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         2007-04-11 LRO: scott    Original;
         2007-11-08 GEO: slavney  Reformatted;
         2008-04-23 GEO: slavney  Revised CHINETAL reference."

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## Instrument Host Overview

For all LRO experiments, data are collected by science instruments on the spacecraft and then relayed via the spacecraft telemetry system to stations of the Space Communications Network (SCN). The following sections provide an overview of the spacecraft, the science instruments, and SCN ground system.

## Instrument Host Overview — Spacecraft

The Lunar Reconnaissance Orbiter is designed for a one-year base mission with a goal of an extended mission of up to four additional years. LRO will be launched on an Atlas 5 401 Launch Vehicle along with its companion spacecraft, the Lunar CRater Observation and Sensing Satellite (LCROSS), into a direct insertion trajectory to the Moon. The on-board mono-propellant hydrazine propulsion system will be used to capture into a polar orbit at the Moon, timed to obtain an orbit plane that enables optimum lighting conditions in polar regions during summer and winter seasons. Additional burns will circularize the orbit and maintain it during the base mission at an altitude of 50 ± 20 km. A lower-maintenance, elliptical orbit of 30×216 km will be used for commissioning and may be used for the extended mission. The spacecraft carries enough fuel to provide over 1300 m/s of velocity change (delta V) for orbit capture and maintenance.

## LRO Description

The orbiter is a 3-axis stabilized, nadir-pointed spacecraft, designed to operate continuously during the primary mission. Four reaction wheels provide attitude control to 60 arc sec and momentum storage of up to 2 weeks, with thrusters providing momentum dumping once per month. Two star trackers and an inertial reference unit provide attitude knowledge of 30 arc sec. Coarse sun sensors provide attitude information in contingency modes, to enable and maintain proper attitude with respect to the sun, keeping the spacecraft power positive and thermally stable.

A 10.7 square meter solar array provides 1850 W end-of-life during the sunlit portion of the orbit. An 80 A-hr Lithium-ion battery maintains the bus voltage and provides operational power during the orbit eclipses and survival power during the rare, long eclipses of the sun by the earth. The power electronics distributes the raw 28+7 V to the instruments and the spacecraft bus electronics, delivering over 800 W average power each orbit.

The flight computer is a RAD-750 processor executing at 133 MHz. Two 100 Gbyte recorders store science data for playback to the earth at 100 Mbps through a 40 W Ka-band transmitter and high-gain antenna. An S-band system provides command, engineering telemetry, and navigation functions. Laser ranging capability provides ~10 cm position precision during four one-hour passes per day. These data, when combined with lunar measurements from LOLA, will improve the orbit determination capability of LRO.

The thermal control system utilizes heat pipes to spread heat and move it to the zenith-facing radiators. A modular structure design enables parallel assembly of the spacecraft.

The total mass of the observatory is less than 949 kg dry and 1846 kg fully fueled.

**Orbiter Features**
**----------------**

The LRO Orbiter has the following key features:

1. Single string with selected redundancy
2. C&DH hosts flight software, handles data interfaces
3. PSE controls battery charging and power distribution
4. PDE controls thrusters, deployment devices, and inhibits
5. S-band transponder and Ka-band transmitter support radio links
6. 2-axis gimbals on array and high-gain antenna track sun and earth
7. Omni antennas provide continuous receive capability
8. Standard interfaces
   a. 1553
   b. Spacewire
9. Modular Assembly
   a. Propulsion module, instrument module, avionics module
   b. Parallel development

**Spacecraft Subsystems**
**---------------------**

The spacecraft subsystems are summarized briefly below.

