Low-Cost Uncrewed Lunar Landers: Mission Design and Preliminary Feasibility Assessment

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Abstract

Lunar exploration is poised for a renaissance. The Artemis program intends to create a permanent human presence on the moon. There has been much research and development interest in launchers, crew vehicles and habitats for lunar outposts. These lunar outposts will necessitate regular two-way cargo deliveries between the Earth and the Moon. Since space transport costs are so high compared to terrestrial shipping costs, it is interesting to investigate cost-effective and suitable ways of delivering cargo to a human lunar outpost. This paper uses a systems engineering approach to identify, evaluate and size a set of five possible mission architectures for unmanned cargo delivery from Earth to the lunar surface. The first three mission architectures are fully reusable, single-stage, two-stage, and three-stage. The fourth uses a reusable electric space tug, and the fifth is a non-reusable Apollo-style architecture. The five mission architectures are sized using statistically derived sizing rules from a database of 42 lunar landers and transfer vehicles compiled from an extensive literature review. Two sizing exercises were carried out for each architecture: (i) a maximalist case to understand the upper bound to the payload delivery mass, (ii) and a two tons payload delivery case comparable to the mass of ISS cargo deliveries. A trade-off is then performed based on seven figures of merit: Landed Payload [kg/trip] Returned Payload [kg/trip], Cost of Landed Payload [$/kg], Cost of Returned Payload [$/kg], Delivery Time [days], System Complexity and Extensibility. The most cost-effective and suitable mission architecture is chosen, and a lander design which can carry two tons of cargo down to the lunar surface and back up again is sized from statistical and parametric sizing rules. This small lander is then costed using TruePlanning software. It is found that the main cost drivers for lunar cargo delivery are fuel carriage and launcher costs. These costs could be reduced by orbital fuel depots in LEO and at the Lunar Gateway, or by ride-sharing on very large launchers.

1. Introduction

This paper takes a holistic approach to the conceptual design of lunar landers. To design a lunar lander, the engineers should consider the overall space mission architecture, in which the lander is just one of many elements. Typical trade-offs consider the launcher's choice, the transfer vehicle's design, the availability of on-orbit refuelling, the trajectory choice, and possible reusability requirements. These different variables impose size and performance limits on lunar landers. Therefore, although the lunar lander is the vehicle of primary interest, this project investigates and analyses diverse mission architectures, each including several building blocks, to answer two key questions:

1. Which is the upper bound of payload mass landers can put on the lunar surface using current technologies?

2. What is the specific cost [$/kg] to regularly deliver and return reasonably sized payloads (two tons) to and from the lunar surface using current technologies?

These two points are answered by evaluating two sets of five mission architectures. The mission architectures for lunar transport propose diverse solutions to deliver and return payload to the lunar surface. They include concepts for space tugs, on-orbit refuelling, and re-usability. The first set investigates the maximalist case of payload delivery. The five mission architectures are sized assuming that the payload to be landed on the moon has to arrive in Low Earth Orbit (LEO) in a single launch. By sizing the different elements of each mission architecture with this assumption, this paper identifies which elements of the mission architecture limit the maximum payload mass deliverable to the lunar surface. The second set investigates a two-ton payload case. This case investigates the requirements to deliver cargo similar

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in mass to that received regularly by the International Space Station (2.5t - 3.5t) [1] but to the lunar surface. Sizing mission architectures and spacecraft for the smaller two-ton case will reveal the cost drivers for regular lunar cargo delivery and return. Moreover, understanding these drivers will allow mission architects to design cheaper missions, thus making space more accessible to potential innovators such as startups and research organizations.

Following the definition of the mission architectures, a preliminary trade-off helps to rank them based on the identified mission drivers. The trade-off is guided by the rapid prototyping of the main elements of the architecture in terms of mass, primarily based on statistical and parametric formulations.

For instance, aircraft design benefits from a wealth of data to draw on when designing new solutions. For lunar landers, however, there are fewer flight models to learn from. In any statistical study of lunar landers, engineers draw mostly from well-documented conceptual designs such as those in [2]. Previous work has been done on databases of crewed lunar landers, as in a study by Ref. [3] or the investigation by Ref. [4]. Seeing the application to uncrewed landers, the authors of this article built their database from various sources, including history books, textbooks, catalogs, original data sheets and user manuals like [5–10]. An initial database version has been analysed in a previous authors' publication [11]. It focused on uncrewed surface elements, encompassing both rovers and landers. The work of [11] is extended in this publication, focusing on lunar landers. This analysis provides a set of rapid prototyping rules to size the landers in the different architectures.

This study is guided by a well-established design methodology of Ref. [12]. The approach is highly oriented toward both functional analysis [12–14], and scenario-based assessments [15, 16]. A critical analysis highlighted in both Ref. [12] and Ref. [13] is related to cost. Thus, cost engineering is a crucial element of space mission design. Without cost estimates or, worse, cost underestimation, a space mission will probably never leave the ground. Cost engineering can be defined as the practice of managing project costs through good cost estimation and cost control. In this project, parametric cost estimation was carried out using TruePlanning [17]. This software uses databases of component costs and historical patterns in project costs to give reasonable cost estimates for the whole life cycle of a space system.

For example, the unit cost of lunar landers is high compared to other similar-sized and complex aerospace systems, such as fighter jets. The unit cost of the Apollo lunar modules is $3.8 billion USD [18] in 2020 USD. The authors of Ref. [19] estimate that an uncrewed lunar lander may have a unit cost of $786 million USD. The F22, widely considered the most expensive fighter jet ever built, had a unit cost of just $377 million USD when the final model was finished in 2011 [20]. This difference is due to the economy of scale present for the F22. Such savings are not available to lunar landers because almost all lunar landers have very short production runs. The longest series of lunar landers produced so far has been the Surveyor series, with seven flight models [21]. This inherent cost disadvantage means that other ways must be found to make lunar landers cheaper.

To analyse all of this open points, the rest of the paper will be organized as follows: Section 2 analyses the design methodology employed in this study, Section 3 summarizes the main results, while Section 4 wraps up the main outcomes of the study and important take-aways.

2. Method and Materials

This section summarises the design steps carried out during the definition and sizing of the different mission architectures. After an initial focus on the functional architecture and derivation of the figures of merit (FOMs), this article dives into the derivation of the sizing rules from the databases of landers and transfer vehicles.

2.1 Design Methodology

In Ref. [13], systems engineering is described as a methodical, disciplined approach to the design, realisation, technical management, operations, and retirement of a system. With the label *system*, systems engineers describe a construct or collection of different elements that together produce results not obtainable by the elements alone. In this project, the *system*, or *building block*, is defined as a combination of elements and their interfaces that function together to meet a need of the mission architecture [13]. On the other hand, *subsystem* indicates a set of components that function together to produce a capability to meet a need of the system, i.e. supply power.

To define which building blocks should be considered in a mission architecture, the first step is to define a mission statement and the related operational capabilities. The mission statement should resume in one sentence the needs of the stakeholders [15]. Beyond the mission statement, another important point to analyse is the operational environment and the constraints that should or may guide the mission. On the other hand, the functional analysis provides the framework to analyse different physical configurations that may answer the mission statement [13]. The idea is to describe the high-level functions of a system and link them to different physical solutions. The best physical architecture is evaluated with trade-off analyses as the one described in Ref. [12].

In this case study, the objective of the proposed mission architecture can be encapsulated by the following mission statement:

To transport payloads to the lunar surface from the Earth's surface and back again in a cost-effective manner.

Moreover, a series of high-level assumptions provide the context of the study [22]:

- The time frame of the mission architectures is to begin development in 2024, launch in 2032 and operate until disposal in 2047.

- The Lunar Gateway, which is set to begin operations in the 2030s, is assumed to be a free gas station with no upper limit to the propellant required from it. This assumption is based on several feasibility studies on the subject of orbital fuel depots around the moon and the Esprit module on the lunar gateway [23], [24], [25].

- There will be no propellant manufacturing infrastructure on the lunar surface in the time frame of the mission architectures: any reusable lunar lander must have enough ΔV to land and return again.

- The down-payload capacity and the up payload-capacity are assumed to be the same for all architectures [1].

Following the steps detailed in [13], the engineers can directly derive a set of operational capabilities (OCs) that answer the needs outlined in the mission statement. Table 1 defines the required operational capabilities. OC5 and OC7 address the need to be *cost effective*. The performance impact of reusability and on-orbit refuelling are illustrated by comparing them to a non-reusable Apollo-style benchmark case.

ID	Operational Capability
OC1	Carry a payload.
OC2	Perform a targeted landing on the Moon.
OC3	Launch to LEO.
OC4	Perform Trans-lunar Injection.
OC5	Be reuseable.
OC6	Rendezvous in LEO.
OC7	Perform on-orbit refuelling.

