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A DISCOVERY-CLASS LUNETTE MISSION CONCEPT FOR A LUNAR GEOPHYSICAL NETWORK

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The *Lunette* mission concept for a network of small, inexpensive lunar landers has evolved over the last three years as the focus of space exploration activities in the US has changed. Originating in a concept for multiple landers launched as a secondary payload capable of regional science and site survey activities, *Lunette* has recently been developed into a Discovery-class mission concept that offers global lunar coverage enabling network science on a much broader scale. A particular mission concept has been refined by the *Lunette* team that would result in a low-cost global lunar geophysical network, comprised of two landers widely spaced on the near side of the moon. Each of the two identical landers would carry a suite of instruments that would make continuous measurements of seismic activity, heat flow, and the electromagnetic environment during the full lunar day/night cycle. Each lander would also deploy a next-generation laser retroreflector capable of improving on distance measurement accuracy by an order of magnitude over those emplaced by the previous Apollo and Lunokhod missions. This paper presents a comprehensive overview of the Lunette geophysical network mission concept, including mission and flight system design, as well as the key requirements and constraints that guided them.

I. INTRODUCTION

The Lunette study began at JPL four years ago as an exercise to develop a low cost lunar lander that could be used for a variety of missions. The initial concept began as a multi-lander mission launched as a secondary payload on a lunar or GTO-bound EELV. The packaging constraints involved in adapting the lander to the envelope available from the EELV Secondary Payload Adapter (ESPA) drove a compact, simple design. Cost was kept low by specifying that available flight qualified subsystems and components be used to the maximum extent possible [1].

The concept evolved over the years as the aims of the US space program have transformed. What was originally conceived as a precursor mission for a lunar base regional site survey at the lunar poles became more broadly applicable to a wide range of scientific missions that could be performed at any location on the moon. The shift to a more capable single lander design was made in steps, first by replacing the six-lander mission's ESPA ring and common braking motor with a dedicated solid rocket motor (SRM) for each lander. Fixed, vertical solar arrays suitable to operations in a polar environment were augmented with deployable horizontal arrays for lower latitude operations, and size was increased to allow accommodation of a wider range of payloads once the ESPA envelope constraints were removed [2].

Recently the Lunette design team has focused its activities on development of the mission and flight system for a lunar geophysical network mission. This mission would be carried out by two landers targeted to geographically diverse lunar terranes with identical instrumentation, operating on the surface for a period of four years.

II. MISSION CONCEPT

The mission would begin with launch on an Atlas V 401 launch vehicle (LV), with both landers integrated into a common launch vehicle adapter. The LV upper stage would place the stack on a trans-lunar trajectory, following which the two landers would deploy from the

