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Lunar Lander Designs for Crewed Surface Sortie Missions in a Cost-Constrained Environment

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The authors have performed a parametric study of lunar lander vehicle designs to transport a crew of 2 between the Earth-Moon L1 or L2 libration points and the lunar surface for a 14-day sortie mission. The trade space of lunar lander designs includes different propellant options and staging configurations. The propellant combinations considered are: oxygen and hydrogen, oxygen and methane, oxygen and kerosene, and nitrogen tetroxide and hydrazine. The staging configurations considered are: a single-stage lander, a two-stage lander with ascent and descent stages, and a two-stage lander with in-space and lander stages. Lander masses and dimensions are presented for each combination of propellant type and staging configuration. The resultant lander masses ranged from 30t to 60t. For each configuration, the Design, Development, Test, and Evaluation (DDT&E) and Theoretical First Unit (TFU) costs, as well as lander vehicle contribution towards the probability of loss of mission, are also presented. DDT&E costs ranged from $6.0B to $7.5B, while TFU costs ranged from $500M to $800M. The total loss of mission for all concepts ranged from 1.9% to 3.2%.

Nomenclature

DDT&E	=	Design, Development, Test, and Evaluation
Isp	=	Specific Impulse
LCH4	=	Liquid Methane (CH_4)
LEO	=	Low Earth Orbit
LH2	=	Liquid Hydrogen (H_2)
LLO	=	Low Lunar Orbit
LOM	=	Loss Of Mission
LOX	=	Liquid Oxygen (O_2)
LRECM	=	Liquid Rocket Engine Cost Model
L1	=	Earth-Moon L1 libration point
L2	=	Earth-Moon L2 libration point
MMH	=	Monomethylhydrazine ($CH_3N_2H_3$)
NAFCOM	=	NASA Air Force Cost Model
NTO	=	Nitrogen Tetroxide (N_2O_4)
RP	=	Rocket Propellant, highly refined form of kerosene
SLS	=	Space Launch System
TFU	=	Theoretical First Unit

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I. Introduction

The United States is considering a number of architecture options for conducting human space exploration beyond low Earth orbit. The authors believe that cislunar space, i.e. the region of space surrounding the Earth and the Moon, is the next logical step for NASA's human space exploration program, with benefits in three areas: commerce, exploration, and science. Development of a cislunar infrastructure will ensure continued U.S. leadership in the international community, allows the U.S. to extend its economic influence beyond LEO, and enable the utilization of the Moon's material and energy resources. In addition, cislunar space and the lunar surface provide nearby proving grounds for new exploration technologies and hardware; cislunar space is also a natural basing point for deep space missions. Finally, the study of the Moon's surface and interior will be useful to the fields of planetary science and solar system formation, and the lunar far side is of great interest to the astronomy community.

A natural and logical step in the development of cislunar space is the advancement of crewed missions to the lunar surface. These missions will depend on a lunar lander vehicle to transport the crew from cis-lunar space to the lunar surface, to support the crew for the duration of the lunar surface sortie, and to return the crew back to cis-lunar space. The trade space of lunar lander designs includes different propellant combinations and staging configurations.

The authors have performed a parametric study of lunar lander designs to better understand this trade space. Propellant combinations considered are: liquid oxygen and liquid hydrogen (LOX/LH2), liquid oxygen and liquid methane (LOX/LCH4), liquid oxygen and kerosene (LOX/RP), and nitrogen tetroxide and hydrazine (NTO/MMH). The staging configurations considered are: a single-stage lander, a two-stage lander with separate ascent and descent stages, and a two-stage lander with separate in-space and lander stages. All of the lander designs were assumed to operate between the Earth-Moon L1 or L2 libration points (L1/L2) and the lunar surface. The libration points offer several advantages as cis-lunar staging locations.

II. Trade Study

The options for lander propellant are shown in Table 1 and Table 2 respectively, along with a summary of the advantages and disadvantages of each option. The propellant options present a trade between performance (i.e. specific impulse) and propellant density; as the density of the propellants increases, their performance decreases. Higher performance propellants require less total propellant mass but larger tanks and vehicle structures. Conversely, lower performance propellants required more total propellant mass but smaller tanks and vehicle structures. The concept of operations for the three vehicle configurations described in Table 2 are shown in Figure 1, Figure 2, and Figure 3 for Option 1, Option 2, and Option 3 respectively.

Table 1. Propellant Options

Option	**Propellants**	**Advantages**	**Disadvantages**
A	LOX/LH2	High performance, commonality with SLS upper stage	Hydrogen boil-off, low fuel density
B	LOX/LCH4	Low boil-off fuel and oxidizer, good fuel density	Few heritage engines, low performance (compared to LOX/LH2)
C	LOX/RP	Storable fuel, low boil-off oxidizer, heritage engines, great fuel density	Low performance (compared to LOX/LH2 or LOX/LCH4)
D	NTO/MMH	Storable propellants, great fuel and oxidizer densities	Lowest performance of all options, toxicity necessitates special handling

Table 2. Vehicle Configurations

Option	**Configurations**	**Advantages**	**Disadvantages**
1	Ascent + Descent Stage	No LLO rendezvous maneuver, single propulsion system	Expendable descent stage, complex vehicle design to share propulsion
2	In-Space + Lander Stage	Fully reusable, common propulsive elements	LLO rendezvous and orbit plane change maneuver required
3	Single Stage Lander	No LLO rendezvous maneuver, fully reusable, single vehicle design	Highest dry and gross mass, largest physical lander size

