thrusters are only used for the lunar orbit insertion maneuvers, and, in that configuration, the 20 N attitude control thrusters are on-pulsed for control. All other maneuvers only use the 20 N attitude control thrusters, operating in an off-pulse configuration.

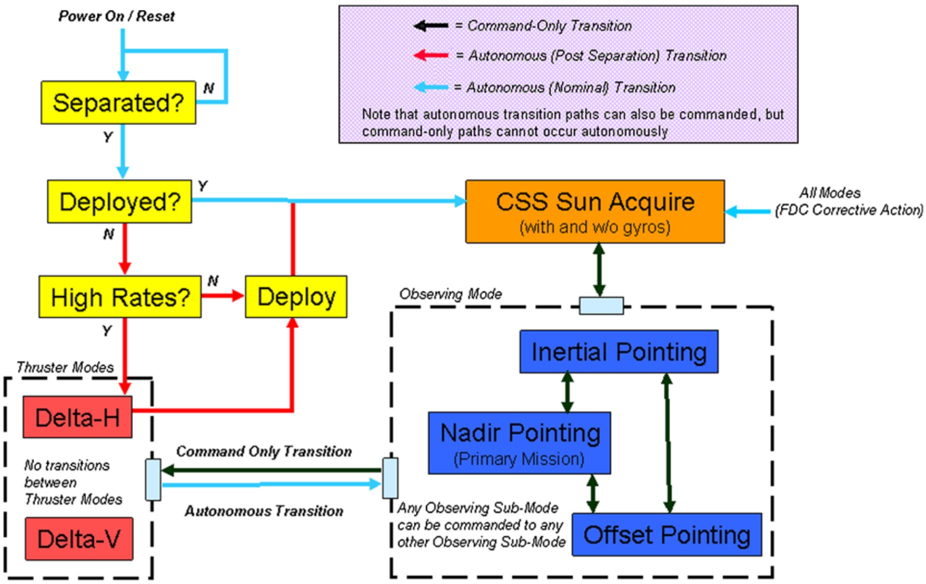


Fig. 25 ACS nominal control mode flow

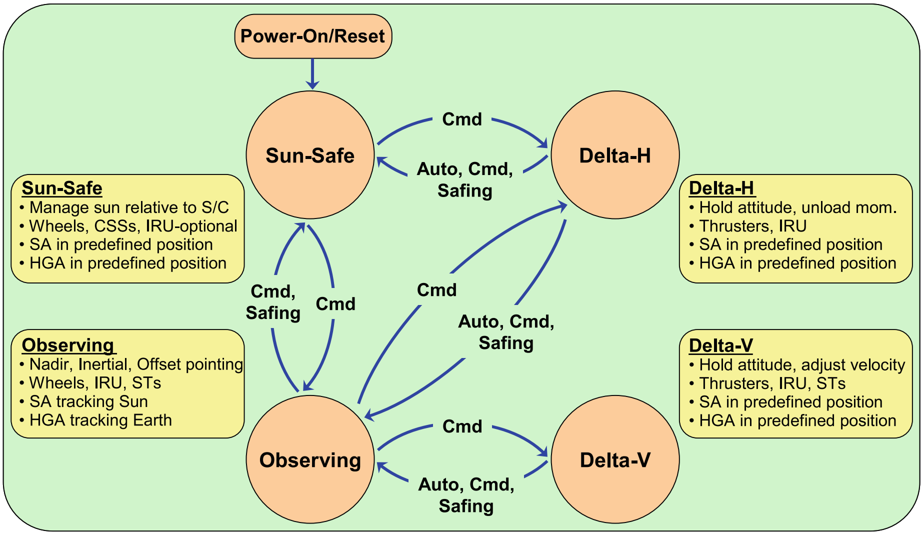


Fig. 26 LRO control modes

Delta-H mode is used to reduce the Orbiter system momentum. This mode holds the commanded attitude by propagating rates measured by the IMU. The 20 N attitude control thrusters fire to maintain attitude as the reaction wheels are commanded to the speed indicated by the operators.

Although the Orbiter autonomously enters a lower-state control mode when an anomaly is detected, ground commands must be provided to move the Orbiter into a higher-state

