# A PROPOSED MICROLANDER FOR LOW-COST LUNAR MISSIONS.

J. M. Kruep[1], W. P. Blase[2], and V. Olliver[3]

[1] TransOrbital, Inc., 11826 Federalist Way, # 12, Fairfax, VA 22030, kruep@erols.com.
[2] TransOrbital, Inc., 6430 The Parkway, Alexandria, VA 22310, pblase@aol.com.
[3] TransOrbital, Inc., 72 Warner Park Avenue, Laingholm, Waitakere, New Zealand, vsop@ihug.co.nz.

## Abstract

Microspacecraft using selected off-the-shelf components are becoming more prevalent as universities, private organizations, the military, and NASA are required to reduce project costs and times simultaneously. In addition, falling component and launch prices are making microsatellite design, construction, and flight projects more and more feasible for university aerospace schools and small governments. After an almost 30 year hiatus, there is a renewed interest in exploring the Earth's nearest neighbor, the Moon. Clementine and Lunar Prospector paved the way, and also demonstrated the possibilities of the low-cost microspacecraft approach. Even more missions are planned for the next decade. These include not only the NASA, the European Space Agency (ESA) and the Japanese Space Agency but also private concerns. In this paper, a low-cost "microlander" microspacecraft is described that is capable of carrying small payloads to the lunar surface and providing power and communications services. The microlander is intended to be a modular, adaptable, common platform that is suitable for a number of purposes, including both landing missions and orbital missions. Several possible missions are described, including a mission to the lunar poles to obtain ground-truth data against which the Lunar Prospector data may be calibrated, and a lunar sample return mission.

## Introduction

A micro-spacecraft is one that weighs on the order of 100 kilograms, as compared to the 1000 kg or more for a traditional communications or earth-surveillance satellite. Typically, they are also low-budget, specialized craft performing limited tasks. The amateur radio community, in particular the Amateur Radio Satellite Corporation (AMSAT), has launched over 30 repeater and relay microsatellites since 1961.[1,2,3]

Over the last decade, NASA and the U.S. Air Force have shown increased interest in "smaller, cheaper, faster" microspacecraft for performing a variety of missions from Earth surveillance, to flocks of microsatellites acting together to form a synthetic aperture radar, to interplanetary exploration. Perhaps the most famous microspacecraft of recent time, *Clementine*, was the first lunar mission launched by the United States in 20 years, and returned a tremendous amount of data on the lunar surface.

Microspacecraft typically perform a limited number of tasks, with 1 to 3 instruments or sensors per craft. However, because they can be constructed relatively inexpensively - especially if a common "bus" design is used - many of them can be launched in rapid succession. These microspacecraft can carpet a target with a large number of similar sensors to make up for limitations due to spacecraft size (or to allow less "hardened" and thus less expensive instruments), or carry a variety of different sensors. [4]

> Microspacecraft of the order of 10 kg, employing miniature sensors and instruments, are uniquely suited for three types of

space science missions: missions that require multiple simultaneous measurements in different locations, missions that require very high launch energy and high risk missions where risk may be reduced by replacing a single large and expensive spacecraft with numerous independent small crafts. [5]

This paper describes a lunar lander which is currently under development and is intended to serve as a common platform for a variety of lunar missions. It has a limited payload and limited lifetime on the lunar surface, but will cost at least an order of magnitude less than previous landers. Also described are several missions that can be performed with this platform or adaptations of it.

## *The Electra Platform*

The *Electra* platform is named for a handmaiden to Artemis, the Greek goddess of the moon, who is identified as the missing "seventh" sister in the Pleides who fled to Artemis following the fall of Troy. It is a microspacecraft platform, based largely on proven technology and components. To minimize costs, it is intended to be launched as a secondary payload on a launch vehicle carrying a commercial payload. It may also be carried on a variety of launchers, including the reusable launch vehicles currently under development.

The *Electra* platform consists of a single bipropellant engine; fuel tanks; attitude control thrusters; an electronics chassis containing the flight computer, communications equipment, and navigation sensors; a multi-purpose imaging system; and a frame with landing gear. Figure 1 shows the spacecraft folded for launch, as it might appear just after ejection from the launcher. During flight, the landing gear would be extended to landing position and locked, but the photovoltaics and imaging system would be kept folded.

The imaging system, which is located behind the right-front landing gear as seen in Figure 1, is positioned so that it can view directly down, or can rotate in elevation over 200 degrees. During flight, this imaging system serves as a navigation sensor and obtains star, Earth, and Moon images for attitude determination. During the landing cycle, the imager can view the lunar surface and give the Earth-bound operators a view of the terrain upon which Electra is landing.

**Figure 1 - Electra Platform, folded for launch**

Figure 2 shows the *Electra* fully deployed, as it would appear on the lunar surface (with the photovoltaic panels removed for clarity). The imager and high-gain antenna have been rotated to their upright positions. The imager sits above the main body of *Electra* so that panoramic images of the surface may be obtained from a height.