**Spacecraft Bus**
**---------------**

The LRO spacecraft bus consists of the following components:

1. Primary Structure (aluminum)
   a. Heating/Cooling panel
   b. Edge members, fittings, and spools
   c. High Gain Antenna System (HGAS) support brackets
   d. Solar Array Substrates (SAS) stanchions
   e. Reaction wheel assemblies brackets
2. Instruments
   a. CRaTER
   b. DLRE
   c. LAMP (on optical bench)
   d. LEND
   e. LOLA (on optical bench)
   f. LROC (on optical bench)
   g. MRF
3. Avionics Components
   a. Reaction wheels
   b. Battery
   c. S-band OMNI antenna
   d. Coarse Sun Sensors (CSSs)

**High Gain Antenna System**
**-------------------------**

The High Gain Antenna System (HGAS) includes a deployment system with two latches requiring mechanical release, and three restraint areas. The articulation system is a two axis gimbal system capable of slightly greater than 180 degrees of rotation about each axis, and two rotating cable wraps. RF components include Ka-band antenna with an S-band patch antenna, an S-band coax cable and a Ka band waveguide, and rotary joints in each of two gimbal actuators. A small laser ranging telescope is mounted on the wave guide with fiber optic cable providing the connectivity to the LOLA instrument. Thermal control is provided through blanketing, T-stats, heaters, and radiators dedicated to the HGAS system.

The latching release assembly utilizes a 400 lb preload, to include a 1.3 gapping factor of safety. Release begins with actuation of the non-explosive actuator (NEA) which initiates the deployment sequence. The principle of operation of the NEA involves a tensile load reacting against a spring-wound split spool. Upon fuse wire initiation, the spring unwinds, allowing the halves of the spool to separate. The slow release of the tensile load minimizes the release of strain energy and therefore minimizes shock.

**Propulsion System**
**-----------------**

The propulsion system has been designed to provide mid-course transit corrections after separation from the launch vehicle, lunar orbit capture, and station keeping for the remainder of the mission. The propulsion system is a monopropellant hydrazine system. Fuel load is 894 kg of hydrazine (~ 1300 m/sec delta-V capability) in two identical titanium diaphragm propellant tanks (40 in OD oblate spheroid TDRSS type tanks in TDRSS configuration)

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. Flow control orifices prevent water hammer surges.

The Helium Pressure Regulated System is a 4200 psi COPV Helium pressurant tank (17 in outside diameter by 29.6 in Length). A two stage regulator (single fault tolerant) is set at 270 psi nominal. Redundant pyro valves provide for high pressure isolation during ground and launch operations with a high pressure latch valve to isolate high pressure source during mission operations.

**Electrical Power System**
**-----------------------**

The LRO Electrical Power System (EPS) provides the following functions:

1. Load power requirements for all nominal mission modes
2. Battery charging and control
3. Interface to the C&DH subsystem for power system performance, monitoring, configuration and control
4. Power interface to the electrical subsystem
5. Capability to respond to Special Commands
6. Power distribution and load switching capability
7. External power interfaces for I&T, ground, and pre-launch operations
8. Bus protection by automatically shedding loads if a defined fault condition occurs
9. Restoration of power to shed loads shall be remotely controllable via ground command.

The power system contains three sub-elements: the solar array, Power Sub-system Electronics (PSE), and the lithium-ion battery.

**Solar Array System (SAS)**
**------------------------**

Three low-CTE graphite composite solar array panels are contained within one articulated wing. The system is mounted on flexures to aluminum spacecraft structure to allow for differential thermal growth. Four NEAs are used to release the array.

Articulation is provided through a two-axis gimbal with four panel hinges, two at each hinge line connecting center panel to two outer panels. When stowed, cells of the central panel face outward. For deployment, panel springs (less than 10 lb of stored force) are driven with viscous dampers to dissipate energy.

The solar array mounts to a single structure. Benefits of such a design include straightforward integration, alignment, and testing. Further, the design provides for reduced and less-sensitive loads on gimbal bearings, while structural loads are carried at cup/cone fittings rather than through more-flexible gimbal paths.