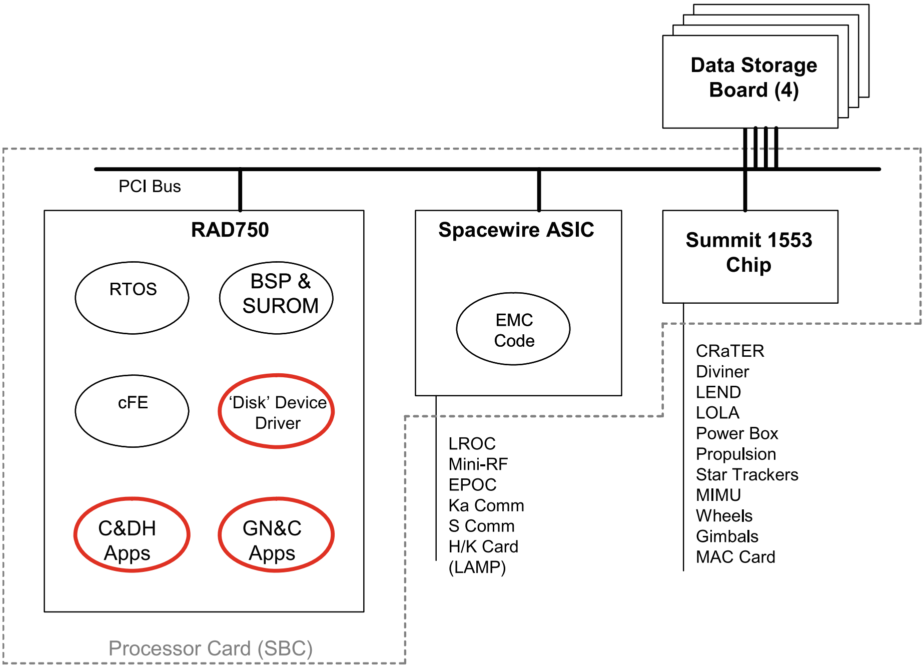


Fig. 27 SBC software context diagram

control mode. Delta-Vcan only be entered from Observing mode, while Delta-H mode can be entered from either sun safe or observing. The thruster modes will return to the mode from which they were called after a normal or anomalous termination.

LRO's ACS includes the Propulsion Deployment Electronics (PDE) component, which provides thruster deployment and control, as well as safety inhibits during launch. It is controlled by the SBC via the MIL-STD-1553 bus.

3.3 Flight Software

The LRO Flight Software (FSW) runs on LRO's RAD750 processor within the C&DH subsystem with a VxWorks Real-Time Operating System (RTOS). It consists of two application components, the C&DH FSW and the GNC FSW, both developed on the Application Program Interface (API) provided by the GSFC Core Flight Executive (cFE) (Wilmot 2006). The cFE provides a mission independent flight software operational environment with a set of services with functional building blocks used to create and host applications. In addition to the RTOS, the cFE, and the applications, the RAD750 also includes a software Board Support Package (BSP) which provides the software interface between the RTOS and the hardware located on the SBC. The software components running within the SBC are illustrated in Fig. 27 and the overall FSW architecture in shown in Fig. 28.

The primary functionality of the LRO FSW can be broken down into 5 areas or Tasks to better understand its overall operation, these which are described below:

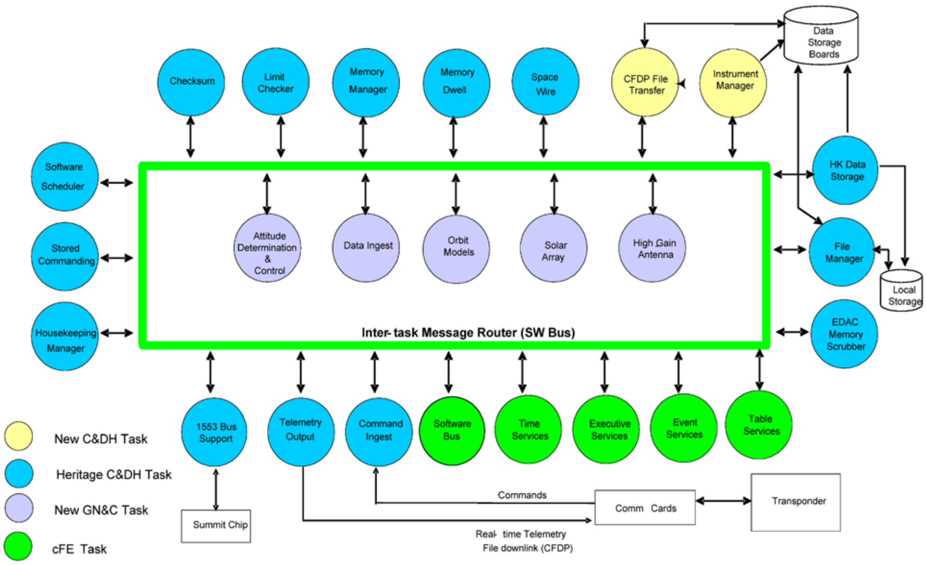


Fig. 28 Flight software architecture with heritage prior to LRO shown

3.3.1 Software Scheduling and Housekeeping

Predictable system activities, such as housekeeping telemetry collection, operate on a schedule which is controlled by the software schedule table. The table allows up to five activities in any 10 ms slot across the one-second scheduling interval and is designed to ensure that tasks can be accomplished in the time available. The schedule begins at the interrupt from the 1 pulse per second (PPS) signal and continues to the next one second tick.

The housekeeping task collects housekeeping data from all of the other software tasks, and it services the processor watchdog. If one of the critical tasks fails to report back in two consecutive cycles, the software will initiate a processor reset.

3.3.2 Commanding and Command Sequences

The majority of LRO's commands are executed by one of two variants of stored commands, although LRO can also execute individual commands from the ground. Absolute Time-tagged Sequence (ATS) commands are executed with reference to Universal Coordinated Time (UTC), and Relative Time-tagged Sequences (RTS) execute stored commands with timing relative to the start of sequence execution. Both types of stored command use a one second resolution. Ground commands (an individual command or a command to start an RTS or ATS) are ingested by the S-Comm card in the C&DH then forwarded to the flight software in the SBC for processing via the SpaceWire interface. Up to eight commands can processed each second.