Table 1: Operational capabilities derived from mission statement

The five mission architecture are investigated in Section 3. They are ranked on seven figures of merit (FOMs). These figures of merit are chosen based on the project's interest in cost, pragmatic concerns for deliverable payload mass, and integration with other lunar infrastructure projects. The identified FOMs are shown in Table 2.

ID	Figure of Merit	Unit
FOM1	Cost	$/kg_landed
FOM2	Cost	$/kg_returned
FOM3	Landed_Payload	kg/launch
FOM4	Returned_payload	kg/launch
FOM5	Delivery_Time	days
FOM6	System_complexity	#rdvs
FOM7	Extensibility	score

Table 2: Figures of Merit for the Mission Architectures

FOM1 and FOM2 are evaluated by dividing the total life cycle cost [$] by the total payload carried over the lifetime of the architecture [kg]. The following assumption drives the definition of these quantities:

1. Since the most re-used space vehicles, the Space Shuttle and Falcon 9 boosters have, at the time of writing, at most 39 and 20 flights, respectively. It is reasonable to assume the systems in the mission architecture should target a reusability target of 20 flights at a frequency of four times per year. To run the mission architecture for a time comparable to the lifetime of manned space stations 15 years, which makes 60 cycles of the mission architectures with 3 flight models of all the systems.

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[1]Down-payload capacity is the amount of payload which can be sent down to the moon. Up-payload capacity is the inverse

2. All launch costs are calculated using the published specific launch cost. This means assuming that every launch has a ride-sharing arrangement allowing for sharing of the total launch cost.

3. During the development of a lunar lander, about 2.4 equivalent production units of prototypes will be needed. This comes from a similar costing exercise done in [19].

The price of the payload depends on the overall life cycle cost that can be divided into four contributions [19]: (i) design and development costs; (ii) architecture production costs; (iii) operating costs; (iv)disposal costs. The design and development costs of the lander are evaluated using TruePlanning [17], an industry-standard cost estimation tool. It uses statistical and parametric cost estimation relationships to estimate the costs of components given their mass and some details about their function.

Each mission architecture's construction and operating costs are assumed to consist almost entirely of launch costs. Those costs are inputs for the software TruePlanning. The cost of construction C_c is given by equation 1.

$$C_c = \frac{m_c}{C_{CL,sp}}[\$] \tag{1}$$

Where m_c is the construction mass. It consists of the total dry mass of the systems to be launched and the propellant mass required to place them in the correct orbits (as described in their mission architecture). $C_{CL,sp}$ is the specific launch cost of the construction launcher. Equation 2 gives the operational cost for each mission architecture cycle.

$$C_o = \frac{m_{prop} + m_p}{C_{RL,sp}} + C_G[\$] \tag{2}$$

Where: (i) m_{prop} is the fuel requirement of the transfer vehicle which performs Trans-lunar injection (TLI); (ii) m_p is the payload mass being delivered from Earth to the lunar surface; (iii) $C_{RL,sp}$ is the specific cost of the regular launcher used for regular operations of the mission architecture; (iv) C_G is the ground segment cost. C_G is estimated to be the cost of 7 to 10 full-time engineers operating the mission architecture. This estimate is based on the personnel required for the early operations phase of an unmanned satellite, which is approximately 20 people. The disposal cost C_D is equal to the cost of bringing all of the systems in the mission architecture back from their orbits around the moon and re-entering them into Earth's atmosphere.

2.2 Sizing Rules Derivation

To quantify the impact of different design drivers [2], the first step suggested by this study is to compute a first set of statistical sizing rules. The created database of lunar landers contains every model of spacecraft (manned and unmanned) which has landed softly on the moon since Luna 9 in 1966 [26]. It also contains several conceptual lander designs as the one presented in Ref. [2]. Table 3 shows a sample of 6 entries in the database. It is available in full on github [3]

Lunar Landers Database											
Year	Organisation	Project name	total mass m_0 [kg]	dry mass m_{dry} [kg]	bloc payload mass m_p [kg]	prop mass m_{prop} [kg]	propellant	1st stage I_{sp} [s]	ΔV [m/s]	type	sources
1966	USSR	Ye-6M Luna 9	1538	847	99.8	591.2	HNO3/Amine	287	2630	pod	[6] [27] [28]
1969	USSR	Ye-8-5M Luna 16	5750	1880	520	3350	HNO3/UDMH	314	1880	2stage	[6] [27] [28]
1969	NASA	Apollo 12	15065	2034	4819	8212	N2O4/AZ49	311	2273	2stage	[29]
1971	NASA	Apollo 15	16447	2626	4795	9026	N2O4/AZ46	311	2250	2stage	[29]
2007	NASA	Gryphon	43501	8500	18634	16367	LOX/LH2	451	2117	1stage	[30]
2021	CNSA	Chang' 5	3800	1200	800	1800	N2O4/UDMH	333	1930	2stage	[31]

Table 3: Sample of 6 rows of lunar landers from the project database. The full database has 43 landers

It is important to define precisely the quantities attributed to each spacecraft. The total mass m_0 in kg is defined as in equation 3:

$$m_0 = m_{dry} + m_p + m_{prop} \tag{3}$$

Where: (i) m_{dry} is the dry mass of the system, (ii) m_{prop} is the propellant mass, (iii) m_p is the *bloc payload mass*. For one-stage landers, m_{dry} is the mass of hardware which is not propellant or payload. For two-stage landers, such as the Apollo LM and others, the dry mass is the dry mass of the descent stage only. The bloc payload mass is an invented quantity to maintain comparability between one-stage *unmanned* landers and two-stage *manned* landers. For one-stage

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[2]The principal mission parameters which influence performance cost, risk, schedule and which the user can control

[3]https://github.com/Conall98/ORLA_Lander_Mission_Architecture/tree/main

unmanned landers, the bloc payload is equivalent to the traditional definition of payload [22] [4]. For two-stage landers, however, the payload mass is considered the total mass of the ascent stage. The different definitions of m_{dry}, and m_p for one and two-stage landers are necessary to maintain comparability between many various types of landers. By defining all of the characteristics of the landers in the database in this way, this report fits the landers to a lander reference model. This is a single-stage, liquid-fuelled, fully reusable lander which carries a payload from lunar orbit down to the lunar surface and returns to lunar orbit. This lander reference model is the basis for the ORLA lander design in the results. The other entries in Table 3 can be characterized as follows:

- 1st stage I_{sp} is the specific impulse of the descent stage only.

- ΔV is the velocity change that the spacecraft is stated to have, or it is implied from its mission profile. Only the ΔV of the descent stage(s) is counted for two-stage landers.

- With *Type*, the authors classify the landers. the three types are 1-stage, 2-stage, and pod landers [5].

On the other hand, data is gathered from literature on several existing upper stages to derive sizing rules for space-tug-like transfer vehicles (here identified as TVs). Table 4 below shows a sample of TVs database for chemically powered vehicles. Another database was constructed for electrically powered vehicles.

Characteristic energy of Upper Stages						
Vehicle name	Dry mass m_{dry} [kg]	propellant mass m_{prop} [kg]	reference payload m_p [kg]	reference ΔV [m/s]	Characteristic Energy [J]	Propellant	sources
DCSS 5m	3,490	27,200	22977	1910.00	4.83E+10	LOX/LH2	[32]
Common Centaur	2462	20830	18850	3076.00	1.01E+11	LOX/LH2	[33]
Falcon 2S	3900	92670	15600	5361.00	2.80E+11	LOX/RP1	[34]
Ariane 5 2S	4540	14700	10865	2350	4.25E+10	LOX/LH2	[35] [36]

Table 4: Sample of the transfer vehicles database

The total mass m_0 of the transfer vehicle is defined similarly to Equation 3. In this case, the reference payload is the largest payload launched on that transfer vehicle which corresponds to the reference ΔV. The characteristic energy is the kinetic energy of the empty transfer vehicle and the payload together.

$$E_k = \frac{1}{2} \cdot (m_{dry} + m_p) \cdot \Delta V^2 \tag{4}$$

Calculating this quantity is helpful because it allows one to see how the transfer vehicles would perform differently if more or less ΔV was required. Curves for this relationship are drawn and discussed in section 2.3. Finally, a subsystem-focused database is defined from the literature [2]. Table 5 shows a sample of the subsystem sizing database. Each of the columns is defined as follows:

- *dry mass:* the unfuelled mass of the system without payload. For two stage manned landers, this is the dry mass of the descent stage only. Furthermore, for two stage landers it is the subsystem mass in the descent stage only which is counted.

- *Structure:* The mass of the structural elements of the system. including frame skin, fasteners and Landing legs.

- *Propulsion:* the mass of the components which contribute to the propulsion of the spacecraft, including tanks, engines, propellant management hardware, and the reaction control system (RCS).