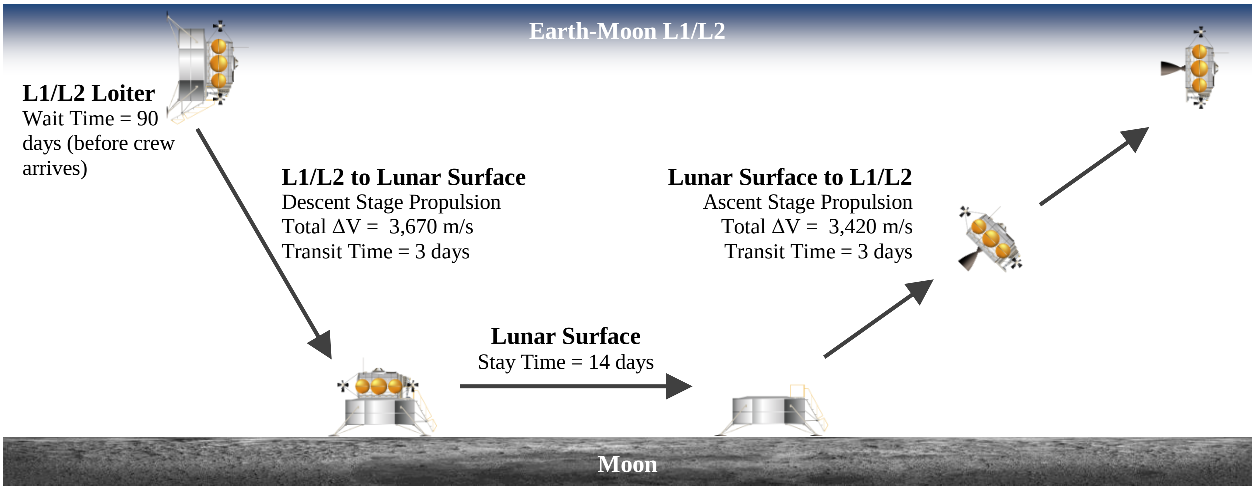

Figure 1. Concept of Operations for Ascent + Descent Stage Configuration (Option 1)

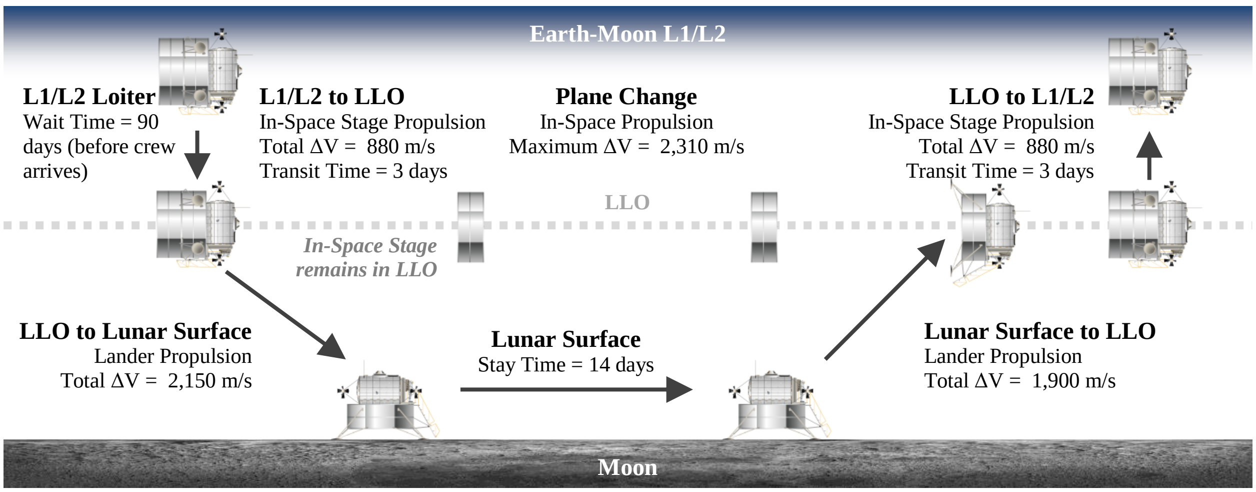

Figure 2. Concept of Operations for In-Space + Lander Stage Configuiration (Option 2)

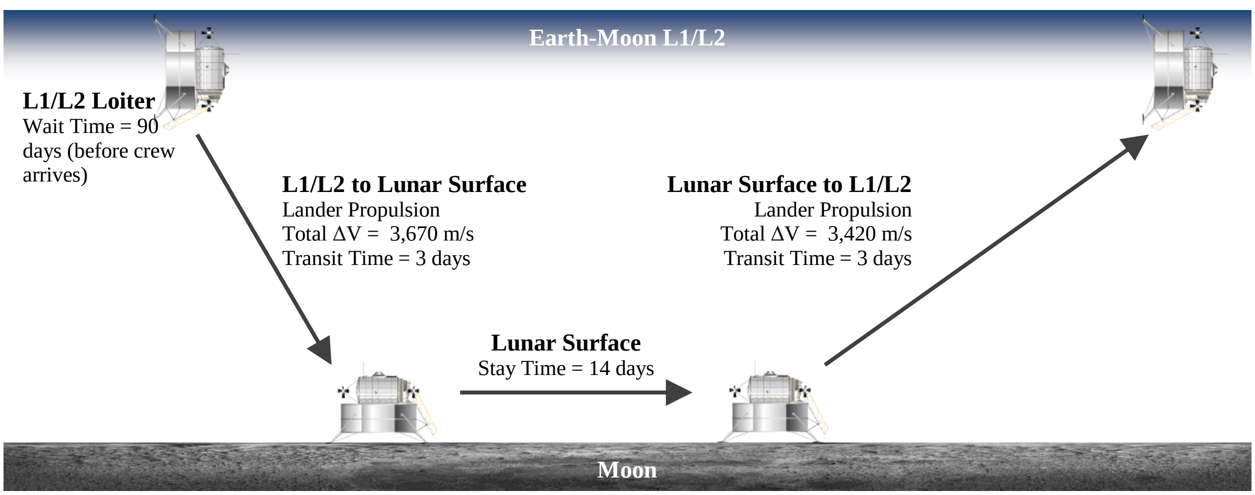

Figure 3. Concept of Operations for Single Stage Lander (Option 3)

