Lunar Lander Configurations Incorporating Accessibility, Mobility, and Centaur Cryogenic Propulsion Experience

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As part of the Vision for Space Exploration, NASA plans to develop a human lunar lander, known as the Lunar Surface Access Module (LSAM). A practical lunar lander must transport astronauts to the Moon, then facilitate their activities on the lunar surface. Propulsion systems using LO_2/LH_2 propellants have several advantages, including high specific impulse main propulsion, the opportunity for synergistic application in fuel cells and life support, and compatibility with In Situ Resource Utilization (ISRU). However, due to their very low boiling points, it is difficult to design systems which retain LO_2 and LH_2 for long mission durations. This paper describes design principles based on development and operational experience with the Centaur upper stage which can be applied to build practical LO_2/LH_2 lunar landers. These include the use of a minimum number of large propellant tanks with common bulkheads and thin-wall structure. Consideration has been given to crew and cargo operations on the lunar surface, which drive needs for lander mobility and ease of transfer between the lander cabin or cargo mounts and the ground. The low density of LH_2 means that LO_2/LH_2 landers will have large fuel tanks which can impede crew access to the lunar surface. Three innovative design concepts are presented which incorporate the suggested propulsion design principles and address mobility and access. Concept 1 is a two-stage Dual Thrust Axis lander, which uses an axial main engine for primary descent, then rotates to land with its long axis parallel to the ground. As a result, the crew and cargo are placed very close to the surface. Concept 2 jettisons the descent propulsion system shortly before landing so that the landed vehicle is much smaller. This concept is particularly suited to a mobile lander. Concept 3 is a single stage lander which descends and ascends using the same propulsion system. It is likely to be the lowest cost approach and could be adapted to a reusable architecture.

Nomenclature

CECE	=	Common Extensible Cryogenic Engine
CEV	=	Crew Exploration Vehicle
ECLSS	=	Environmental Control Life Support System
ESAS	=	Exploration Systems Architecture Study
EVA	=	Extra Vehicular Activity (‘moonwalk’)
GH_2	=	Gaseous Hydrogen
GO_2	=	Gaseous Oxygen
Isp	=	Specific Impulse
ISRU	=	In-situ Resource Utilization
LEO	=	Low Earth Orbit
LH_2	=	Liquid Hydrogen
LIDS	=	Low Impact Docking System
LO_2	=	Liquid Oxygen
LOI	=	Lunar Orbit Insertion
LSAM	=	Lunar Surface Access Module
MMH	=	Mono Methyl Hydrazine
mT	=	Metric Tons
VDMLI	=	Variable Density Multi-Layer Insulation

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I. Introduction

Since the announcement of the Vision for Space Exploration in January, 2004, Lockheed Martin has explored a diverse set of lunar lander designs. Lockheed Martin participated in the Concept Exploration and Refinement (CE&R) studies that served as an input to NASA's internal Exploration Systems Architecture Study (ESAS)[1]. During the CE&R effort, Lockheed Martin studied an extensive set of concepts for a human lunar lander including the exploration of a wide variety of ascent and descent propulsion types[2], horizontal and vertical configurations, and mobility options including driving and hopping capability. Since the CE&R studies were completed and the ESAS results were released, Lockheed Martin has continued internal research and development activities studying designs for human and robotic lunar landers. While parts of the exploration architecture are becoming more defined, much about the LSAM remains to be determined. This paper presents three concepts for human lunar landers which address propulsion design issues associated with the lander's role as a transportation system, and operational issues related to its role supporting lunar surface activities. The three concepts are intended to illustrate different design features and provoke further thought. They are not promoted as final designs, nor intended for comparative trades.

II. Exploration Architecture Tenets

The eventual objective of the Vision for Space Exploration is to land humans on Mars. Initial missions to the Moon should be conceived using a 'Mars Back' philosophy which first considers how Mars missions would be performed, and then works backward to design the lunar mission in a manner which develops necessary technology and demonstrates relevant operational approaches. Most details of a Mars mission are undefined, but two features are common to nearly all recent Mars mission proposals: very long surface stays, and In Situ Resource Utilization (ISRU). Demonstrating long duration surface stays on the Moon will require a lunar surface base. Lunar landers should be designed to perform the initial series of exploratory sortie missions quickly and effectively, and then focus on lunar base operations. A proposed approach for rapid and cost effective exploratory missions is described in a separate paper[3]. Landers designed for lunar base operations should be capable of transporting and unloading large, heavy items like habitat modules to assemble the base and bring supplies. The crew transport function should be thought of as a 'taxi' rather than a 'camper.' The lander should support the crew for short periods of time during descent and ascent, rather than having the crew live in the lander for weeks at a time. The lander should also be compatible with eventual transition to lunar-produced propellants in order to demonstrate ISRU operations for a Mars mission, and to reduce the cost of ongoing lunar missions. The most likely propellants for lunar ISRU are oxygen and hydrogen. Even if polar ice deposits are determined to be a non viable resource, oxygen can be produced from lunar soil, and LO_2/LH_2 uses a higher percentage of oxygen than any other practical propellant combination.

III. Lunar Lander Design Philosophy

A. Cryogenic Design Principles

Figure 1 shows the mass reduction benefit of LO_2/LH_2 propulsion for a two-stage lander with ESAS-like design requirements, accounting not only for the Isp benefit but the penalties associated with insulation and larger tanks. The mass savings are more substantial for the descent propulsion than for ascent. It is also somewhat easier to develop a LO_2/LH_2 propulsion system for descent because the system is larger and the storage duration is shorter than for the ascent propulsion system. For these reasons, cryogenic propulsion is focused on the descent stage in the configurations presented in this paper. Using cryogenic propulsion on any in-space element of the exploration architecture will require managing cryogenic propellants for much longer than the half-day missions that have been performed to date. Fortunately, the lessons learned from developing and flying multiple versions of the Centaur launch vehicle upper stage (Fig. 2), indicate a potential design solution that satisfies a lander's needs. Requirements for minimum mass, low heat leak and efficient Cryogenic Fluid Management drive designs towards large single LH_2 and LO_2 tanks.