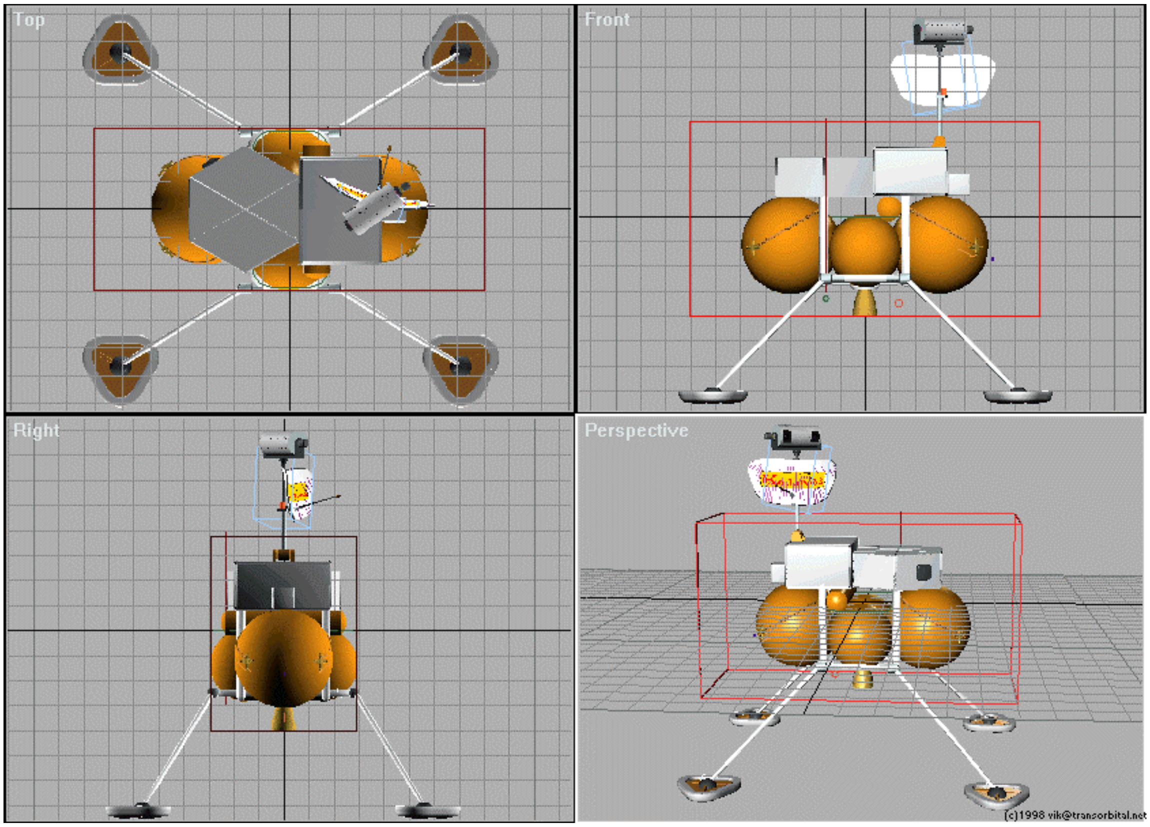


**Figure 2 - Electra Platform, unfolded for landing (photovoltaic panels removed for clarity)**

The hexagonal box in the figures above is the electronics chassis. This is derived from a proven, flight qualified design developed by One Stop Satellite Solutions of Ogden, Utah. The square box is the payload container. Below these boxes are the fuel tanks and the propulsion system. Figure 3 shows the *Electra* with photovoltaics unfolded after landing.

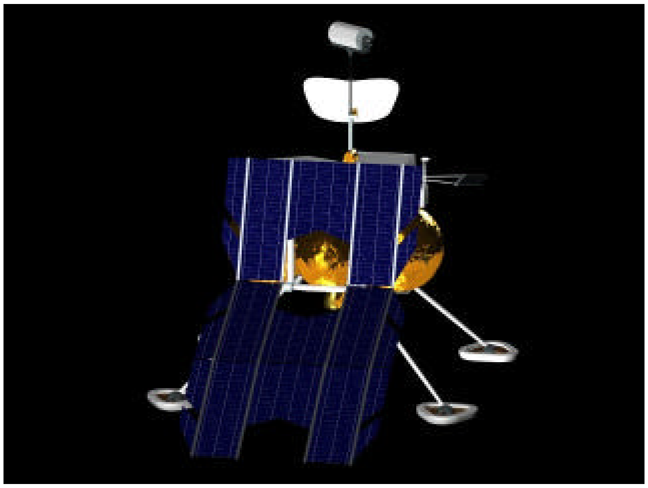

**Figure 3 - Electra, unfolded after landing**

**Spacecraft Specifications**:

| | |
|---|---|
| *Main Propulsion*: | Single bipropellant thruster, 250 - 350 N (55-75 lbf) thrust, utilizing high-purity hydrogen peroxide and RP-1. |
| *Attitude Control*: | 3-axis stabilized via monopropellant hydrogen peroxide reaction thrusters. |
| *Electronic systems*: | Based on off-the-shelf modules developed for the microsatellite industry. Includes GPS receiver, 3-axis inertial sensors, and on-board computer, and may also include sun angle sensors and Earth and Moon horizon sensors. |
| *Communications:* | C band for command uplink and data downlink. Data downlink at 1 Mbit/sec. |
| *Power:* | 100 Watts on surface, from deployable photovoltaic panels; 50 Watts available for payload. |
| *Size:* | 80 cm (31 inches) H & W, 60 cm (24 inches) D.<br>45 kg (100 lbs) dry mass including payload, 200 kg (440 lbs) fueled |

