Lower-Cost, Relocatable Lunar Polar Lander and Lunar Surface Sample Return Probes

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Abstract—Key science and exploration objectives of lunar robotic precursor missions can be achieved with the Lunar Explorer (LEx) low-cost, robotic surface mission concept described herein. Selected elements of the LEx concept can also be used to create a lunar surface sample return mission that we have called Boomerang.

The LEx mission would land a small, re-locatable lander within the lunar South Polar Region to evaluate distribution of volatiles, dust, illumination, provide geologic context, and assess local hazards in both sunlit and permanently shadowed environments. Improved assessment and understanding of lunar polar region cold-trap environmental characteristics, geophysics, and potential resources (water ice) will have direct impact on future robotic and human exploration architecture decisions. LEx represents a highly-focused pathfinder, exploring both sunlit and shadowed sites within the lunar south polar region to provide essential ground-truth for understanding resources and for linking surface observations with those from orbital remote-sensing (i.e., from LRO).[1 2]

The mission is designed to be co-manifested as a secondary payload on any future lunar mission allowing further cost savings and programmatic flexibility. Upon landing within a sunlit south polar region, LEx will perform independent, specific, and quantitative measurements of the accessible lunar regolith. The LEx mission continues with a new and innovative mobility approach that enables a multi-kilometer, powered flight to a nearby permanently shadowed site where a full suite of measurements will be repeated. This provides first-ever landed access to a permanently shadowed region and a "ground-truth" dataset for environmental characterization, potential resources assays, local hazards reconnaissance, as well as validating an innovative near-surface mobility approach. Additional LEx lander flight systems can be acquired at a fraction of their initial cost to serve as an asset in precursor robotic missions such as those envisaged in NASA's new xScout program. The low cost, globally targetable, mobile, precision-landed robotic access to the Moon that can be provided by LEx has significant potential to reduce risk and cost of future landed systems. LEx was originally designed in mid 2005 and dubbed the 'hopper' on the basis of its powered-flight relocation capability, which contrasts with traditional overland roving.

A similar design approach and design was used in developing a NASA Discovery class lunar surface sample return concept (Boomerang). The proposed Boomerang landing region would be within the South Pole Aitken Basin (south polar highlands) where a simple vacuum sample acquisition system acquires a ~100g sample from the uppermost 20cm of the regolith. The Boomerang lander/sample acquisition system would then lift off and deploy a sample canister capsule to low lunar orbit, which is then captured by a simple spacecraft for rendezvous and Earth return and entry. This mission can directly address prioritized lunar precursor robotic mission goals, serve as a pathfinder for enabling future low-cost, risk-tolerant robotic missions, and address high priority NRC Decadal Survey (Planetary), and NRC lunar science (SCEM 2007) objectives. Both missions leverage previous investments, and accept a slightly higher risk posture to reduce cost and shorten development schedules.

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[2] IEEEAC paper #1099, Version 2, Updated October 28, 2010

1.0 SMALL MOBILE LUNAR EXPLORER–LEX

In 2005 NASA's Goddard Space Flight Center (GSFC) proposed the LEx lander mission approach to the NASA LRO co-manifest opportunity. A version was also proposed with partners to an earlier Robotic Lunar Exploration Program mission 2 (RLEP2) opportunity. Both were selected for step two consideration but not for flight implementation. The approach and mission, which was dubbed the 'hopper', are relevant to current discussions of human exploration robotic precursors and potential opportunities for lunar activities in the next 10 years. Specific details of this mission concept have not appeared in public until now. The LEx mission addresses lunar robotic goals and objectives that have been formally discussed in association with Exploration robotic precursors and science (i.e., NRC SCEM, 2007), as follows:

• Demonstrate a lower cost lunar robotic landing approach (Braking-Descent-Landing or BDL), characterization of the lunar polar environment, and evaluation of potential lunar landing sites and their resource potential for human exploration.

• Early assessment and understanding of water ice as a lunar polar resource will have direct impact on robotic and human exploration architecture decisions in the upcoming 10 years.

LEx would demonstrate a new paradigm for the development of low-cost lunar surface access by leveraging extensive Defense Department investments in technologies. These technologies consist of a flight-proven high-thrust liquid propulsion system with exceptional dynamic controllability algorithms. The LEx team would use experience in short development cycles and special payload development efforts, as well as the use of co-located skunk works operations via proven streamlined processes to successfully conduct surface operations at a fraction of the traditional costs.

LEx would serve as an early step toward "go anywhere" lunar surface access with a new class of near-surface mobility for catalyzing and sustaining the possibility of future human lunar exploration. LEx represents a pathfinder, exploring both sunlit and shadowed sites in the lunar south polar region to provide essential ground-truth for understanding resources such as water ice and for linking surface observations with those from orbital remote-sensing (i.e., from LRO).

Upon successful landing in a sunlit south polar region LEx will perform independent, specific, and quantitative measurements to detect water ice within the upper several meters of the lunar regolith at sensitivity levels required for the determination of its resource potential. The LEx instrument suite will provide strategic knowledge of the lunar polar region environment including distribution of volatiles, dust, illumination, geologic context, and local hazards. The LEx mission continues with demonstration of a new mobility approach that enables a multi-kilometer powered flight to a nearby shadowed site where a full suite of measurements will be repeated. This provides first-ever access to a solar shadowed region and a second "ground-truth" dataset for environmental characterization, potential resources, local hazards, as well as validating an innovative near-surface mobility approach. A summary of the LEx baseline mission architecture, including the mission phases, capabilities, relevance to the NASA human exploration objectives and associated mis-sion objectives, is shown in **Table 1**.

Table 1: The LEx mission accelerates NASA's implementation of future human exploration missions to the lunar surface (and other destinations).

Mission Phase	Activity	Capabilities	Relevance to Vision for Space Exploration	LEx Mission Objectives
Baseline Mission				
Landing and Initial Measurement Phase	Precision land at south polar sunlit region in zone of known high hydrogen near shadowed crater (no hazard avoidance required) Conduct first set of measurements: measure water ice potential; characterize hazards, dust, geological context; collect video images Maintain DTE communications link Time at initial sunlit landing site: 1–7 days	High thrust propulsion for precision landing (i.e. ~1 km) and subsequent flight; 4 propellant tanks Instrument suite to conduct *in situ* search for resources and characterize lunar environment Volumetric water ice sensing Onboard DSMAC to acquire data of landing site region	Support identification of future human landing sites; evaluate polar resources Characterize polar lunar environment (resources, dust, illumination) for human exploration Autonomous precision landing Demonstrate operational concepts and technologies to enhance future exploration	Baseline Measurement Objectives 1, 2, 3, 4 (See Table 2)
Fly to Shadow and "Touch the Ice" Phase	Fly into permanently shadowed crater Maintain DTE communications link Conduct 2nd set of measurements in shadow. Time in shadowed region: 2–12 hours	All the above, plus: —Ground processing of DSMAC and radar imagery to ensure safe landing in shadowed region —IMU guidance system to enable precision landing	Develop infrastructure: first new surface mobility approach for future exploration First shadowed region surface access and measurements	Baseline Measurement Objectives 1, 2, 3, 4, 5 (See Table 2)

Table 2: LEx measurements are traceable to strategic knowledge needs and mission science goals.

Measurement Objective	Implementation
(1) Baseline Imaging	MSL MastCam-derived telephoto & video digital panoramic camera (RGB)
(2) Water Ice Electrical Sounding	Low-frequency (Hz-kHz) electrical sounding to measure complex dielectric permittivity
(3) Water Ice Microwave Heating Experiment	Microwave heater and 22 GHz water "line" via optical fiber and quartz microbalance
(4) High Resolution Hydrogen Mapping	Active/passive neutron spectrometer with 0.1–10 Hz neutron generator (engin. model of MSL DAN)
(5) Imaging Reconnaissance	Telephoto & video-mode panoramic imaging system-based on MSL MastCam; plus descent imaging (DSMAC)

We also identify enhanced mission options for consideration with associated increased costs. These mission options retain the baseline mission's sunlit and shadowed landing sites and in situ measurement campaigns, while augmenting capabilities to explore additional sites and validate further mobility options. Additional LEx lander flight systems can be acquired at a fraction of the initial cost. A summary of mission enhancement options and potential program enhancement opportunities are summarized in Section 9.

It has been more than 40 years since a controlled robotic landing on the Moon, and LEx is an attempt to demonstrate a lower cost approach to achieving multiple exploration and science goals at the lunar surface at a higher risk posture.

Objectives—Accessing the never-before-explored lunar south polar region and its unique environments in support of human exploration is a primary goal of the proposed LEx mission. This mission will establish a baseline set of comprehensive measurements of the resource potential of this region with emphasis on water ice existence and abundance. The LEx mission will obtain the first in situ observations of the lunar polar environment, including measurements of local hazards, dust, electrical properties, and geological context. Such measurements will begin the process of opening up the lunar polar frontier to both robotic and future human exploration.

1.1 Mission Objectives

The LEx mission addresses the following objectives, in priority order.

1. South Polar Lunar Landing: Precision–land on the Moon at a south polar site where there is a priori knowledge of hydrogen enrichment (as a possible indication of water ice up to 1-2 wt.% in concentration [LRO and Lunar Prospector]).

2. Water Ice Measurements: Perform calibrated and validated measurements of specific decisive parameters (i.e., hydrogen concentration and water related electrical properties) associated with the existence of water ice within the upper few meters of the lunar surface.

3. Polar Environmental Characterization: Measure polar landing site hazards, physical materials, geomorphology, dust, and solar illumination in support of potential future human access.

4. New Class of Mobility: Demonstrate kilometer scale mobility via a powered flight to a second south polar site in shadow where there is a priori knowledge of hydrogen enrichment (and available LRO or Earth based radar imaging) with subsequent decisive measurements as described in (2) and (3) above.

5. Key Capabilities Demonstration: Demonstrate key aspects of an autonomous, precision-landed robotic flight system (i.e., flight path) in support of Autonomous Landing and Hazard Avoidance Technology (ALHAT) algorithm development, and in support of future human) landings (i.e., by Altair or similar systems).

2.0 MISSION CONCEPT

Our concept for the LEx mission is to send a small, extremely capable lander to the lunar surface in a polar region. The LEx mission utilizes a proto-flight system from a space-qualified exo-atmospheric vehicle assembly line that includes a high thrust propulsion system with four fuel tanks. The LEx lander is shown in its originally proposed design scenario [2005], integrated with the Lunar Reconnaissance Orbiter (LRO) in the payload fairing in **Figure 1**.

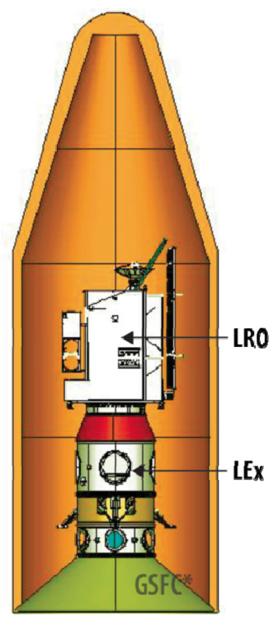

Figure 1: LEx easily fits in the EELV 4 m fairing together with a primary mission (such as LRO).

Baseline Mission—The LEx lander will first land in a sunlit area in the lunar south polar region adjacent to a solar shadowed zone for which there is evidence of a potential water ice deposit. Here the LEx instrument suite will acquire a 360˚ panoramic image, as well as the initial low-frequency Electromagnetic (EM) sounding measurements, active neutron spectrometry, and microwave heating measurements, to begin the process of searching for the presence of water ice.

Safely landed on the lunar surface in continuous sunlight with a Direct-to-Earth (DTE) communications link, the LEx lander will collect its baseline sunlit dataset and downlink it via the Deep Space Network (DSN) to Earth. The baseline dataset will serve as in situ “ground-truth” for LRO and international missions (i.e., Chandrayaan-1, Kaguya) by providing better than mm-resolution imaging, direct measurements of local hydrogen, electrical properties related to water ice, and measurements of any volatiles within the upper 1–2 meters of the polar regolith.