Previous concepts for cryogenic landers, such as the First Lunar Outpost[4] and ESAS designs, have typically clustered many separate propellant tanks around the main engines (Fig. 3). However, Multi-tank designs will pose challenges for long duration cryogenic propellant storage and cryogenic fluid management. The large surface area and numerous penetrations associated with multi-tank designs significantly increase the heat load relative to a large single tank. Pressure control of multiple tanks is also a severe challenge. Independent pressure control requires redundant valves controlling pressurization, venting and fluid outflow for each tank. The increase in number of valves and system complexity seriously degrades the multi-tank system reliability. Without independent tank control, the inevitable small difference in tank heating will lead to the migration of liquids from tanks with high heating to the tanks with lower heating. This will result in unacceptably large propellant outage residuals because some tanks will still have propellant remaining when the first tanks become empty. This multi-tank propulsion complexity results in increased stage mass and cost, and reduced reliability.

To accommodate long mission duration requirements, the lander will need to control the cryogenic boil-off rate to about 0.1% of the initial combined LO_2 and LH_2 mass per day. The flight-demonstrated Titan Centaur boil-off rate was 1.8%/day[5]. Titan Centaur was designed for 8 hour missions and had no need to further reduce boil-off. However, results from the Titan Centaur missions do provide guidance on system improvements that can satisfy the lunar lander thermal requirements. The improved cryo storage can be achieved through a combination of design features including common bulkhead LO_2 and LH_2 tanks, high performance internal bulkhead insulation, external Variable Density Multi-Layer Insulation (VDMLI), sun shields, vapor cooled structures, propellant positional management, and an over sized LH_2 tank to accommodate primarily LH_2 boil-off (Fig. 4). All of these technologies are individually at a high level of development but must be integrated into an effective design solution[6].

A light weight, smooth tank structure is also critical to cryogenic fluid management. Internal tank protrusions, such as on ortho- or isogrid tanks, act as heat exchange fins, complicating both the cryo storage and pressurization of the tanks. Heavy walled tanks not only reduce the mission performance, but also provide substantial thermal capacitance in the tank walls exasperating the cryogenic fluid management. A thin walled, monocoque tank provides a reasonable solution that satisfies the tank structural needs as well as the cryogenic storage and management in an efficient, light weight package.

Due to inherent thermodynamic properties, it is two to ten times more efficient to vent hydrogen than oxygen in terms of the amount of heat removed per pound of vented gas (Table 1). A common bulkhead provides an efficient and reliable method to direct all stage heating to the LH_2

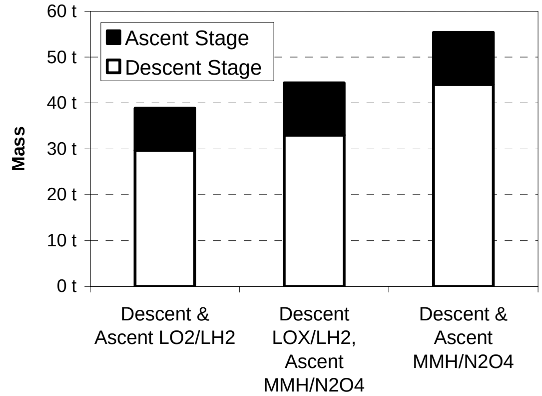

Figure 1. The use of LO_2/LH_2 propulsion for the lander reduces system mass, especially when used for the descent stage.

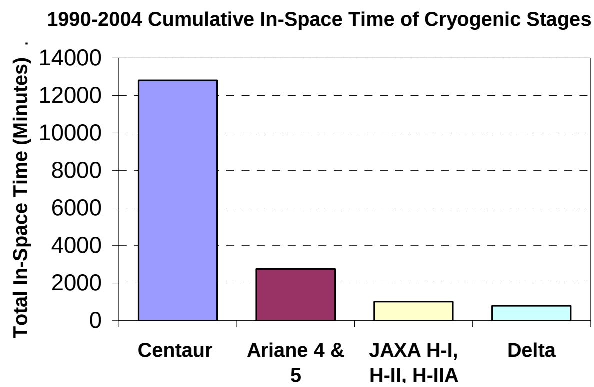

Figure 2. Centaur's 179 flights over 45 years provide unparalleled cryo-fluid management experience that can guide the design of a cryogenic lander.

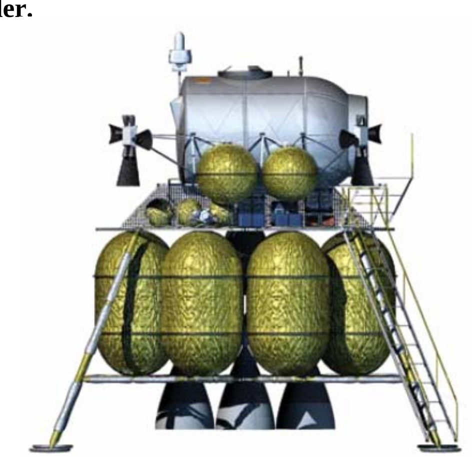

Figure 3. Multiple LO_2 and LH_2 tanks complicate cryogenic fluid storage and handling while increasing weight and complexity of the baseline ESAS lunar lander. Image courtesy of John Frassanito & Associates.

tank, where the energy can be efficiently removed via H_2 venting. The LH_2 tank cools the LO_2 tank to prevent O_2 boil-off. High performance bulkhead insulation, such as glass bubbles, is required to prevent too much heat from passing from the LO_2 to the LH_2.

With proper design, a light weight tank can reliably support more than six months mission duration with passive thermal control. The addition of active cooling, such as cryo coolers, to such a thermally efficient system can support indefinite mission durations.

B. Engines

There are two philosophies for selecting the number of engines for the lander. One approach is to use multiple engines in order to provide an engine-out capability. Typically at least three or four engines are required so that the loss of thrust from a single engine shutdown will not be too severe. Because lander configurations are typically squat, it is often challenging to arrange the engines such that their net thrust vector still points through the lander center of mass with one engine out. The other approach is to use a single engine. This is certainly simpler and less expensive, but it does not provide engine out capability. However, the inherent simplicity of a single engine system, with its associated simplification of valves, manifolds, and controls, provides a reliability benefit over multi-engine systems if the engine is itself sufficiently reliable. It is unclear whether a single engine or multi-engine system with engine out is more reliable. For the lander application described in this paper there is another advantage of a single engine. The landers described below ideally would have about 100–130 kN (22–30 klbf) of thrust. The existing RL10 engine is the right size for this application. Therefore, a single engine lander can use the existing RL10 engine and take advantage of the extensive flight heritage of this engine. A multi-engine system would likely require a new or RL10-derived smaller engine. The configurations presented in this paper use a single RL10 or RL10-class engine. These configurations would also work with multiple engines.