## Propulsion and Reaction Control System

The lander's propulsion system utilizes environmentally benign, or "green", hydrogen peroxide and a hydrocarbon fuel. These were chosen to promote safety and ease of handling during testing and launch preparations. The traditional propellants hydrazine (and its variants) and nitrogen tetroxide ($N_2O_4$) have been in use for decades and are well understood. However, they are both very toxic materials that require storage and handling under dry nitrogen pads/purges and special facilities and permits for testing. High purity (85-90%) hydrogen peroxide and RP-1, on the other hand, require only moderate care during handling, do not require inert atmospheres, can be easily flushed if spilled during handling, and produce a comparable specific impulse performance. [6,7]

The Reaction Control System will utilize monopropellant catalytic-decomposition thrusters fed from the same hydrogen peroxide tank that feeds the main thrusters. The spacecraft will be fully stabilized in three axes.

## Guidance, Power, Control, and Communications Systems

The electronics systems utilized on board the spacecraft will be based on the proven microsatellite platform developed by One-Stop-Satellite-Solutions, the commercial offshoot of Weber State University, Utah, Center for Aerospace Technologies (CAST). This platform, a block diagram of which shown in Figure 4, contains the CPU, power management circuitry, batteries, and navigation instrumentation. It has been flight-proven on OSSS' JAWSAT and other microsatellites. [8]

Due to the unique requirements of the microlander, the reaction control wheels used for the platform's original Low Earth Orbit (LEO) applications will be removed and a solid-state inertial measurement unit (IMU) will be added. This IMU will most probably be the space-rated Litton LN-200 3-axis system, which incorporates fiber-optic gyroscopes and solid-state silicon accelerometers. Additional attitude and position sensors will be added as determined to be necessary during the design process. Currently, TransOrbital is considering adding infrared lunar/earth horizon sensors for attitude determination, and utilizing the on-board camera - which will be taking images of the landing site - as a starfield sensor and to take images of the moon during low lunar orbit. These images would be compared on Earth with calibrated data from the Clementine lunar orbiter for moon-relative position determination. During Earth orbit a Global Positioning System (GPS) receiver, part of the OSSS platform, will provide accurate spacecraft position information. One unique instrument required for landing on the lunar surface is a radar altimeter. In order to minimize cost, TransOrbital is investigating using a space-rated variant of a commercially available altimeter manufactured by Roke Manor Research, Ltd. (UK) for unmanned aerial vehicles.

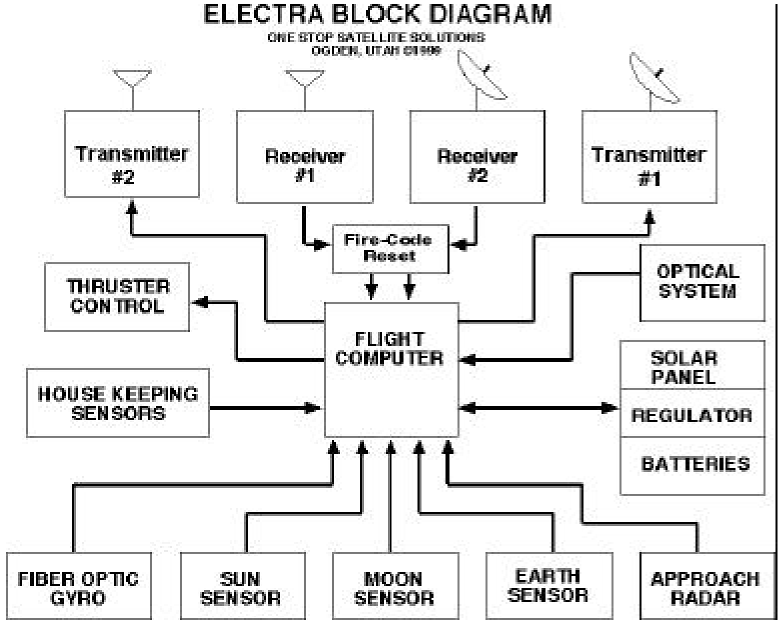


**Figure 4 - Proposed Electra Block Diagram (courtesy OSSS)**

Spacecraft power will be provided by approximately 1 square meter of photovoltaic panels on the exterior of the spacecraft, buffered by silver-zinc batteries. The panels will be split between the two long sides of the spacecraft and will provide approximately 100 Watts of power. During flight, the panels will be flat against the spacecraft and the spacecraft will rotate to optimize the internal thermal environment, exposing each panel to the sun alternately. After landing, the panels will be extended outwards to 45 degree angles so as to catch the sun on its (apparent) path through the sky during the lunar day. (The spacecraft will be oriented along a North-South axis during landing so that one panel faces roughly east, and the other west). The batteries will provide approximately 50 Watt-hours of storage to keep the spacecraft alive during eclipses and thruster burns when the panels cannot be oriented toward the sun.