3.3.3 Limit Checking

The Limit Checker (LC) implemented in the FSW provides the safing functions for the Orbiter. A table defines watch points and action points. The watch points monitor telemetry and are considered "true" when the referenced telemetry data has exceeded a limit. Action points initiate an RTS when one or more watch points meet the criteria indicated by the action point table. The action point table defines which RTS to execute for which watch points, how the watch points should be logically combined, and how long the entire condition should be true prior to execution. This provides great flexibility to define and adjust safing as the Orbiter performance changes with time. The detailed architecture and implementation of LRO's safing design is detailed in Andrews et al. (2008).

3.3.4 Instrument Control and Data Handling

The FSW provides Orbiter level control of each of the instruments, including the handling of commands, housekeeping telemetry, science data and the distribution of time information. In order to take advantage of as much instrument heritage (from previous missions) as possible, each interface has unique characteristics, such as the time message format, command structure, and housekeeping telemetry definition. The FSW handles these unique interfaces and provides the translation to the standard CCSDS interface used by the Orbiter and the ground system. Low-data-rate instruments (CRaTER, DLRE, LEND, and LOLA) communicate with the FSW over the MIL-STD-1553 interface, while SpaceWire is used for the high-rate instruments (LROC and Mini-RF) and for LAMP via the custom serial interface on the HK/IO card. The LROC's high data rate, combined with minimal buffering, requires the FSW to prioritize LROC science data collection and precludes the ability to operate both Mini-RF and LROC simultaneously. The FSW sets up a Direct Memory Access (DMA) transfer to move LROC data to a special location in RAM and then to the DSBs. A similar process is used to transfer data from the DSBs to the Ka-comm card.

All on-board data is stored in files, including configuration tables. The FSW handles the creation of instrument data and housekeeping files. Files are stored in a directory structure and are downlinked by command.

3.3.5 File Transfers

Files are transferred between the Orbiter and ground (in both directions) using CCSDS File Delivery Protocol (CFDP). CFDP code must run on both the Orbiter and the ground system to reliably transfer files. The file sender divides the file into Protocol Data Units (PDU), which are sent to the receiver. The PDUs include information so that the receiver can determine if any are lost. The receiver requests retransmission of any lost PDUs and then provides positive acknowledgment when the entire file is received. LRO will delete a file when positive acknowledgment is received from the ground; otherwise files remain on board, guaranteeing that science data is reliably transferred to the ground. This strategy allows for a large amount of compression of LROC images, without fear of the bit error rate corrupting large portions of an image. LRO also has the capability to downlink data without a corresponding uplink, using a different class of CFDP but the capability will only be used in a contingency.

3.3.6 Time Keeping

LRO's flight software works with the hardware to maintain Orbiter time to within 100 ms of UTC. Post-processed knowledge of time will be accurate to 3 ms. A hardware counter on the HK/IO card counts continuously from the time the card is powered on. This Mission Elapsed Timer (MET) contains 32 bits of seconds and 16 bits of sub-seconds (>136 years before rolling over and a precision of better than 16 μs). The software adds an offset to the MET so that the total corresponds to the time since the LRO epoch (January 1,

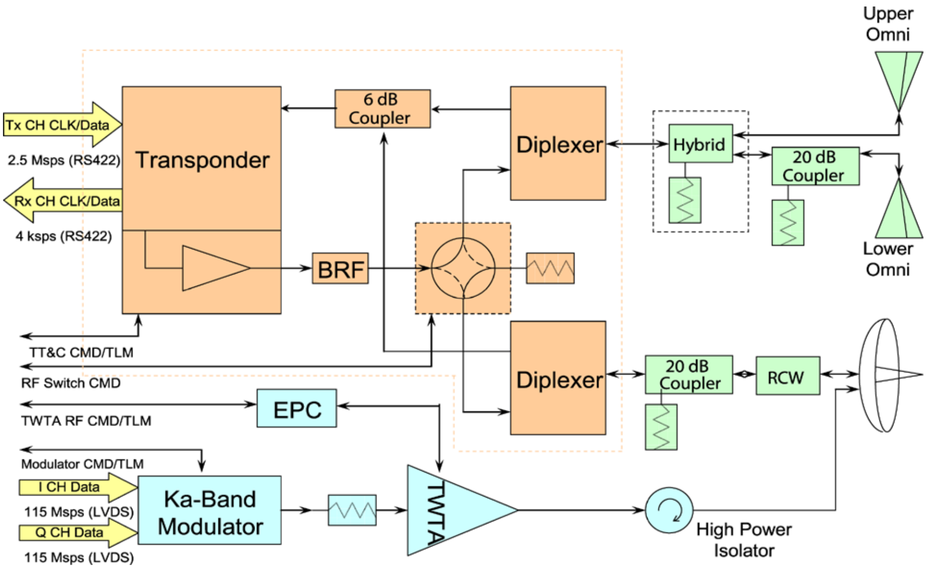


Fig. 29 LRO communications system block diagram

2001 00:00:00 UTC). This time is inserted in the time field of housekeeping and science data packets. The software distributes time with a "time at the tone" message via the instrument data interfaces. The tone is either a 1 pulse per second hardware signal for the 1553 instruments or a SpaceWire time tick for the SpaceWire instruments. The ground adjusts the time on the Orbiter using a feature on the S-Comm card which pulses the HK/IO card and latches the MET when a command is received. The ground compares this MET to the time the command was sent, compensates for the propagation delay, and then sets the software offset accordingly to ensure the correct time on board.