The most significant new functional requirement for a descent stage engine is deep throttle capability. The optimum thrust to weight ratio at the start of the descent burn is in the 0.35 to 0.5 range (using Earth weight). At touchdown, the thrust to weight ratio should be about 0.15 (thrust slightly less than weight in lunar gravity). Since the mass of the lander decreases due to propellant expended, the thrust required at touchdown is only about 10-15% of the ignition thrust, or a throttle ratio of 7:1 to 10:1. Throttling an engine this deeply has been demonstrated but requires additional development before flight. The first two lander concepts presented here use unusual configurations to avoid a requirement for deep throttle of the main engine. They can therefore use the RL10 with minimal modification (some minor changes for long duration and human rating may be needed). Concept 3 will require a deep throttling RL10-derived engine such as Pratt & Whitney Rocketdyne's Common Extensible Cryogenic Engine (CECE).

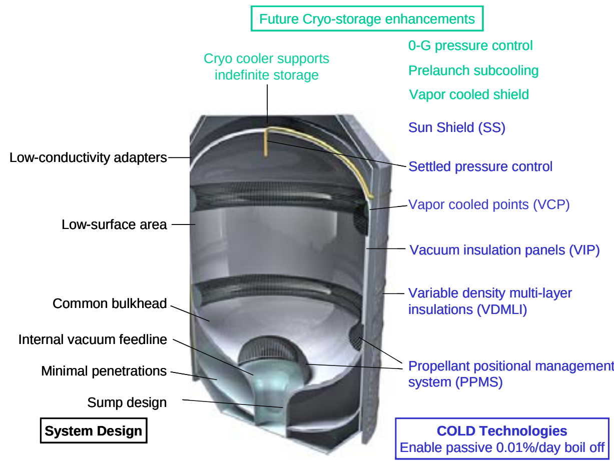

Figure 4. A combination of upfront tank design and cryogenic fluid storage technologies enable efficient passive LO_2/LH_2 storage supporting long duration missions.

Table 1. Venting GH_2 provides 2 to 10 times the thermal efficiency as venting GO_2.

Thermal attribute	LO_2	LH_2	Benefit
Heat of formation	205 kJ/kg (88 BTU/lb)	428 kJ/kg (184 BTU/lb)	H_2 2 times better than O_2
Change in enthalpy from liquid to 400 R gas	335 kJ/kg (140 BTU/lb)	3,256 kJ/kg (1,400 BTU/lb)	H_2 10 times better than O_2

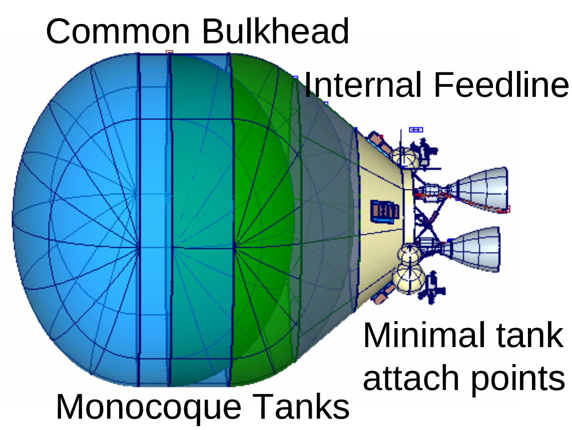

Figure 5. Using two thin walled monocoque tanks to store the LH_2 and LO_2 propellants results in a very light weight design that efficiently supports orbital cryo fluid management and long duration storage.

C. Crew And Cargo Access to Lunar Surface

Once the lander arrives on the Moon, it must support repeated crew access to the surface for EVAs, and enable loading and unloading of cargo. This sounds simple, but it is often quite difficult to configure a large cryogenic lander to do this well. Due to the size of the propellant tanks and engines, many lander configurations place the crew and cargo high above the surface. The lander designed for NASA's 1992 First Lunar Outpost study required the astronauts to climb 12 m (about the height of a four story building) between the crew cabin and the surface. The concepts in this paper were specifically configured to place the crew and cargo as close to the ground as practical. The first two configurations include an inflatable two-person airlock to facilitate EVA. The airlock can be detached to remain on the Moon so that the ascent stage does not carry its mass back to orbit.

The lander must accommodate three general sizes of cargo. Many experiments and supplies will come in small packages, approximately the size of a suitcase, some of which will be in the pressurized crew cabin, and some of which will be mounted to the outside of the lander. A few medium sized cargos, such as unpressurized lunar rovers, will also be carried on crewed missions. Ideally, the astronauts should be able to unload these from the lander while standing on the ground. Small and medium cargo is assumed to have a density of about 200 kg/m^3 (12 lbm/ft^3) based on examples of Apollo and International Space Station cargo items, and can be divided among multiple areas of the lander. The third category - very large cargos – includes items such as large pressurized modules for assembling a lunar base. These items must be landed on dedicated, uncrewed, cargo missions. The same lander system proposed for crew can accommodate these cargo-only missions by replacing the crew cabin with the large cargo element. Very large cargos are assumed to be unitary; meaning the total cargo mass can not be divided between multiple compartments. Cargo must be considered for lunar lander design in part because of the impact it has on Center of Gravity (CG) location. Configurations which require tight control on CG are not conducive to carrying cargo which will be loaded and unloaded, and will change from mission to mission.

D. Mobility

Lockheed Martin has identified three scenarios in which a mobile lander will be useful. During initial exploration sorties the crew will try to explore a wide area around the landing site. Complex target landing site regions, such as Aristarchus, are on the order of 100 km (60 miles) across and include dozens of individual science 'stations' at local features of interest. If the crew is limited to an unpressurized rover returning to a fixed lander at the end of each EVA, they will lack the range to reach many of the features. They will spend much of their time driving out and back over the same territory each day to reach nearby sites. A long range pressurized rover would be desirable but is very large, heavy and expensive, and would require a dedicated cargo launch. Adding limited mobility to the lander instead, in the form of wheels, suspension, and electric motors, provides substantial advantages. While the crew drives out to explore a nearby feature the lander can move slowly (on the order of 1 km per hour would be sufficient) in a direction that reduces the astronaut's return commute and brings the lander closer to the next day's science stations. Over the course of a 7-14 day mission the mobile lander could cover on the order of 100 km (60 miles). The astronauts would be able to explore many sites that would otherwise be unreachable, and would use their time more effectively. For this scenario the lander can have very slow mobility speeds and shallow slope limits, because the astronauts still use the small rover to actually approach specific sites. This enables low power draw, a simple suspension, and teleoperation from Earth, all of which make mobility easier to implement.