The spacecraft's communications system will utilize a split-frequency uplink and downlink. The exact frequencies will depend on assignment by the Federal Communications Commission during the licensing process and the frequencies supported by the ground stations. TransOrbital anticipates using the C band - 6 GHz uplink, 4 GHz downlink. Fixed low-gain antennae will be used during flight, when the spacecraft attitude may exclude use of the high-gain antenna. Currently, TransOrbital is planning for a transmitter output of approximately 20 Watts to allow high-bandwidth video links. The uplink will carry only commands for the spacecraft; the downlink will carry digitized video data from the cameras, as selected by the ground operators, payload data, and spacecraft telemetry.

## Imager

The Imager is derived from "Imager for Mars Pathfinder" (IMP) camera developed by the University of Arizona for the Mars Pathfinder probe[9]. Figure 5[10] shows the IMP as used on the Pathfinder spacecraft. For the *Electra*, the IMP's imager would be replaced by a high-resolution CCD array, and the filters in the wheels would be changed. Some filter positions in the wheels would still be available for scientific imaging, but most of them would be replaced by neutral-density filters to increase the camera's dynamic range.

The on-board imaging system serves three primary purposes. First, the video acts as a navigation sensor, returning images of starfields, the Earth, and the lunar surface during flight to serve as navigation position and attitude data. For the most part, the camera would act as a conventional star-field or horizon sensor, with azimuth/elevation navigation information extracted from the camera pan/tilt position and the location of objects in the field of view. During lunar orbit and the landing cycle, however, it should be possible to calculate altitude and position from triangulation of known features, using the considerable database accumulated by the Surveyor, Apollo, Clementine, and Lunar Prospector spacecraft. Because of the limited on-board processing power available, the images will be transmitted to Earth and processed there. One possible use of onboard processing would be real-time tracking of a known object, e.g. the moon or a bright star, to augment the inertial navigation system during engine burns.

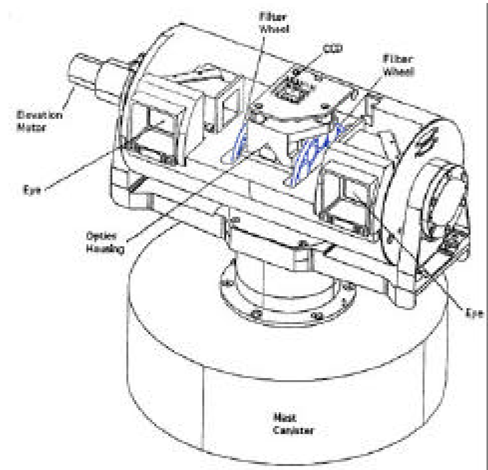


**Figure 5 - Diagram of IMP**

The second purpose of the on-board imaging system is to return real-time images, although not full-motion, of the landing site while the spacecraft is descending and preparing to land. The spacecraft will be capable of hovering for approximately 60 seconds before touching down, allowing time for one or two images to be obtained and transmitted to Earth. This will allow the terrestrial operators to inspect the landing site for hazardous debris and craters and re-direct the landing to a limited extent.

The final purpose of the imaging system is to serve as the surface science/payload imager, returning views of the landing site and payloads deployed to the surface. In this mode, the imager can acquire high-resolution stereo images at approximately 1 image pair per second, or moderate resolution monoscopic images in real time. It can also acquire a high-resolution stereo panoramic view of the landing site.

## Launcher

The primary launch platform for this mission will be a double-micropalette on the Ariane IV or V launch vehicle. Figure 6 shows the single- and double-micropalette configurations, with an outline of the lander inserted into the double-palette. There are 12 micropalette slots, each 600 mm in diameter by 800 mm high, with their centers spaced equally at 30 degrees around a circle 3355 mm in diameter. A double-palette comprises two single-palette slots and the intervening space.

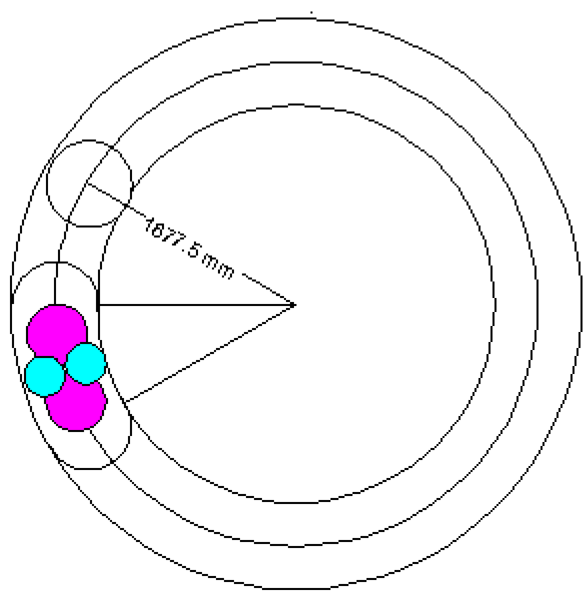


**Figure 6 - Ariane Palette Configurations**

An alternate launch platform is one of the Russian SS-19 derivatives, such as the "Eurokot" marketed by a European/Russian consortium. This launch vehicle would offer the advantages of allowing TransOrbital to better select a launch time and to be the sole payload on the launch (although we could always sublease to other paying customers). However it is somewhat more expensive than an Ariane micropalette, and a solid-fuel upper stage, such as the Thiokol STAR-37, would be required to augment *Electra*'s on-board fuel tanks.