- *Power:* Power generation and power storage equipment.

- *Avionics:* The communications hardware, flight computers and environment sensors.

- *Thermal:* The thermal protection. Assumed to consist of both active and passive systems.

Beyond the sizing-oriented databases, a database of launch costs was compiled from the literature to perform cost estimation. A sample of the launcher database is shown in Table 6.

These databases are the foundation of the sizing rules extraction. At the same time, they provided a clear indication of typical values of mass and cost related to different solutions. More in detail, sizing rules for all parameters of interest

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[4]*Payload is the combination of hardware and software on the spacecraft that interacts with the subject (the portion of the outside world that the spacecraft is looking at or interacting with)*

[5]small pressure vessels with the payload inside. Landed using airbags

Subsystem sizing						
Spacecraft	dry mass [kg]	Structure [kg]	Propulsion [kg]	Power [kg]	Avionics [kg]	Thermal [kg]	Other [kg]
Apollo J-series	2027	460	495	366	29	404	273
9508-HLR-1	1201	533	253	126	120	114	56
8801-EE-1	9823	1681	4258	478	934	2017	455
...	...	...	...	...	...	...	...
Avgs	100.0%	27.6%	33.7%	7.6%	8.8%	13.0%	9.4%
Standard dev.	-	10.9%	12.8%	5.1%	5.8%	7.5%	6.2%

Table 5: Sample of subsystem mass breakdowns database.

Launcher Database				
Launcher	Per Launch cost	Payload to LEO	$/kg	sources
Falcon Heavy	150,000,000	63800	2,351	[37]
Ariane 6 A64	115,000,000	21650	5,312	[38]
SLS B1	2,000,000,000	95000	21,053	[39]
Electron	5,300,000	220	24,091	[40]

Table 6: Sample of the launch Costs Database

are derived by investigating their relation to the system's dry mass or payload mass. For lunar landers. statistical sizing relationships are found by obtaining the linear fit between the m_{dry}, m_p and m_{prop}. A linear fit with an intercept at zero was chosen between these parameters because a re-arrangement of the Tsiolkovsky equation shows they are linearly related (see equation 7).

There are three kinds of sizing rules used in this paper:

- *Statistical sizing rules* which are derived from trend analysis of the databases. eg.

$$m_p = \alpha \cdot m_{dry} \tag{5}$$

  Where α is the coefficient which gives the line of best fit.

- *Parametric sizing rules* derived by inspection or taken from literature. eg.

$$E_k = \frac{1}{2} \cdot m_{inert} \cdot \Delta V^2 \tag{6}$$

  For these, all the parameters are deterministically dependant on one another.

- *Para-statistical sizing rules* are a blend of both. eg.

$$m_{prop} = \beta \cdot (e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p)) \tag{7}$$

  In this example, β is a coefficient found by adjusting the line made by the Tsiolkovsky equation until it lies on top of the linear fit between m_{dry} and m_{prop}.This method is depicted in figures 1 and 2. The ΔV and the I_{sp} are set to the data set's average [4, 41].

By fitting the Tsiolkovsky line to the purely statistical trend line, one finds a more useful sizing relationship which can take the system performance characteristics such as ΔV and I_{sp} into account.

2.3 Sizing Procedure

To effectively estimate the mass of the systems used in each mission architecture, the following procedure can be employed:

1. Choose a mission architecture to size.

2. Use the following ΔV map to evaluate the required ΔV for each system.

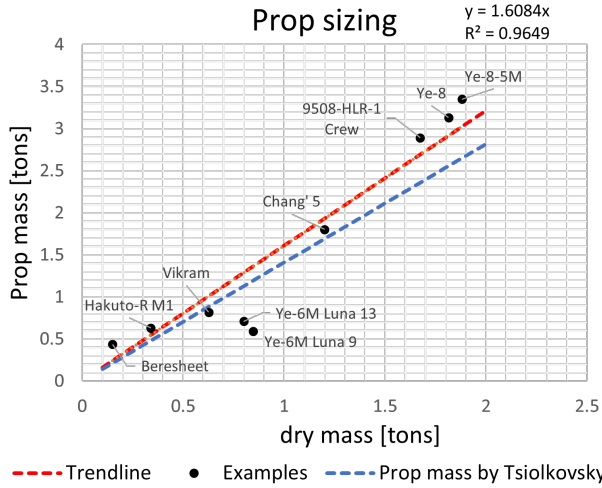3. Set landed payload mass to desired amount.
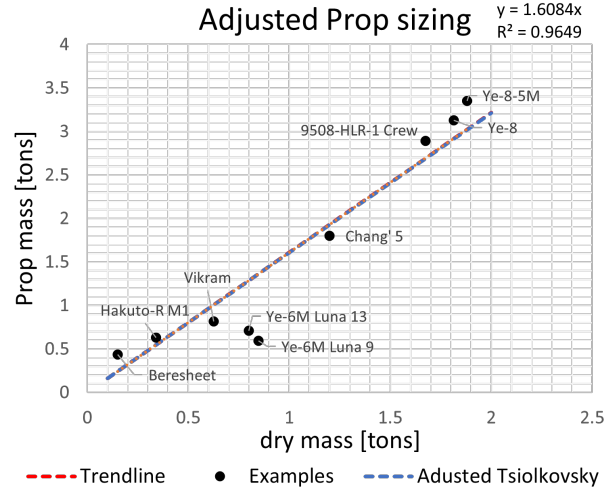
Figure 1: Tsiolkovsky before adjustment

Figure 2: Tsiolkovsky adjusted

4. Size the dry mass of the lander using the relevant sizing rule.

$$m_{dry} = \frac{m_p}{\alpha_1} \tag{8}$$

where α_1 is a coefficient found from the created database (Ranging from 0.3 to 2.2 depending on payload size).

5. Add the payload and the dry mass and use them with the ΔV and a chosen I_{sp} to size the propellant requirement for the Lander.

$$m_{prop} = \alpha_2 \cdot (e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p)) \tag{9}$$

Where α_2 is a coefficient found from the Lander Database. (Usually ranging from 1.6 to 3.69).

6. Use the lander dry mass $m_{dry_{Lander}}$ and sizing rules from the database to size the structure, power, propulsion, avionics, and thermal subsystems.

7. The thrust required to land the total lander mass is:

$$T_{req} = 1.78 \cdot m_{total} \cdot g_{moon} \tag{10}$$

This comes directly from the value used for Apollo missions [42]

8. Use the lander dry mass, $m_{dry_{Lander}}$ or the lander payload mass $m_{p_{Lander}}$ (whichever is larger) as the payload mass to size the transfer vehicle. (It is intended that the lander is placed unfuelled in the correct orbit and the payload is brought later) Use the characteristic energy charts in figure 4 of known transfer vehicles to identify a reference TV. In figure 4 the characteristic energy curves of the transfer vehicles are shifted left by an amount equal to their dry mass such that the payload mass is displayed on the x-axis. The curves are defined by equation 11

$$\Delta V = \sqrt{\frac{2 \cdot E_k}{m_p + m_{dry}}} \tag{11}$$

Where m_{dry} is a constant always greater than zero unique to each TV, E_k is a constant unique to each TV and m_p is the variable of interest. Using the payload mass and the ΔV required for the mission architecture, chose the characteristic energy curve closest to the bottom left corner of the resulting green zone. Set the dry mass of the TV equal to that which has the closest curve. (In the example in figure 4, it would be the DCSS.)

