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Paper Session I-B - Conceptual Design of a Cargo Lander for the First Lunar Outpost

Michael J. Wright
University of Minnesota, NASA Marshall Space Flight Center Summer Fellow

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1.0 Introduction

The Space Exploration Initiative (SEI) encompasses a series of exploratory and colonization missions that will expand human presence to the Moon and Mars via an evolutionary program, whereby each succeeding phase builds on and compliments the previous efforts. The first phase in this initiative, as defined by president Bush, is to "return to the Moon, this time to stay". In order to accomplish this goal by the turn of the century the First Lunar Outpost (FLO) strategy has been proposed. In support of this mission scenario, NASA Marshall Space Flight Center (MSFC) has designed a habitation module and cargo lander concept which offers significant advantages over the current baseline design. This paper will briefly present the Marshall design, emphasizing the cargo lander.

1.1 FLO Guidelines and Assumptions

While it is beyond the scope of this paper to go into detail about the FLO mission scenario, the brief overview which follows will serve to highlight the objectives of the mission, particularly those that pertain directly to the Lunar lander.

The baseline FLO mission will land a self contained habitation module at a predetermined location on the moon. A piloted mission will follow about 60 days after the successful cargo landing and set up, carrying a crew of four that will remain on the surface for 45 days (two Lunar days and one night), carrying out a wide variety of astronomy, physics, and in-situ resource development experiments before returning to Earth. This mission scenario, which will increase our knowledge of our sister planet immensely and will also serve as a testing ground for planned future Mars missions, can be repeated as many times as desired simply by sending another habitation module and piloted follow-up mission.

The habitation module must be capable of supporting a crew of four for 45 days, and must provide power to the cargo lander from Trans-Lunar Injection (TLI) until touchdown on the Lunar surface. The module must also be capable of powering down and remaining in a dormant state for at least 60 days, and possibly up to six months, before the arrival of the piloted mission. In addition, the hab module will provide a navigation beacon for the piloted mission.

The descent stage will utilize a common design for both the cargo (habitation module) and piloted missions, and must be capable of landing 27.5 t on the Lunar surface in all configurations. The lander must also be capable of performing a fully autonomous landing at any predetermined location on the moon, with the option of pilot override, and must provide a landing accuracy of $\pm$ 2 km for the cargo flight and $\pm$ 100 m for piloted flights (with the aid of the navigation beacon). The piloted lander will land approximately 1 km from the habitat.

For a complete list of FLO requirements, assumptions, and guidelines, consult the current version of the First Lunar Outpost Requirements and Guidelines (FLORG) document maintained by the Exploration Program Office (ExPO) at NASA/JSC. The version used for the lander design in this paper was dated June 10, 1992.

1.2 Cargo Mission Description

For the cargo mission, the transit between the Earth and Moon will take about 4 days, including about 4 1/2 hours spent phasing in Low Earth orbit (LEO) prior to the TLI burn. Upon arrival at the Moon, the lander performs the Lunar orbit insertion (LOI) burn, placing it into a 100 km circular parking orbit. Once the lander has reached a stable parking orbit, a maximum of 2 days are allowed for proper orbital alignment with the landing site, at which time the lander will de-orbit, coast to an altitude of 18.5, km and begin powered descent to the Lunar surface. A summary of the ΔV's performed by the lander main propulsion system is presented below in table 1.1:

Maneuver	Delta V (m/s)	Delta V (ft/s)
Mid-course corrections	30	98.4
LOI burn	882	2893.7
De-orbit burn	20	65.6
Descent	1878	6161.4
Totals	2810	9219.1

Table 1.1 Main Propulsion ΔV's

As per mission requirements, the descent will be totally autonomous and the vehicle will land within 2 km of the predetermined location. After touchdown, the only purpose of the descent stage is to provide a support structure for the habitation module.