The configuration options present three different methods for vehicle design. The first option is a traditional two stage vehicle with a dedicated ascent stage and descent stage. The descent stage performs all maneuvers from L1/L2 to the lunar surface for descent; the ascent stage performs all maneuvers from the lunar surface back to L1/L2 for ascent. In this configuration, it is assumed that the descent stage is expendable and left on the lunar surface after descent. In the interest of reducing mass and cost, it is also assumed that these vehicles share a single propulsion system housed on the ascent stage.

The second option is also a two stage vehicle, but for this option an in-space stage performs maneuvers between L1/L2 and LLO, while the lander stage performs maneuvers between LLO and the lunar surface. The in-space stage loiters in LLO during lander stage operation. Once the lunar surface sortie is concluded, the lander stage performs a rendezvous with the in-space stage in LLO. To enable mission abort capability, the in-space stage is designed to carry enough additional propellant to perform a plane-change maneuver in order to rendezvous with the lander stage at any point during the surface sortie regardless of landing site latitude. Because of this additional performance constraint, the propellant requirements between the in-space stage and lander stage are very similar. It is therefore assumed that the two stages in this configuration share a common propulsive core element (engines, tanks, subsystems, structure, etc.). The lander stage only requires the addition of landing gear from the common element; the in-space stage requires additional avionics and communications systems to be capable of operating autonomously in LLO.

The third and final configuration option is a fully-reusable single stage vehicle.

III. Assumptions and Methodology

A. Vehicle Design

The vehicle designs generated by the model assume the use of existing and near-term technologies for structures, propulsion, and subsystems. Cryogenic propellants are assumed to be cooled passively using multi-layer insulation; no active thermal management systems were considered for propellant thermal management. Power generation is provided by advanced stirling radioisotope generators to enable operation during both the lunar day and lunar night. For the ascent + descent stage configurations, the ascent stage provides power for both ascent and descent vehicles. Because it needs to operate autonomously, the in-space stage of the in-space + lander stage configurations uses an independent power system.

The liquid rocket engines selected for use in this study are presented in Table 3. These engines were selected based on applicability to design (e.g. thrust level). The Common Extensible Cryogenic Engine (CECE) is a prototype deeply-throttlable engine currently in development by Aerojet Rocketdyne for use in future human exploration missions; it is being designed to operate using either LOX/LH2 or LOX/LCH4[1]. The RD-58M is an existing LOX/RP engine currently in production by RSC Energia and being flown on the upper stage of the Proton launch vehicle[2]. The Aestus rocket engine is an existing NTO/MMH engine current in production by EADS Astrium and being flown on the upper stage of Ariane 5 ES and GS versions[3].

Table 3. Liquid Rocket Engines

Parameter	**LOX/LH2 (A)**	**LOX/LCH4 (B)**	**LOX/RP (C)**	**NTO/MMH (D)**
Engine Name	CECE[1]	CECE[1]	RD-58M[2]	Aestus[3]
Manufacturer	Aerojet Rocketdyne	Aerojet Rocketdyne	RSC Energia	EADS Astrium
Number per Stage	1	1	1	3
Cycle	Expander	Expander	Gas Generator	Pressure Fed
Vacuum Thrust	66.7 kN (15.0 klbf)	66.7 kN (15.0 klbf)	83.4 kN (18.5 klbf)	29.6 kN (6.6 klbf)
Vacuum Isp	460 sec	360 sec	349 sec	324 sec
Mixture Ratio (O/F)	5.8	3.6	2.5	1.9
Mass	210 kg	210 kg	300 kg	111 kg
Design / Modify Cost*	$260M	$310M	$77M	$77M
Acquisition Cost*	$30M	$31M	$9.4M	$16M
Mean Failure Rate[4]	0.575%	0.575%	0.379%	0.097%

* Costs include hardware, integration, and industry standard wraps, and are presented in $M FY2013

All lander vehicles use a common habitat element designed to support a crew of 2 for 20 days: 3 days from L1/L2 to the lunar surface, 14 days on the lunar surface, and 3 days from the lunar surface back to L1/L2. This habitat provides a cylindrical pressurized volume of 2.5m diameter and 4.0m length, or 20 m^3 total. Two NASA suitlocks with spacesuits provide crew surface access from the habitat, and a NASA docking system provides crew access to other space vehicles (e.g. Orion Multi-Purpose Crew Vehicle). The habitat uses an open loop life support system: all required crew consumables including oxygen, water, and food, are assumed to be outfitted at the beginning of the mission and expended during the mission. Waste carbon dioxide is collected with disposable lithium hydroxide canisters. Waste water is collected and tanked.