$$m_{dry_{TV}} = m_{dry_{TV_{ref}}} \tag{12}$$

Anything in the green zone would be feasible, but it is best to choose a vehicle which is close to your design point such that there is as little scaling to do as possible.
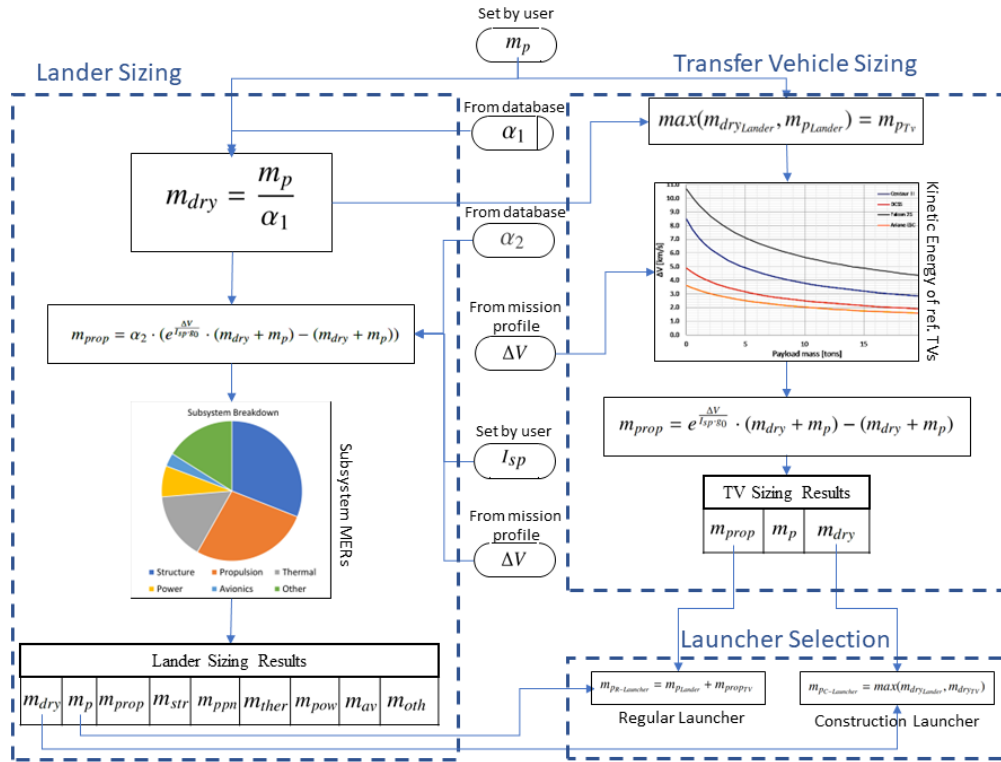
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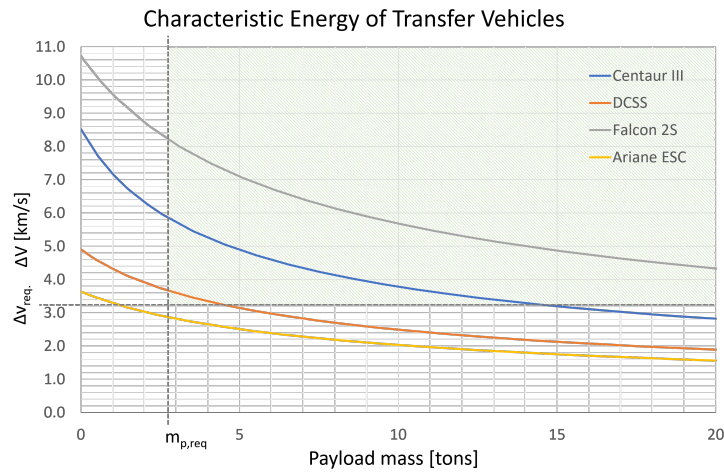**Figure 3:** Flowchart of mission architecture sizing methodology

Figure 4: Constraint diagram for transfer vehicles

9. Add the TV dry mass and the TV payload mass and use Tsiolkovsky's equation to find the propellant requirement.

$$m_{prop} = e^{\frac{\Delta V}{I_{sp}\cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p)$$ (13)

10. Choose the regular launch vehicle by the amount of fuel required by the TV, and the payload mass of the lander. Multiple launches may be necessary to fuel the TV.

11. Choose the construction launch vehicle as the one which can carry the fuel required for the TV(s) for construction and the dry mass of the TV with as little over performance as possible.

3. Results and Discussion

This project defines five mission architectures. They were chosen to cover as much of the lunar transport trade space as possible. The first four, are a one-stage, two-stage, three-stage and electric space-tug concepts. A fifth mission architecture is defined as a benchmark case that drops Operational Capability (OC) 5 and 7 so that it can be a mission architecture similar to the Apollo missions. All five are defined using Bat diagrams. These diagrams are read from left to right. The Earth is at the bottom, the Moon at the top and various orbital destinations in the middle. It describes the concept of operation of a mission architecture.
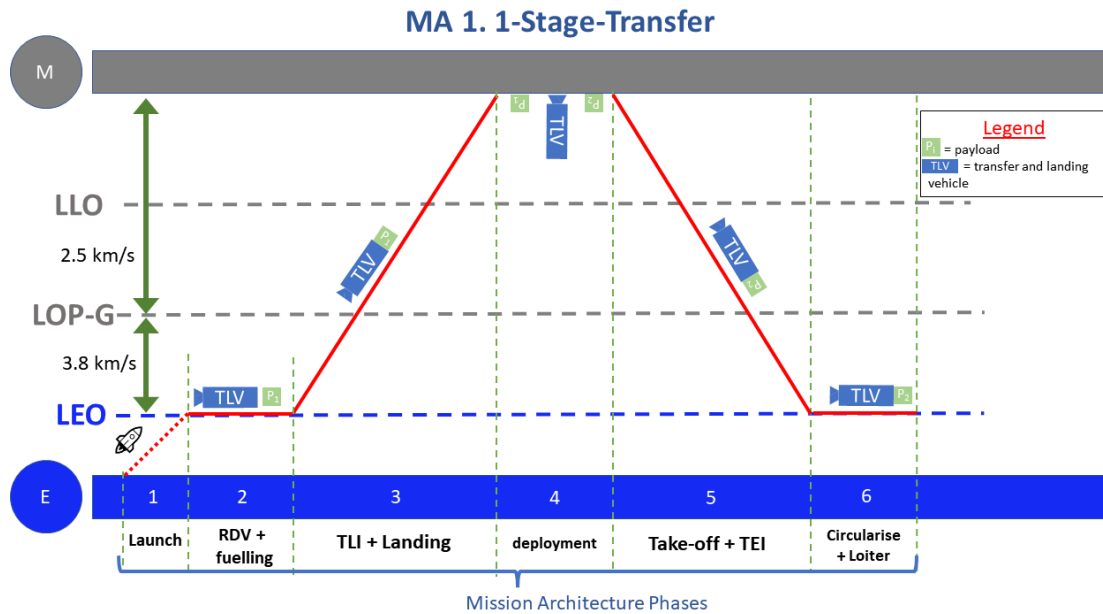
Figure 5: The Magic Carpet Mission Architecture
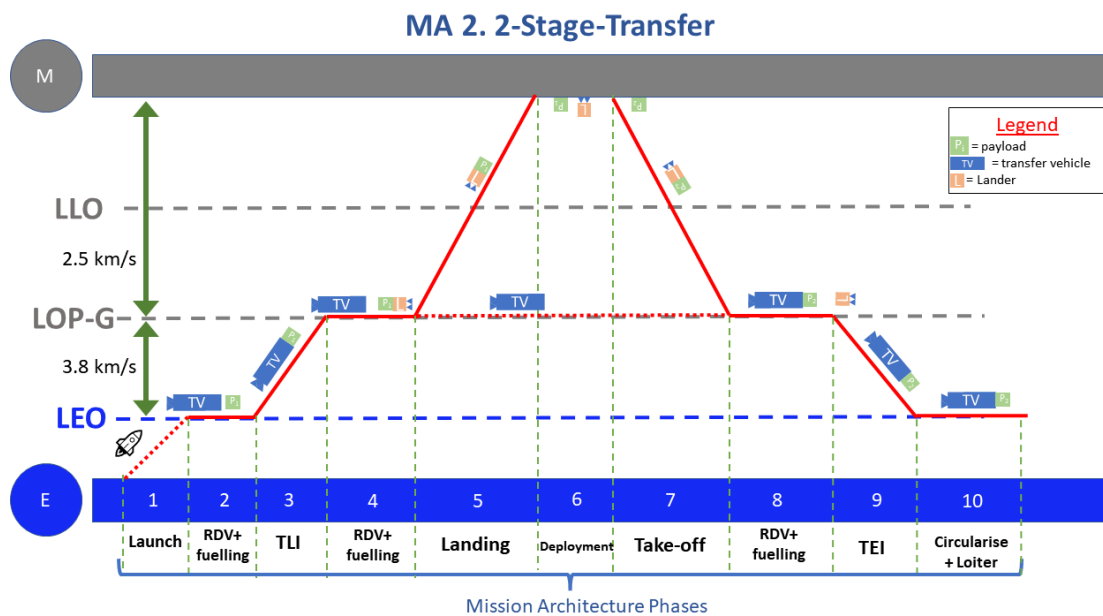
Figure 6: The Space Tug Mission Architecture

The mission architectures (MA-index) are defined as follows (acronyms are defined as they appear):