Lander mobility will also be valuable once a lunar base has been established. When the lander touches down it will kick up high-velocity dust and gravel. In order to protect the base, landings will happen hundreds of meters – perhaps a kilometer – away from the base itself. Unloading and transporting the crew and cargo over this 'last mile' turns out to be a significant challenge. It will likely drive requirements for additional rovers, trailers, and support fixtures. If instead the lander could be driven or towed from the landing point to the 'front door' of the base this would greatly simplify operations. The base will require dozens of lander missions during its lifetime. If each of these landers is left where it touched down, the base will soon be surrounded by discarded descent stages, each of which must be avoided by future landers. Instead, these stages will likely be collected in a 'boneyard' both to get them out of the way and to facilitate scavenging of parts. This process will be simplified if the lander can be driven or towed to a convenient storage location.

The first two landers presented here include wheels and are designed to be towed. Motors and appropriate power supply could be added if desired. In addition to wheels and motors, mobile rovers should also have low centers of gravity to prevent tipover.

E. Additional Ground Rules and Assumptions

The design ground rules and assumptions used in the generation of the following three concepts were derived from the ESAS study, NASA's 2006 Request for Information on Lunar Lander Concepts Studies[7], and internally generated data. Ground rules and assumptions include:

The lander carries 4 astronauts to the lunar surface. The lander is responsible for performing a 1100 m/s (3609 ft/s) Lunar Orbit Insertion (LOI) burn carrying a 20 metric ton CEV. The descent and ascent maneuvers are budgeted at 1911 and 1850 m/s (6270 and 6070 ft/s) respectively. The mission duration includes 30 days in LEO between launch of the lander and the subsequent launch of the CEV, followed by 5 days for the transfer to the Moon and initial orbital operations prior to landing. The active duration on the surface is 7 to 28 days, depending on the configuration. Some configurations can support longer dormant durations with support from a lunar base.

Pressurized free volume (i.e. excluding the volume taken up by subsystems) is sized based on a requirement for 3.4 m^3 (120 ft^3) per person, resulting in a total pressurized volume (including subsystems) of 28 m^3 (1000 ft^3). Landing gear are sized to provide a minimum 0.5 m (1.6 ft) clearance between the vehicle and the ground and a footpad to CG angle less than 49 degrees to prevent tipover.

IV. Concept 1: Dual Thrust Axis Lander

The Dual Thrust Axis lander uses an axial main engine to perform LOI and most of the descent burn, and then rotates until its long axis is parallel to the ground, and lands using a second set of smaller engines. In its landed orientation (Fig. 6), the large propellant tanks and engine are beside the crew cabin, rather than underneath it. This places the crew and cargo very close to the lunar surface for easy access and maintains a low center of gravity.

The key to Dual Thrust Axis landing is the recognition that the primary descent and the terminal landing propulsion have nearly diametrically opposite requirements. The primary descent system requires high thrust, low dry mass and high Isp to maximize landed payload. Deep throttling is not required for the primary descent phase. The landing phase requires a highly responsive multi-axis propulsion system with absolute maximum reliability but with inherently low, throttleable thrust. The landing phase can tolerate lower Isp with minimal system impact because the total impulse required is low compared to the initial deceleration phase.

The Concept 1 lunar lander implements this design approach using the RL10, with high energy LO_2/LH_2 propellants, to perform nearly all the descent propulsion task (Fig. 7). This leaves the lander at low velocity a few thousand feet above the lunar surface. When the RL10 shuts down the vehicle is oriented such that transition to the laterally-facing hypergolic landing thruster system is straight forward. The pressure-fed, throttleable lateral landing thrusters allow precision control of the descent and translation rates. Since nearly all the work of descent was performed using the high efficiency RL10 engines the system has a low gross weight. Even substantial hover and final descent durations using the lateral thrusters do not demand onerous propellant burdens.

The ability to rapidly maneuver is a clear advantage enabling selection of an optimal landing site. The distribution of lateral thrusters around the lander enables management of widely varying centroid locations which are inevitable from mission to mission. It also permits control over residual propellant slosh behaviors as the vehicle maneuvers.

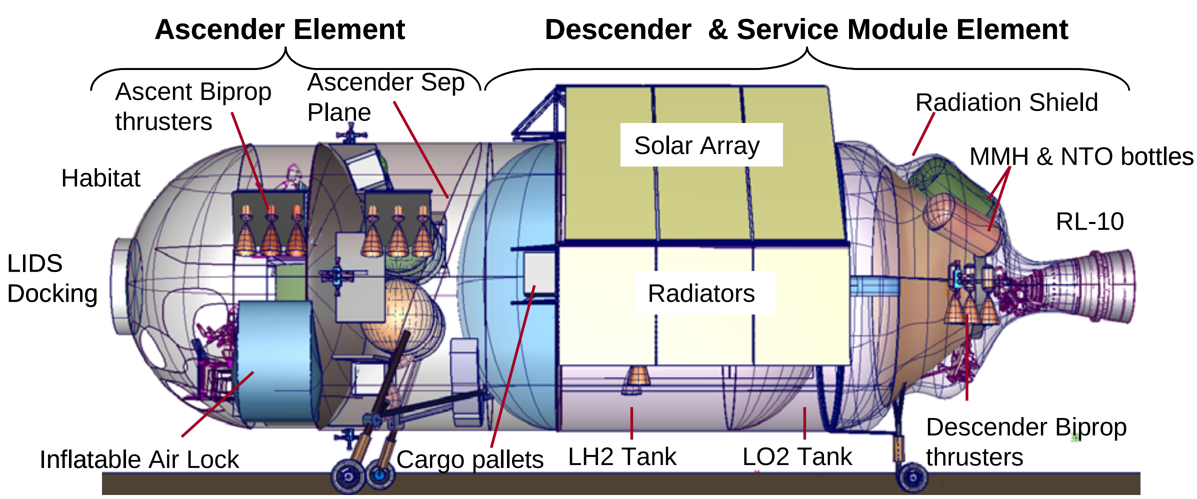

Figure 6. The Dual Thrust Axis lander satisfies the opposing objectives of locating crew and cargo close to the lunar surface with the requirement of large LH_2 tanks and a long, high expansion area nozzle

The loss of a single thruster has minimal impact on system behavior, allowing engine out and increasing system reliability.

