Research Project Report

Low-Cost Uncrewed Lunar Landers: Mission Design and Preliminary Feasability Assesment

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Contents

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 recent years 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 the cost of space transport is 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 5 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 architectures. 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. This database includes every model of spacecraft to land on the moon. A sizing and costing exercise for a 2-ton payload delivery case is carried out. A trade-off is then performed based on 7 figures of merit: Landed Payload [kg/trip] Returned Payload [kg/trip], Cost of Landed Payload [$/kg], Cost of Returned Payload [$/kg], Delivery Time [days], and System Complexity. The most cost effective and suitable mission architecture is chosen and a lander design which can carry 2-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 was 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 and ridesharing on very large launchers.

Introduction

In order to design a lunar lander one must give due regard to the overall space mission architecture. The choice of launcher, the design of transfer vehicle, the availability of on-orbit refuelling to the choice of trajectory and reusability requirements, all of these impose size and performance limits on lunar landers. Therefore, although the lunar lander is the vehicle of primary interest, this project has designed and analysed whole mission architectures including several vehicles to answer two questions:

- What are the upper bounds of payload mass deliverable to the lunar surface using current technologies?
- What is the specific cost [$/kg] to deliver and to return payload from the lunar surface using current technologies?

To tackle these questions, this project builds upon work carried out last year in *Phase-A Sizing Tool for Systems of Systems for Planetary Exploration (C.de Paor, et al.).* Namely the database of lunar landers compiled last year is expanded significantly to include more examples, transfer vehicles, ΔV and I_{sp} data. From this, new parametric and statistical sizing rules are derived.

These sizing rules are then applied to five alternative mission architectures for lunar transport is the mission concept plus a definition of each of the elements of the mission [1]. In this project mission architectures for lunar transport are a diverse set of ways to deliver and return payload to the lunar surface. They include concepts for space-tugs, on-orbit refuelling, and re-usability. The five mission architectures are described in detail in Chapter 2.

Each mission architecture is evaluated on 7 Figures of Merit given a two ton delivery case study:

- Landed Payload [kg/trip]
- Returned Payload [kg/trip],
- Cost of Landed Payload [$/kg],
- Cost of Returned Payload [$/kg],
- Delivery Time [days]
- Complexity
- Extensibility

With the Figures of Merit, a weighted trade-off analysis for a case study where the lander carries two tons of payload is carried out to find which mission architecture should be focused on for a detailed lander design. A subsystem sizing exercise is carried for a lander which would be a part of the winning architecture, using statistical sizing rules from the database. The development, production, and operation cost of the lander design is the evaluated using TruePlanning software.

The lander design is then named and presented as part of the winning mission architecture and is proposed as a candidate cost effective architecture for lunar cargo delivery.

1 State of the Art

1.1 Project context

This project started in February 2022 studying Phase-A sizing of Lunar Space Exploration Systems. Until September 2022, work was done in the area of sizing lunar landers and lunar rovers for a lunar lava-tubes mission case study. The sizing was parametric and statistical and involved building databases of spacecraft and gathering parametric sizing rules. The work of the project up to September was presented at IAC 2022 in Paris in paper entitled *An Integrated Design Platform to Analyse and Size Planetary Exploration Systems Applied to Lunar Lava Tube Exploration*[2]. In September 2022, the work of this project shifted to two main foci:

- The sizing of lunar landers and space mission architectures for lunar transport. The landers databases from before September were expanded from 25 to 43 spacecraft, and the quality of the data was improved by careful literature review.

- Costing of space mission architectures and lunar lander designs in order to find the specific cost per kilogram of delivering cargo to the moon and bringing it back again. Using top-down cost estimation methods and parametric costing software, TruePlanning.

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1.2 Spacecraft Databases

Aircraft design benefits from a wealth of data to draw on when designing new aircraft. For lunar landers however, there are fewer flight models to learn from. In any statistical study of lunar landers, one must draw on well documented conceptual designs such as those in [3]. Previous work has been done on databases of lunar landers. A large database of over 100 spacecraft was constructed by M. Isaji at Georgia Tech in 2018[4]. However, only the metadata for this database was published. In the construction of it's database this project quoted from a wide range of sources including history books, text books, catalogs, original data sheets and user manuals. Some data had to be inferred from the mission profile of certain spacecraft. (inferring ΔV capabilities from orbital characteristics for example.) Some of the stand-out sources which gave good data on a wide variety of landers were