### Trajectory

The primary choice of launch platform for the *Electra* Microlander is as a secondary payload on an Ariane launch to Geosynchronous Transfer Orbit (GTO). The launch window will be chosen/negotiated with Arianespace and the primary payload customer to allow the Microlander to reach lunar orbit in less than six weeks after launch. The orbital sequence the Microlander will follow is illustrated in Figure 7. The trajectory is based upon the GTO to Lunar Orbit trajectory described by Uphoff[11].

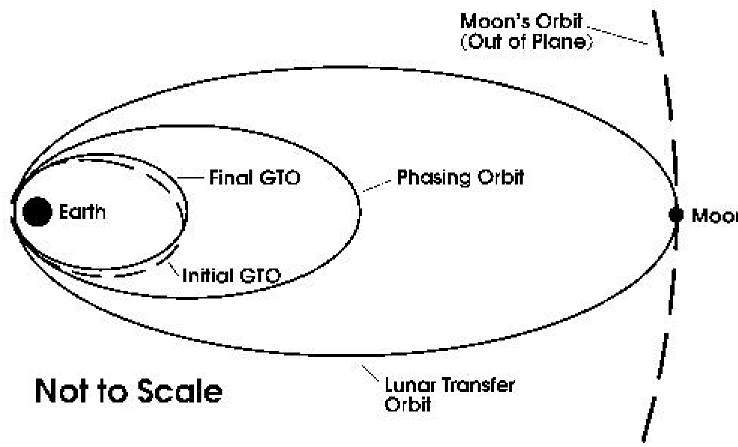


**Figure 7 - Electra Trajectory [After Uphoff]**

After launch, the Microlander will wait as its orbit slowly precesses into the proper alignment to intercept the Moon's orbit. Once the proper alignment has been established, the Microlander's apogee will be raised by a propulsive burn to put it into an intermediate, "phasing" orbit that will allow the Microlander to wait until the Moon reaches the proper point in its orbit for an intercept. The phasing orbit will also avoid some of the high radiation exposure that occurs during an extended stay in GTO. When the proper time has elapsed, the Microlander will again fire its engine to move it into a Lunar Transfer Orbit (LTO), and the lander will be on its way to the Moon.

At least one trajectory correction maneuver is planned during the Earth-to-Moon coast to allow for tuning of the lunar arrival orbit. The target lunar orbit has not yet been chosen, but it will likely be a low-altitude polar orbit. This will permit surface images to be obtained during flyovers of the lunar poles. The Microlander will wait in its Low Lunar Orbit (LLO) until an acceptable sun angle over the target landing site has been reached, then will make its descent to the target on the lunar surface.

## ***Proposed Missions***

### Lunar Microlander

The initial planned mission for the Electra platform is to scout a proposed site for the lunar base being planned by the Artemis Society[12]. The purpose of the mission is to prove the viability of the Electra microspacecraft and its design principles. The goals of the mission are:

- to obtain survey imagery of the Angus Bay area (shown in Figure 8) during landing;
- to safely land at Angus Bay (Mare Anguis, located at 22.6N 67.7E, near to Mare Crisium) and return a panoramic image of the landing site to Earth;
- to deploy a payload to the lunar surface;
- to deploy and return images from a remote camera capable of imaging the lander itself and the payload on the surface;
- and to survive the remainder of the lunar day and continue to return imagery of the lunar surface and of Earth.

**Figure 8 - Proposed *Electra* landing site**

A summary of the preliminary Concept of Operations for this mission follows:

### *Post Deployment*

After deployment from the launch vehicle, the *Electra* will power up its systems and perform a series of self-diagnostics. The Microlander will ascertain its position and velocity through the use of its onboard GPS system, and determine its attitude with its onboard attitude sensors. It will attempt to contact mission control, pressurize its attitude control system, and orient itself so that its solar panels provide power. When commanded to do so, it will jettison any excess pieces of its launch adapter and extend its landing legs.

### *Earth Orbit*

During the period of the Electra's orbit around the Earth, which could last anywhere from a few hours to several weeks, the Electra will primarily perform a series of "housekeeping" duties designed to keep the spacecraft healthy and its orbit on target. Orbital navigation will be maintained using the GPS system while at lower altitudes of the orbit and attitude control will be with the onboard sensors and RCS thrusters. Passive thermal control will be used, probably by giving the craft a small roll rate. Communication with the ground will occur as needed, primarily through the low gain antenna system.

At least one, and probably two, main engine burns will occur to put the Microlander into a phasing orbit, and subsequently into a Lunar Transfer Orbit. During these burns the IMU will be used to ensure that attitude control is maintained and that the proper velocity change is imparted to the spacecraft. Very small orbital maintenance burns may also be required. These will be conducted using only the RCS thrusters.

### *Lunar Transfer Orbit*

Lunar Transfer Orbit (LTO) operations aboard the Microlander will be similar to those conducted in Earth Orbit, with the addition of some supplemental navigation methods for estimating both Earth- and Moon-relative distances once the Microlander has moved outside the useful range of GPS.