2.0 System Layout

Figure 2.1 (following page) depicts the Marshall FLO Lunar lander concept. This lander was designed to accommodate the alternate habitation module concept, shown in the flight configuration with the lander in figure 2.2. The alternate habitat module offers several design advantages over the current baseline, including improved access to the Lunar surface, improved load paths, lower cg location, and an improved interface between lander and habitat. This design provides approximately 1.0 m maximum stroke length for pad penetration and impact attenuation while maintaining a minimum ground clearance of 1.9 m for the airlock.

One of the principle advantages of this design is improved accessibility to the Lunar surface. Since the center hole of the lander truss is kept free of other components (figure 2.1), the habitat module can be designed vertically to take advantage of this space. With this design, the airlock will be only 1.9 m from the Lunar surface, which will greatly simplify ingress/egress procedures. In addition, the design allows for in-line loading of cargo or injured crew members with the help of a simple pulley system.

Another advantage of this design over previous efforts is the folding landing gear. When deployed the legs have a 20 m diagonal spread, but during launch they can be stowed in a manner that will minimize the total "on the pad" stack length of the launch vehicle. In launch configuration the entire vehicle will fit inside a cylinder 10 m in diameter by 11.2 m long, which is consistent with the standard 10 m HLLV shroud as depicted in figure 2.3.

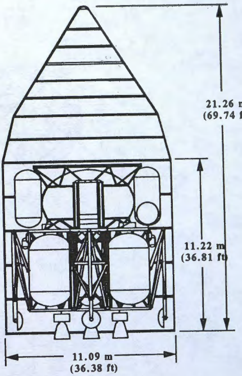


Figure 2.3 Launch Configuration

While this study concentrates mainly on the lander for the cargo and habitation missions, the same lander will also carry the ascent and Earth return stage for piloted missions.

3.0 Lander Total Mass Statement

Table 3.1 (following page) presents the total vehicle mass statement for the Marshall FLO Lunar lander concept, including an engineering contingency of 15%. For a more detailed breakdown of component masses, consult the subsystem mass table located in each section of this document.

Figure 2.1 Cargo Lander Dimensions

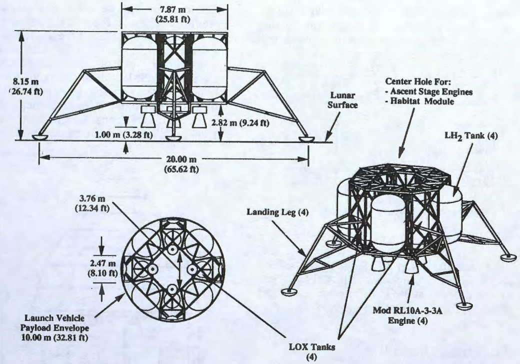
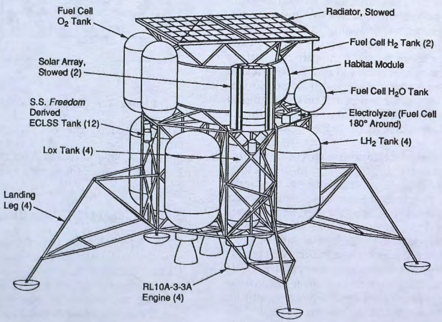


Figure 2.2 Lander/Hab Flight Configuration



Subsystem	Mass (kg)	(lbs)
Structures	4633	10214
Propellant Tanks	1782	3929
Propulsion System	1226	2703
Reaction Control System	140	309
Power Distribution System	342	754
Avionics Systems	282	622
Thermal Control System	522	1151
Contingency (15%)	1339	2952
Total Dry Mass	10266	22634
Propellant Load	39173	86376
Total Vehicle Mass	49439	109010
Payload (Hab Module)	34200	75411
Total System Mass	83639	184421

Table 3.1 Total Vehicle Mass Statement

4.0 Structural Analysis

4.1 Groundrules and Assumptions

There were several guidelines which governed the structural design of the Lunar lander. First, the vehicle was designed for a maximum deceleration during descent of 3 g's, and touchdown velocities not to exceed 1 m/s vertical and 0.5 m/s horizontal. In addition, the worst case Lunar surface conditions were assumed to be an effective slope of 10°, with one leg encountering a 61 cm depression at the bottom of the slope. Finally, a safety factor of 1.4 for all structural components and pressure vessels was included in the design.