- *Soviet Robots in the Solar System* by W. T. Huntress and *Soviet and Russian Lunar Exploration* by B. Harvey were the authoritative sources on the 7 soviet lunar landers included in the database. They are exhaustively detailed and have interesting figures. Their focus on unmanned landers suited the interest of the project [5], [6].
- *After LM: NASA Lunar Lander Concepts Beyond Apollo* by edited by J.F. Conolly and *APOLLO BY THE NUMBERS A statistical reference for the manned phase of Project Apollo* by R,W, Orloff were where the data for the Apollo landers and many other large manned lander concepts were sourced. In order to compare the manned lander data to that of unmanned landers, some parsing of the data was necessary such as only including descent stages and considering the ascent stage to be the payload. *After LM* Has a very large amount of subsystem mass breakdowns for dozens of landers. It was from this source that all of the subsystem sizing rules were derived.
- *The NASA Space Science Data Coordinated Archive* and other web based sources such as *Gunter's Space Page*, *Spaceflight.com* and *Astronautix* often had information on systems which was not available in text books. For example, data was scant for new models of Chinese lunar landers, but could be found, albeit un-cited, on websites such as these [7][8], [9].

1.3 Space Mission Design

Space mission design involves the development of a plan and strategy to design, build, launch, and operate a spacecraft for a specific goal. This process includes mission planning, requirements analysis, system architecture, design optimization, and operations and maintenance. The use of systems engineering methods is important in ensuring the mission is effectively and safely designed and operated. "Space Mission Analysis and Design" by Wertz and Larson provides comprehensive insights into space mission design and the application of systems engineering methods in this field. According to Wertz and Larson there are 8 steps in designing a space mission.

1. Defining mission objectives: identifying the scientific or engineering goals of the mission and the requirements for achieving them.
2. Conceptual design: developing a high-level plan for the mission, including the spacecraft architecture, launch vehicle, trajectory, and mission operations.
3. Requirements analysis: breaking down the mission objectives into specific functional and performance requirements for the spacecraft, instruments, and ground systems.
4. System architecture: developing a detailed system architecture for the spacecraft and ground systems, including the selection of subsystems, components, and interfaces.
5. Design optimization: analyzing the trade-offs between different design options to optimize the mission performance, cost, and risk.
6. Verification and validation:
7. Operations and maintenance: .
8. Decommissioning: safely disposing of the spacecraft.

This project focused on steps one to five for designing mission architectures for lunar transport. In particular, it focused on the space segment as this is an interesting research area from a cost-engineering perspective.

1.4 Cost Engineering

Cost engineering is the practice managing project cost through good cost estimation and cost control. It is a crucial element of space mission design. Without good cost estimates, worst of all, cost underestimation, a project will be doomed to never leave the ground. In this project, parametric cost estimation was carried out using TruePlanning. This software uses databases of component costs and historical patterns in project costs, to give reasonable cost estimates. It has been used to cost lunar systems for Telespazio [10], lunar space stations at the Space Station Design Workshop and several other significant aircraft and spacecraft development projects [11].

1.5 System Engineering Definitions

Before any work on mission architecture design or database building, this project's working definitions of systems engineering terms must be clearly described. They are based on system levels described in the NASA System Engineering Handbook. and the Space Mission Analysis and Design Textbook.

System
The combination of subsystems that function together to meet a need of the mission architecture. For example: Lander, Transfer vehicle. Based on the definition of System in the NASA System Engineering Handbook.

Subsystem
A combination of Components that function together to produce a capability to meet a need of the System.

Components
The devices/hardware which make up the subsystems of the spacecraft.

System Drivers The principal mission parameters which influence performance cost, risk, schedule and which the user can control. This comes directly from the Space Mission Analysis and Design Textbook.

2 Method and Materials

In this section the methodologies employed at each stage of the project are defined namely:

- Definitions and Constraints
- Database Definition and Sizing Rule Derivation
- Mission Architecture Definition and Evaluation
- Trade off analyses.

2.1 Functional Analysis

With these definitions. We can now tackle the mission architecture design. The overall function of all 5 mission architectures examined in this project is:

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

From this mission objective, We can derive seven operational capabilities for the unmanned mission architectures. These are high level functions derived directly from the Mission Objective.

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 Objective

Some of these operational functions warrant some explanation. OC5: to be reusable, and OC7: to Perform on-orbit refuelling were chosen to address the "cost effective" part of the mission statement. The effect of these two capabilities on the cost of the mission architectures will be illustrated by comparing them to a non-reusable Apollo style benchmark case.