A. Lunar Surface Access

Once on the lunar surface the crew can access the surface without ladders or other impediments. In similar fashion the unpressurized scientific cargo, stored on side mounted racks, is immediately accessible for deployment. Much of it is at chest level and each of the ten 200 kg (440 lbm) pallets can be lowered directly to the surface via simple mechanisms and without the crew engaging in elaborate deployment procedures. Concerns for work beneath suspended loads are minimized.

B. Ascender Crew Cabin and Ascent Vehicle

The Ascender (Fig. 8 and 9) provides a basic capability for extended crew operations on the lunar surface. It is 4.5 m (15 ft) in diameter with nearly 28 m^3 (1000 ft^3) habitable volume divided between the 7.5 m^2 (80 ft^2) lower flight deck area and the 8.5 m^2 (92 ft^2) upper deck area. The primary pressurized structure is composed of efficient axisymmetric elements that benefit from internal pressure stabilization. Behind the pressure compartment is a simple cylinder for supporting main terminal descent and ascent propellants, pressurants, avionics, and ECLSS hardware. The main thrust loads, distributed by the multiple thrusters, are efficiently reacted into the Ascender cylindrical elements tangent to the structure. The ascender interfaces to the Descender propulsion stage via a simple cylinder optimized for low thermal conductivity and weight.

To the extent practical, systems not required for ascent are mounted on the Descender stage. Until the ascent to lunar orbit, the Ascender receives all of its power from the Descender as well as breathing oxygen, water and cooling capacity. With the copious power available it is practical to pump down the airlock instead of simply venting the gas during each airlock cycle supporting a demanding lunar surface schedule. To replace atmospheric nitrogen, redundant Nitrous Oxide (N_2O) tanks provide up to 136 kg (300 lbm) of N_2O fluid which is catalytically reacted to form supplemental breathing air (oxygen and nitrogen) and also nitrogen to pressurize the N_2O_4 propellant tanks. Hundreds of airlock cycles can be accommodated with onboard stores of O_2 and N_2O.

To return to lunar orbit after the surface mission the Ascender propellant tanks are brought to pressure by the onboard GH_2 and N_2O pressurization systems (Fig. 10). With Descender systems stowed and umbilicals retracted the Ascender thrusters are brought to 35% power to achieve positive upload at the separation interface. Commanding separation, the Ascender can then come to hover and translate away from the Descender before applying full power for ascent. In this way the Descender can be preserved without damage for potential future use.

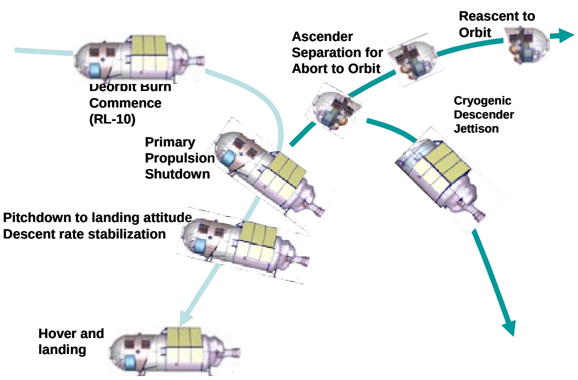

Figure 7. The proposed design lends itself to efficient nominal landing trajectories, while protecting the ability to accommodate mission aborts.

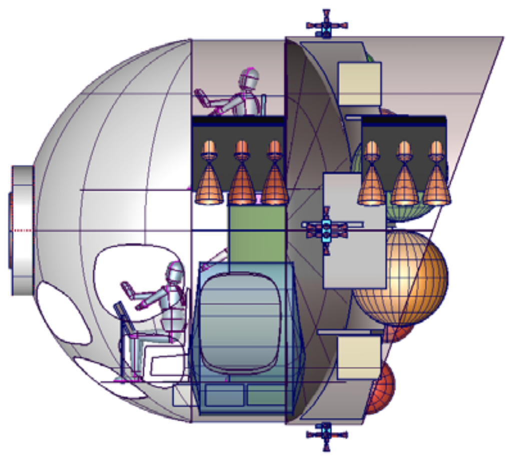

Figure 8. The Ascender integrates the crew cabin and ascent propulsion into a compact, efficient structure.

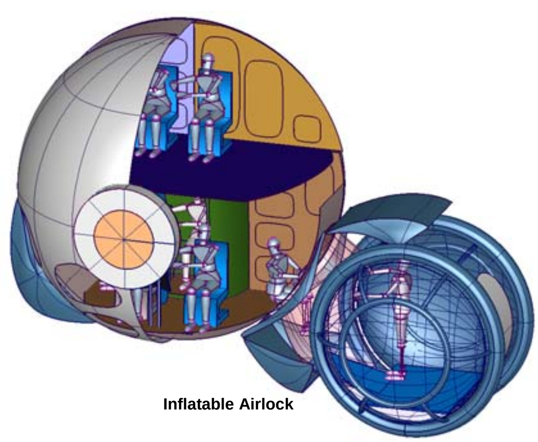

Figure 9. The crew cabin accommodates four astronauts on the lunar surface and includes a large inflatable airlock at ground level.

The arrangement of propellants and thrusters on the Ascender minimize CG movement and permit widely varying amounts of residuals or up-cargo.

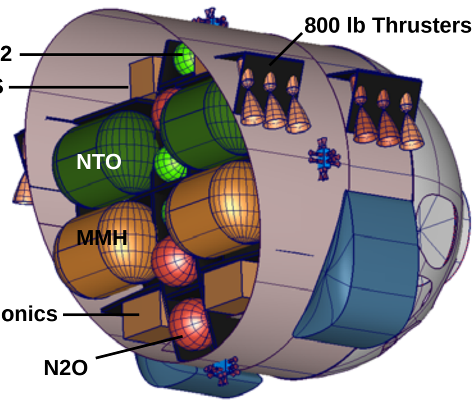

Figure 10. The terminal landing and ascent propellants are packaged on the back end of the ascender along with external equipment such as ECLSS and some avionics boxes.