Deployment is initiated through a software command delivered to the propulsion and deployment electronics (PDE), which then fire non-explosive actuators (NEA's) with a design similar to that shown for the high gain antenna. Separation nuts release rods that restrain the array in the stowed configuration. Restraining bolts are pulled by strings to the outer panel and captured panel-to-panel latches prevent panels from immediately unfurling. Firing the prime and redundant actuators on one unit at the same time, then moving sequentially on to the next unit, ensures that the actuators will release the solar array in a deterministic order. Individual panel deployment is controlled by cam latches. The three panels unfold perpendicular to the spacecraft x-axis until all three panels are coplanar with panel surfaces, perpendicular to the x-axis.

Once fully deployed, the wing rotates about two axes, with power going through Gimbal Actuators at each axis. Rotation required is 180 degrees for azimuth actuator and 90 degrees for elevation actuator. Position knowledge is obtained via incremental encoders at actuators and potentiometers at panel hinges.

## Power Sub-system Electronics

The basic design and operation of the LRO PSE architecture is modeled after the Microwave Anisotropy Probe (MAP) and utilizes the solar arrays to convert sunlight energy into electrical energy while in the sunlit, or solstice, period. 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. In order to achieve minimal electrical losses due to power converters as well as a maintaining a stable voltage range, the battery will be connected directly to the electrical bus.

Any excess power from the solar arrays not needed for battery charging or spacecraft loads will be shunted back to the solar array. The PSE will perform all of the functions related to power distribution as well as battery charging and will be designed for single fault tolerance while still being capable of meeting all mission requirements.

## Lithium-Ion Battery

A single Lithium Ion battery will provide power to the spacecraft. The battery energy storage capacity shall be greater than 80 Ampere-hours at Beginning of Life (BOL) within a 24 — 34V voltage range. The battery dimensions are (L × H × W) 700 × 300 × 180 mm (max value). The battery is attached to the spacecraft structure at 15 locations around the perimeter of the battery, roughly evenly spaced. Thermal control is provided at the spacecraft level though a heat pipe assembly.

The battery is constructed from cells arranged in blocks. The basic block size is 8s12p (i.e. 12 strings of 8 cells each) and seven blocks are used, to give a total of 8s84p. Within each block the cells are connected in strings with eight cells in each string in series. Four blocks are arranged to create the top deck, with three blocks in the lower deck. Four positive connections for each block in a deck are led out to a 25-way positive power connector. Four negative connections for each block and half-string taps in a deck are led to 35-way negative/telemetry connectors, In addition, PRT's and thermistors installed in the battery are run out to the internal harness connectors.

Internal 25- and 35-way harnesses are connected to the power distribution system in the lower deck. Voltage monitoring lines are protected with resistors. The positive side of the battery is connected via two relays arranged in parallel to give redundancy against open-circuit failure of relays. The relay system also includes status-monitoring lines and may be commanded from several sources.

Command lines are isolated from each other with diodes.

The SONY US18650HC Li-Ion Cell is 18 mm diameter by 65 mm high with a mass of 40.3 grams. The cell is a steel can (anode) containing a roll of interleaved electrodes soaked in electrolyte, with the necessary connections attached and cell safety features in the top cap (cathode). The insulation desks are used to isolate the electrode roll from the can top with electrical connections provided by tabs and the central anode pin. The cathode cap is crimped to the can. The cells are each identical SONY 18650HC from the same manufacturing lot, with a high level of uniformity in performance characteristics.

## Command and Data Handling (C&DH)

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The command and data handling sub-system includes a single board computer (SBC), a mass storage system, a space wire (SpW) interface which serves as a high speed interface for the instruments, and a 1553B Mil-standard low speed bus. Each has some level of heritage with previous NASA and/or DoD missions.