2.2 Definition of Databases

The databases of lunar landers, transfer vehicles and launcher costs were constructed in order to find statistical sizing rules for vehicles which can fulfil the operational capabilities (OCs) for the mission architectures.

2.2.1 Lunar Lander Database

The database of lunar landers contains every model of spacecraft (manned and unmanned) which has landed softly on the moon since Luna 9 in 1966. It also contains several spacecraft which were launched but failed to land softly, and several conceptual lander designs. Table 2 shows a sample of 6 entries in the database. It is available in full in the Appendix.

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	[5] [12] [13]
1969	USSR	Ye-8-5M Luna 16	5750	1880	520	3350	HNO3/UDMH	314	1880	2stage	[5] [12] [13]
1969	NASA	Apollo 12	15065	2034	4819	8212	N2O4/AZ49	311	2273	2stage	[14]
1971	NASA	Apollo 15	16447	2626	4795	9026	N2O4/AZ46	311	2250	2stage	[14]
2007	NASA	Gryphon	43501	8500	18634	16367	LOX/LH2	451	2117	1stage	[15]
2021	CNSA	Chang' 5	3800	1200	800	1800	N2O4/UDMH	333	1930	2stage	[16] [17]

Table 2: Sample of 6 rows of lunar landers from the project database

It is important to define exactly what is meant by all of the quantities attributed to each spacecraft. The total mass m_0 in kg is defined as:

$$m_0 = m_{dry} + m_p + m_{prop} \tag{1}$$

Where m_{dry} **is the dry mass of the system**. For one-stage landers this 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.

m_{prop} **is the propellant mass.**

m_p **is the bloc payload mass**. The bloc payload mass is an invented quantity to maintain comparability between one stage *unmanned* lander and two stage *manned* landers. For one-stage *unmanned* landers, the bloc payload is equivalent to the traditional definition of payload in *Space Mission Analysis and Design*: "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)" For two stage landers however, the payload mass is considered to be the total mass of the ascent stage.

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 it's mission profile. For two stage landers, only the ΔV of the descent stage(s) is counted.

Type is the classification of lander. the three types are 1-stage, 2-stage, lander.

The different definitions of m_{dry}, and m_p for one and two stage landers are necessary to maintain comparability between a many various types of landers. By defining all of characteristics of the landers in the database to be comparable with one another, this report is in fact comparing all of the landers to a reference lander 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.

2.2.2 Transfer Vehicle Database

Data was gathered from literature on several existing upper stages to derive sizing rules for space-tug-like transfer vehicles (TVs). Table 3 below shows the TV database for chemically powered vehicles. Another database was constructed for electrically power vehicles.

Characteristic energy of Upper Stages							
Vehicle name	Dry mass [kg]	propellant mass [kg]	reference payload [kg]	reference ΔV [m/s]	Characteristic Energy [J]	Propellant	sources
DCSS 5m	3,490	27,200	22977	1910.00	4.83E+10	LOX/LH2	[18]
Common Centaur	2462	20830	18850	3076.00	1.01E+11	LOX/LH2	[19]
Falcon 2S	3900	92670	15600	5361.00	2.80E+11	LOX/RP1	[20] [21]
Ariane 5 2S	4540	14700	10865	2350	4.25E+10	LOX/LH2	[22] [23]

Table 3: Transfer vehicles database

The reference payload and the reference ΔV are found by watching the telemetry feeds of launch broadcasts on YouTube.

2.2.3 Launcher cost database.

To perform cost estimation, a database of launch costs was compiled from literature. A sample of the launcher database is shown in Table 4

Launcher Database				
Launcher	Per Launch cost	Payload to LEO	$/kg	sources
Falcon Heavy	150,000,000	63800	2,351	[24]
Atlas V 551	73,000,000	18831	3,877	[25]
Ariane 6 A64	115,000,000	21650	5,312	[26]
Falcon 9 FT	67,000,000	22800	2,939	[24]
SLS B1	2,000,000,000	95000	21,053	[27]
Vega	37,000,000	3300	11,212	[28]
Electron	5,300,000	220	24,091	[29]

Table 4: Launch Costs Database

2.2.4 Lunar Lander Subsystems

To size the subsystems of the lander, consistent and coherent subsystem mass breakdowns of several examples are required. One source in particular has a lot of subsystem mass data: The Beyond Apollo Catalog. It is the source of all the systems in Table 5.