3.4 Communication Subsystem

LRO's communications system consists of an S-band system to provide tracking, telemetry and commanding (TT&C) and a high data rate Ka-band downlink only system for telemetry and science data transfers. The system design is shown in Fig. 29.

The S-Band system has a fixed forward link data rate of 4 kbps, and a selectable on orbit return link data rate between 125 bps and 1093 kbps. It consists of one Spacecraft Tracking and Data Network (STDN) compatible transponder, an S-band Radio Frequency (RF) Switch, and the RF paths to and from the two Omni-Directional antennas and the S-band feed on the High Gain Antenna (HGA). The transponder downlinks at a frequency which is phase-locked to the uplink, providing two-way Doppler tracking information to the ground with an accuracy of better than 1 mm. It also repeats uplinked range tones with a fixed delay, allowing the ground to determine the distance to the Orbiter within 15 m.

The Ka-band system includes a Ka-Band modulator, a 40 W Traveling Wave Tube Amplifier (TWTA) consisting of a traveling wave tube (TWT), an Electronic Power Conditioner (EPC), and a High Power Isolator. The Ka-band return link is also selectable on orbit and varies from 25 Mbps to 100 Mbps

Although the Ka-band system uses only the High Gain Antenna, the S-band system can utilize either the Omni-Directional or the High Gain Antenna for transmit and always uses

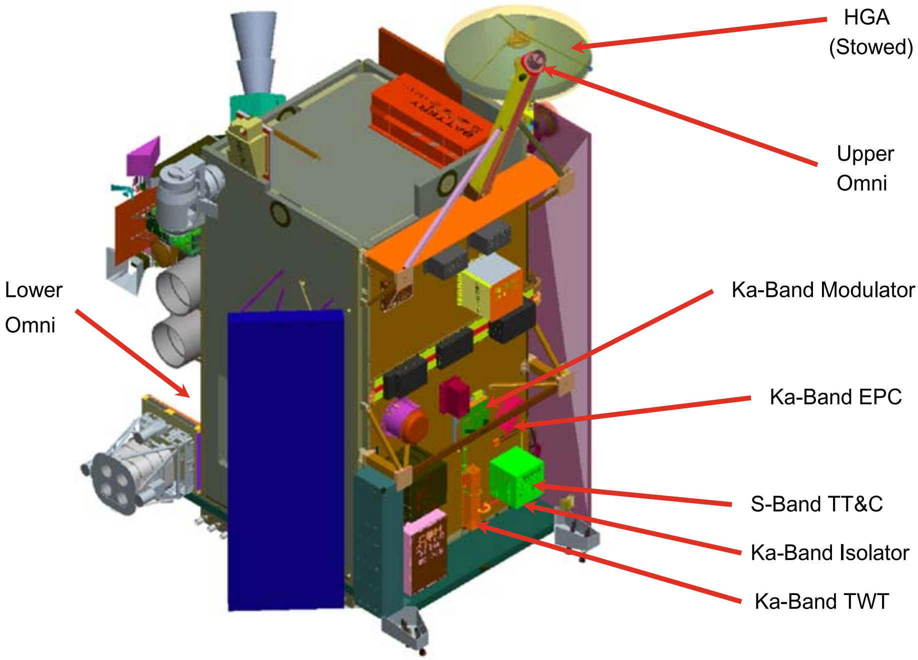


Fig. 30 Communications system components (Avionics radiator not shown)

Table 7 LRO communication system summary

LRO Communication Subsystem Characteristics		
Parameter	S-Band System	Ka-Band System
Frequency	2271.2 ± 2.5 MHz (Transmit)	25.65 GHz ±150 MHz (Transmit)
	2091.3967 ± 2.5 MHz (Receive)	
Mass	8.8 kg	8.4 kg
Polarization*	LHCP (Omni), RHCP (HGA)	LHCP
RF transmit power	5.8 Wat Diplexers	41.9 Wat TWTA Output

*L/RHCP—Left/Right Hand Circular Polarization

both paths for receiving. The ground selects the uplink path via polarization, as the HGA has opposite polarization from the omni antennas for this reason.

Major components of the communications system are shown in their installed location on the avionics panel of the spacecraft bus in Fig. 30. Specifications for each of the S- and Ka-Band subsystems are shown in Table 7.

3.5 Propulsion Subsystem

LRO's propulsion system was designed to provide mid-course transit corrections after separation from the launch vehicle, lunar orbit capture and then momentum dumping and station keeping for the remainder of the mission. Figure 31 illustrates the overall design and shows