- In MA-1 (Figure 5), the magic carpet, the transfer landing vehicle is launched unfuelled into LEO. Its operations
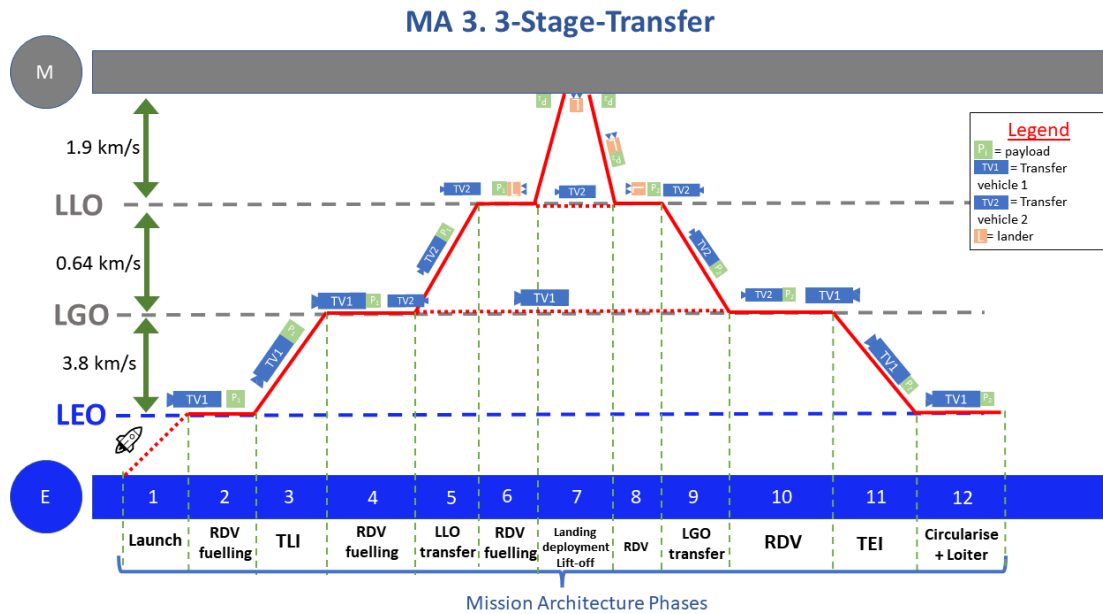
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MA 3. 3-Stage-Transfer

Figure 7: The Double Space Tug Mission Architecture
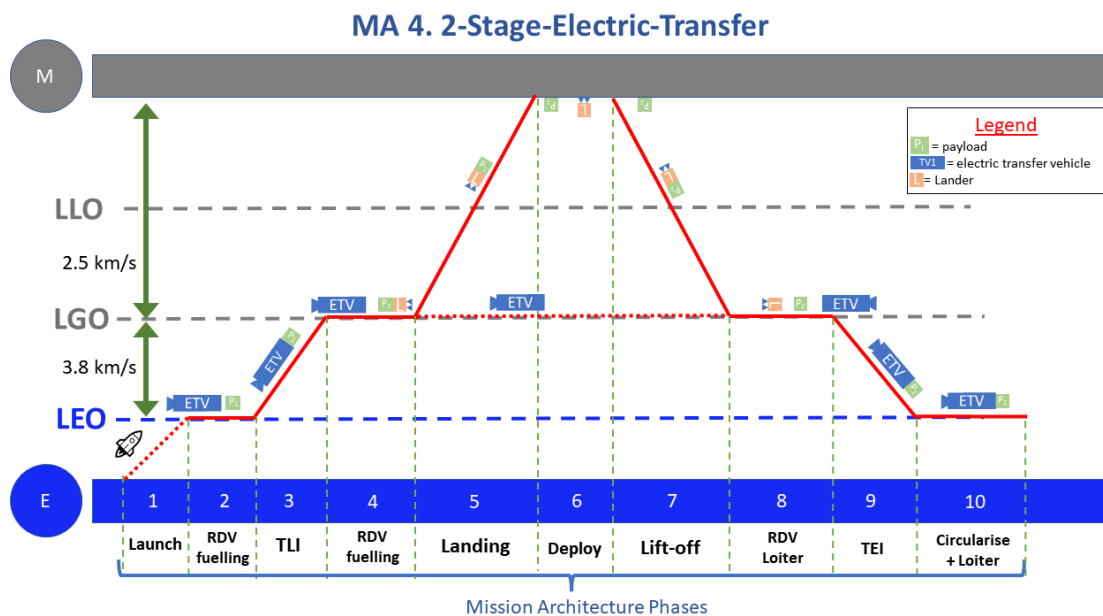
MA 4. 2-Stage-Electric-Transfer

Figure 8: The Electric Space Tug Mission Architecture

per cycle [6] can be summarized as: (i) launch of the payload and fuel; (ii) rendez-vous (RDV) and transfer of the fuel and payload; (iii) translunar injection to a Selenographic trajectory and landing; (iv) deployment of the payload and taking on a return payload; (v) take-off and Trans-Earth injection; (vi.a) circularisation to LEO, (vi.b) payload delivery, (iv.c) loiter for next mission. Named the magic carpet because a single vehicle which can go from LEO to the lunar surface and back again would have to be almost as performant as a magic carpet. But it is still interesting to compare the characteristics of a single stage transfer to two and three stage transfers in the other architectures. It's advantage is that it minimises the number of orbital rendezvous and construction launches.

- MA-2 (Figure 6) Its constructions proceeds as follows: The space tug the transfer Vehicle is launched unfuelled

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[6]One cycle is one round trip from LEO to the moon and back again

MA 5. Non-reusable reference

Figure 9: The Old Fashioned Way
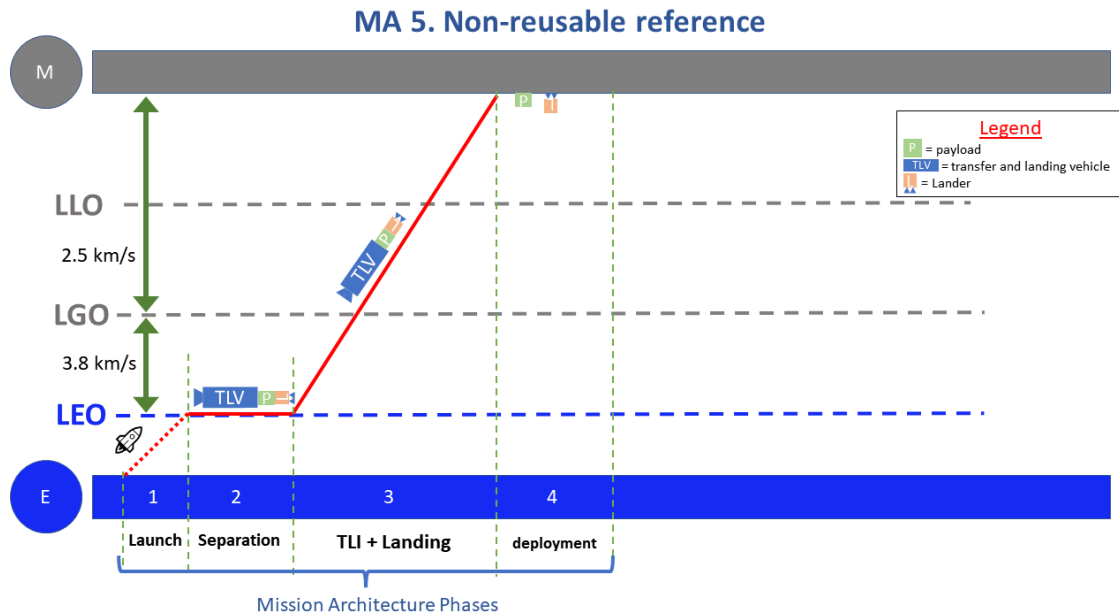
into LEO. Then the lander is launched unfuelled into LEO. Its operations per cycle can be summarized as: (i) payload and fuel launched to LEO; (ii) Rendezvous (RDV) and transfer of the fuel and payload to the Transfer Vehicle (TV); (iii) TV performs Trans-lunar injection (TLI); (iv.a) RDV with Lunar Outpost - Gateway (LOP-G); (iv.b) payload passed to the lander, (iv.c) lander fuelled from LOP-G; (v) Landing; (vi) payload deployment and pick-up; (vii) take-off; (viii.a) RDV with LOP-G, (viii.b) payload passed to TV; (ix) TV performs Trans-Earth Ínjection (TEI); (x) TV arrives in LEO, delivers payload and loiters for next cycle.