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
9205-FLO-1	9320	1770	4969	385	307	420	1469
0503-CE&R-8	5858	2822	1749	204	1083	0	0
Robotic Lunar Lander	3485	1186	918	663	259	459	0
Low Cost Lunar Lander	354	45	191	41	76	0	0
0507-ESAS-D	7542	2015	2425	553	594	1072	883
0507-ESAS-F	7412	1985	2347	553	594	1066	867
0507-ESAS-H	5970	1877	1125	553	594	1049	772
0507-ESAS-J	5912	1860	1095	553	594	1046	764
Avgs	100.0%	27.6%	33.7%	7.6%	8.8%	13.0%	9.4%

Table 5: Subsystem mass breakdowns. [14]

Each of the columns are 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 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.

2.3 Sizing Rule Extraction

Sizing rules for all parameters of interest were derived by investigating their relation to the dry mass or the payload mass of the system. For lunar landers. statistical sizing relationships were found by finding the linear fits between the m_{dry}, m_p and m_{prop}. A linear fit was chosen between these parameters because the Tsiolkovsky equation shows that they are linearly related.

$$m_{prop} = e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p) \tag{2}$$

There were three kinds of sizing rules extracted from the database, derived by hand, or found in literature. *Statistical*

$$m_p = \alpha \cdot m_{dry} \tag{3}$$

Where α is the coefficient which gives the best fit according to excel. *Parametric:*

$$E_k = \frac{1}{2} \cdot m_{inert} \cdot \Delta V^2 \tag{4}$$

where all the parameters are deterministicaly dependant on one another.

Para-statistical

$$m_{prop} = \beta \cdot (e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p)) \quad (5)$$

Where β is a coefficient found by adjusting the line made by the 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 average of the data set.

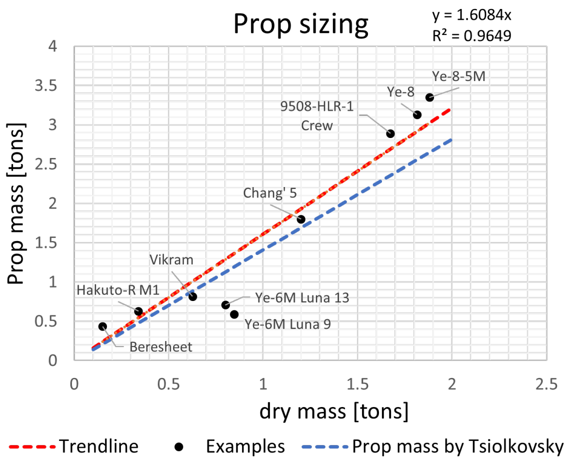

Figure 1: Tsiolkovsky before adjustment

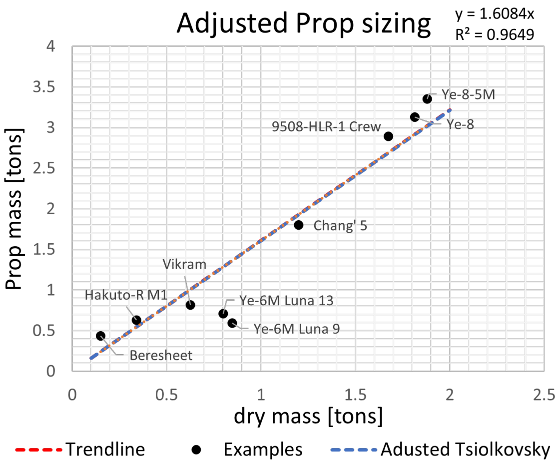

Figure 2: Tsiolkovsky adjusted

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

2.4 Mission Architecture Definition

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 which drops Operational Capability 5 (OC5) and OC7 so that it can be a mission architecture similar to the Apollo missions. All five are defined using Bat diagrams. These diagrams are read 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.