The mission will continue with a 1–4 km powered flight to a shadowed site (within a PSR), guided by existing high resolution radar imaging and lidar altimetry. The data acquired onboard by the Digital Scene Matching Area Correlator (DMSAC) system derived from DoD cruise missile technology. The LEx lander maintains continuous contact with the DSN and will be transmitting relocation engineering critical event information. The LEx lander will perform its second complete set of measurements at the shadowed site, including active neutron spectroscopy, EM sounding, microwave heating, and a 360˚ panorama using an illuminator. This data will be transmitted to Earth via the DSN communication link within ~2 hours. With the completion of this phase, the LEx mission will have sampled two independent measurement sites within a region of known hydrogen enrichment. Orbital neutron spectrometry and radar imaging illustrates such regions, including for example, the shadowed rim of the crater Shackleton. LEx will establish two independent data sets from which to better interpret existing orbital remote-sensing data from LRO, Lunar Prospector, and Clementine, as well as recent orbital (LRO, Chandrayaan-1) and Earth based radar imaging, thus enhancing our understanding of the resource potential of the lunar poles.

2.1 Baseline Mission Profile

2.1.1 Launch and Injection

Following the mission launch and injection to translunar trajectory by an EELV, the LRO spacecraft is separated first from the co-manifest enclosure just as it would from the EELV had there been no co-manifest. Next the EELV upper stage performs a collision avoidance maneuver and reorients to separate and release the co-manifest enclosure.

After another collision avoidance maneuver and reorientation, the EELV releases the secondary. The separation switches activate the LEx. The LEx propulsion is activated and LEx acquires a safe attitude that optimizes solar power, thermal state, and antenna pattern for initial ground communications.

2.1.2 Cruise Operations

The proven coherent two-way ranging and Doppler measurements over the lander communications link are used for precise trajectory navigation during the translunar cruise. A trajectory estimate is developed from DSN tracking data, with expected accuracy on the order of 800 meters and ~3 cm/sec at the lunar distance. Communications link analysis indicates that a 32 kbps data rate is attainable with a 5 W transmitter and existing DSN ground stations. All mission objectives can still be met with a significantly lower data rate.

2.1.3 Braking Maneuver

The landing sequence (**Figure 2**) starts about six hours

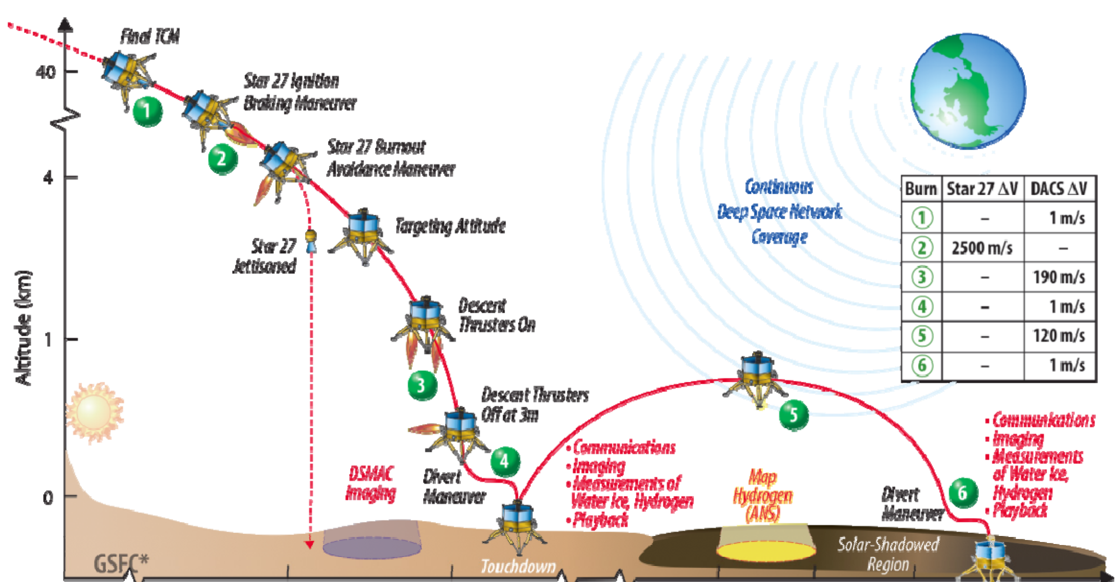

Figure 2: LEx lunar surface operations meet all mission science and exploration objectives.

before projected landing time with the last (nominally, second) TCM. Following the TCM execution, LEx is slewed to the proper attitude for the SRM braking maneuver. This attitude is verified and the post-TCM trajectory is estimated with radio tracking, then the final state vector and landing timeline are updated. The entire LEx system is spun up to ~60 rpm. At approximately T-25 seconds prior to surface contact, the ATK Star 27 braking stage is ignited, burns for ~35 s and reduces the approach velocity from 2500 m/s to <100 m/s at an altitude of 4-8 km. The time until touchdown is ~60 s at the Star 27 burnout. The spent braking stage is jettisoned and a separation and despin maneuver is performed by the Divert and Attitude Control System (DACS).

2.1.4 Landing Zone Targeting

Existing datasets from LRO (imaging, lidar, radar), Lunar Prospector, Clementine high resolution imaging, and recently acquired Earth-based radar imaging (Arecibo-Greenbank) provide adequate information for targeting the polar LEx landing. Existing visible and SAR imaging datasets offer coverage of regions of high hydrogen at adequate resolution for precision targeting and to facilitate avoidance of hazards. Local hazards are well known in the candidate landing region using from new LRO datasets at 20 cm to 2 m scales. We will be able to conduct our flight mobility tests on the basis of these datasets which are already in hand thanks to LRO.

LEx reorients to the targeting attitude and starts acquiring the DSMAC and landing radar data. The guidance, navigation and control (GNC) software commands the DACS to maneuver LEx to a predetermined position and velocity over a landing target, using radar and Inertial Measurement Unit (IMU) inputs.

2.1.5 Terminal Phase

During the terminal landing phase, the LEx Lander nulls horizontal and vertical velocity to <0.5 m/s at ~3 meters altitude. The lander assumes landing attitude and descent propulsion is turned off while attitude is controlled until the landing touchdown. The lander impacts the surface at residual velocity of <3 m/s (verti-cal) and <0.5 m/s (horizontal). The landing legs absorb energy from the landing impact with crushable material, thus alleviating probability of lander tipover and reducing landing impact shock. The crushable aluminum honeycomb is pre-compressed into a range where the material properties are linear. The material dimensions are chosen to allow two landings with margin before maximum crush depth is reached. **Figure 2** illustrates this mission architecture, including the breaking-descent-landing (BDL) phase and describes the delta-V budgets.

3.0 Example Instrument Suite, and Landing Site Selection

Understanding lunar polar resources, environments, and physical processes are directly linked to many international lunar science and exploration goals. Given evidence of potentially appreciable lunar polar volatiles in the form of water ice from hydrogen observations, confirmation via in situ measurements of hydrogen and water ice content of the lunar polar regolith are an important early step. In addition, early landed measurements will support the in situ calibration and validation of orbital remote sensing observations from NASA's LRO and other missions. More direct measurements of potential polar resources, geologic landing hazards, dust related processes, and local context will serve as a pathfinder for future landed missions and priority in situ lunar science.

Table 3: LEx advances the future of human exploration of the Moon.

Advancements	LEx Contribution
Advance lunar science by providing data or knowledge	First landed measurements of polar volatiles, geologic context at human scales
Characterize the lunar environment	Landed polar surface lighting, temperatures, dust, electrical properties; ground-truth for orbital remote-sensing (i.e., LRO)
Support identification of sites for future human missions as well as the potential utility of those sites	Landed resources evaluation, polar landing hazards, polar illumination, temperature, dust
Test or demonstrate technology that could enhance future exploration	Precision landing, descent imaging, km-scale mobility, hazard avoidance
Demonstrate operational concepts in support of exploration activities	Precision landing, descent imaging, mobility, hazard avoidance, DTE telecommunications
Develop or emplace infrastructure in support of exploration activities	Mobility asset for future landed applications, potential as passive optical beacon
Advance commercial opportunities	Significant industry & small business participation, new partnerships
Collect engineering data to support development activities of the constellation elements	Landed geotechnical data of polar surface site(s), temperature, dust, illumination geometry, quantitative ground-truth for remote-sensing

The primary focus of the LEx lander example measurement suite is detection of polar volatiles and their geologic context, in support of resources objectives at relatively low cost. Three instruments using independent methodologies investigate the nature of polar volatile deposits and validate decisively the presence of subsurface water ice at the landing locations:

1. *Active, pulsed neutron spectrometer (ANS)*
2. *Low-frequency Electromagnetic Sounder (EMS)*
3. *Microwave-heating/cold-finger water detector (MHP)*

In addition, a fourth instrument, the Panoramic Imager (PanI), provides geological context for the other instruments, detects any potential geomorphologic signatures of subsurface ice and supports other investigations including a lunar horizon dust survey. This instrument is based upon the flight certified MSL MastCam imaging system, adapted for the thermal environments of the Moon. For ITAR and space reasons only a small part of the detailed design of the PanI instruments is shared below.

3.1 Active, Pulsed Neutron Spectrometer (ANS)

Hydrogen in the lunar regolith results in thermalization of neutrons and in emission of the 2.2 MeV gamma ray line. Measurements of the die away emission curves of neutrons will definitively detect hydrogen in the upper meter of the lunar regolith. The LEx lander includes an active, pulsed neutron spectrometer (ANS) as part of its water ice detection suite. The ANS sensor field of view encompasses about one cubic meter of regolith beneath the LEx lander.

Two ANS sensors measure thermal (STN) and epithermal (SETN) neutrons. Each LEx landing site measurement requires emission of ~ 104 pulses at a typical 1 Hz pulsing rate, for a ~3 hr integration time (~1 hr if at 10 Hz). The ANS instrument can sense ~50 ppm hydrogen concentration, resulting in a water ice mass concentration sensitivity of ~0.045 wt.%. The value is far lower than current estimates of economically viable water ice (~3-5 wt.% mini-mum) as a resource for human exploration. The ANS method can also distinguish between solar wind implanted hydrogen and water ice, depending on the concentration and vertical distribution of the hydrogen bearing phases within the regolith.

3.1.1 Measurements of Water Ice Content via Neutron Albedo with ANS in Active Mode

Thermal neutrons are the most sensitive for measurement of hydrogen concentration (water ice) in the lunar regolith. The ANS instrument includes two similar sensors with 3He counters, one with and another without a Cd shield. When the water ice content varies from 0.045 wt.% (~50ppm H) up to 10 wt.%, the average flux of induced thermal neutrons increases by approximately a factor of 10.

Table 4 presents the required measurement time for "decisive" detection of water ice in the lunar polar

Table 4: The LEx ANS instrument will detect water ice relative to a "dry" reference site in less than a minute at its highest operating frequency of 10 Hz and in less than an hour at 0.1 Hz.

Water ice content (wt%)	Time for detection with ANS operating at 10 Hz (sec)	Time for detection with ANS operating at 0.1 Hz (sec)
0.1	30.0	3000
0.3	2.1	210
1.0	0.34	34
3.0 (resource)	0.11	11
10.0 (resource)	0.05	5

regolith at the LEx shadowed measurement site in comparison with a worst-case sunlit landing site with a small concentration of ~0.045 wt.% of hydrogen (3 detection threshold). Neutron activation methods are very sensitive to different water ice deposit layering geometries within the shallow lunar subsurface. Diffusion and moderation processes of neutrons in a substance with different water ice stratigraphies are associated with distinctive time profiles of the die away emission of induced neutrons.

The layering structure of the lunar polar regolith can be investigated at each of the LEx lander measurement sites (sunlit and shadowed) with active ANS operations. We anticipate requiring at least 1 hour for reliable counting statistics in the LEx "shadowed measurement site" at 10 Hz to ensure adequate discrimination of water ice scenarios.

3.1.2 In-flight Passive Operations of the ANS for Ice Detection

ANS will also be used in a passive mode of operation (i.e., with no emission of pulses from its PNG generator) during the powered flight of LEx above the lunar surface as it transits from its initial sunlit landing location to a shadowed site. Neutron detectors within ANS will detect the natural neutron emission of the lunar surface due to galactic cosmic rays. We intend to operate ANS as LEx transits up to ~100 s from the initial sunlit landing site to a shadowed site several kilometers away.

For ANS operating in its passive mode, the counting rate of epithermal neutrons is ~10 counts/s above a canonical desiccated lunar regolith. When the content of water ice is ~10 wt.%, the flux of epithermal neutrons decreases by a factor of 5 (to ~2 counts/s). This decrease would be readily detectable by ANS in only tens of seconds of flight.