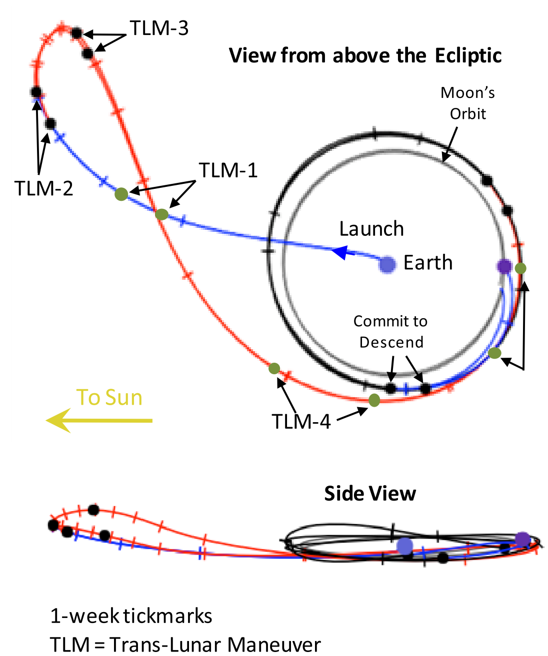


Fig. 1. Trans-lunar Trajectory

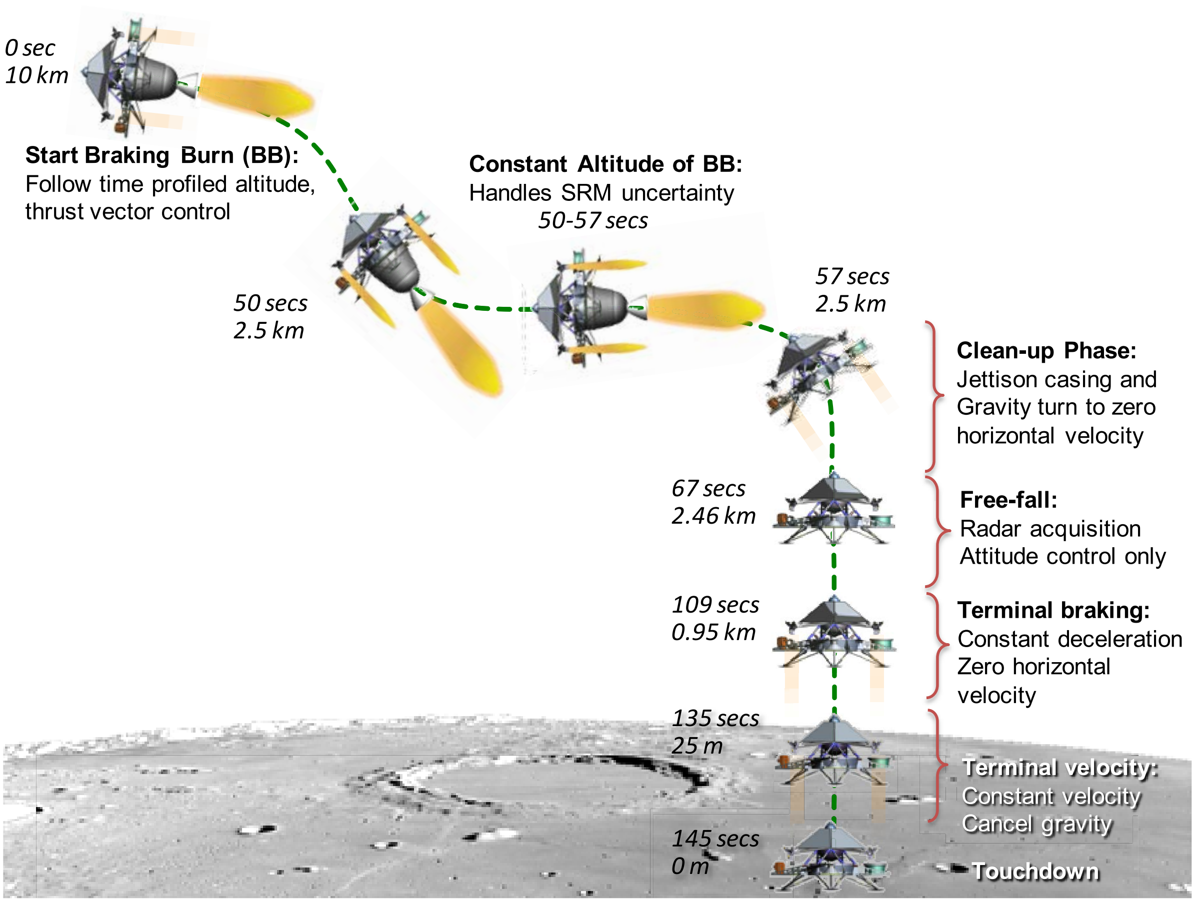


Fig. 2. Lunette landing scenario

upper stage. After deployment the landers would pursue similar but independent low energy trajectories to staging orbits at the earth-moon L2 point (LL2, Fig. 1).

The landers would remain in their staging orbits in preparation for lunar approach and landing. This would commence about seven days prior to landing and each lander would be timed to arrive at its target in the local morning to ensure sufficient time for post landing checkouts and instrument deployment. The two landers would land at least one month apart, allowing navigation and operations teams to focus on each landing and initial surface operations phase independently.

The major landing burn would be accomplished using an SRM beginning at an altitude of about 10 km and a downrange distance of about 70 km from the landing site (Fig. 2). The SRM would remove most of the horizontal velocity from the lander before being jettisoned at about 2.5 km altitude. From this point an integral hydrazine monopropellant system would take over for final braking and lunar descent and landing.

Landing and initial vehicle checkouts would be followed by a landing site survey using fixed wide field-of-view engineering cameras mounted on the lander body. This survey would be used by the operations team to determine optimum placement of the science instruments on the surface. Four types of science instruments would be carried on each lander consisting of:

- SEIS (SEISmometer): An extremely sensitive instrument combining a 3-axis triad of SP (Short Period) sensors and a 3-axis set of LP (Long Period) sensors, to be placed with its environmental shield on the surface;
- HP^3 (Heat flow and Physical Properties Package): A pair of self-penetrating Moles, each carrying thermal and physical sensors at least 3 m below the surface to measure the heat flow from the lunar interior;
- L2R3 (Lunar Laser Ranging Retro-Reflector): A high-precision, high-performance corner cube reflector (CCR) for laser ranging between the Earth and the Moon;
- EMS (ElectroMagnetic Sounder): A set of directional magnetometers and electrometers that together probe the electrical resistivity and conductivity of the interior.