Table 4. Maneuver Delta Vs[5, 6]

Maneuver	Delta V (m/s)
L2 Departure/Arrival	240
LLO Departure/Arrival from L2	640
LLO Descent to Surface	2,150
Surface Ascent to LLO	1,900
L2 Descent to Surface	2,790
Surface Ascent to L2	2,540
LLO Plane Change Contingency	2,310

B. Mass and Performance

Lander vehicles were sized using a combination of a parametric mass and sizing model and a trajectory performance model. The mass and sizing model is built on combination of historical mass estimating relationships, physics-based equations, and empirical data. Inputs to the mass and sizing model are propellant requirements (determined by the trajectory performance model), liquid rocket engine selection and number of engines, and a set of top-level vehicle design parameters including number and diameter of tanks, structures, and subsystems assumptions. From these inputs, a mass breakdown statement of the lander vehicle is determined. A 30% mass growth allowance is carried for all vehicle dry masses.

Propellant requirements for the landers are generated from the trajectory performance model, which determines required Delta V values for each maneuver from literature and in-house trajectory models. The Delta Vs assumed for vehicle sizing are presented in Table 4. These represent the ideal Delta V for each maneuver, including any gravity or thrust vectoring losses.

C. Cost

The NASA Air Force Cost Model (NAFCOM) 2011 was used to estimate the cost of all lunar lander stages. In general, the landers were assumed to be developed by a traditional aerospace prime contractor, with a typical level of government oversight. All technologies on the landers were considered to be at a technology readiness level of six or above. A moderate amount of design and requirements changes are expected to occur during the program and modern manufacturing techniques were assumed for most subsystems. All costs shown are in millions of fiscal year 2013 dollars and include cost wraps reflective of a large complex program (12% fee, 11% program support, 30% contingency, and 6% vehicle integration). Launch, facilities, software, mission operations, and fixed costs are not included in the estimates.

All non-propulsive subsystems were considered to be a new design, and four system test hardware (STH) units were included for each. In the case of the two stage lander (in-space stage and lander stage), the bulk of the DDT&E was carried on the lander stage. Because it is assumed that the lander and in-space stage share common elements where possible, commonality credits were taken on the in-space stage DDT&E based on the assumption that limited design work and fewer STH units were required.

Engine costs were estimated using the Liquid Rocket Engine Cost Model (LRECM) embedded within NAFCOM. The CECE engine costs considered the hot fire testing that has already occurred on the LOX/LH2 version. The RD-58M and Aestus engines were assumed to be existing, but required minor modifications.

All habitat subsystems were considered to be a clean sheet design. The reported habitat cost includes suitport hardware development and suitport suit development.

D. Reliability

A fault tree analysis and event sequence diagram methodology was used for each lander design. The event sequence diagram represents a high-level analysis of various endpoints for each scenario. These endpoints are either mission success, or a failure leading to Loss of Mission (LOM). Once the probability of success and failure for each event are determined, the subsequent endpoints may be calculated. The reported reliabilities are the mean values on a 4th flight maturity level.

The engine failure rates shown in Table 3 were estimated based on mean failure rates of historical engines. The CECE reliability was derived from the RL-10 engine, the LOX/LH2 expander cycle engine from which the CECE is evolved. The RD-58M reliability was derived from the F-1, another LOX/RP gas generator engine. The Aestus engine reliability was derived by from the the Orbital Maneuvering Engine, a similar NTO/MMH pressure-fed engine. A 50% improvement in overall reliability over the historical data for all three engines was assumed to account for modern engine design and manufacturing quality control.[4]

Catastrophic failure rate of each engine was assumed to be 25% with the additional assumption that a non-catastrophic (benign) failure would lead only to a LOM, not a loss of vehicle. The lunar lander habitat module's reliability is computed by means of its subsystems' failure rates. A life support malfunction was assumed to fall into two categories: benign failure and critical failure.

IV. Results

A. Performance Results

A summary of the performance results are shown in Figure 4. Total vehicle mass ranged from 30t to 60t. Detailed mass breakdowns for each lander design are shown in Table 5 for the ascent + descent stage configuration, Table 6 for the in-space + lander stage configuration, and Table 7 for the single stage configuration.

Vehicle mass increases as Isp decreases; the LOX/LH2 designs are the lowest total mass (1A- 3A), while the NTO/MMH designs are the highest total mass (1D- 3D). As expected, the single stage configurations are all significantly more massive than the two stage configurations (3A-3D).

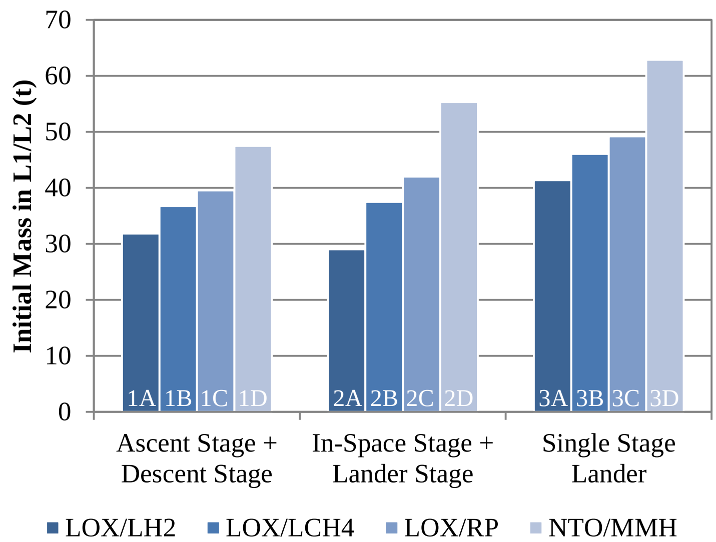

Figure 4. Performance Results

Among the two stage configurations, the in-space + lander stage designs (2A-2D) show a stronger sensitivity to propellant choice than the ascent + descent stage designs (1A-1D). This is not unexpected, because the total Delta V requirement on the in-space + lander stage designs is higher due to the plane change requirement on the in-space stage. The LOX/LH2 and LOX/LCH4 designs for the ascent + descent and in-space + lander configurations (1A and 2A, 1B and 2B) are roughly comparable in terms of mass. The LOX/RP and NTO/MMH designs for the ascent + descent stage configuration (1C and 1D) are appreciably lower mass than these same propellant options for the in-space + lander stage configuration (2C and 2D).