- In MA-3 (Figure 7), the double space tug mission architecture. Its constructions proceeds as follows: The Transfer Vehicle 1 (TV1) is launched unfuelled into LEO. Then the Transfer Vehicle 2 (TV2) and fuel for TV1 is launched unfuelled into LEO. TV1 takes TV2 as its payload to LOP-G. TV1 refuels in LOP-G, leaves TV2 and returns to LEO. The lander (unfuelled) and fuel for TV1 is launched into LEO. TV1 then takes the lander as its payload to LOP-G where it is passed to TV2. TV2 fuels at LOP-G and goes to Low Lunar Orbit (LLO) where it leaves the lander and returns to LOP-G. At the end of construction, the TV1 is in LEO, the TV2 is at LOP-G and the lander is in LLO. Its operations per cycle can be summarized as: (i) payload and fuel for TV1 are launched into LEO; (ii) TV1 RDVs with launcher and takes on fuel and payload; (iii) TV1 performs TLI to LOP-G; (iv) TV1 RDVs with TV2 at LOP-G; (v) TV2 takes the payload and fuel for the lander to LLO; (vi) TV2 RDVs with the lander and passes fuel and payload; (vii.a) Landing, (vii.b) payload deployment and taking on new payload, (vii.c) take-off; (viii) Lander RDV in LLO with TV2 and passes payload; (ix) TV2 takes payload to LOP-G; (x) TV2 RDV with LOP-G and payload passed to TV1. TV1 fuelled at LOP-G; (xi) TV1 performs TEI; (xii.a) TV1 arrives in LEO, (xii.b) delivers payload and loiters for next mission.

- In MA-4 (Figure 8), the electric space tug is similar to the MA-2 but it employs electric propulsion. Its operations per cycle can be summarized as: (i) Payload and fuel launched to LEO; (ii) RDV and transfer of the fuel and payload to TV1; (iii) ETV performs TLI; (iv) ETV RDVs with Lander at LOP-G and refuels; (v) landing; (vi) payload deployment and pickup; (vii) take-off; (viii) lander RDVs with ETV at LOP-G and passes payload; (ix) ETV takes payload to TEI; (x.a) ETV circularises to LEO, (x.b) delivers payload and loiters for next mission.

- In MA-5 (Figure 9), the old-fashioned way, no construction is necessary. All systems are launched fully fuelled on the first launch. The Old Fashioned Way as the name implies has a similar mission architecture to almost all unmanned lunar landers in the past: (i) all Systems launched into LEO; (ii) TV-Lander-Payload stack separate from launcher; (iii.a) TV performs TLI onto selenographic trajectory, (iii.b) TV separates from lander, (iii.c) Landing; (iv) the lander deploys its payload.

These mission architectures have been sized using the sizing rules derived primarily from the database. These sizing rules are shown in Table 7.

Major mass estimation relationships		
Lander Class	eqaution	R^2
Small: $m_p < 2t$	$m_p = m_p$	1
	$m_{dry} = 3.011 \cdot m_p$	0.826
	$m_{prop} = 1.14 \cdot (e^{\frac{\Delta V}{I_{sp}g}}[m_{dry} + m_p] - [m_{dry} + m_p])$	0.9649
Medium: $2t < m_p < 5t$	$m_p = m_p$	1
	$m_{dry} = 0.494 \cdot m_p$	0.9874
	$m_{prop} = 1.11 \cdot (e^{\frac{\Delta V}{I_{sp}g}}[m_{dry} + m_p] - [m_{dry} + m_p])$	0.9947
Large: $m_p > 5t$	$m_p = m_p$	1
	$m_{dry} = 0.4585 \cdot m_p$	0.8151
	$m_{prop} = 0.95 \cdot (e^{\frac{\Delta V}{I_{sp}g}}[m_{dry} + m_p] - [m_{dry} + m_p])$	0.9378

Table 7: Collected sizing rules for lunar landers

These sizing rules are used to size the lander of all the mission architectures. The other systems in the mission architectures are sized and costed according to the methods described in section 2.1. Each mission architecture has the following mass breakdown.

Mission Architecture Sizing Results - 2t case									
Mission Architecture	System	reference vehicle	mtotal [kg]	mdry [kg]	mp [kg]	mprop [kg]	Ec [J]	Prop	Isp [s]
MA 1	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	TLV	bespoke	59,338	6,022	2,000	51,316	3.29E+11	LOX/LH2	450
MA 2	Construction Launcher	Atlas V 551							
	Regular Launcher	Atlas V 551							
	TV	Common Centaur	22,189	3,490	5,910	12,789	6.79E+10	LOX/CH4	451
	Lander	bespoke	23,214	1,963	1,963	15,341	9.84E+10	LOX/LH2	450
MA 3	Construction Launcher	Atlas V 551							
	Regular Launcher	Falcon 9							
	TV1	Ariane ESC	25,011	4,540	6,056	14,416	7.65E+10	LOX/CH4	348
	TV2	Falcon 2S - scaled	11,934	2,140	6,056	3,739	6.71E+09	LOX/CH4	348
	Lander	bespoke	18,124	6,056	2,011	10,057	5.82E+10	LOX/LH2	450
MA 4	Launcher	Anteres							
	ETV	Gateway PPE	15,210	7,230	6,015	1,965	9.56E+10	Xenon	2,800
	Lander	bespoke	23,627	6,015	1,998	15,614	1.00E+11	LOX/LH2	450
MA 5	Launcher	Falcon Heavy							
	TV	falcon 2s	114570	3900	18000	92670	1.58E+11	LOX/CH4	348
	Lander	bespoke	18000	6021	2000	7664	2.51E+10	LOX/LH2	450

Table 8: Sizing results for the 2t case

With the mission architecture sized, the other Figures of Merit can be evaluated for each MA.

In order to perform a trade-off on the mission architectures, we must first define which of the figures of merit are the most important. We do this with the pairwise comparison in Table 10 following the guidelines of Ref. [12]. This analysis, shows clearly that the cost-related FOMS are the most important in this project. Due to their very low score, system flexibility and extensibility are excluded from further analysis.

In the maximalist case, the upper bounds for the deliverable payload to the lunar surface are calculated. 6.3t is the maximum deliverable payload using modern launcher capabilities in an Apollo style architecture as in MA5 - The Old Fashioned Way. [7]

Compared to the original Apollo descent stage which landed a payload (the fully fuelled ascent stage) of 4.4t-4.8t, this is an improvement. Compared to the other mission architectures however, its performance is poor.

Using a two stage electric transfer as in MA4 is the most performant in terms of mass delivered. The high efficiency of electric propulsion allows Up to 62t of payload to be delivered. The required lander however, has a dry mass larger than any other lander in the project database by a factor of 2. This may make this mission architecture unfeasible.

MA2 and MA3 show more reasonable estimates of the upper bound of mass deliverable to the lunar surface by a re-usable lunar lander: 36t. This is a similar payload mass to the second largest lander in the database: 9205-FLO-1 Crew (payload 36t), and more than the latest lunar lander design: Blue Moon, a lander proposed by Blue origin for the Artemis 5 mission has a projected payload mass of 20t [43]. Note that MA3 gains no improvement in payload mass delivered from having a second space tug stage. This shows that the limiting factor on the upper bound of deliverable payload is the payload of the first transfer vehicle. MA5 has such a high payload because of its efficient transfer vehicle.

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[7]Note that SLS has been excluded from consideration from this paper about low-cost lunar landers because of its gargantuan cost.

Mission Architecture Sizing Results - Maximalist Case									
Mission Architecture	System	reference vehice	mtotal [kg]	mdry [kg]	mp [kg]	mprop [kg]	Ec [J]	Prop	Isp [s]
MA1	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	TLV	bespoke	688,293	29,254	63,800	595,239	7.64E+12	LOX/LH2	450
MA2	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	TV	Falcon 2s	123,101	3,900	36,543	82,657	2.92E+11	LOX/CH4	348
	Lander	bespoke	157,159	36,543	16,756	103,860	6.66E+11	LOX/LH2	450
MA3	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	TV1	Falcon 2s	123,101	3,900	36,543	82,657	2.92E+11	LOX/CH4	348
	TV2	Falcon 2s	55,858	1,849	36,543	17,465	3.15E+10	LOX/CH4	348
	Lander	bespoke	119,753	16,756	36,543	66,453	3.85E+11	LOX/LH2	450
MA4	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	ETV	Gateway PPE (scaled)	80,057	7,230	62,484	10,343	5.03E+11	Xenon	2800
	Lander	bespoke	268,720	28,650	62,484	177,586	1.14E+12	LOX/LH2	450
MA5	Construction Launcher	Falcon Heavy							
	Regular Launcher	Falcon Heavy							
	TV	Falcon 2s	117,570	3,900	21,000	92,670	1.8E+11	LOX/CH4	348
	Lander	bespoke	21,000	3,116	6,311	9,008	2.95E+10	LOX/LH2	450

Table 9: Sizing results for the maximalist case

FOM Raw Values - Maximalist case					
MA	Cost [$/kg_landed]	Cost [$/kg_returned]	Landed_Payload [kg]	Returned_payload [kg]	Delivery_Time [days]	System_complexity [#rdvs]
MA1	26,567	26,567	5,800	5,800	6.44	22
MA2	8,620	8,620	18,272	18,272	5.91	8
MA3	8,209	8,209	18,272	18,272	6.21	11
MA4	4,801	4,801	62,484	62,484	270.68	8
MA5	23,768	NA	6,311	NA	3.31	1

Table 10: Results of evaluations of the Figures of Merit for the maximalist case

FOM scores - Maximalist case							
MA	Cost_landed	Cost_lifted	Landed_Payload	Returned_payload	Delivery_Time	System_complexity	Total Scores
weightings	0.27	0.15	0.19	0.15	0.12	0.12	1.00
1-stage Transfer	0.00	0.00	1.78	1.43	11.25	0.00	14.45
2-stage Transfer	18.16	10.38	5.61	4.49	11.27	7.33	57.24
3-stage Transfer	18.58	10.61	5.61	4.49	11.26	5.76	56.31
2-stage Electric Transfer	22.02	12.58	19.20	15.36	0.00	7.33	76.50
Old Fashioned Way	2.83	0.00	1.94	0.00	11.38	11.00	27.15

Table 11: Trade-off results for the maximalist case

The way to increase the payload mass of MA2 would be to increase the characteristic energy of the vehicle. Once that is done, the designer may run into size issues on the lander design due to the structural loads induced by carrying large volumes of liquid fuel during the landing phase. It is worth noting that the specific costs of the maximalist case are *much lower* than those of the 2t case.