The functions of the C&DH system include:

1. Hosting the attitude control system (ACS) and flight software (FSW)
2. Command and data handling functions (command acceptance and distribution, telemetry collection, science data storage) for the instruments
3. Provide the interface for low rate telemetry, and command and control of spacecraft subsystems
4. Provide the interface to the spacecraft communications transmitter and transponder for high speed telemetry (selected instruments and subsystems requiring high data transfer rates)
5. Science data formatting

The C&DH subsystem has the following features:

1. Ten Printed Wiring Assemblies (PWAs) in a single housing.
2. The extensive use of SpaceWire European Cooperation for Space Standardization (ECSS-E-50-12A) and MIL-STD-1553 allows expandability and scalability.
3. S-Band Communication (SComm)/ Ka-Band Communication (KaComm) cards are two independent functions, allowing the addition or deletion of either without redesign. They are referred to as a single assembly, Communications (Comm) card, since they share a common backplane connector.
4. The Single Board Computer (SBC) is a British Aerospace Engineering (BAE) off the shelf product, with the addition of MIL-STD-1553 Bus Controller (BC)/Remote Terminal (RT) and 4-port SpaceWire interface.
5. Four Data Storage Boards (DSBs) operate as a mass storage system for the spacecraft. The DSB boards are designed to interface with the SBC via a Compact Peripheral Component Interconnect (cPCI) backplane interface.
6. The Housekeeping / Input Output (HK/IO) card provides an interface to the LAMP instrument as well as providing synchronization to all instruments on the spacecraft.
7. The Multi-function Analog Card (MAC) has been custom designed to manage the wide range of thermal environments in the lunar orbit.
8. The Low Voltage Power Converter (LVPC) provides power to all the subassemblies except the Primary Ultra Stable Oscillator (USO).
9. The backplane provides interconnectivity to all the cards within the C&DH enclosure.
10. The primary and redundant USOs are components of the C&DH subsystem, but are externally mounted separate assemblies.

## Guidance, Navigation, and Control (GN&C) Attitude Control System (ACS)

The GN&C Attitude Control System (ACS) controls the pointing of the LRO spacecraft. Through the use of Star Trackers and Coarse Sun Sensors, the ACS will determine where the spacecraft is currently pointing in fine and coarse accuracies, respectively. A three axis, ring laser gyro (called an Inertial Measurement Unit) measures the rate at which the spacecraft is rotating. The use of Reaction Wheels allows the spacecraft to smoothly point into 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 available to provide steering (attitude control) during large thruster firings (80 N) for Lunar Orbit Insertion (LOI). Additionally, the small thrusters are available to use every two weeks to zero out the momentum buildup in the Reaction Wheels and to perform station-keeping maneuvers while in the mission orbit.

The Attitude Control System (ACS) will determine spacecraft attitude, guidance to reach the desired pointing vector, and use actuators to achieve the desired pointing vector. Additionally, the ACS will provide pointing support for High Gain Antenna and Solar Array.

The ACS consists of:

1. Sensors
   a. Star Trackers (2)
   b. Inertial Reference Unit (1)
   c. Coarse Sun Sensors (10)
2. Actuators
   a. Reaction Wheels (4)
   b. 20-Newton ACS thrusters (8)

The Propulsion Deployment Electronics (PDE) component will provide thruster control as well as inhibit during launch. GN&C Flight software (FSW) will be executed within the C&DH Single Board Computer (SBC).

Four reaction wheels have the primary purpose of managing nadir pointing requirement for the spacecraft, and will be used initially as the primary means of orienting the spacecraft after separation from the launch vehicle. The mass of each wheel is 13.2 kg, 18 inches in diameter, and 6 inches in height. The fourth reaction wheel allows for redundancy throughout the mission, such that any one reaction wheel failure will allow the mission to continue. Momentum dumping is forecast at approximately two-week intervals throughout the mission, performed by the attitude control system (small) thrusters.

The wheels are mounted on the spacecraft structure. Operationally they will be 'off' at launch and will not be turned 'on' until after launch vehicle separation. At that time, the wheels will be powered and will begin to adjust the angular momentum vector that results from the separation sequence.

## Thermal Control System

The Thermal Control system basic design contains four major elements: the spacecraft bus (avionics, reaction wheels, and battery), instrument module, propulsion module, and deployables (high gain antenna and solar array). The overall orbiter is designed with mainly zenith directed radiators providing minimal exposure to the lunar environment.