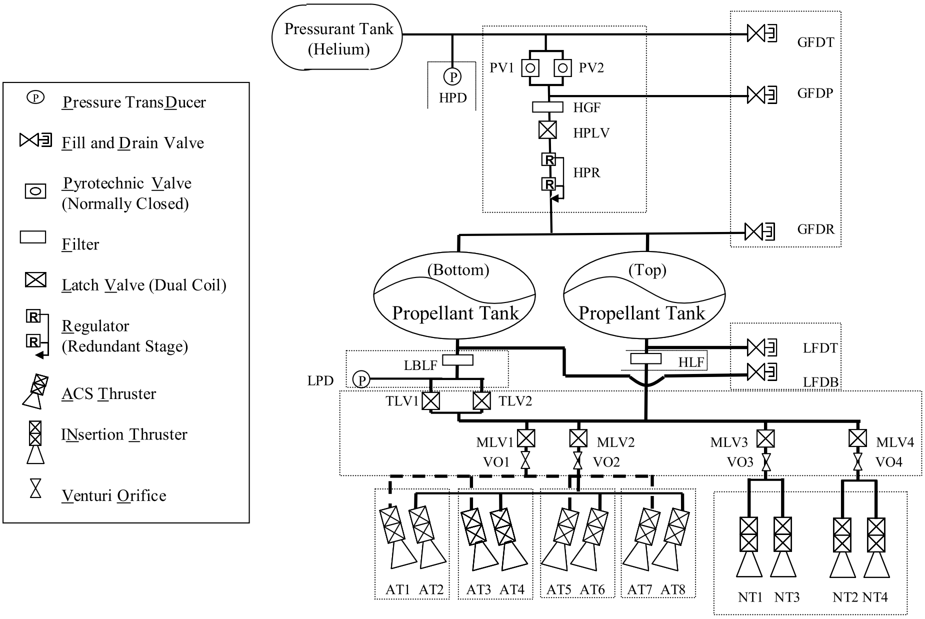


Fig. 31 Diagram of propulsion system

the redundancy of the propulsion system, and Fig. 32 shows the flight Propulsion Subsystem prior to the buildup of the Orbiter around it.

The propulsion system is a monopropellant hydrazine (N_2H_4) system, chosen for its simplicity. It carries a hydrazine fuel load of 895 kg ($>$1270 m/s ∇V capability) in two identical titanium diaphragm type 28,144 in^3 oblate spheroid propellant tanks. These tanks were specifically chosen because of their availability from a previous program, the cancelled NASA X-38 mission. The system is Helium pressure regulated, with the pressurization provided by a single composite overwrapped tank filled to 4200 psi at launch.

The system includes twelve dual coil catalytic hydrazine thrusters, four of which are on-axis 80 Newton-class insertion thrusters located around the spacecraft center of gravity (in the x-axis). Eight canted 20 Newton-class attitude control thrusters provide attitude control, lunar orbit maintenance maneuvers, and momentum dumping. Isolation valves with redundant coils are used to isolate thruster banks in the event of a thruster failure. The LRO Propulsion subsystem forms the physical core of the LRO spacecraft and the integration of the spacecraft was essentially the process of building up the Orbiter around it.

3.6 Power Subsystem

The LRO Electrical Power System (EPS) employs Direct-Energy-Transfer (DET) architecture and consists of three sub-elements: the Solar Array, Power System Electronics (PSE), and a Lithium-Ion Battery. The LRO Power System block diagram is shown in Fig. 33.

3.6.1 Solar Array

LRO's solar array unfolds to a single wing of three panels as shown in Fig. 1, and is continuously gimbaled in two axes to track the Sun. The array configuration consists of three

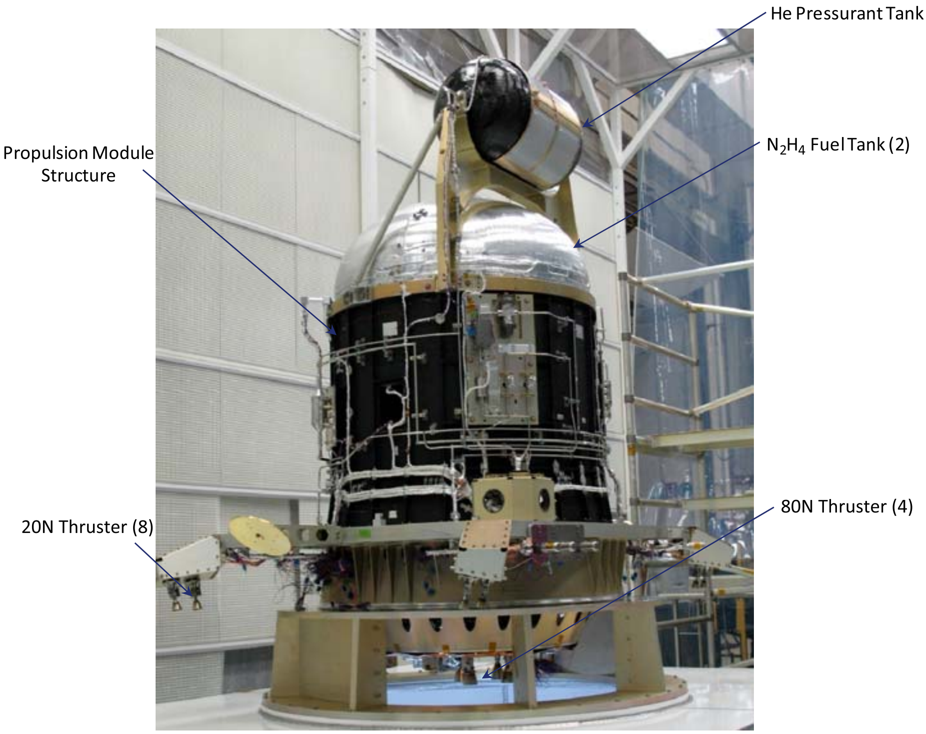


Fig. 32 LRO propulsion subsystem (on support stand)

aluminum honeycomb panels populated with 3,696 Advanced Triple Junction solar cells, for a total area on the order of ten square meters, and a power generating capability of 1,849 W at end of life.