Recent studies predict that the baseline SLS Block 1 configuration can deliver 27t of payload to L1/L2, insufficient for any of the landers considerd in this study[7,8]. The authors estimate that SLS configurations with the advanced boosters and/or new upper stage to SLS can deliver between 33t and 42t of payload to L1/L2[9,10].

An SLS configuration that provides 33t of payload to L1/L2 would enable either of the two stage LOX/LH2 lander designs (1A and 2A). A 42t payload configuration would enable the non-hydrogen two stage LOX/LCH4 designs (1B and 2B) and LOX/RP designs (1C and 2C), as well as a single stage LOX/LH2 design (3A). All of the configurations were designed to fit within the SLS Block 1 8.4m fairing, with the sole exception of the single stage LOX/LH2 design (3A), which requires the SLS Block 2 10m fairing.

Illustrations of all twelve individual lander designs are shown in Figure 5. These illustrations are all created to the same scale to show the difference in size between the different lander designs. The non-hydrogen options all provide a significant volume advantage compared to the LOX/LH2 designs. This allows the habitat element to be placed closer to the lunar surface, reducing the distance required by the crew to reach the surface from the suitlock. Note that the three Aestus engines are in-plane in these illustrations, and are not all visible from the selected viewpoint.

Table 5. Mass Breakdowns for Ascent + Descent Stage Configuration

Element	Item	LOX/LH2 (1A)	LOX/LCH4 (1B)	LOX/RP (1C)	NTO/MMH (1D)
Habitat	Inert	3.0 t	3.0 t	3.0 t	3.0 t
	Crew/Consumables	1.0 t	1.0 t	1.0 t	1.0 t
Ascent Stage	Inert	2.6 t	2.2 t	2.3 t	2.6 t
	Usable Propellant	6.3 t	7.5 t	8.0 t	9.4 t
Descent Stage	Inert	2.6 t	1.8 t	1.8 t	2.1 t
	Usable Propellant	16.3 t	21.2 t	23.3 t	29.3 t
Total	***System***	***31.8 t***	***36.7 t***	***39.5 t***	***47.4 t***

Table 6. Mass Breakdowns for In-Space + Lander Stage Configuration

Element	Item	LOX/LH2 (2A)	LOX/LCH4 (2B)	LOX/RP (2C)	NTO/MMH (2D)
Habitat	Inert	3.0 t	3.0 t	3.0 t	3.0 t
	Crew/Consumables	1.0 t	1.0 t	1.0 t	1.0 t
In-Space Stage	Inert	2.0 t	1.9 t	2.0 t	2.5 t
	Usable Propellant	10.1 t	14.6 t	16.6 t	23.2 t
Lander Stage	Inert	2.6 t	2.6 t	2.9 t	4.2 t
	Usable Propellant	10.2 t	14.3 t	16.5 t	21.4 t
Total	***System***	***29.0 t***	***37.4 t***	***42.0 t***	***55.3 t***

Table 7. Mass Breakdowns for Single Stage Configuration

Element	Item	LOX/LH2 (3A)	LOX/LCH4 (3B)	LOX/RP (3C)	NTO/MMH (3D)
Habitat	Inert	3.0 t	3.0 t	3.0 t	3.0 t
	Crew/Consumables	1.0 t	1.0 t	1.0 t	1.0 t
Lander	Inert	6.7 t	4.8 t	4.9 t	6.0 t
	Usable Propellant	30.7 t	37.2 t	40.3 t	52.8 t
Total	***System***	***41.3 t***	***46.0 t***	***49.2 t***	***62.8 t***