All of the lander structural components were designed using Martin Marietta's lightweight Weldalite-049 Al-Li alloy, except the tank support struts, which were designed using G-10 fiberglass/epoxy to minimize heat transfer. The Weldalite alloy was chosen primarily because of its strength, low density, and excellent mechanical properties at cryogenic temperatures. The masses of the thrust structure, propellant tanks, and tank support struts were driven by the launch loads, which were calculated for a representative HLLV based on estimated quasi-static launch and random vibration accelerations. The landing gear mass, on the other hand, was driven by the forces encountered during landing as well as the required crushing strength of the shock absorbing material in the struts.

4.2 Propellant Tanks and Struts

The propellant tanks for both LOX and LH_2 were sized using thin shell formulas, assuming cylindrical design with elliptical bulkheads. In both cases the required thickness of Weldalite was less than the minimum gage thickness for the material, so minimum gauge was used Meteoroid/debris shielding requirements were investigated, but it was determined that no shielding was necessary, due primarily to the short exposure time of the lander to orbital debris. A summary of the tank design features is presented below in table 4.1:

	LOX tanks (4)	LH2 tanks (4)
H, cm (in)	249 (98)	191 (75)
D, cm (in)	170 (67)	282 (111)
Thickness	0.102 (0.04)	0.102 (0.04)
Ullage, %	3%	3%
Vol, m^3 (ft^3)	7.5 (264.5)	20.2 (713.6)
Mass, kg (lb)	81.9 (180.6)	152.1 (335.4)
Rings, Stiffeners	33.6 (74.0)	62.4 (137.5)
Growth, %	35%	35%
Total Mass	155.9 (343.7)	289.6 (638.4)

Table 4.1 Propellant Tank Characteristics

Finally, tank support struts were sized by calculating buckling loads and axial stresses. As stated above, the struts are constructed of G-10 fiberglass/epoxy to minimize heat transfer to the propellant. There are a total of 8 struts per tank, with a total mass of 342 kg for the LOX and 125 kg for the LH_2 struts.

4.3 Thrust Structure

The thrust structure was analyzed using a NASTRAN model, resulting in a total mass of 3205 kg, including necessary fittings and a 35% growth factor. The final design consists of 384 members constructed of Weldalite alloy round tubing. The main support members are 15.24 cm OD and all other members are 10.16 cm OD. The structure was designed to provide extra support for the heavier LOX tanks while maintaining overall symmetry and clear load paths.

4.4 Landing Gear

A preliminary stability analysis of the cargo lander under the most extreme Lunar surface conditions (10° slope) was performed, using data from the Apollo Lunar Excursion Module (LEM) under similar conditions. This analysis results in a maximum allowable cg height of 8.15 m for the current landing gear diagonal spread of 20 m, assuming a 2-2 landing configuration, which was chosen as a worst case scenario. While the cg has not yet been accurately calculated, preliminary computations show that the cg height will be somewhat less than 8.15 m. Further stability analysis will be done to optimize the landing gear spread once the cg has been determined with greater accuracy.

The landing gear was sized by calculating maximum forces in the struts at impact, using the groundrules stated in section 4.1. Where necessary, Apollo LEM design data was scaled to meet the FLO requirements. The primary struts (1 and 3 in figure 4.1 below) are constructed of 20.3 cm diameter Weldalite hollow tubing, while all secondary struts are 15.25 cm in diameter. In addition, the three shaded members contain Hexcel crushable honeycomb impact attenuation material to help lessen landing loads. The footpads will be similar to those used on the Apollo LEM, scaled to handle the higher landing mass.