3.6.2 Power Subsystem Electronics (PSE)

LRO's PSE performs the functions of power distribution and battery charging. The basic design and operation of the LRO PSE architecture is modeled after the architecture used on the Microwave Anisotropy Probe (MAP) and uses the solar arrays to convert sunlight energy into electrical energy while in the sunlight. The electrical power is then transferred to the PSE, where it is conditioned and directed to all of the electrical loads connected to the spacecraft bus. During the eclipse seasons, the PSE will also direct a portion of the sunlight-generated electrical power to the battery for energy storage recharging. During the eclipse portion of the orbit the battery will provide all of the energy to the spacecraft. Any excess power from the solar arrays not needed for battery charging or spacecraft loads is shunted back to the solar array. The PSE architecture can be seen within Fig. 33. The PSE is designed for single fault tolerance while still being capable of meeting all mission requirements.

3.6.3 Lithium-Ion Battery

A single Lithium-Ion battery provides power to the spacecraft when illumination of the solar array is insufficient. In order to minimize electrical losses due to power converters

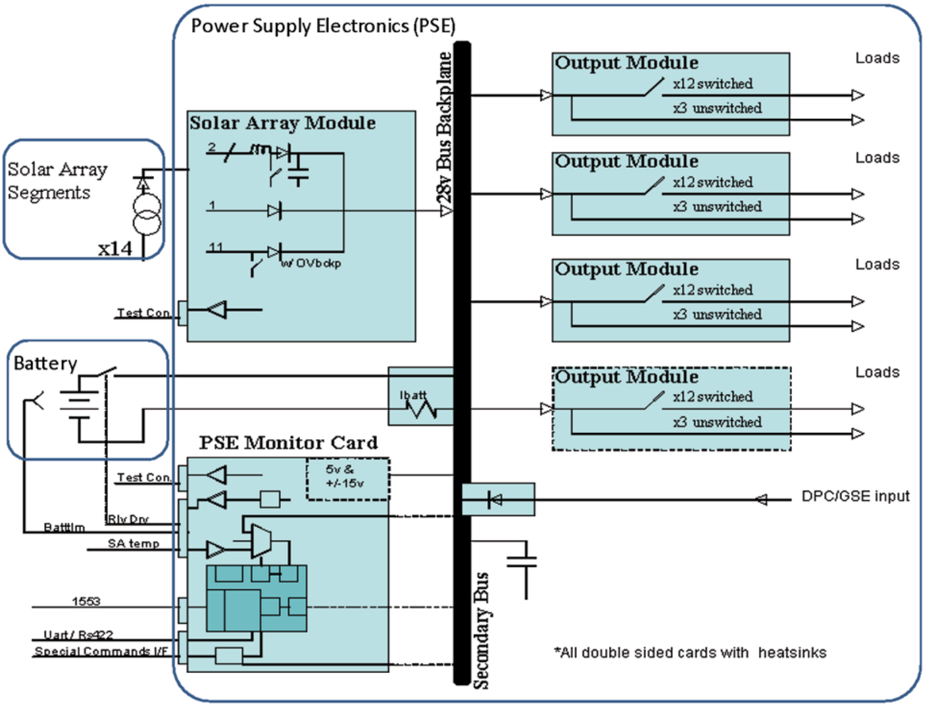


Fig. 33 Power system block diagram

and maintain a stable voltage spacecraft bus range, the battery is connected directly to the electrical bus. The battery energy storage capacity is 100 Ampere-hours at Beginning of Life (BOL) within a 24–34 V voltage range. The battery was selected to meet a requirement of storage capacity of more than 80 A-hr at End of Life (EOL) and keeping the Depth of Discharge (DOD) to less than 30% during normal operations. This limited DOD enables the battery to survive over the 5000 cycles expected during the nominal mission with a significant margin. Thermal control of the battery is provided at the spacecraft level though a heat pipe assembly which connects the battery directly to a zenith facing radiator.

The battery is constructed from small cells arranged in strings and blocks. The basic block size is 12 strings of 8 cells each and seven blocks are used, to give a total of 672 cells. Within each block, the cells are connected in strings with eight cells in each string in series. This design ensures that cell failures (both open-circuit and short circuit) result in benign events. An open circuit failure results in the loss of a cell string, and a battery capacity loss of $\sim$1% of capacity. A short circuit failure activates a cell internal protection device leading to an open-circuit failure and a similar loss in capacity. The battery weighs 36.4 kg and is shown in Fig. 34.