C. Descender Propulsion, Power, and ECLSS Systems

The Descender stage includes both the LO_2/LH_2 primary descent and the MMH/N_2O_4 lateral touchdown propulsion systems. The Descender stage also acts as a service module to the Ascender stage, providing power and life support commodities. The primary propellant tanks can store large quantities of O_2 and H_2, which supply fuel cells, provide oxygen and water for ECLSS requirements, and provide cooling capability. Combining the surface O_2/H_2 requirements with the propulsive cryogenics also allows the mission designers more flexibility in assigning margin for both the propulsive and the ground phases of a lunar mission. The primary propulsion system will leave hundreds of kilograms of unusable outage residuals and performance reserve propellants on most missions which can be used during surface operations.

Power is generated by an integrated solar and fuel cell system. The solar arrays provide 10 kW of power under peak solar illumination. The fuel cell system is sized to provide 8 kW continuous power for 14 days so that the mission can be extended beyond the lunar daytime to encompass a complete lunar day/night cycle. An integrated thermal rejection system can reject these high power levels even at solar noon and can be modulated to match the actual power consumption and local environment.

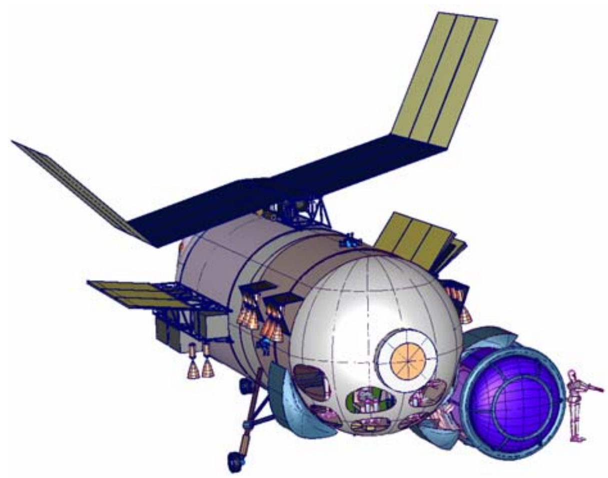

Figure 11. The solar arrays are designed to gimbal to maximize power even at low sun elevations while also shading the cryogenic propellant tanks and radiators to enhance storage duration and heat dissipation.

The horizontal lander configuration is optimal for the simple deployment and subsequent orientation of the solar panels. During flight the solar panels and radiators are covered with low-emissivity optical covers to minimize heating during orbital phases of flight, when power demand is low, and to guard the panels from landing generated dust (Fig. 6). Once on the ground the optical covers swing back and the entire solar panel/optical cover assembly can be pivoted to face the sun (Fig. 11). The panel/cover angles can be optimized to increase power generation by increasing incident light or to reduce vehicle heating as required.

Once the solar power system is enabled the heat rejection system can be similarly deployed. Each double-sided radiator panel is anchored to structure by a simple hinge and is sandwiched by two low-emissivity optical reflectors/shields. This arrangement allows heat to be efficiently rejected to space with minimal incident radiation from the surrounding terrain or the vehicle itself. Six radiator panels, three on each side of the vehicle, allow not only ample dissipation capability (roughly 26 m^2 effective area) but also the ability to modulate the heat rejection as power loads and the local environment change.

The fuel cell system is redundant with each unit having independent ambient temperature GH_2 and GO_2 reactant tanks. These tanks are launched full and are used during RL10 operation for main tank prestart pressurization as well as to command various vehicle pneumatic valves. This eliminates the need for helium on the vehicle and hence contamination on the vehicle propellants with inert gas. These warm gases are also used as pressurants for the aft bi-propellant propulsion system during terminal descent. Control valves allow cryogenic ullage gasses or liquids stored as residuals in the main propellant tanks to refill these ambient vessels as required. Potable water is stored in each of the three modules with a total capacity of 136 kg (300 lbm) - roughly 1.5 days of production at 50% efficiency. High pressure gaseous oxygen is also delivered to the Ascender to replenish breathing air and to permit reloading of portable life support systems.

V. Concept 2: Retro-Propulsion Lander

Concept 2 is similar to Concept 1 in that a large RL10-powered LO_2/LH_2 propulsion system performs LOI and primary descent but not the touchdown maneuver. The primary difference is that the cryogenic propulsion system is shut down and jettisoned several thousand feet above the surface. The remaining Crew Module stage of the lander performs final descent and touchdown as well as ascent using a pressure-fed N_2O_4/MMH propulsion system with multiple small engines (Fig. 12). In this regard, Concept 2 is similar to the scheme used for the Surveyor landers of the 1960's. The Crew Module stage combines the functions of approach and landing, ascent propulsion, and the crew cabin. Utilities such as power and life support are integrated on the Crew Module stage rather than the cryo propulsion element as in Concept 1 (Fig. 13). If the lander is visiting a surface outpost, the release of the Retro stage must be arranged so that the impact of the stage does not threaten the outpost. Trajectory simulations indicate that jettison should occur either when about 25% of the descent delta V remains to be performed so that the impact point will be far away from the landing site, or near the very end of descent during the final approach when the dispersion in the impact location will be small and the retro stage can impact in the landing zone.

Concept 2 shares several advantages with Concept 1. The cryogenic propulsion stage is designed with the same features for structural efficiency and cryogenic fluid management. It does not require a deep-throttling engine, and it puts the crew and/or cargo in close proximity to the ground.

Jettisoning the cryogenic propulsion stage provides several advantages compared to Concept 1. The landed stage is much smaller. This eases the addition of mobility features (i.e. wheels, suspension, and electric motors) which enable the crew module to move from the landing site either to multiple science sites or to a parking spot near a surface habitat. The Retro Propulsion Stage does not carry hardware for landing or surface operations, making it a more generic large propulsion stage, which could also be used for other high-energy missions. Concept 2 as shown here lands horizontally. However the approach is also attractive for a vertical landing because the descent propulsion system will not be in the way on the surface. The vertical landing approach maintains the same thrust axis during retro burn, landing, and ascent, reducing concerns associated with rotational maneuvers, propellant slosh, and center-of-gravity management which may complicate the Dual Thrust Axis concept.