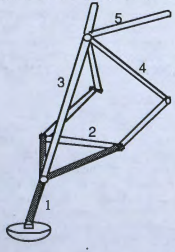


Figure 4.1 Landing Gear Design

4.5 Structures Mass Summary

Table 4.2 below summarizes the masses for the various lander structural components:

Component	Mass (kg)	(lbs)
LH2 Tanks	1158	2553
LOX Tanks	624	1376
Tank Struts	467	1030
Thrust Structure	3205	7066
Landing Gear	961	2118
Total System Mass	6415	14143

Table 4.2 Structures System Masses

5.0 Propulsion Subsystem

5.1 Main Propulsion

A trade study was performed between several possible engine designs for the main descent propulsion system, including the RL10 derivative family, the Integrated Modular Engine concept (IME), and the Advanced Space Engine (ASE). The principle drivers of this trade study were cost, reliability, performance, and the desire to use existing hardware where practical.

The ASE was eliminated after the initial performance screening as it was felt that the additional performance gained by use of this advanced engine concept was not required for this lander and therefore the initial DDT&E cost required to implement this design was not warranted. The IME offered many benefits over the other options, including improved performance, engine out capability, increased throttlability, and attitude control. However, while it is recommended that future FLO mission studies continue to explore the possible use of IME engines, the higher development risk and front end cost associated with this design caused us to turn to the RL10 series. This engine family offers several advantages over the other types considered, most notably the extensive flight experience and reliability testing during its 34 years of use on the Atlas/Centaur launch vehicle and the continuing research underway at Pratt & Whitney to improve the design.

The engine model baselined for this study was the RL10A-3-3A, which is a 16.5 klb (73.4 kN) thrust engine currently in use on the Atlas/Centaur upper stage; however, it is recommended that a LOX/Methane RL10 derivative be considered in the future as a possible evolutionary engine to make use of in-situ propellant production on Mars. To meet stated mission requirements, the RL10A-3-3A engine must be modified to provide 5:1 throttling capability for Lunar descent, which is significantly higher than the 3:1 throttlable engine which will undergo preliminary testing in 1994 on a McDonnell Douglas DCX. In addition, the engine must have a gimbal capability of ± 6° to perform terminal descent maneuvers. While the current design has a maximum gimbal angle of only ± 4°, it is felt that a 6° capability could be achieved with minimal modifications. Table 5.1 below presents the specifications of this modified RL10A-3-3A engine.

Thrust vac, klbf (kN)	16.5 (73.4)
Propellants	LOX/LH_2
Isp, vac (sec)	444
Max Throttle Ratio	5:1
Mixture Ratio	6:1
Expansion Ratio	61:1
PC, psia (MPa)	465 (3.21)
Mass, lb (kg)	365 (166)
Exit Dia, in (cm)	40 (102)
Length, in (cm)	70 (178)
Gimbal Angle (deg)	± 6°
Thrust to Weight	5.41

Table 5.1 Modified RL10A-3-3A

Four of these modified RL10A-3-3A engines are required, providing a maximum thrust of 66,000 lbf (293.6 kN) at full throttle. The eight propellant tanks (4 LH_2 and 4 LOX) supply the main engines with LOX and LH_2 at a mixture ratio of 6:1 (by mass). All feed line valves are electromechanically actuated and are designed in parallel to provide fail-op/fail-safe (FO/FS) operation. The design of the RL10 lends itself readily to autogenous pressurization of the tanks, but a small amount of gaseous helium (GHe) is still required for prepress as well as system purges before restarting the engines. A preliminary breakdown of the total system mass is shown in table 5.2 below:

Component	Mass (kg)	
Engine System	954.4	
Mod RL10A-3-3A (4)		662.3
Hardware		292.1
Feed System	85.0	
Feed Lines		16.0
Gimbal Joints, Valves		69.0
Tank Pressurization	186.8	
Valves, Regs (44)		48.4
Helium Bottles (3)		114.0
Support/Install (15%)		24.4
Total System Mass	1226.1	

Table 5.2 Main Propulsion Mass Statement

5.2 RCS Propulsion System

A bipropellant N_2O_4/MMH system was chosen over hydrazine and GOX/GH_2 mainly for commonality with current technology used on the space shuttle orbiter. Future work will explore more fully the advantages of a GOX/GH_2 system, which would provide the highest Isp as well as commonality with the main propulsion system, at the cost of higher complexity.

The RCS system will consist of 16 25 lbf thrusters, which are similar in size and thrust to the vernier thrusters currently used on the shuttle orbiter. The fuel and oxidizer are stored in 61 cm diameter (including a 5% ullage volume) spherical titanium tanks, which are pressurized to 2.5 MPa with GHe, which is stored in a spherical tank at an initial pressure of 25 MPa. A relief valve and burst disk is located in each pressurant line to prevent overpressurization. This system provides FO/FS redundancy through a combination of parallel valve paths and redundant thruster pairs. Total system mass is presented below in table 5.4:

Component	Mass (kg)	Mass (lb)
Thrusters (16)	25.4	56.0
Propellant Tanks (2)	23.2	51.2
Other Systems	73.1	161.1
Lines, Support (15%)	18.3	40.2
Total System Mass	140.0	308.5

Table 5.4 RCS Mass Statement

6.0 Avionics and Power

6.1 Guidance, Navigation & Control

The listed GN&C equipment for the lander (table 6.1) meets all mission requirements and provides FO/FO/FS redundancy for all mission critical components. Included is a Hexad inertial navigation system (INS) with internal ring laser gyroscopes, inertial measurement units (IMU's), and accelerometers. This system provides internal FO/FO/FS redundancy while replacing multiple components with a single unit. Also included are three Celestial Sensor Assembly star trackers with sun shields and four standard precision sun sensors, which are distributed symmetrically around the lander to allow for continuous viewing.

Component	Power (W)	Mass (kg)	Mass (lb)
Hexad INS (1)	100	22.7	50.0
Sun Sensors (4)	10	7.0	15.4
Star Trackers (3)	30	13.0	28.7
Landing Radar (1)	123	38.1	83.9
Video Cameras (2)	20	14.0	30.9
Control Electronics (1)	100	50.0	110.3
Totals	383	144.8	319.2

Table 6.1 GN&C Equipment List

Landing equipment includes two video cameras and Apollo based landing radar. The video cameras send slow frame rate data back to Earth for possible revectoring and could also be used in the future to provide images to an on-board scene recognition system which, when coupled with an expert system, would provide a true autonomous landing capability. These video cameras are also used by the C&DH subsystem for landing site inspection and visual verification of problems as discussed the section 6.2.

The RCS system consists of 8 thruster pairs spaced equally around the base of the lander thrust structure, under the lower attachment points of the landing gear (figure 6.1). This arrangement will help to insure that the exhaust plumes from the thrusters do not impinge on the lander structure or propellant tanks. This system will provide 2 and 3-axis attitude and thrust vector control during LOI, time spent in parking orbit, and descent.

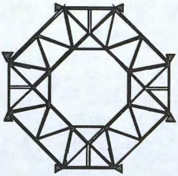


Figure 6.1 RCS Thruster Placement

Analysis indicates that about 254 kg of propellant will be required to perform the necessary maneuvers, including a 15% contingency and a 2% residual.

6.2 Communications & Data Handling

The communication system must be able to provide a data link between the lander and Earth continuously during all post TLI mission phases, and provide for the acquisition and transmission of engineering data at a rate of 200 Kbps. All major data links will use Ka-band frequency to provide commonality with future Mars missions, and will take advantage of the capabilities of the existing Deep Space Network (DSN). In addition, the lander will also require a low gain X-band antenna and transponder, which will be used for the acquisition and transmission of tracking data.