### *Low Lunar Orbit*

The Low Lunar Orbit (LLO) phase of operations will begin with a main engine burn to insert the *Electra* into stable orbit around the moon. This insertion burn will take place at perilune (the point of closest approach to the moon), which will likely be on the far side of the moon. Therefore, no direct communication with Mission Control on Earth will be possible. When the Microlander emerges from behind the moon, its first order of business will be to inform Mission Control of its status. Once its orbit and health have been verified, *Electra* will also return numerous images and/or video of the surface of the moon. Navigation and attitude control methods may need to be modified to accommodate the proximity to the Moon. High data rate communications will be used more frequently than during the previous flight phases.

### Descent and Landing

The spacecraft will begin its descent with the engine pointed in the direction of orbit. The main thruster will fire continuously during descent, with rotational adjustments and minor course corrections being accomplished with the RCS thrusters. The descent will be initially programmed using position data obtained from examination of starfield and lunar surface imagery. The onboard IMU will be used continuously during descent as the primary means of tracking spacecraft location. At approximately 1 km in altitude, the radar altimeter will obtain lock on the lunar surface and be able to provide the exact distance to the surface. As orbital velocity is lost, and the lander drops towards the surface, the spacecraft will rotate until it is hovering approximately 100 meters above the landing site, with no horizontal velocity. Just prior to hover, the on-board camera will return an image of the landing site to Mission Control. The lander will be able to sustain a hover for up to one minute. During this time the operators will examine the landing site for obvious hazards (e.g. large rocks and deep craters) and have the opportunity to modify the exact landing location. Other than this opportunity for intervention, the descent will be accomplished autonomously by the spacecraft.

### Surface Operations

Weight on the landing gear at touchdown will result in automatic engine-shutdown. Following touch down, the lander will fine tune the high-gain antenna position for maximum signal strength, deploy the photovoltaic panels to surface configuration, and await commands from Mission Control.

Further surface operations will all be commanded from Mission Control. Following is the initial agenda.

- Deploy the panoramic/stereo imager to full height.
- Acquire a panoramic image of the landing site.
- Deploy the remote camera and return an image of the lander on the surface.
- Deploy the cargo, returning live video imagery from the panoramic imager and the deployed camera.
- Continue to return imagery - of the cargo, of the lunar surface and of the Earth, at regular intervals.
- When the lunar night approaches, Mission Control will place the lander into a sleep mode. The cameras will be shut down and data transmission will cease, power will be primarily routed to heaters in order to keep the batteries and electronics above their minimum temperatures, and any propellants in the tanks will be allowed to freeze. The electronics will be put in a minimum power usage configuration, with nothing active except for an interrupt timer on the CPU.
- The next lunar morning (336 hours later), assuming that all systems survive the night, the spacecraft will reawaken when power is available from the photovoltaic panels. The spacecraft will then await additional commands from Mission Control.

Additional (non-polar) landers are also planned. These missions would be comparable to the initial Electra Microlander mission, but would carry more surface scientific equipment and would be enhanced for longer on-surface durations. They could be deployed anywhere on the near side of the moon that is deemed of interest. With the use of a communications relay satellite, they could be deployed to the far side as well.

## Lunar Sample Return

Recently, TransOrbital submitted a proposal to a commercial customer for a lunar sample return mission using a spacecraft based on the Electra Platform. The spacecraft, shown in Figure 9, consists of a core vehicle, the sample return re-entry vehicle which it carries, and a surface module. The ensemble is capable of carrying a 10 kg customer payload to the lunar surface, and of returning 15 kg of lunar material to the Earth.

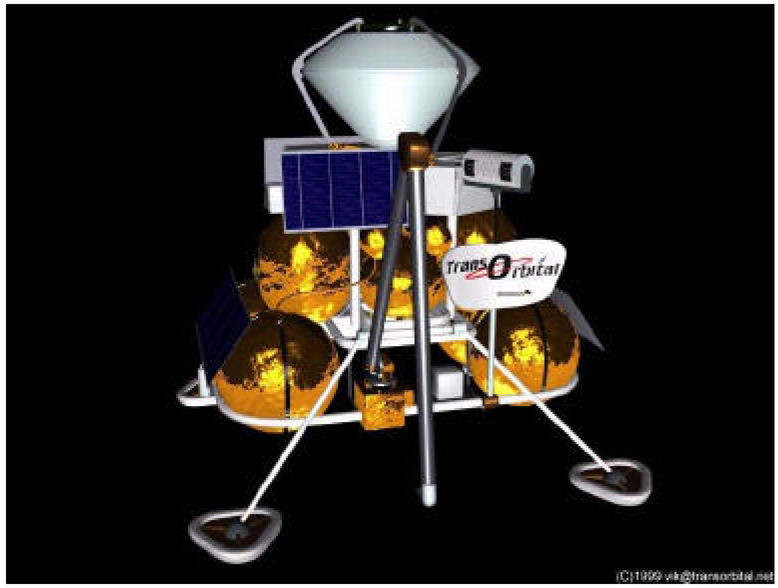

**Figure 9 - Lunar sample return spacecraft**

**Electra Core Vehicle.** The core vehicle is a modified Electra lander which contains the flight electronics and the engines. In its tanks, it has sufficient fuel to carry itself and the loaded sample return capsule from the lunar surface to the Earth, launching from the surface module in a fashion similar to the Apollo LEM. In order to minimize mass for the return journey, the Electra's landing gear, imager, altimeter, and high-gain antenna will be moved to the surface module. In addition, the spacecraft's propulsion system will be upgraded to use three engines instead of the normal single engine. The core unit will mass approximately 30 kg dry, 120 kg fully fueled, not including the sample return capsule.