Concept 2 has some disadvantages as well. It requires a separation event and propulsion system start event during final descent. The failure of either of these could be catastrophic. Because the same propulsion system is used for touchdown and for ascent, there is a risk that the propulsion system could be damaged on landing, either from scattered debris, or from contact with the surface as occurred on Apollo 15, and hinder a safe ascent. These risks are mitigated

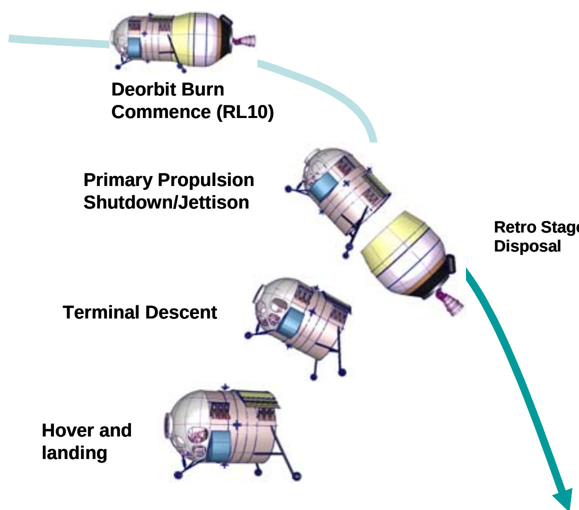

Figure 12. Concept 2 is similar to Concept 1, but jettisons the large cryogenic propulsion stage following RL10 shutdown, prior to actual landing on the lunar surface in order to minimize the soft landed mass.

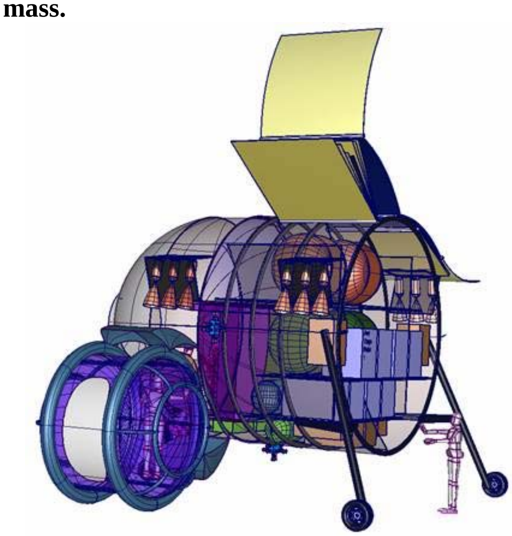

Figure 13. Concept 2 integrates all of the power, ECLSS and propulsion onto the ascent stage.

by using several small engines which are highly reliable because they are pressure fed and use hypergolic propellants. The engines can be placed relatively far above the ground contact plane of the lander because they are sized only for touchdown and ascent thrust, and are therefore much smaller than a full descent engine.

Concept 2 suffers a mass penalty because the Crew Module carries back up to orbit many components of the power, thermal management, and life support systems which were left behind on the surface in Concept 1. It gains a small mass advantage over Concept 1 because the touchdown propulsion and landing gear do not have to be sized to accommodate the large retro propulsion system. However, because the retro propulsion tanks are jettisoned rather than landed intact, Concept 2 does not benefit from the use of residual LO_2 and LH_2 for power generation or thermal management during the surface mission. For this reason the Concept 2 lander is configured for a 7 day surface stay duration, not a longer mission.

VI. Concept 3: Single Stage Lander

Concept 3 takes advantage of the high Isp of LO_2/LH_2 propellants to enable a single stage lander to perform LOI, descent, landing, and ascent to lunar orbit. Unlike the previous concepts, concept 3 uses a single RL10-class engine to perform touchdown and ascent (Fig. 14). This requires deep throttle capability for the main engine. The CECE engine, with its 10:1 throttle capability, is baselined for this concept.

In the past, most lunar lander concepts have been two-stage systems, following the tradition of Apollo. However, Lockheed Martin has concluded that a single stage cryogenic lander would probably be less expensive and not much more massive – in fact a single stage lander can even be lighter than a two-stage lander. Many engineers would instinctively be skeptical of a single stage system because on Earth it has been very difficult to design Single Stage to Orbit (SSTO) launch vehicles with high enough propellant mass fraction. However, even if the lander is responsible for the LOI burn, the lander mission requires only about half the ΔV of an Earth to orbit launch, so that both the allowable mass fraction and the sensitivity to errors in mass fraction are much lower than for a launch vehicle. Building a single stage lander is fundamentally no more difficult than a two-stage system. Conventional staging theory says that, assuming constant mass fractions, dividing a rocket vehicle into multiple stages reduces total mass and therefore a two stage system would still be preferable. But, in practice staging requires additional interfaces and duplication of hardware (such as an engine for each stage rather than a single engine). Detailed subsystem-level modeling of the lander shows that single stage designs can be less massive than a comparable two stage design depending on requirements. Since the single stage lander will use LOX/LH_2 for both descent and ascent, it is particularly competitive against two stage landers which use lower Isp propellants for the ascent stage.

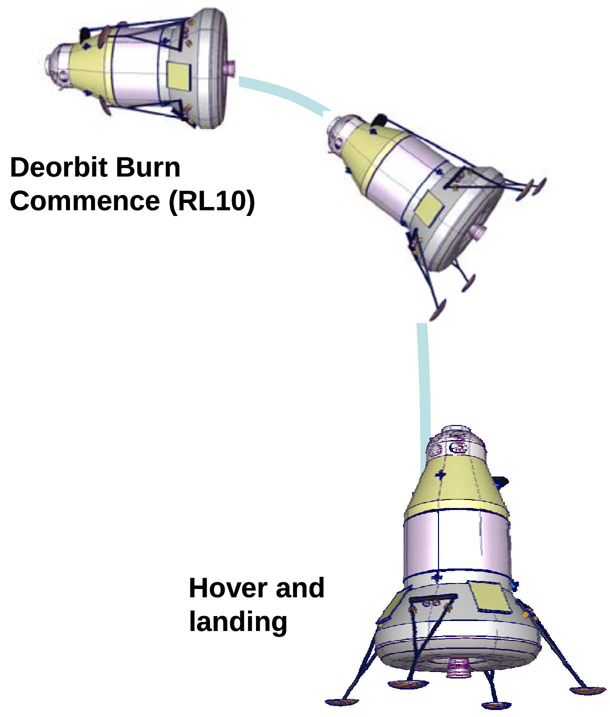

Figure 14. Concept 3 uses deep throttling of the RL10 land on the lunar surface.