3.2 EM Sounder (EMS) for Water Ice Detection

Studies by Onsager & Runnels [1969], Olhoeft [1998] and oth-ers indicate that water ice deposits, even at low mass fractions (~0.1%), can be robustly discriminated from dry regolith by measuring the dielectric relaxation time constant of the subsurface regolith. **Figure 3** illustrates this unique water ice signature as a function of frequency.

The LEx lander EMS experiment operates by applying a voltage sine wave across two of the lander electrodes (landing footpads) and measures the value of alternating current flowing through the circuit that is established. The two electrodes create a capacitor, and the regolith serves as the dielectric of this capacitor. The instrument relies on capacitive coupling because of the known extremely high resistivity of the lunar regolith [Heiken et al., 1991].

The EMS instrument produces a point measurement by sensing a weighted average signature of the subsurface regolith layer. Both the EMS and ANS sensors generate volumetric measurements beneath the LEx lander without any physical subsurface access (i.e., drilling).

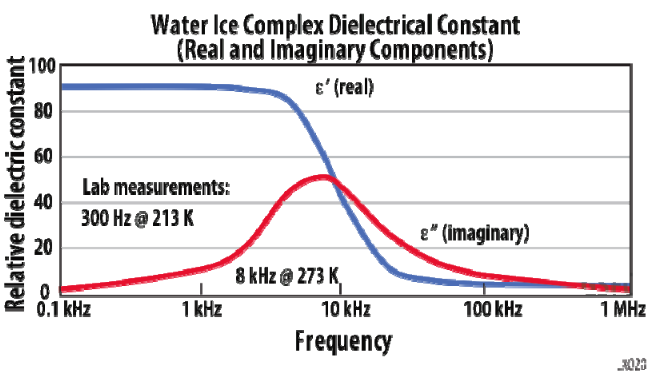

Figure 3: Low frequency electrical signature of water ice

3.3 Microwave Heating Probe (MHP)

This unique experiment injects microwave energy into the regolith to a depth of tens of cm. The regolith is heated and the vapor pressure of any volatiles that are present is raised, thus causing vapor pressure increased diffusion through the porous regolith material.

The instrument transmitter is a miniaturized integrated assembly. The transmitter is mounted on a payload shelf and pointed downward at the regolith beneath the lander, and is relatively height insensitive.

Any evolved volatiles produced by the microwave heating are readily detected by simple and commonly used sensors attached to a cold finger (passively-cooled).

3.4 Panoramic Imager (PanI)

The PanI consists of a camera head and a digital electronics assembly, connected by a cable. The camera head consists of a zoom/focus optical assembly, a focal plane assembly and the camera head electronics. The optical assembly consists of the optical elements, their housing and the zoom/focus mechanisms. It provides a focal length range from 6.5 mm to 100 mm, corresponding to a field of view ranging from 90˚ to 6˚. The lens images onto CCD detector with an integral color filter array to provide RGB color imaging capability. The PanI has two imaging modes: still-frame and high definition video.

The PanI will extend knowledge of lunar regolith development to an environment (the lunar polar region) never before seen. The spatial resolution (0.5 mm/pixel) permits quantitative grain size measurements from a millimeter scale near the LEx lander to tens of cm at distances of 1 km. In addition to characterizing the impact craters and rock populations over an area of several hundreds of m2, PanI will look for unique landforms and regolith textures that may reflect polar processes.

PanI also features internal image compression capabilities to facilitate optimized transfer of image data to Earth.

3.5 DSMAC Imaging (during BDL)

As part of LEx activities, we plan to utilize the DSMAC system to experiment with hazards mapping of solar shadowed regions of interest to exploration, as illuminated via its built-in illuminators from altitudes above terrain as high as 1000 m. On approach to the initial sunlit touchdown site, we will activate DSMAC in an overflight of a shadowed region and acquire zero-phase angle images in stereo (along-track) for post-landing analysis.

3.6 Laser Reflector

The laser reflector to be flown on the LEx lander will enable the location of the lander to be determined by LRO's laser altimeter (LOLA) or via a similar orbiting lidar instrument. This can be accomplished to an accuracy level of about 10 m in a global lunar geodetic network and independently of the LEx lander being in sunlight or permanent darkness. This will provide the location of any scientific or exploration observations obtained by LEx and for this location to be revisited at any time in the future. Further, the reflector, which can receive and return laser signals from any direction within 60˚ of zenith and about 50 km distance (assuming an uninterrupted view), could be used as a marker for precise navigation back to the same location.

Measurements Operations—The guiding principles associated with the measurement operations outlined below, center around acquisition of all necessary data as quickly as possible after the initial landing and any subsequent powered-flight landings. We recognize that acquisition of a complete set of observations as described in the measurement objectives will require several hours in the shadowed, cold-trap environment and have built measurement activity sequences around minimal time acquisitions and efficient data downlink to ensure success.

Measurement operations will commence during the terminal phase of descent to the lunar surface. The DSMAC imaging system will acquire at least three images during the final 100 m of BDL, before the final divert maneuver to provide sub-meter scale geologic context of the initial sunlit landing site. Upon touchdown, and after establishment of DTE telecommunications, landed measurements for exploration will be initiated.

Initial landing site measurements include a 360° panoramic image via the PanI system. An additional telephoto image will be acquired of the surface landscapes at the margin of the shadowed region where LEx will later fly. At least two independent sets of EMS measurements will be acquired and stored on board the LEx lander for playback to Earth. After EMS measurements, the Microwave Heater Prober (MHP) will be activated for a series of active microwave heatings of the lunar regolith beneath the lander. Following the MHP heatings, the ANS neutron spectrometer will be operated in a passive mode for ~1 hour followed by downlink of all data. The second phase of measurements will commence with activation of the ANS pulsed neutron generator. Currently, we are baselining 1 hour of high pulse rate ANS operations (at 10 Hz), but we will evaluate the merits of operating for a longer period but at a lower pulse rate during power budget trade analysis activities.

Upon activation of powered flight, ANS will be activated in passive mode in an effort to detect the edges of enhanced hydrogen deposits at sub-km spatial scales. During the terminal phases of its powered landing in the second, shadowed site, the DSMAC system (with illuminators) will acquire imaging data as part of geologic context and for on-the-ground hazards assessments. We anticipate LExoperations within the shadowed region will require at least 1 hour, and another hour for complete downlink to Earth. Additional measurements will be considered on an "as-needed" basis within the remaining power budget at the shadowed region site.

Landing Site Contamination—Any lander that relies on powered descent must address the issue of potential surface contamination from thruster effluent. The LEx mission strategy for addressing this issue is centered on two mitigation measures:

1. The LEx lander executes a divert (horizontal) maneuver at the initiation of the terminal landing phase, at an altitude of 3.5-5 m (depending on the radar altimeter performance). Approximately ~1 second later, the lander executes the opposite divert thruster firing to null the horizontal velocity. This divert maneuver offsets the landing site location by ~15 m from the intermediate descent aiming point. Thus, the actual landing site will never be directly contaminated by descent thrusters.

2. The last firing of the LEx lander descent thrusters occurs at an altitude of ~3 m prior to divert, thus minimizing any downward flow of propulsion contamination. The lander is designed to free fall from that height, though its attitude is still actively controlled by top-mounted firing ACS thruster to assure stable landing.

3.7 Selection of Candidate Landing Sites

Existing lunar remote sensing datasets (LRO, Kaguya, Chandrayaan-1, Lunar Prospector, Cle-mentine, Earth based Arecibo-Greenbank S-band Radar) have identified several promising landing sites for the baseline LEx mission. The baseline option used for this example mission design is a distinctive region (**Figure 4**) that maximizes potential

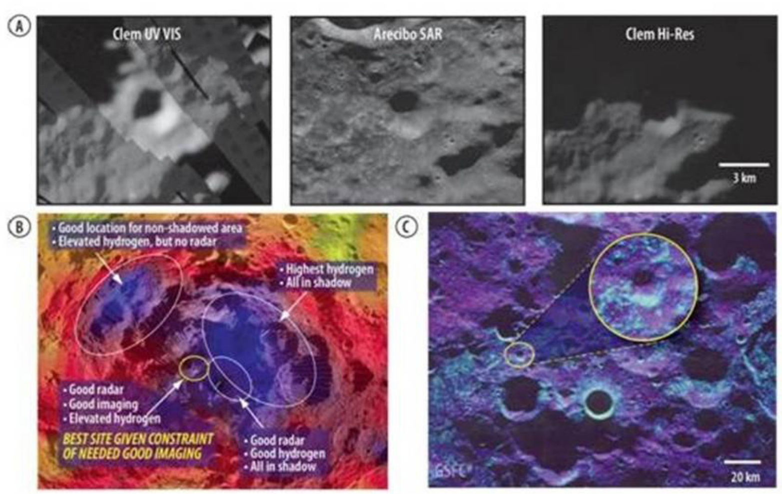

Figure 4: (A) Best available Clementine and Earth-based S-band radar imaging of a candidate LEx landing site with elevated hydrogen and shadowed terrain. (B) Image of lunar south polar region from Clementine with superimposed hydrogen concentration data in color from LP, with blue and purple as highest levels (100's ppm). One example of low-risk LEx landing zone is illustrated in the circle to the left of center, which is ~ 40 km from the rotational pole. (C) Currnetly available circular polarization ratio (CPR) radar imaging data (D. B. Campbell et al., 2006) for example LEx landing region: purples represent smooth surfaces with few rocks > 25 cm in scale. There are several candidate LEx landing ellipses within the south polar region that will be evaluated with these datasets and others (i.e., LRO) as they become available.

mission success. This region consists of a smooth plateau that remains sunlit most of the lunar year with an elevated LP and LRO hydrogen signature and lower levels of landing hazards.

Figures 4A and **4B** clearly indicate that sunlit, smooth regions exist that meet the LEx landing engineering constraints, including DTE visibility for telecommunications, known elevated hydrogen signature and less chance of major landing hazards. We recognize that others exist (e.g., the distal rim region of Shackleton crater) and the mission design meets the requirements for many of these other targets, and any of these could be a target for a LEx mission.

4.0 LUNAR EXPLORER BUS

The LEx bus design leverages mature and flight proven propulsion, guidance and software elements from DoD and NASA GSFC, resulting in an affordable bus with low developmental risk, high reliability, and precision landing capabilities. Leveraging flight tested components from the successful EKV vehicles, coupled with the real time scene matching capability creates an innovative, low risk design capable of providing a controlled soft landing.

The LEx bus design emphasizes modularity by allowing integration of the avionics module, propulsion module, and landing leg structure to occur in parallel, a key attribute to meeting a future a secondary payload schedule. A modular bus design integrated with a Star 27 braking SRM is shown in **Figure 5**.

A key for precision mobility is a high thrust-to-weight ratio propulsion system with high dynamic controllability. Improved payload mass fraction is achieved through multiplicative effects of high specific impulse (Isp) and larger thrusters than have been used previously for planetary missions. This enables efficient ‘fly to’ mobility. For space and ITAR and reasons, only some of the detailed design specifics are discussed below.

4.1 Technical Resources

The LEx mission has more than adequate technical resources to achieve its mission. The power system has 30-40% margin depending on mode and over 300% margin on time needed in the shadowed region. The communications system maintains 3 db margin on all links and has a 200% margin on telemetry rates. The total V for the DACS system is 353 m/s with 120 m/s unused or 34% reserve.

The LEx mass contingency is 17.4 kg plus 14 kg of margin for an overall margin of 32%. The LEx lander is shown in relative scale in **Figure 5**, mass estimates in **Table 5**, and power estimates in **Table 6**. The height from landing gear footpad to the top of the solar panels is 1.6 m. The diameter of the circle that circumscribes the landing gear footpads is 2.7 m.

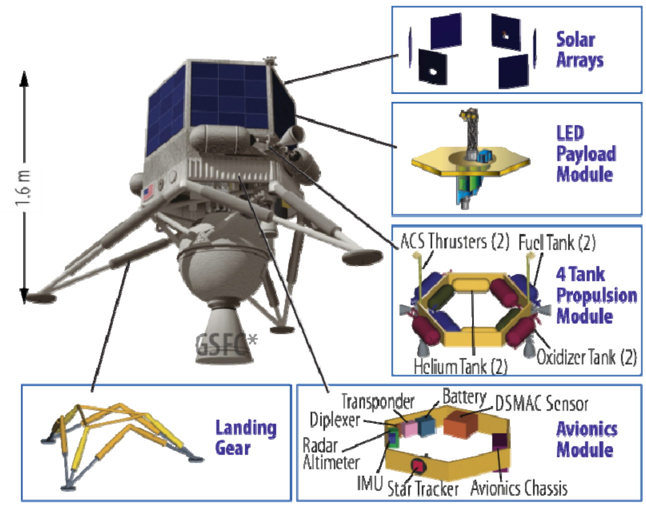

Figure 5: Modular Bus Design integrated with a Star 27 Braking SRM.