The SEIS , HP^3, and L2R3 instruments would be deployed on the lunar surface by a robotic deployment arm, while the EMS instruments deploy directly from the lander. Following deployment, which should be completed within the first six earth days after landing, the instruments begin collecting geophysical data for a period of four years.

III. FLIGHT SYSTEM DESIGN

The Lunette flight system represents a straightforward design consisting of a lander and an SRM-based braking stage, following a formula first proved in the Surveyor program. An overview of the lander is shown in Figure 3. Design simplicity is enhanced through the avoidance of deployable appendages and mechanisms. Solar arrays and landing legs are fixed, and a catalog S-band telecom system provides ample data return capability from the lunar surface using a fixed omni antenna. The propulsion subsystem has been designed using catalog thrusters and tanks that can be integrated and tested as a complete subsystem before delivery to system ATLO. Likewise, the design of the warm electronics box (WEB) is such that avionics, batteries, and instrument electronics can all be integrated and tested as a complete unit prior to start of ATLO. Instruments and the deployment arm are mounted on external platforms for flight, with instruments linked by tethers to their electronics in the WEB.

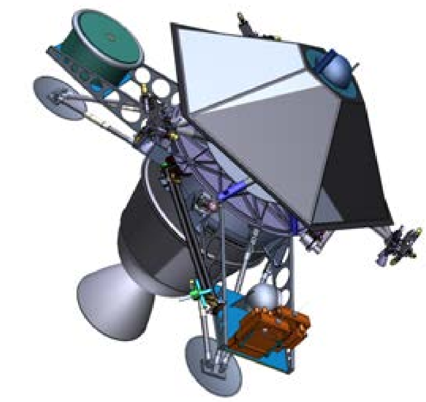
The integral propulsion system uses Phoenix heritage monopropellant landing thrusters in a pulse-modulated throttling design, with two 60 lbf landing thrusters mounted on each of three thruster clusters. In addition, a cluster of three reaction control system (RCS) 0.2 lbf thrusters are co-located with the landing thrusters. The propulsion subsystem is assembled as an integrated unit, with thrusters, support structure and fuel and pressurant tank integrated and tested together before integration into the lander. A Star 30BP SRM is used for the braking burn.

Fixed solar array panels are mounted on top of the propulsion system and are angled to provide a relatively uniform power output of about 200 W (at a 40° latitude landing site) over the course of the lunar day. Power output is about 260 W during cruise.

Operation of instruments over the lunar night was a driving requirement on the flight system design. Lunette has addressed this using a systems engineering solution that combines operations constraints, thermal design, and power management to allow continuous data taking using a minimum of power. Nighttime operations are supported by a nominal 4500 Whr LiIon battery contained in the lander WEB.

To minimize power consumption over the lunar night the C&DH and other subsystem elements are shut down, with the exception of the instruments and an Event Timer Module (ETM) providing timing and data storage and transfer functions. The ETM serves as an interface between the instruments and the C&DH sub-

Cruise Configuration



Landed Configuration

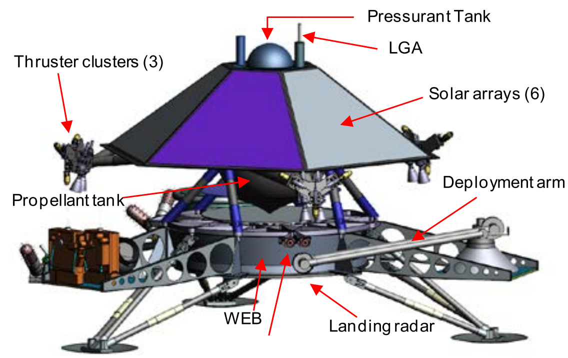


Fig. 3. Lunette Flight System Configuration

system. Data collected by the instruments is passed to the ETM for one minute out of every hour over night, where it is stored in non-volatile memory and time-correlated by the ETM's highly stable chip-scale atomic clock. At dawn, when solar power is once again available, lander subsystems are re-activated and science data is passed to the C&DH for processing and return to Earth.