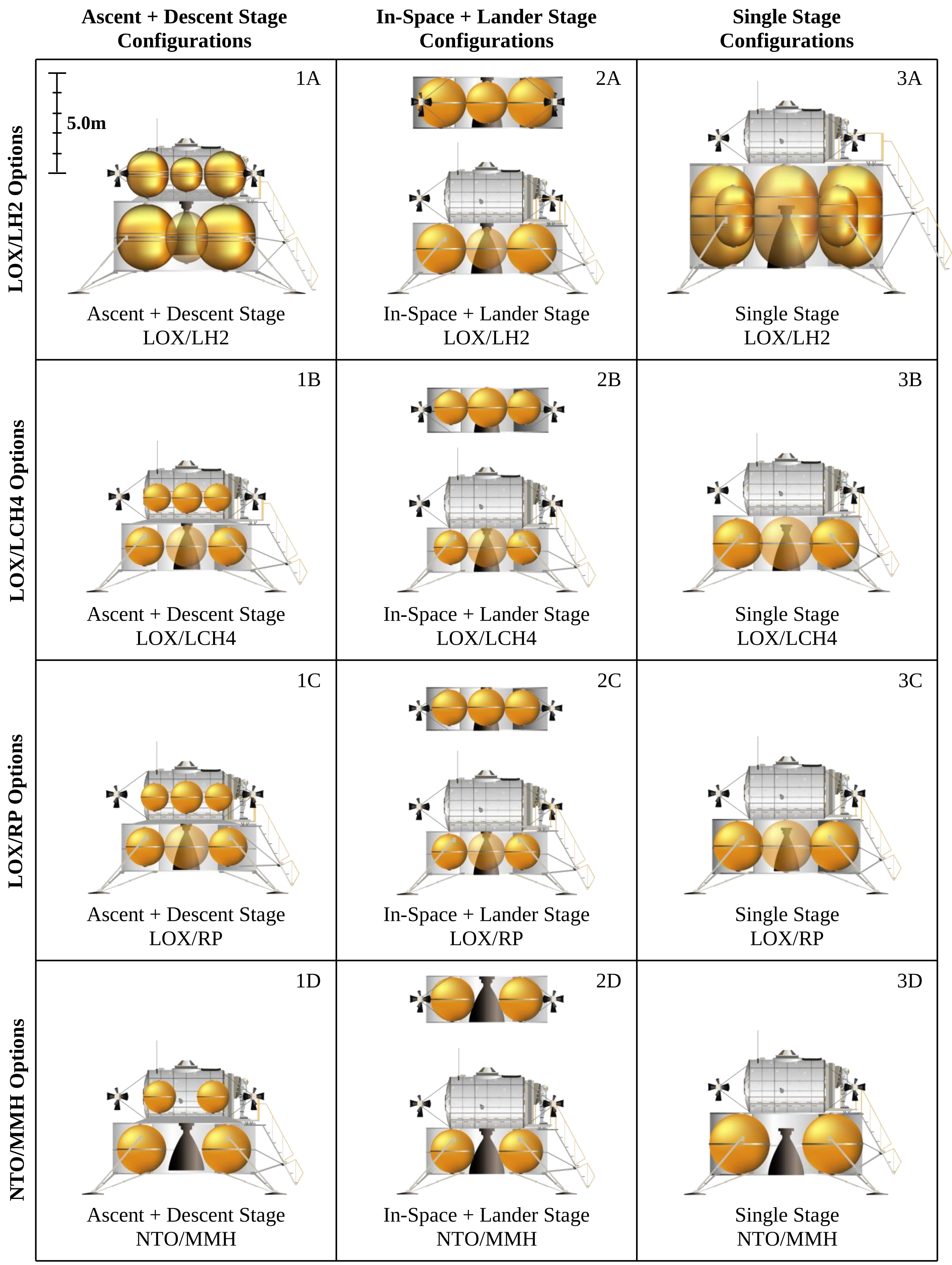

Figure 5. Illustrations of All Configurations (to relative scale)

B. Cost Results

The DDT&E cost and TFU cost results for all configurations are shown in Table 8. DDT&E costs ranged from $6.0B to $7.5B, while TFU costs ranged from $500M to $800M. A summary of total cost (DDT&E + TFU) results is shown in Figure 6.

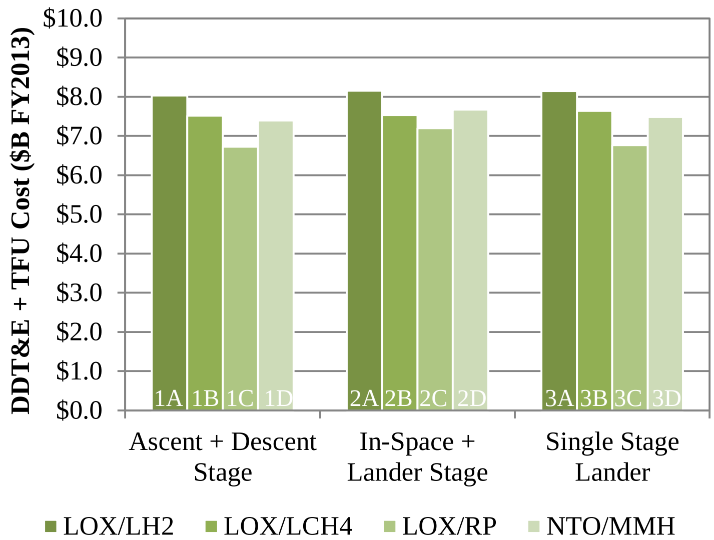

Figure 6. Cost Results

The two stage ascent + descent stage configurations (1A-1D), across similar propellant choices, show a slight cost advantage over the other two options, with the single stage configurations (3A-3D) coming in second, and the two-stage in-space + lander stage configurations (2A-2D) yielding the highest costs. The use of a shared propulsion system between the ascent and descent stages allows this two stage configuration to provide the most affordable solution. Despite having significant commonality between the two stages of the in-space + lander stage configurations, in-space stage vehicle still requires some design work as well as test hardware for engineering/structural test articles and other mock-ups. Though the single stage options are significantly more massive, they still yield lower costs compared to the two stage in-space + lander stage configuration. Having only a single stage reduces the subsystem hardware and integration costs required.

Comparing the propellant options, across similar stage configurations, the LOX/RP designs (1C-3C) yield lower costs than LOX/LH2 (1A-3A), LOX/LCH4 (1B-3B), and NTO/MMH (1D-3D) designs. This is largely due to the lower engine DDT&E and unit costs required of the flight-proven RD-58M engine. The LOX/LH2 designs yield the highest costs due to the large stage dry masses, a direct consequence of the significantly higher tank volumes required to store LH2. The LOX/LCH4 and NTO/MMH designs fall in between, with the LOX/LCH4 designs benefitting from the smaller overall vehicle sizes, and the NTO/MMH designs benefitting from the flight-proven Aestus engine.