Most 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 radiator that is separately mounted on the zenith surface of the spacecraft.

For the reaction wheel assemblies, two baseplate heat pipes are connected to two header pipes, that are then routed to the back of the Iso-Thermal Panel through a hole in the structural panel.

The Lithium Ion battery is maintained on a separate CCHP network with two heat pipes mounted on its sidewalls, and an additional pipe coupled to the sidewalls to provide additional reliability (one fault tolerance).

A de-coupled instrument optical bench (low thermal distortion) uses low coefficient of thermal expansion (CTE) M55J material, is fully blanketed, and is heated with low density heaters to maintain cold limit temperatures. The CRaTER and LEND instruments are provided with individual radiators that are thermostatically controlled. Mini-RF electronics are attached to the avionics header. The Diviner electronics box is mounted on the iso-thermal panel, while the remaining instruments are thermally isolated.

Within the propulsion module, most components are attached to a temperature controlled cylinder that is part of the spacecraft bus structure, to include the upper propellant tank, pyro 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 heaters to maintain temperatures within allowable limits. The gimbal assembly has four heater circuits and two radiators to serve the actuators and rotary joints. The laser ranging telescope (mounted on the HGA) has a dedicated heater circuit with MLI thermal isolation. Optical fiber from the laser ranging telescope has a copper strap coupled to the high gain antenna structure. The solar array assembly uses MLI and heater control to maintain actuators and dampers.

## Communications System
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The communications system consists of S-band forward and return links to support command, communications, tracking and telemetry, and a Ka-band return link for telemetry and science data transfer.

S-Band forward link data rate is fixed at 4 kbps. The S-Band return link data rate is selectable on orbit and varies from 125 bps to 256 kbps. The Ka-band return link is also selectable on orbit and varies from 25 Mbps to 100 Mbps.

The Ka-band sub-system includes a Ka-Band modulator, a Traveling Wave Tube Amplifier (TWTA) consisting of a traveling wave tube (TWT), an Electronic Power Conditioner (EPC), and a High Power Isolator. The S-Band subsystem consists of one Spacecraft Tracking and Data Network (STDN) compatible transponder, an S-band RF Switch, and the RF paths to and from the TT&C omni-directional antennas and the S-band feed on the High Gain Antenna (HGA).

The Ka-band uses only the High Gain Antenna. The S-band can utilize either the Omni-directional or the High Gain Antennae as controlled through the RF transfer switch.

Diplexers allow the transponder's receiver and transmitter to connect to a common antenna port that in turn connects to the appropriate RF network for each antenna system. The diplexers also include band pass and band reject filters in the transmit channel to suppress any receive signal component in the output of the transmitting power amplifier.

The traveling wave tube amplifier (TWTA) operates in a bandwidth of 300 MHZ (± 150 MHz) with a voltage standing wave ratio (VWSR) of 2:1. Output power is a minimum of 40 Watts. On/off commands are generated through a 28V signal into the electronic power conditioner (EPC).

## Data Storage Boards (DSB)
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The four DSB cards are designed as a file system which handles the storage and retrieval of files. The DSBs provide 384 Giga bits (Gbits) at Beginning of Life (BOL) of memory capacity for incoming data files for a 17.5 hour minimum science data and HK data collection.

The DSB receive dedicated +3.3VDC MEM power for the main power source and +15 SWITCH power to power the switches that allow the SBC to turn off individual cards. Power is received via the backplane cPCI connector.

Data transfer to and from the SBC is via a cPCI interface on the backplane.

**Instrument Module**

The instrument module includes a graphite composite optical bench with honeycomb panels, inserts, and spools. Flexures for mounting the instruments are titanium. Three instruments are mounted on the optical bench (LAMP, LOLA, LROC) and the remainder are on the spacecraft bus (Diviner, Crater, LEND, and the mini-RF antenna). Star Trackers complete the instrument module.