The lander will also include a video system consisting of two color TV cameras with pan/tilt units and a video recorder, which will be used to relay images during landing, provide images of the chosen landing site, and provide visual confirmation of problems that develop during any phase of the mission. The addition of such necessary equipment as flight processors, data storage units, and remote voter units brings the total mass of the C&DH subsystem to 137.4 kg.

6.3 Electrical Power

The electrical power system (EPS) must provide continuous operational power to the lander from insertion into LEO until touchdown on the Lunar surface. Power is assumed to be generated by the habitat module fuel cells and distributed to the various lander subsystems as required. Total peak power requirements for the various subsystems are estimated at 2 kW, including a 30% contingency factor to handle future subsystem growth and ensure commonality with the piloted mission.

The power generation system consists of three hydrogen-oxygen fuel cells which are resident on the habitat module. These fuel cells will be capable of extended operation using propellant grade reactants; therefore it is assumed that propellant grade fuels supplemented by cryogenic boiloff will be used to power the fuel cells until touchdown on the Lunar surface. Between 100 and 150 kg of water will be produced by the fuel cells during steady state operation, which can either be stored in tanks for use on the surface or vented to space as it is produced.

The electrical power distribution and control (EPD&C) system provides triple redundancy with the main distribution assemblies (MDA's) and the power control assemblies (PCA's), and will supply both direct current at 28Vdc and alternating current at 115V.

The EPS mass statement is listed in table 6.4. The fuel cells are displayed for completeness but are not included in the total system mass.

Component	Mass (kg)	Mass (lb)
Fuel Cells (3)	(89.1)	(196.5)
MDA,PDA (3)	53.0	116.9
Electrical Integration	288.8	636.8
Totals	341.8	753.7

Table 6.4 EPS Equipment List

The mass allotted for integration includes all cabling, harnesses, and supports as well as all interconnects between the EPD&C system and other subsystems.

7.0 Thermal Control Subsystem

The insulation on the cryogenic propellant tanks will be a combination of spray-on foam insulation (SOFI) and multi-layer insulation (MLI). The optimum MLI thickness for each propellant was determined based on minimizing the overall vehicle wet mass assuming worst case conditions, resulting in an MLI thickness of 1.0 cm for the hydrogen tanks and 2.5 cm for the oxygen tanks. In addition, both tanks will be coated with 1.25 cm of SOFI for ground hold purposes. The total weight of this tank insulation is 203 kg of MLI and 119 kg of SOFI.

The method of subsystem heat rejection was not analyzed in detail for this study; however, there are several passive and active approaches available to implement this thermal control. Passive thermal control techniques include optical coatings, heat pipes, MLI, and changing vehicle orientation, while active options include cold plates, evaporators, radiators, and heat pumps. For the purposes of this study it was assumed that a combination of periodic rotation of the spacecraft during transit along with an active control system sized to reject 2.0 kW will suffice, but it is recommended that more detailed analysis be performed in the future to determine the best combination of available options. This thermal control system will require about 200 W of power for steady state operation as is estimated at 73 kg.

Finally, it was determined that 1.25 cm of SOFI (about 40 kg) would provide adequate protection for the propellant tanks and habitat module from engine exhaust during main propulsion system operation.

8.0 Conclusions

This paper represents the results of a top level in-house study to come up with a preliminary design of a direct descent Lunar lander to service the MSFC FLO alternate habitation module concept. The results were carried through only one iteration and as such are not to be considered final values. In the future it is recommended that the results of this study be used to generate new scaling equations, and that the design be further iterated and optimized based on both the cargo and piloted configurations. Finally, there are several areas of interest mentioned in this report that must be explored in more detail, including main engine selection, EPD&C commonality, RCS maneuver requirements, subsystem thermal control, and boiloff utilization.