3.7 Thermal Control Subsystem

The LRO Thermal Control subsystem basic design accommodates four major elements: the Spacecraft Bus (Avionics, Reaction Wheels, and Battery), Instrument Module, Propulsion Module, and Deployables (high gain antenna and solar array). Because the lunar surface temperature varies from less than $-140°$C to greater than 125°C, the heat load on Orbiter $+Z$ face (nadir) varies from only a few W/m^2, when flying over lunar midnight, to

Fig. 34 LRO battery, top view showing cell interconnections

1330 W/m^2 from the Moon at lunar noon. A similarly high load is experienced from the Sun at lunar noon near the equator on the Orbiter $-Z$ face (zenith), further details are given in Fig. 35 and Table 8. This variation occurs in the span of less than an hour when the Sun-line is near the plane of the orbit. When the Sun is nearly perpendicular to the orbit plane, LRO experiences its nominal (routine) cold operational case, even though it does not pass through the lunar shadow, since the Moon surface at the terminator is near $-140°C$ or colder. The most extreme cold case occurs during Type 4 lunar eclipses, discussed in the mission design section earlier in the paper in that in addition to a cold moon, the solar input is blocked by the earth for extended periods causing the thermal system to drive the battery sizing.

The LRO Thermal Subsystem was designed and qualified to ensure the survival of the Orbiter in the most extreme lunar eclipse cold case.

The overall orbiter is designed primarily with zenith directed radiators providing minimal exposure to the lunar environment (Baker et al. 2009). These radiators are covered with Optical Surface Reflectors (OSR), which reflect most of the optical energy of the Sun, but still emit in the infrared. These mirror-like OSR's allow LRO's radiators to run cold, even in the direct Sun.

Most of LRO's avionics are thermally coupled into an embedded Constant Conductance Heat Pipe (CCHP) aluminum honeycomb panel. Dual bore header heat pipes couple the isothermal panel to a CCHP avionics radiator that is separately mounted on the zenith surface of the spacecraft. The reaction wheel assemblies have a dedicated heat pipe system that couples the assemblies to the zenith avionics radiator. Coupling all of the avionics and the wheels to a common radiator and wrapping that around the propulsion system provides a large thermal mass that minimizes temperature variation over the span of an orbit. To survive long passes through the Earth's shadow during lunar eclipses, this thermal mass is pre-heated, using software-controlled heaters, so that minimal heater power is required during eclipse scenarios. The Lithium-Ion battery is maintained on a separate heat pipe network

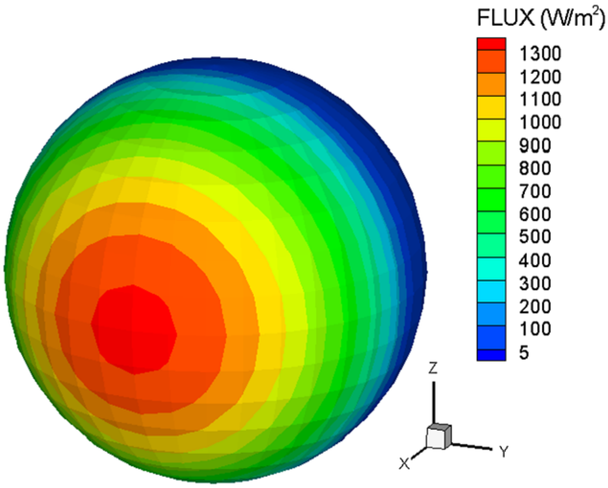


Fig. 35 Modeled lunar thermal environment

Table 8 LRO design lunar thermal environment

LRO Design Lunar Thermal Environmental Parameters		
Parameter	Hot Case	Cold Case
Solar Flux (Q_{sun})	1420 W/m^2	1280 W/m^2
Lunar Albedo Factor (F_{albedo})	0.13	0.06
Peak Lunar IR Flux ($Q_{lunar\text{-}max}$)	$Q_{lunar\text{-}max} = Q_{sun}(1 - F_{albedo})$	
(lunar sub-solar point)	1335 W/m^2	1114 W/m^2
Minimum Lunar IR Flux	5 W/m^2	5 W/m^2
($Q_{lunar\text{-}min}$) (dark lunar surface)		
Lunar IR Flux to Spacecraft	$Q_{s/c\ lunar} = [(Q_{lunar\text{-}max} - Q_{lunar\text{-}min})\cos(\beta)\cos(\theta)] + Q_{lunar\text{-}min}$	
($Q_{s/c\ lunar}$)	where β = solar beta angle & θ = angle from sub-solar point	

and a dedicated radiator on the zenith face of the Orbiter. An oversized radiator combined with heaters ensures tight control of its temperature.

A decoupled instrument optical bench (the Instrument Module) accommodates the instruments requiring high accuracy alignments (LOLA, LROC, LAMP, and the Star Trackers). It is constructed of low coefficient of thermal expansion (CTE) composite faced honeycomb material, is fully blanketed, and is heated with low density heaters to maintain cold limit temperatures. The CRaTER, Diviner, Mini-RF and LEND instruments are mounted directly to the main Orbiter structure and are individually controlled with heaters and radiators.

Within the Propulsion Module, most components are attached to a temperature-controlled cylinder that is part of the spacecraft bus structure including the upper propellant tank, pyrotechnic valves, and high and low pressure panels. Thruster valves are coupled to the spacecraft and tailored to survive soak-back after burn sequences. The lower propellant tank is thermally coupled to the aft deck for structural reasons. Remaining lines not on the structural cylinder are independently heated.