**Sample Return Capsule.** The sample return capsule is a bi-conical aeroshell similar to the Stardust sample return capsule[13]. It is capable of carrying 15kg of lunar samples safely through reentry to the Earth's surface. It is carried by the core spacecraft and released just before reentry. The capsule will be spin-stabilized. Samples are carried internally in a cylindrical "vault" that is hermetically sealed prior to launch from the lunar surface. After the samples are placed in the vault by the surface-module's robotic arm, a cushioning system will be deployed to retain them in the base of the cylinder. This will ensure the proper capsule center of gravity and protect the samples during reentry and landing. A layer of thermal protection materials surround the entire capsule. An ablative material such as PICA[14] will be used on the capsule forebody, and thermal blankets will cover the aftbody.

The sample return capsule will also contain a parachute, derived from a proven NASA design, a radio transmitter, and a Global-Positioning-System receiver. The parachute will be deployed based upon a combination of accelerometer readings and elapsed times, with a barometric altimeter serving as backup. After reentry and parachute deployment, the capsule will begin transmitting its coordinates, encrypted, to the recovery team. At this time, a water splash-down is anticipated, due to expected reentry licensing restrictions, and the capsule will be capable of floating for a considerable length of time. The return-capsule masses approximately 17 kg empty and is approximately 60 cm in diameter and 41 cm in length.

**Surface Module.** The surface module carries the fuel for the outgoing portion of the mission, the landing gear, and all instruments and cargo which are to be left on the lunar surface. These include the sampling arm, most of the photovoltaic panels, the imager, the high-gain antenna, and the payload. The robotic arm is derived from the arm developed by NASA for the Mars Surveyor 2001 Lander. [15]

Figure 10 shows the core module lifting off from the surface module for the return trip, carrying the loaded sample return capsule. Quick-disconnect couplings carry fuel and electrical signals from the surface module to the core. The surface module masses approximately 63 kg dry, 460 kg fully fueled.

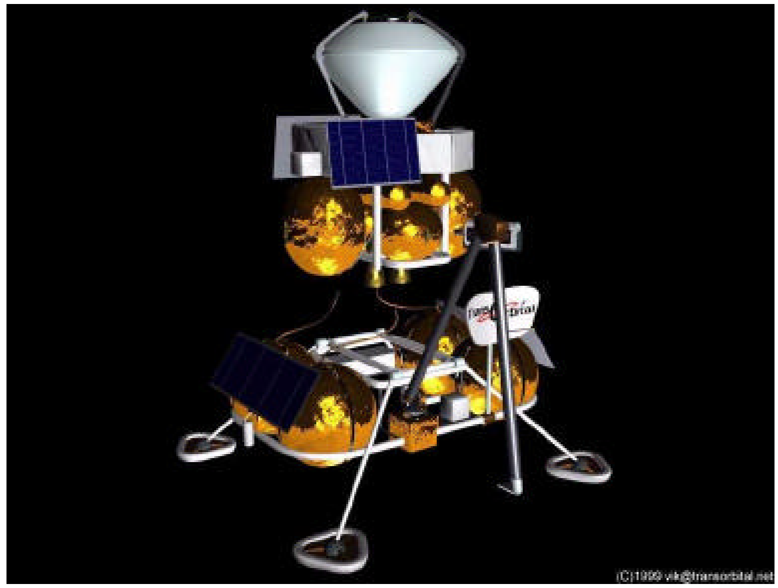


**Figure 10 - Core module launching for return trip to Earth**

## Lunar Polar Lander

The Polar Lander mission has been developed in response to the data returned from the Clementine and Lunar Prospector probes[16] suggesting that there may be large quantities of water buried in the regolith near one or both of the lunar poles. For this mission, a modified Electra would be sent to a region near one of the poles to attempt to confirm the presence of water. Specifically, the goals of the Lunar Polar Lander mission are:

- To search for the presence of water ice buried in the regolith near the lunar poles
- To characterize the concentrations of any water ice found

Two methods for accomplishing this mission have been proposed. Both methods would involve using microprobes similar to the Deep Space 2 probes found on NASA's Mars Polar Lander.[17] These microprobes have been designed survive atmospheric entry and impact on the surface of Mars. Once they have hit ground, they will deploy small drills and will search for water beneath Mars' surface. For a lunar impact, the aeroshells that surround the Deep Space 2 probes would be removed, and replaced with a propulsion package to control the final descent to the surface.