The Instrument module (IM) is of a 'Wine Box' design, which reduces mass (fewer fasteners) and increases stiffness with continuous bond lines (not just connected at fasteners). The entire structure has a low coefficient of thermal expansion (CTE) made of graphite material (M55/CyanateEster Q.I. sheets). Insert material is Titanium with BR127 electrical conductive primer.

**LRO SCIENCE INSTRUMENTS**

LRO investigations provide the following high-priority measurement sets:

1. Characterization of the deep space radiation environment in lunar orbit, including neutron albedo (especially at energies in excess of 10 MeV, as well as:
   a. Characterization of biological effects caused by exposure to the lunar orbital radiation environment
   b. Characterization of changes in the properties of multifunctional radiation shielding materials caused by extended exposure to the lunar orbital environment
2. Geodetic lunar global topography (at landing-site relevant scales)
3. High spatial resolution hydrogen mapping of the Moon's surface
4. Temperature mapping in the Moon's polar shadowed regions
5. Landform-scale imaging of lunar surfaces in permanently shadowed regions
6. Identification of putative deposits of appreciable near-surface water ice in the Moon's polar cold traps
7. Assessment of meter and smaller-scale features to facilitate safety analysis for potential lunar landing sites
8. Characterization of the illumination environment in the Moon's polar regions at relevant temporal scales (i.e., in terms of hours)

Six instruments were selected for LRO to provide these measurement sets as well as ancillary datasets relevant to numerous outstanding lunar science questions. These instruments are:

*1. Cosmic Ray Telescope for the Effects of Radiation (CRaTER):*

CRaTER will investigate the effect of galactic cosmic rays on tissue-equivalent plastics as a constraint on models of biological response to background space radiation. Specific science and measurement objectives are:

a. Measure and characterize the Linear Energy Transfer (LET) spectra of galactic and solar cosmic rays (particularly above 10 MeV) in the deep space radiation environment most critically important to the engineering and modeling communities to assure safe, long-term human presence in space.
b. Develop a simple, compact, and comparatively low-cost instrument, based on previously flown instruments, with a sufficiently large geometric factor to measure LET spectra and its time variation globally in the lunar orbit.
c. Investigate the effects of shielding by measuring LET spectra behind different amounts and types of areal density materials, including tissue-equivalent plastic.
d. Test models of radiation effects and shielding by verifying/validating model predictions of LET spectra with LRO measurements, using high-quality galactic cosmic rays (GCR) and solar energetic protons (SEP) spectra available contemporaneously with ongoing/planned NASA (ACE, STEREO, SAMPEX) and other agency spacecraft (NOAA-GOES).

*2. Diviner Lunar Radiometer Experiment (DLRE):*

DLRE will chart the temperature of the entire lunar surface at approximately 500 meter horizontal scales to identify cold-traps and potential ice deposits. Specific science and measurement objectives are:

a. Map Global Day/Night Surface Temperature
b. Characterize Thermal Environments for Habitability
c. Determine Rock Abundances Globally and at Landing Sites
d. Identify Potential Polar Ice Reservoirs
e. Map Variations in Silicate Mineralogy

*3. Lyman-Alpha Mapping Project (LAMP):*

The Lyman-Alpha Mapping Project (LAMP) will observe virtually the entire lunar surface in the far ultraviolet. LAMP will search for surface ices and frosts in the polar regions and provide frost abundance, landform and surface UV spectral maps of permanently shadowed regions illuminated only by starlight and interplanetary Lyman alpha. Specific science and measurement objectives are:

a. Identify and pinpoint surface exposed frost in Permanently Shadowed Regions (PSRs).
b. Map all permanently shadowed regions with resolutions down to 100m.
c. Demonstrate the feasibility of natural starlight and Lyman-Alpha sky-glow illumination for future lunar surface mission applications.
d. Assay the lunar atmosphere and its variability.