Thermal control of the WEB is achieved through use of multilayer insulation and low conductivity materials to isolate the WEB from lander structure, and thermal switches to isolate the WEB from its heat rejection radiator at night. Modeling shows that these features will keep nighttime heat leakage to less than 3W total, minimizing the need for heater power.

Instruments are mounted on platforms above the legs for cruise. After landing the instruments are deployed using a four degree of freedom (DOF) robotic arm. The arm has a reach of 2 m and an operational range of 240° as shown in Figure 4. The SEIS, HP^3, and L2R3 instrument are deployed by the arm via a "crows foot" end effector which interfaces with a ball and wire interface on each instrument as developed for the 2001 Mars lander. The L2R3 retroreflector instrument requires one additional degree of freedom which is included in the end effector. This rotational DOF allows the retroreflector to be aimed within 1° of Earth after placement on the lunar surface.

The EMS instruments are deployed directly from the lander. These consist of three electrometers which are launched from spring-based launchers to a distance of about 20 m. A fourth electrometer is mounted on a one-m vertical mast extending from the top of the lander. A flux gate magnetometer and a search coil magnetometer complete the EMS package. They are deployed from the lander on a horizontal arm as shown in Figure 4.

The flight system total launch mass is 2343 kg. This is divided between two flight systems at ~958 kg apiece, plus a custom launch vehicle adapter, estimated at 428 kg. Each flight system carries 78 kg of hydrazine propellant at launch, as well as 495 kg of solid propellant in each Star 30BP SRM. Each lander's dry mass at landing is about 292 kg.

Injected mass capability of the Atlas V 401 launch vehicle to the required C3 for this mission is about 3465 kg, leaving over 1100 kg of margin on the launch vehicle for the two lander mission. This extra mass could conceivably be used to fly a third lander on the same launch vehicle and with moderately higher performance launch vehicles it would be possible to accommodate four or more landers on a single launch.

IV. CONCLUSION

The Lunette concept has evolved considerably over the last four years as the targeted missions have changed. Though additional capabilities and performance have been incorporated in the flight system design the basic philosophy of simplicity, low cost and low risk have been maintained and many subsystems have been brought forward into the current design essentially unchanged from the initial concept. Flexibility also remains a key Lunette feature, with a mission and flight system design that can access and operate in almost any region of the moon without significant modifications. The current architecture has been tailored for the lunar geophysical network mission, but the basic lander can be adapted to a range of scientific and precursor missions that will continue to be assessed by the team as the Lunette study continues.

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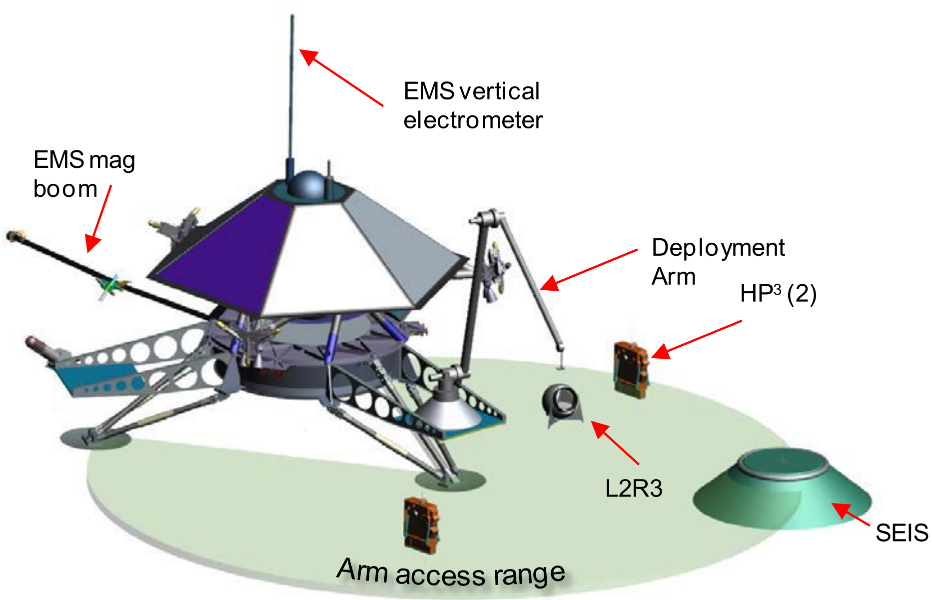

Fig. 4: Instrument Deployment