4.2 LEx Bus Subsystems

4.2.1 Avionics

The flight proven Power PC provides the main computational capability. An I/O Circuit Card Assembly (CCA) interfaces with the PCI Bus to provide interfaces to several I/O devices (e.g., IMU, instruments, star tracker). The Valve Driver CCA provides interfaces for all the propulsion for controlling squibs initiation, battery activation, divert thruster and GN&C control. The gyros have a very high level of accuracy with a bias repeatability of 1 deg/hr. A power control unit CCA provides all the power regulation and switching to the electronic subsystems from the battery bus. All electronic components are selected to meet the total dose and single event effects over the mission lifetime.

Table 5: Mass breakdown for LEx lander.

LEx Subsystems	CBE (kg)	Max Expected (kg)
Navigation and Attitude Determination	9.9	11.2
Command and Data Handling (C&DH)	4.9	5.7
Communications System	5.3	6.2
Harness	3.2	4.2
Power	20.2	23.9
Structure	23.9	31.1
Thermal Control	7.4	8.1
DACS Propulsion System	21.4	23.2
Trapped Propellant and Helium	1.9	1.9
Total Dry Mass without Payload & System Reserve	98.1	115.5
Payload Mass	20.0	20.0
System Reserve	14.0	14.0
Total Lander Dry Mass	132.1	149.5
DACS Usable Propellants	28.8	28.8
Wet Mass (post-SRM jettison)	160.9	178.4
Braking SRM Dry Mass & Interface Hardware	34.3	39.5
Descent Stage Mass at SRM Burn-out	195.2	217.8
Braking SRM Propellant—ATK Star 27	310.4	310.4
Wet Mass (pre-SRM braking burn)	505.6	528.3
DPAF and Adapters	350	455.0
Mass Total (Total Launch Vehicle Separated Mass)	855.6	983.3

Table 6: Preliminary power estimates by mission phase.

Average power [w]	Mission Phase					
	Cruise	Braking	Landing	Srfc. Ops (Sunlit)	Flying	Srfc. Ops (Cold-Trap)
Lander	85	86	185	95	186	90
Payload	0	0	0	30	12	30
Contingency	34	34	74	52	79	48
Required Power w/5% Distribution Loss	125	126	273	186	292	177
Available System Power	210	300	300	210	300	3000
Power Reserve	85	174	27	24	8	2000

4.2.2 Power

The LEx bus is powered by a fixed solar panel assembly during the Earth-Moon cruise and surface operations at the sunlit landing site. Six flat rectangular solar panels are positioned circumferentially around the LEx bus structure to permit a wide range of LEx orientations. Each panel has a 0.4 square meter surface area and weighs approximately 1.1 kg, providing 100 W per panel and an average total system power of 200 W.

The stored energy system consists of a secondary battery for operations in sunlight and a primary battery for operations in shadow. The secondary battery is a single rechargeable Lithium-Ion 6 A-hr 28V (160 W-hr) battery which provides adequate current (10 A, 75%) and energy capacity (32 W-hr, 80%) for the cruise and terminal landing phase.

Upon flight and landing in the solar shadowed region, a dedicated primary (one-time-use) battery powers LEx. With 60 Watts of continuous heater power in the shadowed region total consumption is maintained below 200 Watts providing over 15 hours of battery life when only 2 hrs are required.

4.2.3 Communications

The communications subsystem uses a standard RS422-compatible (5 W) S-Band transponder and two switched low-gain antennas. This provides 4 steradian coverage with a minimum data rate capability of 32 kbps at all times.

4.2.4 Braking and Landing Propulsion

The LEx propulsion subsystem uses two stages. The first is a flight qualified, ATK-supplied STAR 27 solid propellant motor. This motor provides ~2500 m/s V at the total available mass. The STAR 27 uses the same ATK propellant used in many other STAR motors and generates a delivered vacuum Isp of 290 seconds.

Following burnout and separation of the STAR 27, the precision landing is accomplished using a four tank helium pressurized bipropellant system derived from the flight qualified Divert and Attitude Control System (DACS). Two divert thrusters provide axial thrust and two provide lateral thrust. The propellant mass of 28.8 kg provides adequate V margin for the initial precision landing and subsequent flight up to 4 km.

The propulsion system is mounted on a modular structure for integration and testing independent of the LEx bus. After subsystem testing, the propulsion module is integrated to the LEx bus for final testing and fueling operations.

Digital Scene Matching Area Correlator (DSMAC)—The LEx bus design uses proven correlation based recognition techniques for navigation by utilizing the DSMAC during descent phase immediately after SRM burnout to provide accurate position and heading updates to the GNC subsystem. A reference map with a given cell size, image resolution, and other mission data are placed into the mission file that is downloaded onto the spacecraft before flight. The LEx flies over an area, acquires images, and compares images to the stored reference map, as shown in **Figure 6**. LEx motion during the flight is incorporated to improve the correlation peak detection over the scene by combining multiple image correlation results.

The LEx team conducted a study to demonstrate applicability of the DSMAC to the lunar terrain images. Utilizing both 100 m/pixel, 10 m/pixel, five trajectories over five separate scenes, and 40 images per trajectory with no scene overlap were processed with the current version of the DSMAC algorithms.

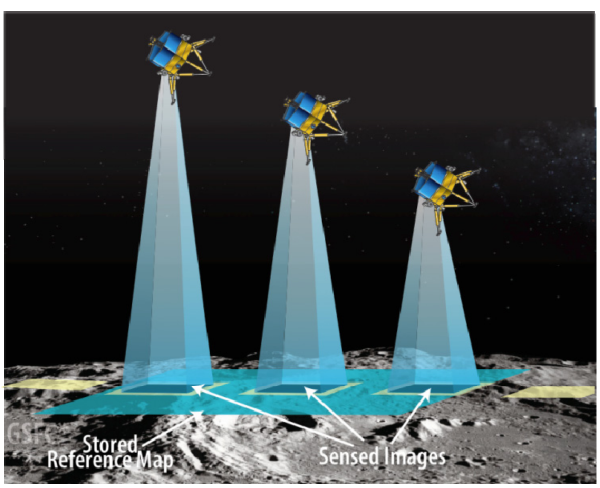

Figure 6: Navigation using lunar feature-based correlation with DSMAC descent images.

4.3 Braking, Descent and Landing (BDL) Operations and GNC Approach

The BDL approach is designed to complement the Human landing system work already accomplished by the Advanced Landing and Hazard Avoidance Technology (ALHAT) program. The LEx team will modify the ALHAT baseline to accommodate differences: direct entry, solid motor braking and descent, vehicle dynamics and propulsion.

The LEx lander utilizes autonomous GNC during the Braking, Descent and Landing (BDL) phase. Prior to spin up for braking, the LEx GNC system will receive a state vector update from GFSC Operations via DSN. Since the braking maneuver is executed by a spin-stabilized SRM burn, this GNC update will dictate the initial vehicle attitude and SRM ignition time. The timing of this burn is critical to control of the downrange landing location. The guidance responsibility hand-off occurs after ground verification of the SRM braking pre-burn attitude.

After SRM burnout and separation and after an attitude update via the star tracker, the GNC system will command divert maneuvers using the on board navigation system's current estimate of the vehicle state relative to the pre-loaded target. Divert maneuvers at this point will allow the vehicle to minimize the known SRM trajectory dispersions. An "all inertial" navigation solution based on a Deep Space Network (DSN) initial condition prior to the braking maneuver and maintained by Inertial Measurement Unit (IMU) and star tracker data can be expected to produce navigation horizontal position errors of ~1 km, 3σ, at the surface.

As the DSMAC correlation and landing radar data become available, the GNC system can improve estimates of the vehicle state and perform maneuvers to minimize trajectory and velocity errors resulting in a much more precise touchdown. Based on currently available data, this is on the order of <100 m. **Table 7** shows the LEx navigation sensor utilization from SRM ignition to landing assuming the inclusion of the DSMAC updates into the blended navigation solution.

Table 7: LEx Navigation Sensory Utilization.

			Navigation Sensors in Use					
Mission Phase	Phase Duration	Primary Navigation Mode	Ground Navigation Update	IMU	Star Tracker	Landing Radar	DSMAC Optical Option	DSMAC Optical With Illuminator Option
Earth-Moon Transit	100 hours	Inertial w/Ground Update	Y	Y	Y	N	N	N
Descent	1-2 hours	Inertial w/Ground Update	Y	Y	Y	N	N	N
SRM Braking	30-40 seconds	Inertial	N	Y	N	N	N	N
Post-Braking Terrain Acquisition	5 second	Inertial	N	Y	Y	Y	Y	N
Midcourse Approach	20-30 seconds	Inertial w/Ground Update and OPTIONAL DSMAC Update	N	Y	N	Y	Y	N
Terminal	5-10 seconds	Inertial w/Ground Update and OPTIONAL DSMAC Update	N	Y	N	Y	Y	N
Landing	5 seconds	Inertial	N	Y	N	Y	N	N
Hopping (Lift-off)	1-5 seconds	Inertial w/Ground Update and OPTIONAL DSMAC Update	Y	Y	Y	Y	N	Y
Hopping (Transit and Landing)	30-40 seconds	Inertial w/OPTIONAL DSMAC Update	N	Y	N	Y	N	Y

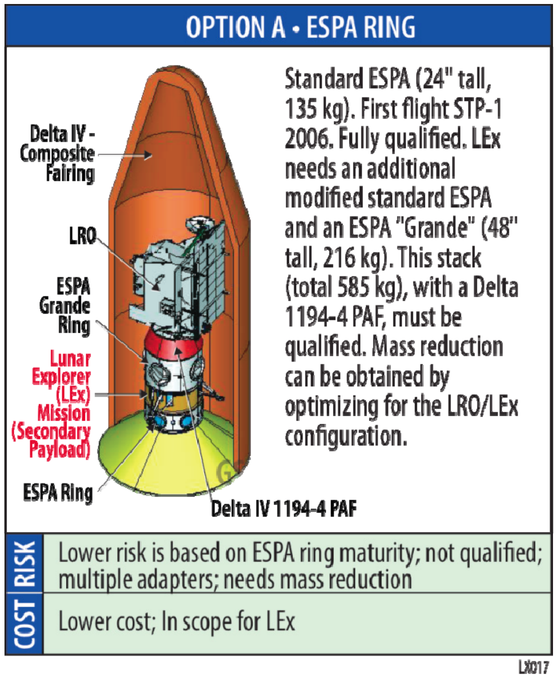

Figure 7: Multiple Dual Payload Adapter Fitting (DPAF) options exist.

4.4 Secondary Payload Adapter

GSFC pursued four options to accommodate LEx as a secondary payload. The mission was originally designed to be a secondary payload on an EELV launch. Of the options, two were chosen as the primary ones for the baseline design: Option A (**Figure 7**) is an EELV Secondary Payload Adapter (ESPA) based solution. All other solutions fit within LEx mass, cost and schedule constraints.

Instrument Accommodations

The Instrument Suite attaches to the LEx bus at a simple match-drilled bolt circle. The Instrument Suite is assembled on a common deck comprised of a mechanical interface plate. For the example payload shown in this paper all instruments are mounted below the plate, with PanI and the Laser Reflector mounted to a camera mast above the plate. A simple instrument Data Handling Unit (DHU) serves as the data and power interface between the instruments and the LEx bus. The DHU distributes commands, buffers data, and switches power to the ANS, PanI, EMS, and MHP instruments. The Instrument Suite accommodation is depicted in **Figure 8**.

The LEx bus provides a separate survival heater bus to thermostatically controlled heaters distributed across the instruments during instrument non-operational modes. While operating, the instruments will be thermally stabilized by the use of heaters, thermostats, and multilayered insulation.