*4. Lunar Exploration Neutron Detector (LEND):*

LEND will map the flux of neutrons from the lunar surface to search for evidence of water ice, and will provide space radiation environment measurements that may be useful for future human exploration. Specific science and measurement objectives are:

a. Determine hydrogen content of the subsurface at the polar regions with spatial resolution of 10km and with sensitivity to concentration variations of 100 parts per million (ppm) at the poles.
b. Characterization of surface distribution and column density of possible near-surface water ice deposits in the Moon's polar cold traps.
c. Global mapping of Lunar neutron emissions at an altitude of 30-50 km above Moon's surface, with a spatial resolution of 5 km (pixel radius) at the spectral range of thermal energies up to 15 MeV.

*5. Lunar Orbiter Laser Altimeter (LOLA):*

LOLA will determine the global topography of the lunar surface at high resolution, measure landing site slopes, surface roughness, and search for possible polar surface ice in shadowed regions. Specific science and measurement objectives are:

a. Global Geodetic Lunar Topography.
b. Characterize Polar Region Illumination.
c. Image Permanently Shadowed Regions.
d. Contribute to the assessment of meter-scale features to facilitate landing-site selection.
e. Identify surface polar ice, if present.

*6. Lunar Reconnaissance Orbiter Camera (LROC):*

LROC will acquire targeted narrow angle images of the lunar surface capable of resolving meter-scale features to support landing site selection, as well as wide-angle images to characterize polar illumination conditions and to identify potential resources. Specific science and measurement objectives are:

a. Landing site identification and certification, with unambiguous identification of meter-scale hazards.
b. Mapping of permanent shadows and sunlit regions.
c. Meter-scale mapping of polar regions.
d. Repeat observations to enable derivation of meter-scale topography.
e. Global multispectral imaging to map ilmenite and other minerals.
f. Global black and white morphology base map.
g. Characterize regolith properties.
h. Determine recent small impactor rates by re-imaging regions photographed with the Apollo Panoramic Camera (1-2 meter m/pixel).

**Instrument Host Overview — SCN**
**================================**

The primary function of the SCN is to provide two-way communications between the Earth and the LRO spacecraft. S-band is used to support commanding (4 kbps), telemetry, and tracking. Ka-band telemetry (100 Mbps) is used to downlink science instrument data. During operations after commissioning, the SCN consists of the White Sands 1 (WS1) ground station, S-Band Sites, and Laser Ranging (LR) site. WS1 transmits s-band commands to the spacecraft and receives both s-band telemetry and Ka-band data from the spacecraft. The S-Band Sites consist of 4 Universal Space Network (USN) sites and Deep Space Network (DSN) as back-up, for transmitting s-band commands to the spacecraft and receiving s-band telemetry from the spacecraft. The Laser Ranging facility transmits laser pulses to the spacecraft as part of the capability of computing the range to the spacecraft.

The WS1 facility includes an 18-meter S/Ka-band antenna at White Sands Complex (WSC), New Mexico. The USN facilities include: a USN 13-meter s-band antenna at South Point, Hawaii; a German Aerospace Center (DLR) 15-meter s-band antenna at Weilheim, Germany; a Swedish Space Corporation (SSC) 13-meter s-band antenna at Kiruna, Sweden; and, a USN 13-meter s-band antenna at Dongara, Australia. The DSN facilities include: a 34-meter s-band antenna at Goldstone, California; a 34-meter s-band antenna at Madrid, Spain; and, a 34-meter s-band antenna at Canberra, Australia.

The LR site in Greenbelt, Maryland transmits 532 nm laser pulses to the LRO spacecraft. The receiver telescope on the spacecraft High Gain Antenna System (HGAS) provides the Lunar Orbiter Laser Altimeter (LOLA) instrument with the LR signal via LOLA channel 1. LR range data are sent from the LOLA instrument to the spacecraft and are then included in LOLA telemetry sent to the LRO Mission Operations Center (MOC), which in turn provides these data to the LOLA Science Operations Center (SOC). The SOC computes the range to the spacecraft."

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