Table 8. DDT&E and TFU Costs for All Configurations ($M FY2013)

Configuration	Cost	LOX/LH2 (A)	LOX/CH4 (B)	LOX/RP (C)	NTO/MMH (D)
Ascent + Descent Stage (1)	DDT&E	$7,390	$6,930	$6,210	$6,770
	TFU	$640	$590	$510	$630
	Total	$8,030	$7,520	$6,720	$7,400
In-Space + Lander Stage (2)	DDT&E	$7,430	$6,840	$6,560	$6,910
	TFU	$720	$690	$630	$760
	Total	$8,150	$7,530	$7,190	$7,670
Single Stage Lander (3)	DDT&E	$7,460	$7,030	$6,190	$6,820
	TFU	$690	$610	$570	$660
	Total	$8,150	$7,640	$6,760	$7,480

C. Reliability Results

The reliability results are shown in Figure 7. The total LOM for all concepts ranges from 1.9% to 3.2%. Detailed reliability results are presented in Table 9 for the ascent + descent stage configuration, Table 10 for the in-space + lander stage configuration, and Table 11 for the single stage lander configuration.

The NTO/MMH designs (1D-3D) yield the highest reliability for two reasons. First, because of the size of the Aestus engine and the small propellant volume requirements, the NTO/MMH configurations are able to use three engines and enable engine-out capability – all maneuvers are feasible with only two engines. Second, being pressure-fed and hypergolic, the Aestus engine is more reliable than the CECE or RD-58M engines. The LOX/RP (1C-3C) designs yield higher reliability than the LOX/LH2 (1A-3A) and LOX/LCH4 (1B-3B) designs due to the higher reliability of the RD-58M engine compared to the CECE engine. Finally, the LOX/LH2 designs were slightly less reliability than their LOX/LCH4 counterparts despite using the same engine, because of the additional complexities associated with LH2 boil-off and venting.

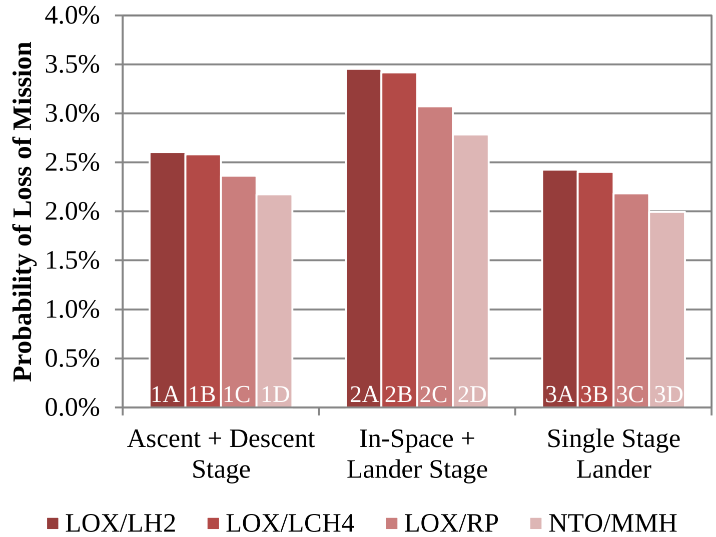

Figure 7. Reliability Results (4th flight)

Comparing the three configuration options, the single stage lander designs (3A-3D) yield the high reliability due to the lack of separation events, compared to the two-stage configurations. The in-space + lander stage designs (2A-2D) yield the lowest reliability due to the additional rendezvous maneuver between the two stages in LLO, and the additional engine ignitions and burns to arrive at and depart from LLO.

Table 9. Loss of Mission Contributors for Ascent + Descent Stage Configuration (Option 1)

Item	LOX/LH2 (A)	LOX/LCH4 (B)	LOX/RP (C)	NTO/MMH (D)
Propulsive Maneuvers	0.68%	0.65%	0.43%	0.24%
Habitat	0.63%	0.63%	0.63%	0.63%
Lunar Landing	0.66%	0.66%	0.67%	0.67%
Rendezvous	0.45%	0.45%	0.45%	0.45%
Separations	0.18%	0.18%	0.18%	0.18%
Total LOM	**2.61%**	**2.58%**	**2.36%**	**2.18%**

Table 10. Loss of Mission Contributors for In-Space + Lander Stage Configuration (Option 2)

Item	LOX/LH2 (A)	LOX/LCH4 (B)	LOX/RP (C)	NTO/MMH (D)
Propulsive Maneuvers	0.67%	0.65%	0.43%	0.24%
Habitat	0.63%	0.63%	0.63%	0.63%
Lunar Landing	0.66%	0.66%	0.66%	0.66%
Rendezvous	0.90%	0.90%	0.90%	0.91%
Separations	0.23%	0.23%	0.22%	0.22%
Total	**3.10%**	**3.08%**	**2.85%**	**2.66%**

Table 11. Loss of Mission Contributors for Single Stage Lander Configuration (Option 3)

Item	LOX/LH2 (A)	LOX/LCH4 (B)	LOX/RP (C)	NTO/MMH (D)
Propulsive Maneuvers	0.68%	0.65%	0.43%	0.24%
Habitat	0.63%	0.63%	0.63%	0.63%
Lunar Landing	0.66%	0.66%	0.67%	0.67%
Rendezvous	0.45%	0.45%	0.45%	0.46%
Separations	0.00%	0.00%	0.00%	0.00%
Total	**2.43%**	**2.40%**	**2.18%**	**1.99%**