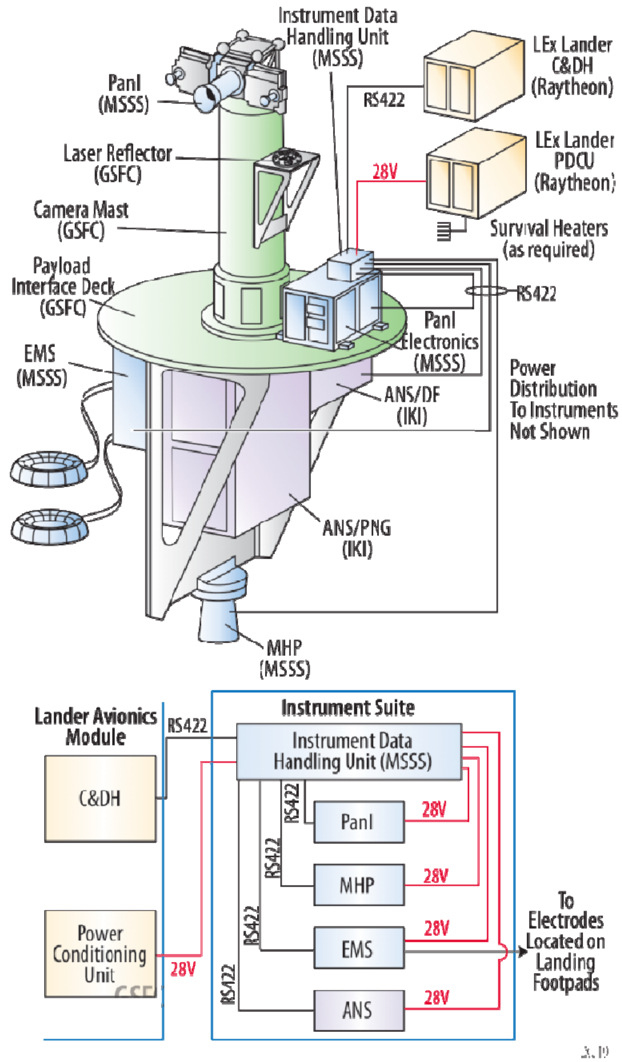

Figure 8: LEx instrument interfaces are straightforward and low risk.

To minimize any impact to the LEx bus design, the Instrument Suite will be thermally isolated from the LEx bus interface.

4.5 Maturity and Risk

The DACS propulsion system that LEx is based on has been successful in multiple space flight tests to date. There are currently dozens of nearly identical propulsion units in place. The DACS propulsion unit hover tests are shown in **Figures 9A** and **9B**. The components selected for the Lander Avionics module leverages existing hardware.

DSMAC, a terrain-based navigation technique used for decades to guide long-range missile to meter-accuracy targets, provides precision navigation capability. This navigation system enables landing capability limited only by the resolution of a reference map.

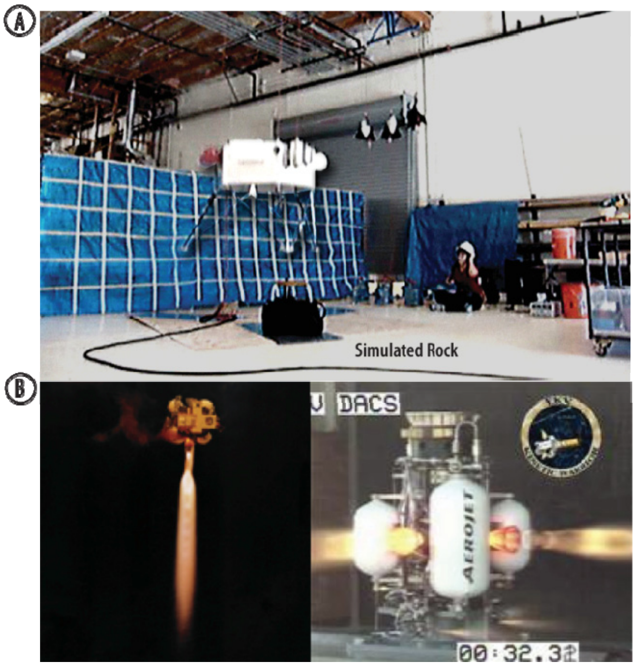

Figure 9: (A) Landing Gear Test—Dynamics of Landing on a Simulated Rock. (B) Integrated, Autonomous GNC, Sensor and Propulsion Unit in Closed Loop Strap Down and Hover Tests.

There are inherent risks in every design including this one. The LEx mission falls within the characterization of risk classification for Class D per NPR 8705.4 based on the nature of this secondary payload opportunity and the limited cost goal. The majority of the LEx flight system has been developed to standards that are Class B/C compliant. Specific risk mitigations not discussed here and bringing a class D mission to completion, requires a team with extensive flight experience.

5.0 MANUFACTURING, INTEGRATION AND TEST

To facilitate rapid, in process, interface verification, subsystem providers will provide interface simulators of their respective components.

The fully assembled LEx bus will undergo acceptance tests (a flight sequence test, interface verification, and EMI/EMC). Standard processes will be supplemented to meet the specific environments of this mission (e.g., SEUs, materials outgassing, thermal).

Completed subsysytems will be moved to GSFC main I&T buildings for acceptance tests and mechanical and electrical integration onto the GSFC supplied payload interface deck. Instrument suite testing will include functional and interface testing. The mission testing sequence includes functional tests, EMI/EMC, comprehensive performance tests (pre- and post-environments) vibration, thermal vacuum and thermal balance, mass properties measurements and spin balance. The I&T flow is summarized in **Figure 10**

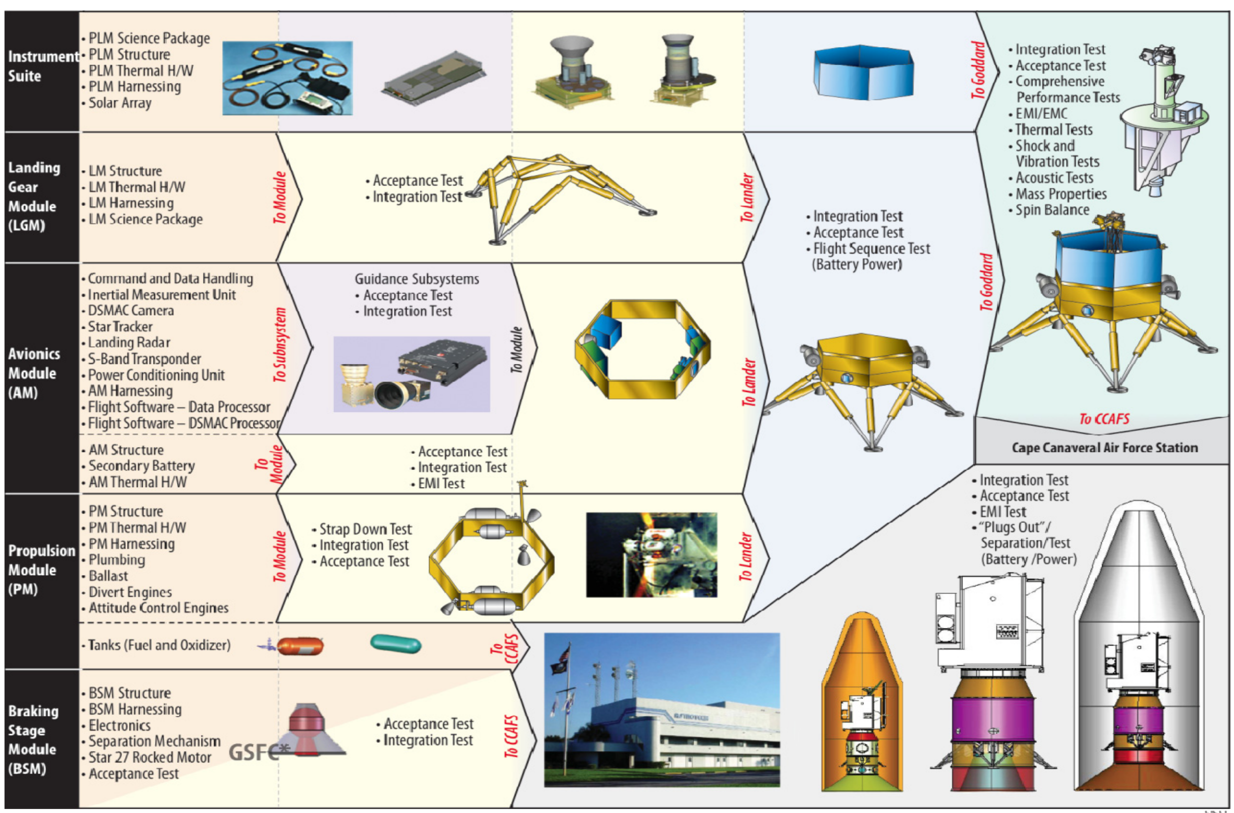

Figure 10: The LEx modular architecture facilitates straightforward integration.

6.0 Mission Operations, Ground and Data System

LEx will utilize TDRSS single access (SA) to cover launch phase critical events, then switch to the DSN 34 m antennas to perform TT&C during the cruise phase, descent, landing and initial sunlit site operations. The DSN 70 m dishes will be utilized during the flight into the solar shadowed site and subsequent operations there to provide 120 kbps communication. DSN provides large apertures, accurate ranging and tracking, and has enough antennas to support multiple missions simultaneously. LEx uses the DSN to cover all critical events from initial acquisition after launch until the end of the mission including initial checkout, trajectory correction maneuvers, landing, and transition to the solar shadowed location. Prior to and during separation, LEx will use TDRSS SA to provide telemetry coverage until DSN acquisition. DSN will provide real time support during the critical BDL phase of the mission to the lunar surface. Contingency support will be provided by redundant DSN antennas and the NASA ground station network. LEx uses the DSN 34 meter antennas to downlink data at a rate up to 32 kbps. It will take about 15 hours to transfer mission data from the sunlit sites. LEx uses a DSN 70 meter antenna to recover the data from the shadowed site to reduce the downlink time to less than 15 minutes (due to both increased data rate and lower data volume). The space to ground communications network is depicted in **Figure 11**. The communications timeline for the mission is shown in **Figure 12**.

The software will be moved after I&T to the GSFC SMEX Multi-Mission Control Center to utilize the existing secure facility and communications (data, voice, and video) infrastructure.

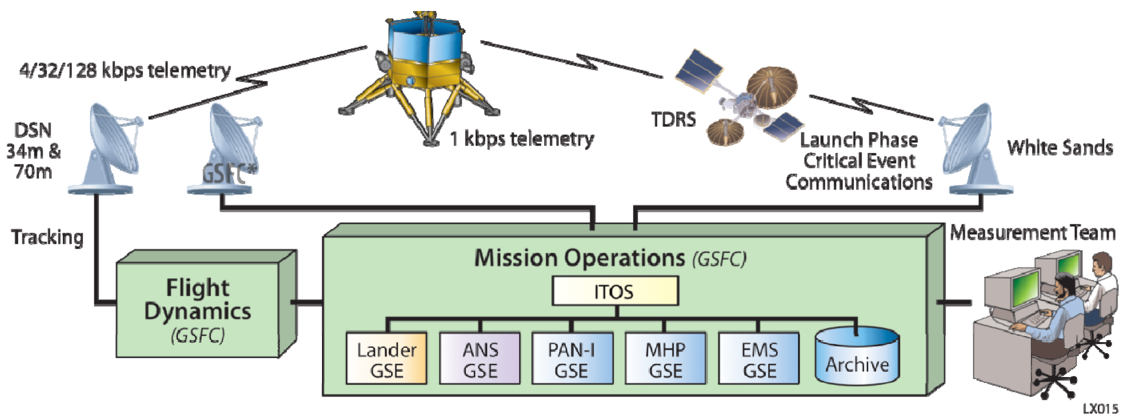

Figure 11: The LEx Flight to Ground communications are straightforward and well understood.