The High Gain Antenna uses multi-layer insulation (MLI), thermal isolation, and numerous heaters and radiators to maintain temperatures within allowable limits. The solar

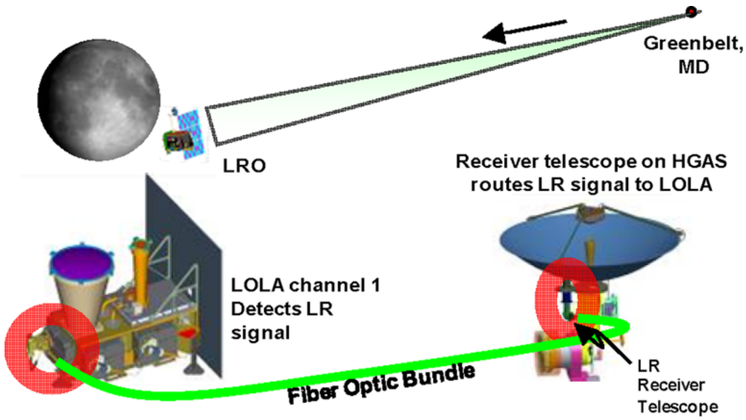


Fig. 36 LRO active one-way laser ranging system elements

array assembly uses MLI, dedicated radiators and heater control to maintain actuators and dampers.

Prior to launch the Thermal Subsystem performance was verified and optimized during Orbiter level Thermal-Vacuum testing which simulated the expected lunar environment and has meet all requirements since launch.

3.8 Laser Ranging Subsystems

3.8.1 Active One-Way Laser Ranging System

During the preliminary design phase an active on-way (uplink) laser ranging system was added to LRO to improve the tracking, and thus geodetic accuracy of the data products, over that possible with the S-Band tracking system alone. Figure 36 shows the elements of the active laser ranging system, beginning with the ground station at GSFC, which transmits a 532 nm pulse at 28 Hz, this pulse is captured by the laser ranging telescope mounted on the high gain antenna and then transmitted via fiber optic cable to the detector assembly located within the LOLA Instrument. Time-stamped arrival and departure times allow for precise range measurements from Earth to the LRO spacecraft, thus helping to determine precise position relative to the Earth's surface. That data, combined with known positioning of the spacecraft and LOLA data gathered from the Moon, will provide enough information to improve the lunar gravity modeling which in turn will be used to improve the orbit determination.

3.8.2 Passive Two-Way Laser Ranging System

In addition to the active laser ranging system LRO carries a retro-reflector assembly consisting of twelve 1″-diameter solid retro-reflective prisms in a square array mounted on the $-Z$ face (zenith radiator). This retro-reflector will allow LRO to perform two-way laser ranging with high-power terrestrial laser ranging sites. This ranging will be conducted as an experiment of opportunity late in the mission. The retro-reflector assembly is visible on the Orbiter in Fig. 37.

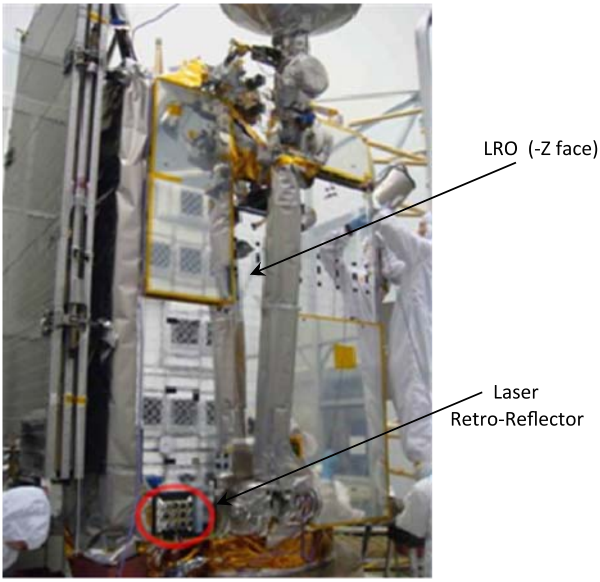


Fig. 37 Laser retro-reflector corner cube array (indicated in oval) on LRO

3.9 Mechanical Subsystem

The spacecraft structure and mechanical subsystem is of modular design, with four (4) distinct structural units (modules) for propulsion (PM), spacecraft bus (SB), avionics (AM), and instruments (IM), and two deployable systems with mechanisms for deployment and articulation, these being the Solar Array System (SAS) and the High Gain Antenna System (HGAS).

LRO structural construction is primary a combination of machined metal elements and metal honeycomb panels with bonded-in bolted fittings. The exception is the IM which utilizes composite face-sheet honeycomb construction for thermal stability. The major structural elements can be seen in Fig. 20.

Both the SAS and HGAS are deployed via spring-hinge-damper mechanisms with release controlled by the activation of Non-Explosive Actuators (electrical melt wire devices). Both systems have two-axis gimbal systems that employ stepper motors paired with Harmonic-Drive gear reducers. Position and rate telemetry from the gimbal assemblies is used by the GNC FSW to drive the motors and control the pointing of the Solar Array and High Gain Antenna.

4 Summary

LRO is an ambitious mission engaged in exploring the lunar surface and environment like never before in preparation for the human return. Its data products will enable future lunar exploration for decades to come. It is a challenging mission with several key driving factors, all of which have been met with robust design margins and innovative operations concepts. The mission development was completed within budget and on schedule. LRO is currently in lunar orbit executing its primary exploration mission with all of the spacecraft systems and science instruments performing flawlessly. Additional information about the LRO mission and its current status can be found at the NASA LRO website (www.nasa.gov\lro).

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