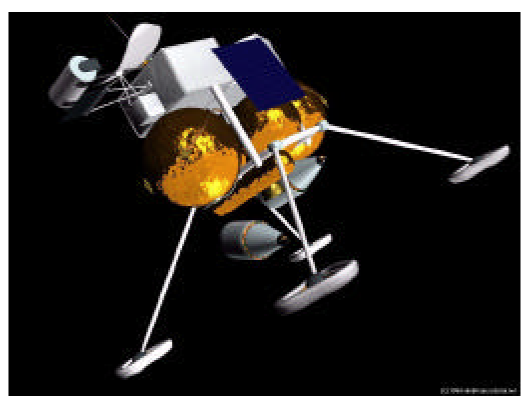

**Figure 11 - Electra configured for polar mission**

The first method for conducting the Lunar Polar Lander mission, shown in Figure 11, would involve using a slightly modified version of the basic Electra platform. For this method, the Electra would be outfitted with two microprobes. The mission profile would be very similar to the basic Microlander mission, and the Microlander would be put into a polar orbit of the moon. The primary variation from the Microlander mission profile is that during the last portion of the descent to the surface, the two microprobes would be deployed. These two probes would use their small propulsion packages to provide the final few hundred meters per second of velocity change required to impact the lunar surface at the proper angle and velocity. The Electra would land a short distance away, and would act as a communications relay for the two probes. Since the near-polar landing site may provide little, if any sunlight or direct line-of-sight communications, the Electra core would be outfitted with more batteries and a communications system capable of over-the-horizon transmission.

The second method for the Lunar Polar Lander mission would use an orbital-only version of the Electra platform. This orbital-only Electra would also be put into a polar orbit. However, this Electra would lack everything needed for landing, such as the radar altimeter and landing gear, and would carry less propellant. In place of these things, the Electra would carry 2 or 4 microprobes with large propulsion modules. These modules would contain enough propellant to remove most of the orbital velocity that the microprobe posses, and direct them to a proper impact on the surface. In this manner, they would be similar to the seismic penetrators planned for Japan's Lunar A mission[18]. Probes could be landed at both poles, or widely spaced near one pole. After the probes have acquired data, the orbiting Electra will relay the data they have collected back to Earth. Since the Electra will be in a polar orbit, the probes should have a clear line of sight transmission nearly every orbit.

---

[1] AMSAT home page: http://www.amsat.org/

[2] Davidoff, Martin , *The Satellite Experimenter's Handbook*, 2nd edition, The American Radio Relay League, Newington, CT., 1990

[3] Baker, Keith and Jansson, Dick, "Space Satellites from the World's Garage -- The Story of AMSAT," National Aerospace and Electronics Conference, Dayton, Ohio, 23-27 May 1994.

[4] Jones, Ross M. "Think small - in large numbers", *Aerospace America*, Vol. 27, pp 14-17.

[5] "Micro Pathfinder: Microspacecraft, Microinstrumentation and Microelectronics", NASA Goddard Space Flight Center on-line document, http://library.gsfc.nasa.gov/Webliographies/Microp~1.htm

[6] Anderson, W. , et al. "Low Cost Propulsion Using a High Density, Storable, and Clean Propellant Combination." AIAA 98-3670. Presented at the 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, July 13-15, 1998.

[7] Whitehead, John C. "Hydrogen Peroxide Propulsion for Smaller Satellites." Presented at the 12th Annual AIAA/Utah State University Conference on Small Satellites, August 31-September 3, 1998.

[8] OSSS home page: http://www.qcontinuum.com/~osss/

[9] Smith, P. H., *et al*, "The Imager for Mars Pathfinder (IMP) Experiment", *JGR-Planets* Special Mars Pathfinder Issue April 21, 1996 Revised. (http://www.lpl.arizona.edu/imp/science/JGR.html)

[10] "How Does The IMP Work", Department of Planetary Sciences, Lunar and Planetary Laboratory, University of Arizona, http://www.lpl.arizona.edu/imp/how.does/how.does.html. Used with permission, credits: NASA/JPL, U of Arizona.

[11] Uphoff, Chauncey. "Practical Aspects of Transfer from GTO to Lunar Orbit." Presented at the Flight Mechanics Theory/Estimation Symposium at Goddard Space Flight Center, 1992.

[12] Artemis Society International home page: http://www.asi.org

[13] Desai, P. N., Mitchtree, R. A., Cheatwood, M. F. "Entry Dispersion Analysis for the Stardust Comet Sample Return Capsule." AIAA 97-3812. Presented at the 1997 AIAA GNC, AFM, and MST Conference and Exhibit, August 11-13, 1997.

[14] Mitcheltree, R. A., Kellas, S., Dorsey, J. T., Desai, P. N., Martin, C. J. "A Passive Earth Entry Capsule for Mars Sample Return." AIAA 98-2851. Presented at the 7th AIAA/ASME Joint Thermophysics and Heat Transfer

[15] H. Das, X. Bao, Y. Bar-Cohen, R. Bonitz, R. Lindemann, M. Maimone, I. Nesnas, C. Voorhees, "Robot manipulator technologies for planetary exploration," in Proceedings of the Smart Structures and Integrated Systems Symposium, Newport Beach CA March 1-5, 1999.

[16] Spudis, P. D., Nozette, S., Bussey, B., Robinson, M., "Lunar Exploration: The next step." *Aerospace America*, February 1999, Vol. 37, No. 2.

[17] "Deep Space 2 – Mars Microprobe," NASA Jet Propulsion Laboratory on-line document, http://nmp.jpl.nasa.gov/ds2/.

[18] Davis, N. W., "Moon and Mars top Japan's ISAS Agenda." *Aerospace America*, March 1999, Vol. 37, No. 3.