V. Observations

A summary of the results for each option and configuration across performance, cost, and reliability is shown in Table 12. The lowest mass (2A), lowest cost (1C), and highest reliability (3D) solutions are highlighted. From a mass perspective, the higher Isp + lower density propellants yield lower mass vehicles than the lower Isp + higher density propellants – the mass savings from higher propellant performance outweigh those from smaller tank volumes. Therefore the LOX/LH2 designs are the lowest mass, and the NTO/MMH designs are the highest. The two stage configurations both yield similar stage masses and are significantly less massive than the single stage configurations. Propellant choice is a much greater cost driver than configuration. The LOX/RP designs are the lowest cost, while the LOX/LH2 designs are the highest. This is driven in some part by the high DDT&E cost associated with the CECE compared to the RD-58M. Further development of the CECE will improve the relative cost of the LOX/LH2 and LOX/LCH4 options. Comparing vehicle configuration options, all three configurations yield similar cost results, with the in-space + lander options costing slightly more than the other two options. For reliability, the single stage configurations provide the highest reliability and the in-space + lander stage designs provide the lowest.

Table 12. Results Summary

Configuration	Metric	LOX/LH2 (A)	LOX/CH4 (B)	LOX/RP (C)	NTO/MMH (D)
Ascent + Descent Stage (1)	Mass (Initial in L1/L2)	31.8 t	36.7 t	39.5 t	47.4 t
	Cost (DDT&E+TFU)	$8,030	$7,520	**$6,720**	$7,400
	Reliability (LOM)	2.61%	2.58%	2.36%	2.18%
In-Space + Lander Stage (2)	Mass (Initial in L1/L2)	**29.0 t**	37.4 t	42.0 t	55.3 t
	Cost (DDT&E+TFU)	$8,150	$7,530	$7,190	$7,670
	Reliability (LOM)	3.10%	3.08%	2.85%	2.66%
Single Stage Lander (3)	Mass (Initial in L1/L2)	41.3 t	46.0 t	49.2 t	62.8 t
	Cost (DDT&E+TFU)	$8,150	$7,640	$6,760	$7,480
	Reliability (LOM)	2.43%	2.40%	2.18%	1.99%

Considering all of these metrics together, several observations can be made. First, while the single stage configurations are the most reliable, they do not provide any cost advantage, and are significantly more massive than the two stage options. Unless the selected launch vehicle has a very high payload capability, the single stage options are likely infeasible. Second, the NTO/MMH designs are substantially more massive than the other propellant options, despite the advantage of having dense, non-cryogenic fuel and oxidizer. Though the NTO/MMH designs perform well in cost and reliability, the low Isp performance is likely prohibitive from a mass standpoint, regardless of engine selection.

Comparing the two stage configurations, the in-space + lander stage configurations are both less reliable and more expensive than the ascent + descent stage configurations, and with the exception of the LOX/LH2 design, more massive as well. However, the in-space + lander stage configuration is fully reusable. If the cislunar infrastructure and mission architecture can take full advantage of this reusability, this configuration may offer performance, cost, and reliability advantages that are outside of the scope of the present study.

The LOX/LH2, LOX/LCH4, and LOX/RP propellant options each offer advantages and disadvantages. The LOX/LH2 designs are the lowest mass but the highest cost, while the LOX/RP designs are the highest mass but lowest cost; the LOX/LCH4 designs strike a balance between the two. Inflexible constraints on either payload mass or system cost may ultimately prove to be the deciding factor for propellant choice.

The ranking of each configuration and propellant option versus performance (initial mass in L1/L2), cost (sum of DDT&E cost and TFU cost), and reliability (probability of LOM) are shown in Table 13.

Table 13. Configuration and Propellant Option Rankings

Configuration	Metric	LOX/LH2 (A)	LOX/CH4 (B)	LOX/RP (C)	NTO/MMH (D)
Ascent + Descent Stage (1)	Mass (Initial in L1/L2)	2	**3**	**5**	9
	Cost (DDT&E+TFU)	10	**6**	**1**	4
	Reliability (LOM)	8	**7**	**4**	2
In-Space + Lander Stage (2)	Mass (Initial in L1/L2)	1	4	7	11
	Cost (DDT&E+TFU)	12	7	3	9
	Reliability (LOM)	12	11	10	9
Single Stage Lander (3)	Mass (Initial in L1/L2)	6	8	10	12
	Cost (DDT&E+TFU)	11	8	2	5
	Reliability (LOM)	5	6	3	1

VI. Conclusions

Based on the results in Table 13, Option 1C, the LOX/RP version of the two-stage ascent + descent configuration, is the best performer against the three metrics considered. The low cost and high reliability of the flight-proven RD-58M, combined with the high density and zero boil-off properties of RP, make it a strong choice for near-term crewed exploration of the lunar surface.

Considering the broader context of human exploration, however, the LOX/LCH4 version of this same configuration (Option 1B) is an interesting alternative. Compared to kerosene, methane is more strongly synergistic with Mars exploration. Long term exploration of Mars will likely involve methane-fueled vehicles relying on In-Situ Resource Utilization on the Martian surface. Furthermore, LOX/LCH4 offers an appreciable performance advantage over LOX/RP, which translates to more payload margin on a launch vehicle.

While the LOX/LCH4 designs are both higher cost and lower reliability than their LOX/RP counterparts, this difference is driven solely by the different in cost and reliability between the flight-proven LOX/RP RD-58M engine compared to the prototype LOX/LCH4 CECE engine. Further investment into the development of exploration-focused LOX/LCH4 engines would make these options more competitive in both cost and reliability, and likely provide benefits to other exploration missions as well.

Therefore, the authors recommend that Option 1B, the LOX/LCH4 version of two-stage ascent + descent configuration, be given serious consideration for future crewed missions to the lunar surface.

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