In the two ton case, the MA 2 is the clear winner. This paper will proceed to investigate the sizing and costing of this architecture in more detail. In particular, the unmanned lander is designed to interface with this architecture. It is worth noting before continuing that mission architecture five, the non-reusable Apollo style benchmark, is the worst performing in terms of the cost even before you include the unit costs of 60 landers for 60 payload deliveries. For subsystem sizing of the lunar lander used in the chosen mission architecture, see Table 14 and Figure 11. Since in the literature, most landers with a subsystem mass breakdown have a name. This lander shall be named as well. The lander shall be known as the ORLA Lander. Derived from the Irish Gaelic name Órfhlaith, meaning golden princess.

Table 15 and Figure 12 present the first cost breakdown of the ORLA Lander. Development cost is the development cost of the ORLA lander only. Although they would be smaller due to the higher number of flight models of the reference TV in service, The development costs of the TV for mission architecture 2 would be interesting to consider also. The production cost is the cost of producing three lander models. This takes into account the learning curve in manufacturing cost. the third lander is 43% the cost of the first. The steady upwards trend of the operations cost after 2027 is due to inflation. Assumed to be 3.9%. The two spikes in cost in 2027 and 2032 are the construction costs of replacing the TV and lander after they have been re-used 20 times each. These spikes can be attenuated if the operator

FOMs raw values - 2t					
MA	Cost [$/kg_landed]	Cost [$/kg_returned]	Landed_Payload [kg]	Returned_payload [kg]	Delivery_Time [days]	System_complexity [#rdvs]
MA1	63,020.28	63,020.28	2,000.00	2,000.00	5.13	3
MA2	23,167.75	23,167.75	1,962.71	1,962.71	5.90	8
MA3	24,986.28	24,986.28	2,011.05	2,011.05	6.19	11
MA4	22,854.27	22,854.27	1,997.63	1,997.63	270.68	8
MA5	64,449.44	0.00	1,999.57	0.00	3.31	1

Table 12: Results of evaluations of the Figures of Merit for the 2t case

FOM scores - 2t case							
MA	Cost_landed	Cost_lifted	Landed_Payload	Returned_payload	Delivery_Time	System_complexity	Total Scores
weightings	0.27	0.15	0.19	0.15	0.12	0.12	1.00
1-stage Transfer	0.60	0.00	19.10	15.28	11.30	8.38	54.65
2-stage Transfer	17.22	9.71	18.74	14.99	11.27	3.14	75.08
3-stage Transfer	16.46	9.27	19.20	15.36	11.26	0.00	71.55
2-stage Electric Transfer	17.35	9.79	19.07	15.26	0.00	3.14	64.61
Old Fashioned Way	0.00	0.00	19.09	0.00	11.38	10.47	40.95

Table 13: Trade-off results for the 2t case

| FOMs weighting	Cost of landed payload	Cost of returned payload	Landed Payload	Returned payload	Delivery Time	System complexity	Flexibility	Total+1	Weighting
Cost of landed payload		1	1	1	1	1	1	7	0.25
Cost of returned payload	0		0	0	1	1	1	4	0.14
Landed Payload	0	1		1	0	1	1	5	0.18
Returned payload	0	1	0		0	1	1	4	0.14
Delivery Time	0	0	1	1		0	0	3	0.11
System complexity	0	0	0	0	1		1	3	0.11
Flexibility	0	0	0	0	1	0		2	0.07

Figure 10: Pairwise comparison of the seven FOMs

is willing to accept an interruption of cargo delivery service for one year. The TruePlanning project file is available on the project GitHub [8]. In comparison to other lunar lander designs, the project cost of this lander are low. The study in [19] has a combined development and unit production cost of $784 million. This is on the same order of magnitude as the same for the ORLA lander (DDTE +unit cost = 1379$ million). However on a per-kg basis, with a dry mass of 1010kg the lander in Ref. [19] is in fact three times more expensive ($776,000/kg for Ref. [19] versus $230,000/kg for ORLA). The Gryphon lander in [30] has a combined development and unit production cost of $6.5 Billion. Much more expensive. Because it is much heavier at 6,539kg. per kg that is 990,000$/kg which is also more expensive than the ORLA design. For reference, the program cost per kg of ISS structure is $\frac{150 Billion}{420t} = 350,000\$/kg$. We can say then that our cost estimations for the ORLA lander and the mission architecture 2 are, generally speaking, lower than the state of the art. Dimensioned conceptual drawings of the ORLA Lander are available in the appendix.

4. Conclusion and Future Work

This project has explored the upper bounds of lunar lander payload capacity and the cost drivers of lunar cargo delivery. Five mission architectures were assessed based on seven figures of merit, and a trade-off analysis was conducted for two cases: A maximalist case where everything was sized to maximise delivered payload, and a 2t case where everything

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[8] https://github.com/Conall98/ORLA_Lander_Mission_Architecture
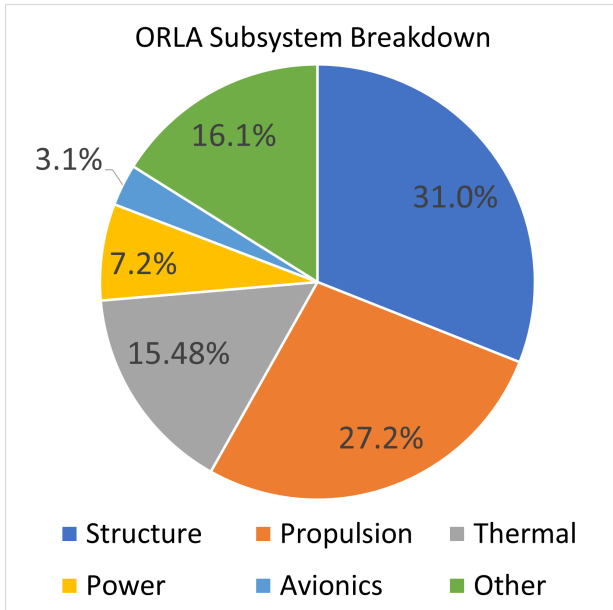
Figure 11: Mass breakdown of the lunar lander in mission architecture 2

ORLA Lander Summary		
Property	unit	value
Structure	[kg]	1832
Propulsion	[kg]	1605
Thermal	[kg]	915
Power	[kg]	424
Avionics	[kg]	183
Other	[kg]	951
total dry mass	[kg]	5910
Propellant (LOX/LH2)	[kg]	15340
payload	[kg]	1962
total mass	[kg]	24394
I_{sp}	[s]	450
ΔV	[m/s]	5000
Thrust	[kN]	66.9
Power	[kW]	10.8
Development cost	[$]	731,000,000
Unit Production Cost (3)	[$]	256,000,000
Delivery Cost w/ MA 2	[$/kg]	38,054

Table 14: Design Summary of ORLA Lander
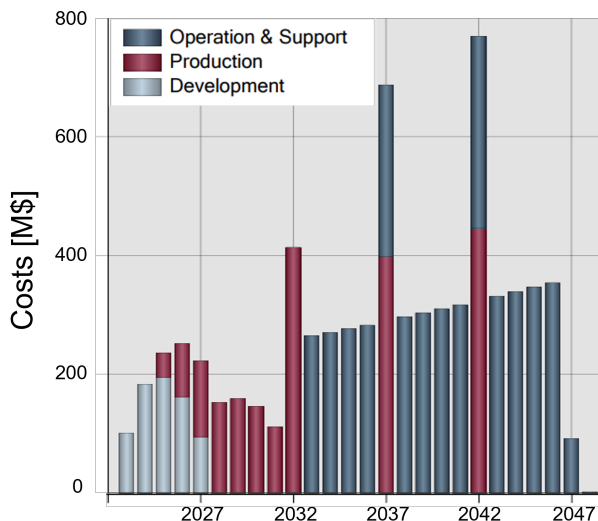
Figure 12: Costs distribution from 2023 to 2047.