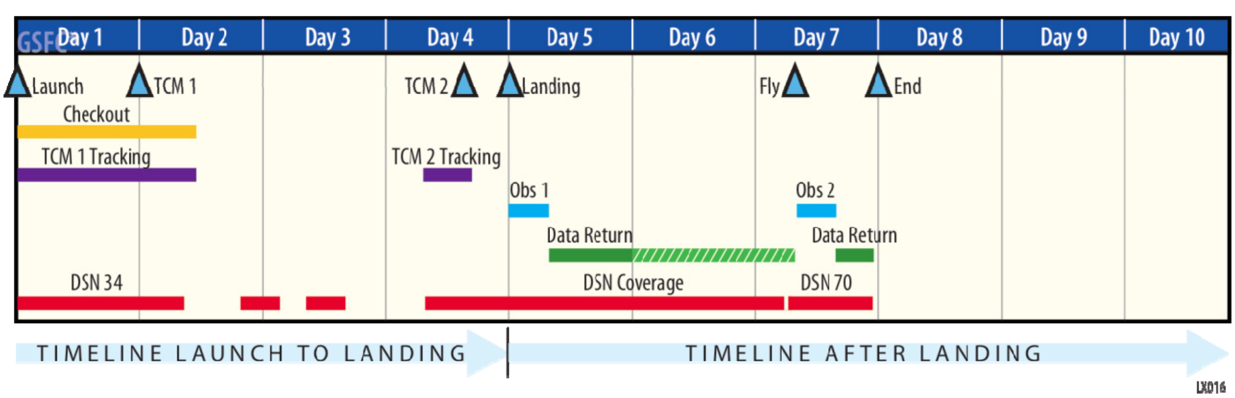

Figure 12: The LEx timeline maximizes measurement return.

LeX Mission Team

The LEx team is highly cognizant of the deliberate management approach required to successfully achieve mission success for this challenging mission within the given cost and schedule constraints. The LEx management approach will be built strongly on a foundation of successfully proven streamlined management processes for low cost missions with rapid development cycles. LEx is envisioned as a PI-led mission in which NASA's GSFC would lead project management, science, and systems engineering, with a carefully selected team, and use efficient partnerships to complete a higher risk, lower cost mission.

7.0 Schedule

The LEx mission master schedule shows an approximately 3 year timeline for delivery. LEx is delivered within tight constraints dictated by a co-manifest approach.

The master schedule ties together the build, test, and delivery of the flight qualified LEx lander; build, test and delivery of the instruments; and final integration and test of the entire LEx lander system (with instruments) at GSFC.

Preliminary analysis of the integrated Master schedule currently identifies the mission critical path to I&T as the LEx Lander Avionics Module. The LEx mission carries a total of 60 calendar days of funded schedule reserve. Extensive development risks and mitigation issues and scenarios have been created and analyzed.

8.0 Cost

The LEx team designed the mission with low cost as the primary goal and has costed the mission in several ways. While meeting the low cost goal will be challenging, we believe that the mission can be achieved because of the current production nature of most of the LEx Lander subsystems and the unique opportunity to capitalize on NASA instrument spare parts and recent development efforts. For the baseline estimate, all costs were estimated in a grassroots exercise that is keyed to our WBS.

During the study we refined our cost estimate to account for the architecture development and decisions that we have made. The cost estimate for the LEx Lander is a combination of firm quotes from material suppliers and detailed labor estimates. The labor estimates are based upon the high fidelity schedule developed for the lander and actual costs from similar and/or equivalent efforts on other programs with almost the same hardware.

This design will leverage the large US government investments in subsystem design, development, and system test capabilities. The instruments are comprised of mature components resulting in credible high fidelity cost estimates.

Independent cost validation via parametric modeling tools at has yielded consistent results with our grass-roots estimate. The cost was estimated as being in the \$100M to \$200M range in FY 2009 dollars.

9.0 Mission Enhancements and Additional Return on Investment

The LEx mission offers several straightforward mission enhancement options as well as program enhancement opportunities for consideration. These are summarized in Table 8.

Table 8: Enhanced LEx mission and future program options.

Mission Enhance-ments	Activity	Capabilities	Relevance to Vision for Space Exploration	LEx Mission Objectives
Enhanced Mission Options—Enhanced mission will include Baseline Mission plus one or both of the following:				
Option 1: Flight to Additional Shadowed Site	Additional flight (~100-1000 m) to second solar shadowed site Maintain DTE communications link Complete 3rd set of measurements (2nd ground-truth of ice potential) With last remaining fuel, conduct low-altitude powered flight for ~30–60 s with ANS in passive mode to search for concentrated ice "blocks" in shadowed region; land and relay data to Earth (DTE)	6 propellant tanks Engine fully qualified for cold restarts Additional battery Advanced landing struts	Develop infrastructure: validates new surface mobility approach 2nd solar shadowed region surface access and measurements	Extends baseline measurement objectives 1-5 (See Table 2)
Option 2: Flight to Additional Sunlit Site	Additional flight (~1 km) to second high hydrogen sunlit site Maintain DTE communications link Complete 3rd set of measurements Monitor solar illumination ('watch the light') until EOM	8 propellant tanks Engine fully qualified for cold restarts Additional battery Advanced landing struts	Longer term surface access and imaging	Extends baseline measurement objectives 1-5 (See Table 2)

At the mission level, modest increases in capabilities (i.e., additional fuel tanks, advanced landing struts, etc.) would allow for an extended sampling of an enhanced hydrogen shadowed region at the lunar surface. Enhanced mission options provide for opportunities to utilize the LEx flight surface mobility for access to either additional sunlit or shadowed sites, with associated measurements. Enhanced Mission Option 1 would extend the LEx baseline mission to acquire a third set of measurements at an additional shadowed site via a second powered flight of at least 100 m. This additional independent set of comprehensive measurements in solar shadow to evaluate polar resources (water/ice) increases confidence in the quantitative assessment of hydrogen-bearing materials within a lunar cold-trap. Such observations would begin a localized assay of specific lunar resources at low cost. At the conclusion of this option, the LEx Lander could provide adequate resources for an optional ultra-low altitude (~300 m) hydrogen mapping overflight campaign using the neutron spectrometer in a passive mode. This end-of-mission option would entail a low altitude powered flight of several tens of seconds across a shadowed region in search of hydrogen signatures associated with large blocks of buried water ice. Enhanced Mission Option 2 would extend the LEx baseline mission to acquire additional measurements of an alternate south polar site via a second powered flight (~1 km) from the shadowed site to a second sunlit site. This option would explore another, independent site to evaluate the region's suitability for sustained future human operations. By flying to an additional sunlit site, LEx can extend its surface operations for weeks, and search for time variable phenomena possibly associated with lunar dust.

On the basis of the LEx flight mission design summarized above, engineers and scientists at GSFC (co-authors of this paper) designed a lunar surface sample return mission that leverages the LEx lander design concept. This concept was nicknamed "Boomerang" and it is described below.

Boomerang Lunar Sample Return Summary—

The Boomerang mission concept is based on the LEx Lander design effort that NASA GSFC initiated in 2005. In the spring of 2007 NASA Headquarters requested that NASA Goddard Space Flight Center formally investigate a low cost lunar surface sample and return mission. There was an interest in developing novel approaches to lunar surface sample and return (LSSR) that could potentially fall within the NASA Discovery Program cost cap (< $425M without Launch Vehicle). At this time the Minotaur Launch Vehicle was ramping up with a promise of lower cost access to space. It was estimated that the Minotaur -5 launch vehicle would be so inexpensive that it would be significantly cheaper to launch 2 Minotaur-5 vehicles than 1 single Medium Class Expendable Launch vehicle.

NASA Goddard's Boomerang Mission was conceived using such a two launch vehicle approach in an attempt to capitalize on this new economy. It is important to note that the Boomerang mission could also be accomplished using a single launch vehicle. While the configuration would be slightly different, most of the architecture would remain the same and many of the trades would converge upon the same solutions. The systems used for the independent launches would be combined only for the single launch phase and once on course would return to acting independently. Here we will focus on the two launch approach that was studied as the Boomerang LSSR baseline for NASA Headquarters in 2007.

The Boomerang study was funded as a short turnaround concept study by an experienced team of engineers and scientists. This was adequate to establish the basic technical details behind the concepts proposed and to provide a reasonable basis for a first order cost estimate. The intention was to identify tall poles, high risks, and long lead time items; not to define detailed requirements interfaces or more detailed design.

10.0 SCIENCE OVERVIEW

The primary goal was to define a low cost (Discovery Class) lunar surface sample and return mission. Lunar surface samples address all of the major science themes of the National Research Council's Scientific Context for Exploration of the Moon report which was published in 2007. These include the early Earth/Moon system, terrestrial planet differentiation, solar system impact record, and lunar environment. We chose to target the return of ~100 grams of sampled materials, on the basis of the community-based assertion that as little as 10 grams was still considered compelling, depending on the sample site. Boomerang was designed to sample South Pole Aitken Basin highlands soils to a depth sufficient to acquire necessary lithic fragments for detailed chronology and related analyses back on Earth. These samples from a major polar-region impact basin (SPA) would provide first time information from areas not sampled by Apollo or the Russian Luna missions.

The Boomerang mission design concept addressed the following science priorities:

1. Fundamental Solar System Science

- Characterize and absolute age date the impact flux (early and recent) of the inner solar system. *[Boomerang samples provide absolute chronology to quantify flux, timing]*

- Determine the internal structure and composition of a differentiated planetary body. *[Boomerang samples constrain composition of crust/sub-crust unlike Apollo]*

- Determine the compositional diversity (lateral and vertical) of the ancient crust formed by a differentiated planetary body. *[Boomerang samples [SPA] sample most ancient crust and sub-crust, not sampled by Apollo, Luna]*

• Characterize the volatile compounds of polar regions on an airless body and determine their importance for the history of volatiles in the solar system.

2. Planetary Processes

• Determine the time scales and compositional and physical diversity of volcanic processes. *[Boomerang samples address composition of SPA impact melt vs known volcanic materials from near-side maria]*

• Characterize the cratering process on a scale relevant to all inner solar system planets.

• Constrain processes involved in regolith evolution and decipher ancient environments from regolith samples. *[Boomerang samples characterize the polar regolith for first time, within SPA Basin → sub-micron dust for VSE and science]*

• Understand processes involved with the atmosphere (exosphere) of airless bodies in the inner solar system.

11.0 Mission Overview

Boomerang is a two launch vehicle lunar surface sample and return mission. The first launch vehicle sends a LEx-like lunar lander on a TransLunar-Injection (TLI) course with a C3=(-2.12 to -1.9) km2/s2 to the Moon. In about 4.5 to 5 days this lander system will arrive at the Moon and land at the predetermined target location (i.e., within SPA Basin). After landing it will collect a subsurface lunar regolith sample of at least 10 grams. Once the sample collected is verified it will be launched from the surface of the Moon contained within a small spherical capsule and placed into a low lunar orbit. Only after the location of the Sample Return Capsule (SRC) has been verified back on Earth will the second launch commence. The second launch will contain a capture and return vehicle. This vehicle will be placed into the same low lunar orbit as the SRC and will commence a rendezvous and capture sequence. Once capture has been verified this vehicle will leave lunar orbit and head back to Earth. Upon arriving at Earth, the lunar sample contained in the SRC now held in the reentry vehicle will be deployed for reentry over the desert of Utah (UTTR). An overview of this mission architecture can be seen in **Figure 13** (the mission architecture chart).