Cost modeling indicates that the single stage lander could be less expensive than a multi-stage design due to reduced mass, parts count, and complexity. Substantial cost savings would result from developing a single new engine and propulsion system, rather than two different propulsion systems. The single stage system is attractive for its evolutionary potential. Unlike a two stage system which leaves hardware behind, it is relatively straightforward to adapt the single stage lander into a fully reusable system which would be refueled initially in orbit with propellant brought from Earth. Eventually, the system could take advantage of in-situ production of LO_2 and/or LH_2. The cost savings of reusability and ISRU may enable more extensive exploration and utilization of the Moon than would otherwise be possible. Multi-stage systems, especially those which use storable ascent propulsion, are not easily adapted for reusability, or to take advantage of ISRU. A few reusable, single-stage landers could support many missions to a lunar base without leaving dozens of discarded stages scattered around the facility.

The single stage system does have several drawbacks. In a two-stage system, the ascent stage provides a crew escape/abort capability back to orbit during the descent phase. A single stage vehicle does not have this capability. During ascent, neither the single nor two stage concepts have an escape capability. The Concept 3 configuration shown here has a small crew cabin on top of the vehicle. Since it would be difficult for the crew to climb up and down for repeated EVAs, this design is best suited for outpost support missions so that the crew would not live for long periods of time in the lander cabin.

The Concept 3 lander has plenty of room for small and medium cargo around the base of the vehicle (Fig 15) where it would be easy to unload. However, this concept is not as well suited for delivering habitats and other large elements. The main engine limits the ability to place such elements close to the ground, and unloading large, heavy payloads from the top of a large cryogenic propulsion system would require cranes.

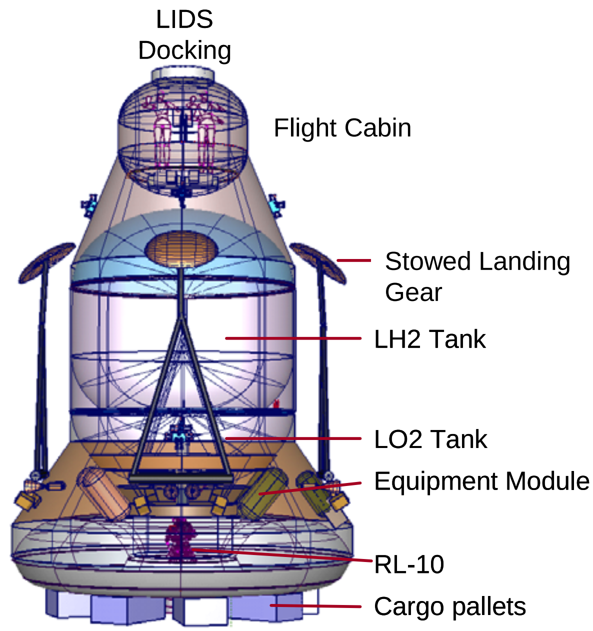

Figure 15. The Concept 3 Lander has a crew cabin on top, wide, common bulkhead tanks in the middle, and the engine and deployable cargo bay on the bottom.

Because the single stage concept will have the cryogenic tanks during ground operations, it offers similar fluid integration benefits as Concept 1. The cryogenic storage for a single stage vehicle presents a significant challenge for long duration missions due to the need to retain the LO_2/LH_2 propellants for ascent. The single stage system will require better thermal management and may be limited to shorter mission durations than a two stage system with storable ascent propellants. The cost and performance advantages of the single stage lander must be weighed against its drawbacks.

VII. Conclusion

The use of high energy LO_2/LH_2 propellants has significant benefits for the lunar lander. Minimum mass, practical heat leak and cryogenic fluid management issues drive the system design to two tanks to store the two propellants. For operational effectiveness, landers should also be designed for easy crew and cargo transfer to the ground, and for limited mobility. The three lander concepts presented in this paper address these requirements with novel solutions. Each of the concepts have different advantages and disadvantages, which make them better suited to specific applications or design priorities.

1) The Concept 1 Dual Thrust Axis Lander places the crew and cargo close to the ground and provides a built in service module with efficient storage of O_2 and H_2 for breathing, water and fuel cell reactants. Although the use of an auxiliary propulsion system requires a transition between propulsion systems prior to landing, the use of pressure fed hypergolic propellants should ensure a reliable transition. Concept 1 trades the added complexity of an additional propulsion system to avoid the requirement for deep throttling of the large cryogenic engine. The Dual Thrust Axis landing also places the large RL10 nozzle away from the lunar surface, avoiding one of the key geometric issues of a cryogenic vertical lander. The large service module can readily accommodate large solar arrays which provide double duty as sun shields for the cryogenic tanks and radiators. The available solar power and O_2 and H_2 commodities make this concept attractive for longer mission durations, in this case, up to 28 days. With its low profile, the Dual Thrust Axis landing concept can readily deliver large payloads to the lunar surface.

2) The Concept 2 Retro-Propulsion Lander jettisons its descent stage prior to touchdown, resulting in a very compact lander with easy surface access. This approach is attractive for a mobile lander, or if a vertical landing is desired to avoid the rotation of required for Concept 1. Because this configuration does not retain substantial cryogenic reactants for fuel cells, it is best suited to daylight-only missions using solar arrays, or access to an emplaced base

3) The Concept 3 Single Stage Lander is potentially the least expensive configuration. It provides a ready path to a fully reusable architecture, particularly benefiting from in-situ propellant production. However, it has fewer abort options than a two stage lander and requires better thermal management to retain cryogenic propellants for ascent

propulsion. Similar to concept 1, retention of the cryogenic stage allows synergy between the cryogenic propulsion system and the power and ECLS systems, allowing long duration lunar stays, even through the lunar night. One major disadvantage of concept 3 is that it is not well suited to the robotic delivery of large structures such as habitats.

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