ORLA Lander Summary		
Cost	unit	Total
Development	[B$]	0.7
Production	[B$]	2.1
Operations and Support	[B$]	4.4
Total	[B$]	7.2

Table 15: General cost results of the ORLA Lander

was sized to deliver 2t of cargo as cheaply as possible.

It was found that the main cost drivers for lunar cargo delivery are fuel carriage and launcher costs, and potential solutions to reducing these costs include the use of orbital fuel depots in LEO and at the Lunar Gateway, as well as ridesharing on very large launchers. Additionally, the project highlights the importance of considering the overall space mission architecture when designing a lunar lander.

The winning mission architecture was MA2: 2-stage Transfer. A reusable space tug, and reusable lander design capable of carrying 2 tons of cargo down to the lunar surface and back up again. The mission architecture was sized using statistical sizing rules derived from a database of lunar landers and transfer vehicles compiled from an extensive literature review. Its life cycle costs were evaluated using TruePlanning software. The lander was named ORLA.

Further development should focus on: automating the sizing of mission architectures and use more advanced iterative sizing methods; A TRL study of the subsystems of the mission architectures along with cost estimations of the technology development; calculation of the payback period of a reusable lunar lander would be a useful exercise; Equipment layout concepts and drawings of the ORLA Lander.

LOW-COST UNCREWED LUNAR LANDERS

Acknowledgements

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5. Appendix

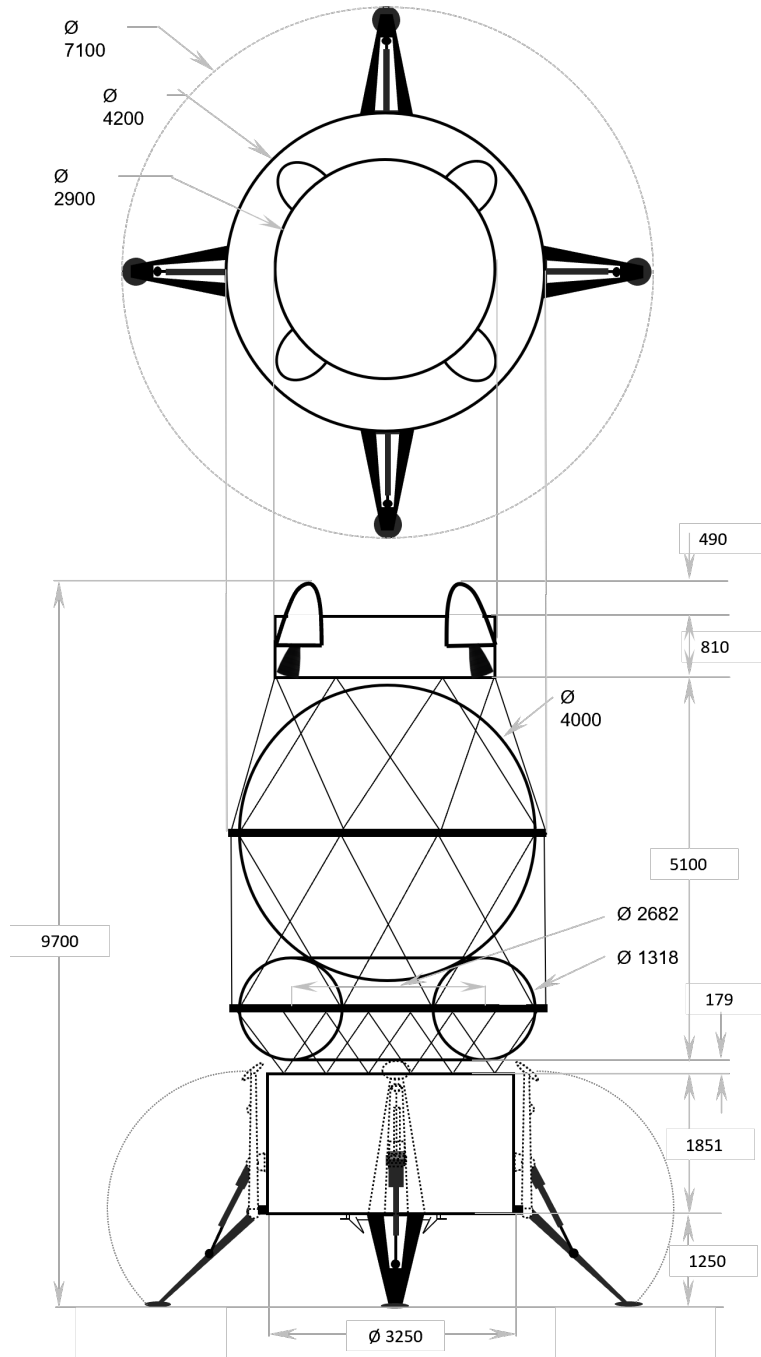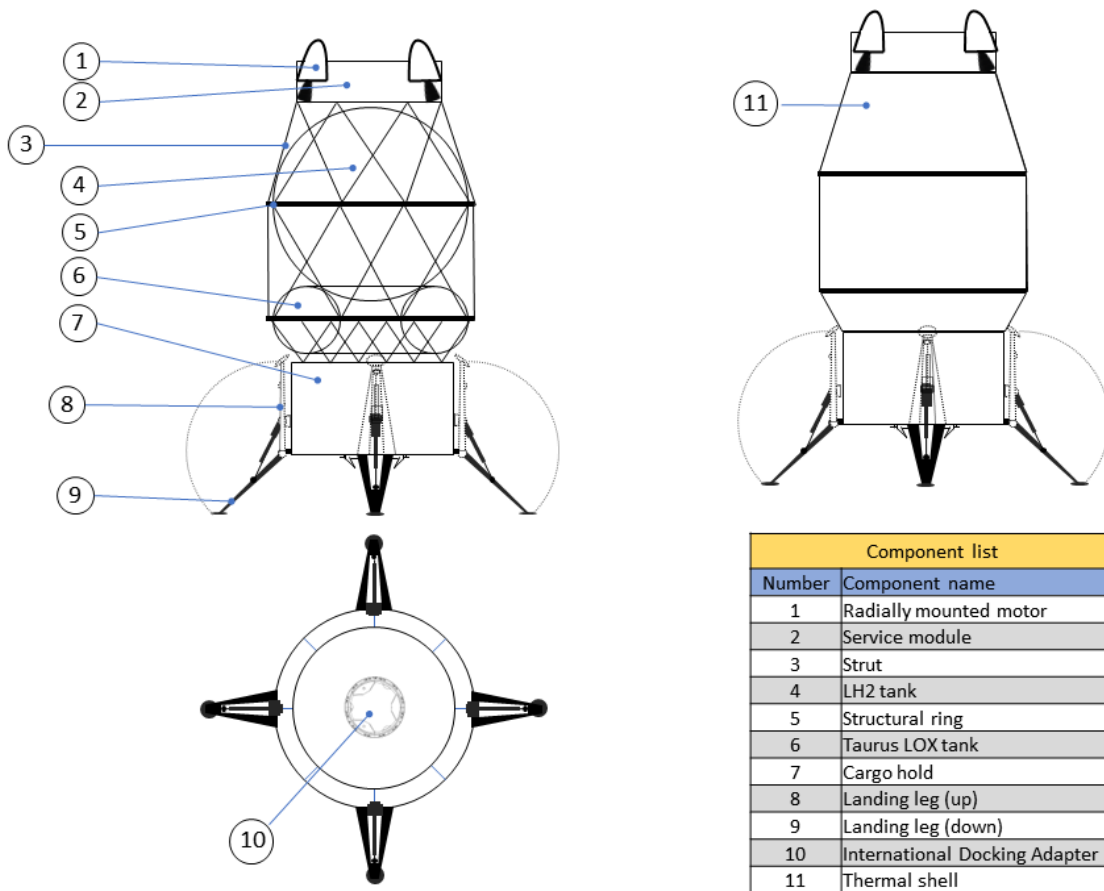### 5.1 Conceptual Drawings of the ORLA Lander

Figure 13: Drawing of ORLA Lander. Dimensions in mm

Component list	
Number	Component name
1	Radially mounted motor
2	Service module
3	Strut
4	LH2 tank
5	Structural ring
6	Taurus LOX tank
7	Cargo hold
8	Landing leg (up)
9	Landing leg (down)
10	International Docking Adapter
11	Thermal shell

Figure 14: ORLA Lander Components