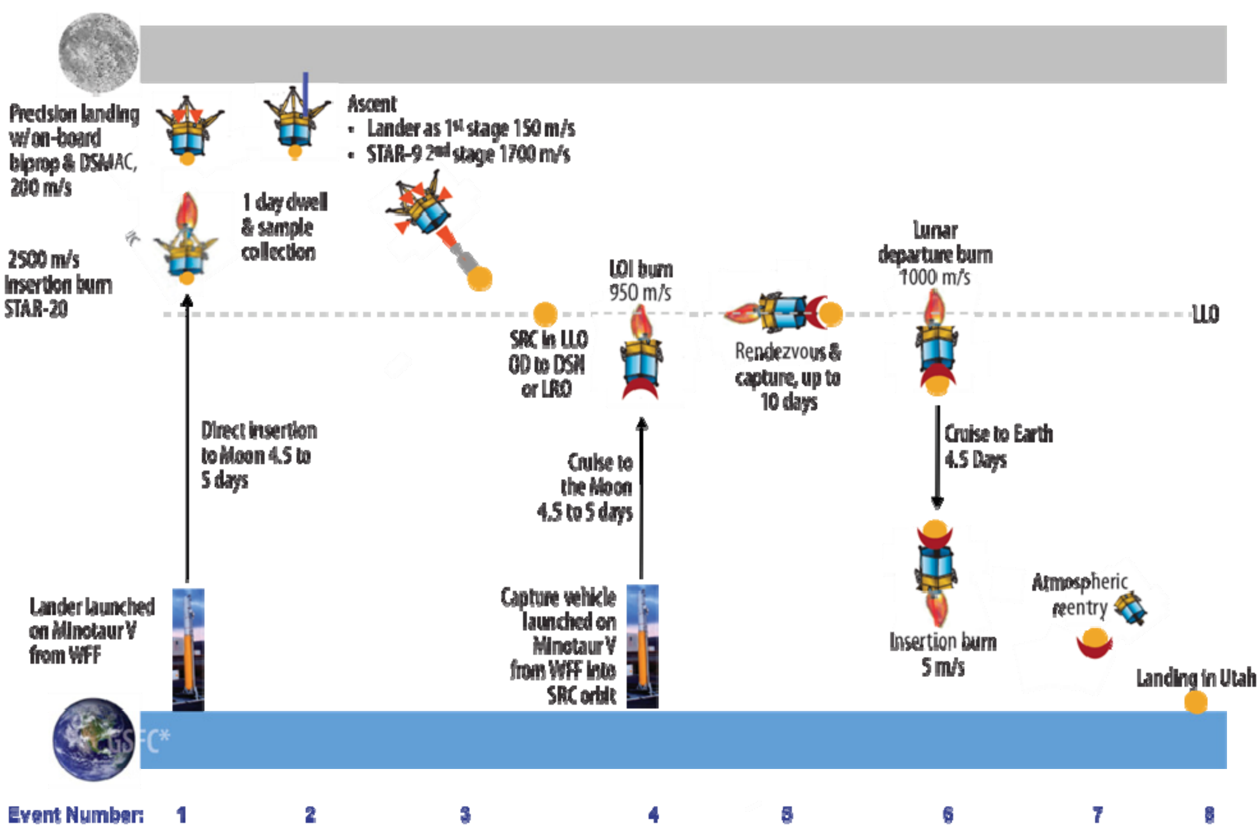

Figure 13: Boomerang mission architecture illustrating each mission sub-phase.

The launch vehicle chosen for this mission was the Minotaur V. At the time of this study NASA deemed this an appropriate risk class vehicle for the Boomerang mission study. The Minotaur V had a predicted capability to trans-lunar injection (TLI) of ~450 kg launched from Wallops (WFF) and 517 kg launched from Kennedy (KSC). The total wet mass for the lander and assent vehicle combined including the Star-20 descent solid-rocket motor is ~477 kg including a 30% mass margin. The payload for the second launch including the capture and return vehicle has a total wet mass of ~445 kg including a 30% mass margin.

The Boomerang mission is capable of landing anywhere on the Moon during lunar daylight conditions. Launch opportunities therefore exist for ~1-2 day windows that repeat every couple of weeks. It takes approximately 4.5 days to arrive at the Moon and conduct BDL operations in order to land. Once the landing site is verified, a sample is collected. By the sixth day the ascent vehicle has placed the sample into a polar low lunar orbit. Collection will occur in the 7th-19th day. The lunar longitude of the sample site determines how long before the collection. The second launch vehicle waits for the sun-Earth conjunction to rotate into place for maximum full sunlit orbits for rendezvous and capture. Adequate orbit determination of the sample in low lunar orbit only takes about 12 hours using DSN stations and standard RF tracking. Once the sample is captured it is returned to Earth where the reentry vehicle is deployed for final touchdown in the Utah dessert.

12.0 Boomerang Lander Overview

The Boomerang lander (**Figure 14**) is by design a slightly modified LEx lander with its payload replaced by the sample collection system and ascent vehicle system. The LEx lander was previously discussed in detail in the first section of this paper. One primary difference in the Boomerang version of the LEx lander is a 50% mass reduction in the LEx power system. This is achieved by taking advantage of differing mission driving requirements. LEx was designed with the capability to land and then hop into a permanently shadowed region (PSR), analyze a surface site, and then transmit that data back to Earth via a relay. This required the LEx lander to carry a larger battery. Boomerang, on the other hand, need only operate long enough to acquire and contain it's sample and launch it in the ascent vehicle to a low lunar orbit. Boomerang will use the LEx DSMAC precision landing system in order to land precisely at the desired sampling location with the SPA Basin.

Figure 14: Boomerang sample acquisition and "spinning jump shot" lander configuration.

The lander system will have the ability to determine the sample site location by means of ranging and local imaging to a degree that is adequate for scientific purposes. The Boomerang lander will also provide the pressurized gas from the propulsion system pressurant residual to power the sample collection system (i.e., via a vacuum-like backflow). In addition, the Boomerang Lander will provide the launch platform and initial attitude and launch timing for the Ascent Vehicle (inside of which the sample is contained).

13.0 Sample System

The sample system is required to be light weight and simple, yet capable of acquiring a sample well below the surface (> 10 cm deep) and transferring it up into the Sample Return Capsule (SRC) (**Figure 15**). To accomplish this task we take advantage of the 10,000 psi helium pressurant already available as a residual in the LEx EKV derived propulsion system, which is used for BDL operations. The design uses the valves and valve control electronics already in use in the propulsion system to enable an effective, vacuum-like sample collection system.

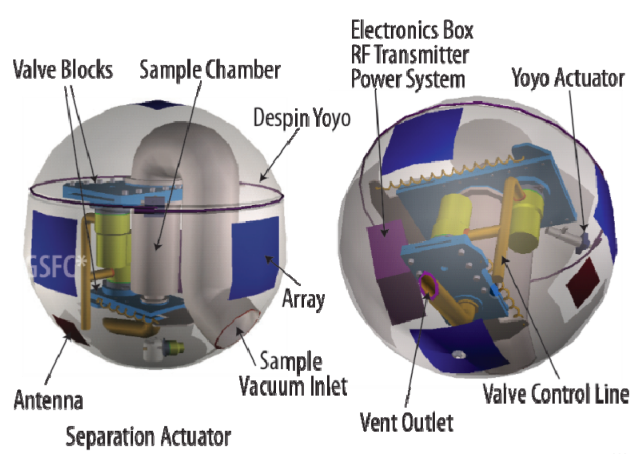

Figure 15: Sample Return Canister for Boomerang (SRC); this system will be launched to a low lunar orbit and captured by the "chase vehicle" and carried to Earth on the Earth Return Vehicle.

The design uses a simple "jack hammer"-like device that will drive a hollow tube 20 cm into the lunar surface. This jack hammer was powered by the 10,000 psi helium and has a titanium cutting ring at its tip.

Once the tube is driven ~20cm below the surface, the jack hammer is deactivated and the same high pressure helium is directed upward from the tip of the tube back into the sample return capsule. This creates an air flow much like in a vacuum cleaner hose; except in this case the vacuum is provided for free by open space. Thus, the lunar regolith sample is vacuumed up from the surface and transferred into the sample return capsule which effectively acts like a vacuum cleaner bag and entraps the sample in front of a micro-screen. This also solves the non-trivial problem of transporting the sample from the collection tip into the sample return capsule with a minimum number of mechanisms. The sample chamber is approximately 7 cm in diameter and 14 cm in height. It will hold approximately 0.5 kg (500 grams) of lunar regolith. The chamber is made of titanium for robustness in case of a failed parachute high G-force landing at the UTTR. In addition, the chamber is aligned with the vertical thrust axis of the ascent vehicle to minimize dynamic imbalance regardless of how much sample is collected or how it might clump within the sample chamber. A simple electronic circuit monitors the change in the dielectric properties of the sample chamber to verify that sample collection was successful.

The sample is hermetically sealed in the chamber by two vane type cutoff valves. This design is based upon existing hardware for hermetically sealed cutoff valves in other applications. The power to actuate these valves is also provided by the pressurized helium controlled from the Lander vehicle. By closing the valves in space, the sample will arrive vacuum sealed. Additionally the design is such that the Earth's atmospheric pressure will serve to hold this valve seal closed even tighter, minimizing contamination or leakage.

During ascent the sample return capsule must separate from the Boomerang lander and that includes the sample collection system. To achieve this sequence of operations, the sample inlet hose as well as the valve pressure control lines all have simple O-ring seals on them and simply pull apart during the very first motion of the ascent vehicle. During this motion the ascent vehicle is still within its guide sleeve of the Lander/launcher such that tip off concerns are minimized.

14.0 Ascent Vehicle

The ascent vehicle is designed to be as simple and as light weight as possible. There are two very unique design features which enable this architecture. The first enabling design is what we call the "spinning jump shot." In this maneuver the Boomerang Lander itself becomes the launcher. Utilizing the attitude control electronics on the Lander we are able to achieve a very accurate initial pointing vector for the launch of the ascent vehicle. It also provides a free spin table. By utilizing the Lander to spin up the ascent vehicle and place it on its initial launch trajectory at a precise ground specified time we are able to greatly reduce the complexity required of the ascent vehicle (see **Table 9**).

Table 9: Boomerang sub-system masses

Ascent Vehicle Mass [kg]			
Subsystem	**CBE**	**Cont %**	**Alloc**
Sample Return Capsule			
Sample Canister and plumbing	1.00	30	1.30
Sample Canister Seal System	0.50	30	0.65
Dielectric Sample Presence Sensors	0.10	30	0.13
RF Antennas	0.30	30	0.39
RF Electronics	0.15	30	0.20
S/A Total Mass incl. Mounts	0.20	30	0.26
Lilon Battery	0.25	30	0.33
Connectors, Harness, Misc.	0.20	30	0.26
Structure	3.00	30	3.90
SRC Ejection System	0.40	15	0.46
Despin Yoyo	0.19	30	0.25
SRC Down (Empty) Mass	***6.29***		***8.12***
Lunar Sample	0.50	30	0.65
SRC Up (Full) Mass	**6.79**		**8.77**
SRC PAF			
SRC PAF Base	0.60	30	0.78
AV Sep System - AV half	0.20	30	0.26
Extension Tubing, T-0 "O"-rings	0.30	30	0.39
SRC PAF Mass	**1.10**		**1.43**
STAR 9 Inert Mass	4.13	1	4.17
Total Ascent Vehicle Dry Up Mass	**12.02**		**14.37**
STAR 9 Propellant	11.81	1	11.93
Total Ascent Vehicle Wet Up Mass	**23.83**		**26.30**
STAR 9			
Required delta-v [m/s]	1700		
Isp [sec]	289.1		
Propellant as a fraction of dry mass [%]	82.218		
Dry mass [kg]	**14.4**	**31.7**	
Required Propellant [kg]	11.8	26.0	
ACS required propellant [kg]	0		
Finite Burn Penalty [%]	0		
Total Propellant mass [kg]	**11.812**	**26.042**	
Pressurant [kg]	0		
Wet Mass	**26.179**	**57.715**	

The second unique feature of the Boomerang ascent vehicle architecture is to take advantage of the Moon's natural physics (gravity field). Specifically because the lunar center of gravity is not at the center of the lunar sphere, there are low lunar orbits that naturally stabilize themselves. Typically in order to launch into a maintainable orbit about a spherical body requires either thrust vector control or a second circularizing (or at least 'ellipticallizing') burn. However, NASA Goddard analyzed and developed orbits that would remain stable for weeks with a single, one vector directional burn. Indeed, from most longitudes we could achieve orbits that will remain stable for longer than 45 days. This can be accomplished by launching the Boomerang SRC into a 10km by 200km elliptical orbit that utilizes a narrow (few degrees) window directed over lunar gravity concentrations (mascons) for the initial thrust vector.

The lunar gravity concentration provides a circularizing force that would cause the ascent vehicle to crash into the surface if it were centered on the lunar sphere. However, since it is not centered this gravitational force pulls the ascent vehicle into an elliptical orbit, thereby providing the secondary thrust vector required without any additional propulsive maneuver.

Maintaining its "keep it simple" approach, the Ascent Vehicle communicates only via the tracking beacon within the SRC sphere. The sphere will have an S-band transceiver that will feed 4 patch antennas via a 1 Watt power amplifier. Each antenna will provide -3dBi of gain over +-80 degrees. The system will automatically transmit a return signal to the antenna that has the highest received signal strength.

15.0 CAPTURE VEHICLE OVERVIEW

Once the orbit of the SRC has been verified, the Capture and Earth return recovery vehicle can be launched. However, this launch may be held until the SRC orbit has rotated into a more optimal sunlit plane for rendezvous.

The capture vehicle has two primary functions. First, to capture the SRC, and second to initiate its return in the Earth reentry vehicle. To simplify flight systems, the SRC is captured into the Earth reentry vehicle itself. The initial Earth reentry vehicle design had a door to close and contain the capsule during reentry. Later it was determined that the magnet-based system used for final capture was so strong that the door would not be needed. In fact the magnetic bond was so strong that after recovery of the reentry vehicle from the Utah test range part of the reentry shield may have to be cut away in order to allow the reentry vehicle to move in shear relative to the SRC in order to separate the two.

The capture vehicle is placed into a Trans-Lunar-Injection (TLI) orbit by a second Minotaur V. A significant propulsion system was required in order to both capture at the Moon and to have adequate delta-V to leave lunar orbit for the return to Earth. This system is designed as a MMH/NTO bipropellant system. The propellant mass was ~220 kg with a pressurant mass of 2.25 kg at 3600 psig BOL (see **Table 10**).

Table 10: Boomerang Capture vehicle mass breakdown.

Capture Vehicle Subsystems Mass [kg]

Subsystem	CBE	Cont %	Alloc
Reentry Unit (Aeroshell w/ TPS, Beacon, etc.)	10.00	30	13.00
Rendezvous APS Startracker + Electronics	2.50	30	3.25
Rendezvous RF Ranging	3.00	30	3.90
Rendezvous Floodlights	1.00	30	1.30
Capture Magn and Mech Equipment	5.00	30	6.50
Navigation and Attitude Determination	1.70	30	2.21
Command and Data Handling (C&DH)	14.30	30	18.59
RF Comm	11.10	30	14.43
Power, incl. Harness	10.35	30	13.46
Thermal Control	5.30	30	6.89
Bypropulsion System dry	41.20	30	53.56
Lander s/c Structure	59.20	30	76.96
Total LOI and Rendezvous dry mass	**164.65**		**214.05**
Sample Return Capsule	6.79		8.77
Total Capture Vehicle dry after Capture	**171.44**		**222.81**
Propellant expanded for TEI			98.29
Total Capture Vehicle mass in LOI before TEI			**312.34**
Propellant expanded for LOI and Rendezvous			133.92
LV to CV Sep System CV portion	6.32	30	8.22
Total Capture Vehicle wet after TLI			**445.71**

Biprop TEI	kg	lbs
Required delta-v [m/s]	1050	
Isp [sec]	320.0	
Propellant as a fraction of dry mass [%]	39.769	
Dry mass [kg]	**222.8**	**491.2**
Required Propellant [kg]	88.6	195.4
ACS required propellant [kg]	5	
Finite Burn Penalty [%]	5	
Total Propellant mass [kg]	**98.291**	**216.692**
Pressurant [kg]	0	
Wet Mass	**321.103**	**707.903**

Communication for the chase vehicle was provided via an X-band system with a 0.3 meter diameter HGA which provides a 10 kbps data rate from lunar orbit. Additionally, a small 1 watt power output S-band system was used to range to the SRC when in lunar orbit. This system requires a medium gain antenna with no less than a 30 degree cone angle beam width for locating and tracking the SRC (**Figure 16**).

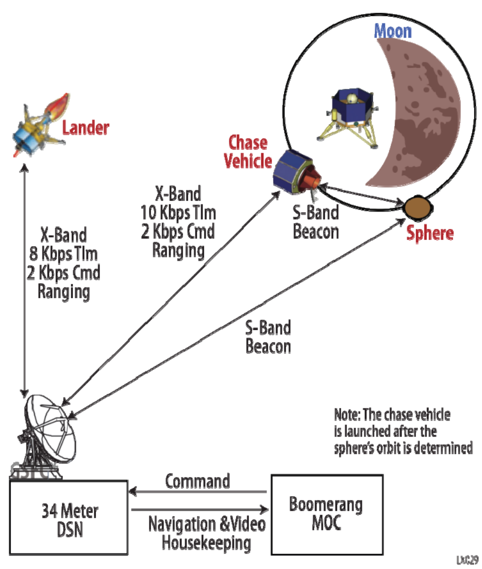

Figure 16: Boomerang telecommunications architecture.

16.0 Rendezvous and Capture

The SRC will be launched into a 10 km perilune by 200 km apolune orbit with a 45 day lifetime. Utilizing the center of gravity to center of volume discontinuity at the Moon to help propel the SRC, it's perilune will naturally rise to 25 km after 20 days.

The Capture Vehicle (CV) will then be placed into a 100 km circular "co-elliptical" phasing orbit over the SRC. The CV can then be brought into capture position through a complex series of rendezvous and proximity operations. The delta-V budget for these operations is contained in **Table 11**.

Table 11: Boomerang delta-V budget for critical maneuvers.

Maneuver Description	Height Diff (m)	Range	Closure rate after maneuver (m/orbit)	Delta-V (m/s)
Rendezvous & Prox Ops	*5000*	*Arbitrary phase angle*	*47124*	
Height Adjust to 500m below	5000			10
Co-elliptic (match ecc at 500m below)	-500	75km	4712	10
Out-of-plane adjust	-500			4
Height Adjust to 100m below	-100			0.2
Co-elliptic (match ecc at 100m below)	-100		942	0.2
Out-of-plane adjust	-100			0.05
Maneuver to Rbar approach	-30	30m		0.15
Sub Total				***24.6***
Docking attempt 1				
Hold at 30m	-30	30m	0	0.5
Approach from 30m to 1m			5 cm/s	5
Hold at 1m	-1	1m	0	0.1
Sub Total				***5.6***
Retreat				
Back out to 30m				5.6
Maneuver to vbar (safety ellipse?)				1
Hold on vbar (~.2m/s/hour)				1
Sub Total				***7.6***
Docking attempt 2				**5.6**
Retreat 2				**7.6**
Docking attempt 3				**5.6**
SUBTOTAL				**56.6**
AVAILABLE				**80**
CURRENT DELTA V MARGIN (%)				**41.343**

Orbit determination of the sample sphere (SRC) is accomplished via ranging from the Deep Space Network (DSN). Performance specification of ~100 meters is achievable after 55 hours of clock time using a minimum of 30 minutes of ranging every 2 hours on a 34 meter DSN dish. We examined a number of options for ranging including dual-doppler and two way time of flight measurements. One approach could include placing a simple S-band phase-discriminator (an elementary PLL) in the sample communications system and then requiring the DSN to transmit interrupted CW. A custom timer interface box at the ground station could then be used to determine time of flight measurements. Ultimately we decided that simplified ground based ranging to a few km was more than adequate for the hand off to the chase vehicle (rendezvous vehicle).

The Chase Vehicle utilizes a custom star tracker design GSFC has recently developed. This system allows optical tracking for a range interval from 500 meters down to 2 meters. In order to close within 500 meters, the chase vehicle performs its own RF Doppler ranging in S-band from 10 kilometers to 500 meters. This system utilizes a 512 x 1024 rad-hard CMOS detector. The total mass is only 2 kg and it requires 4.5 watts of power. A dual head optical system with a 12 degree FOV for long range and a 24 degree FOV for short range is required. This system is capable of determining bearing knowledge to 10 arc seconds and has an accuracy of 150 meters at 500 meters and 1 cm at 1 meter. It runs on a 100 kRad-hard signal processing system. This system could also provide a dual monopulse local radar capability; however it was determined that this would not be required. This system benefits from designs for Hubble Space Telescope (HST) Servicing Mission 4 (SM4) and the Inter-spacecraft Ranging and Alarm System for the Magnetospheric Multiscale (MMS) mission.

Once the chase vehicle is within 1-2 meters of the sample sphere (SRC) magnetic traction is engaged. An ultra-strong permanent magnet at the base of the capture cone (which was also the bottom of the reentry vehicle aeroshell) would pull the sample sphere into a locked position. It was determined by analysis that magnetic traction will function from much greater distances but at a much slower rate. The holding force of this magnet is so large that a locking mechanism is not required to hold the SRC in the aeroshell during reentry. Crushable aluminum honeycomb structure is added to the resting location of the SRC at the magnet in order to safely decelerate the incoming SRC because its velocity increases greatly in the final moments before ground contact (**Figure 17**).

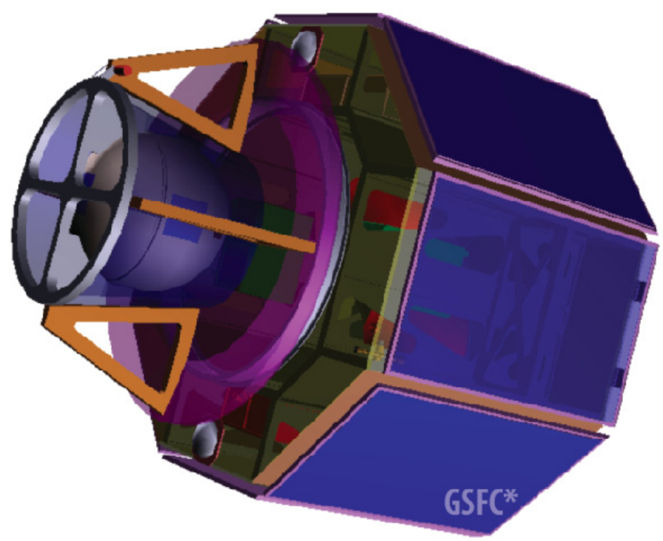

Figure 17: Boomerang Capture and Earth Return Vehicle.

Once the SRC is safely captured within the descent vehicle, the chase vehicle becomes the Earth return vehicle. The Earth return vehicle releases the reentry capsule over the Utah Test and Training Range (UTTR). The reentry interface is approximately 125 km in height and 11.5 km/s. A spin rate of only 2 rpm is needed for stability. Stardust came in at 13 rpm due to its high center-of-gravity (CG). A drogue chute will be used which opens in the target zone to

50 km x 30 km (landing ellipse), similar to *Stardust*. (**Figures 18** and **19**).

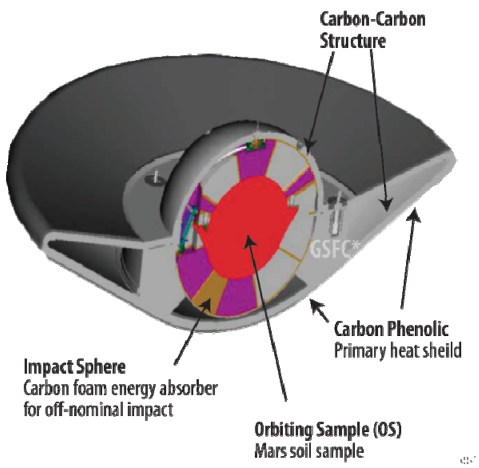

Figure 18: Boomerang Earth re-entry vehicle with SRC inside.

Figure 19: JPL Mars reentry test vehicle.

We also considered a Mars sample return concept which represents yet another low cost concept worthy of future consideration.

17.0 CONCLUSION

The LEx mission is designed to achieve a low cost robotic landing within a polar region of the Moon in the shortest amount of time (i.e., development). The proposed LEx mission is a pathfinder for a cost effective high return-on-investment surface access flight system and a new class of "fly-to" mobility (hopper). The LEx concept would enable a traverse of an unexplored locality and provide vital knowledge regarding the lunar south polar region's suitability as a human exploration landing region, which itself could be a tremendous step toward establishing a sustainable foothold on the Moon. Other approaches for water ice detection from orbital remote sensing or via

Impact related systems (i.e., LCROSS) cannot validate remote sensing measurements accurately enough to provide timely and unassailable strategic knowledge.

The LEx mission concept could provide a substantial foundation for NASA as a pathfinder for low cost lunar surface access. The return on investment will include removing the NRE cost of future LEx-class landers. The use of multiple LEx landers for future NASA (i.e., ESMD's xScout) or related programs would facilitate characterization of multiple landing sites, while their low cost would allow deployment of additional robotic assets.

The Boomerang lunar surface sample return mission offers the potential of returning at least 100g of precision targeted lunar materials from the SPA Basin at a cost and complexity that is "in family" with NASA Discovery class missions. It leverages the mission concept design and definition conducted by NASA's GSFC in 2005 for the LEx lander, which was bid to the ESMD LRO Co-manifest competition.

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Joe Burt is the Deputy Program Manager Technical for the Explorers and Heliophysics Projects Division which currently includes the Planetary Science Projects Division at NASA's Goddard Space Flight Center. This includes oversight of over 15 missions at various stages of development including, MAVEN, SAM, OSIRIS-Rex, Solar Probe Plus, LADEE, SDO, and MMS. Prior to this position Joe was the chief engineer for the Goddard Lunar Program Office which developed and managed LRO. Prior to that he served for over 6 years as the Chief Mission Systems Engineer for the James Web Space Telescope. Joe has been at Goddard for 25 years.