2.4.1 MA 1: The Magic Carpet

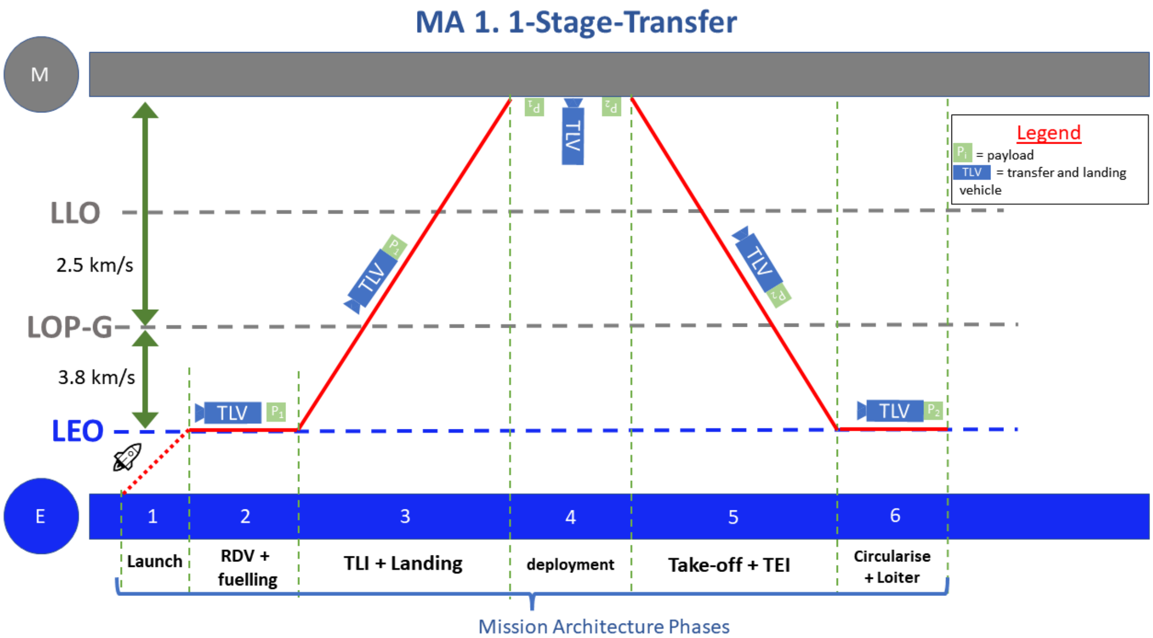

Figure 3: The Magic Carpet Mission Architecture

Construction: The Transfer Landing Vehicle is launched unfuelled into LEO.
Operation: 6 phases per cycle

1. Launch of the payload and fuel
2. RDV and transfer of the fuel and payload
3. Translunar injection to a Selenographic trajectory and landing
4. Deployment of the payload and taking on a return payload
5. Take-off and Transearth injection
6. Circularisation to LEO, payoad delivery and 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.

2.4.2 MA 2: Space-Tug

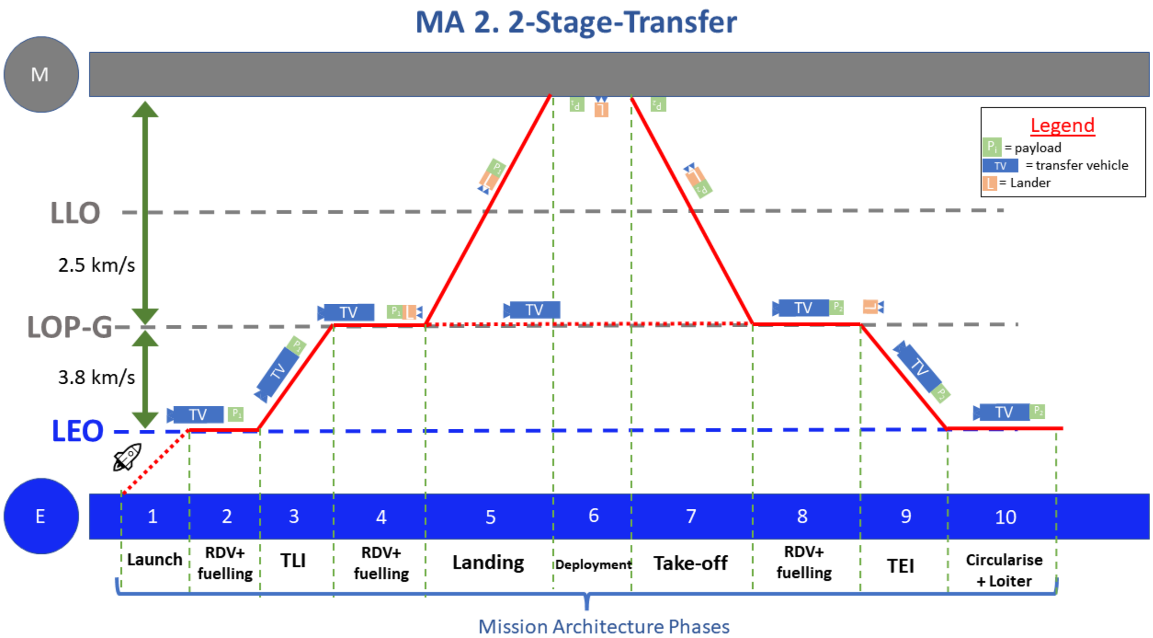

Figure 4: The Space Tug Mission Architecture

Construcion: The Transfer Vehicle is launched unfuelled into LEO. Then the lander is launched unfuelled into LEO .
Operation:

1. Payload and fuel launched to LEO
2. RDV and transfer of the fuel and payload to TV
3. TV performs TLI
4. RDV with LOP-G. Payload passed to the lander. Lander fuelled from LOP-G
5. Landing
6. Payload deployment and pick-up
7. Take-off
8. RDV with LOP-G. Payload passed to TV
9. TV performs TEI
10. TV arrives in LEO and loiters for next cycle

2.4.3 MA 3: Double Space-Tug

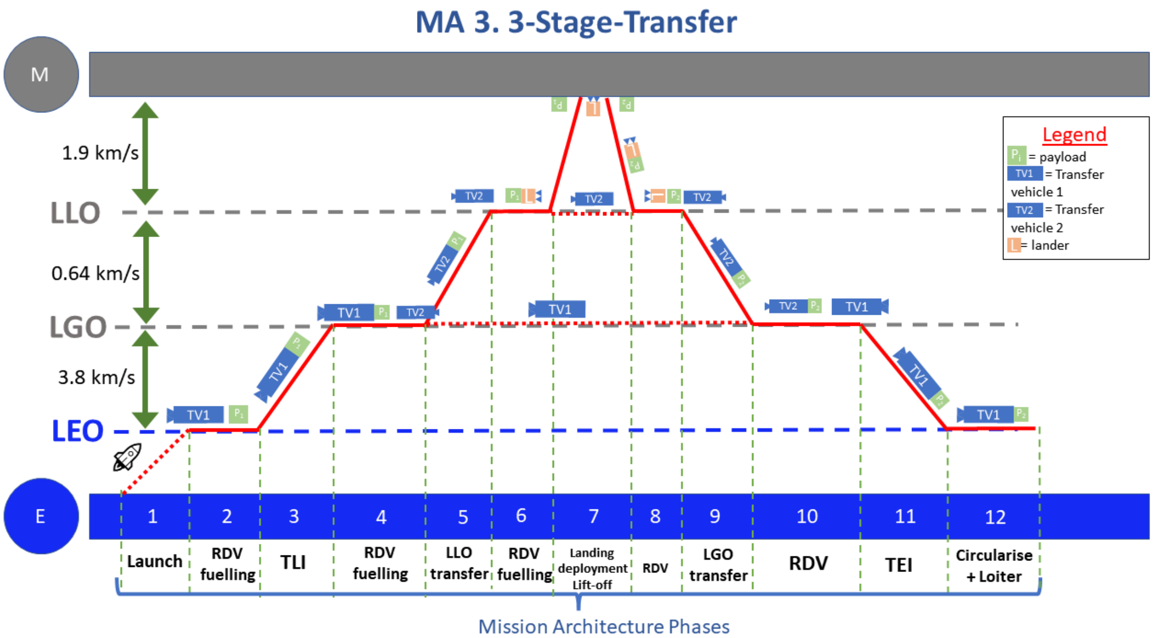

Figure 5: The Double Space Tug Mission Architecture

Construction The Transfer Vehicle 1 (TV1) is launched unfuelled into LEO. Then the Transfer Vehicle 2 (unfuelled) 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 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.
Operation: This architecture has 12 phases.

1. Payload and fuel for TV1 are launched into LEO
2. TV1 RDVs with launcher and takes on fuel and payload
3. TV1 performs TLI to LOP-G
4. TV1 RDVs with TV2 at LOP-G
5. TV2 takes the payload and fuel for the lander to LLO
6. TV2 RDVs with the lander and passes fuel and payload
7. Landing, payload deployment and taking on new payload, take-off.
8. Lander RDV in LLO with TV2 and passes payload.
9. TV2 takes payload to LOP-G.
10. TV2 RDV with LOP-G and payload passed to TV1. TV1 fuelled at LOP-G
11. TV1 performs TEI
12. TV1 arrives in LEO, delivers payload and loiters for next mission.

2.4.4 MA 4: Electric Space Tug

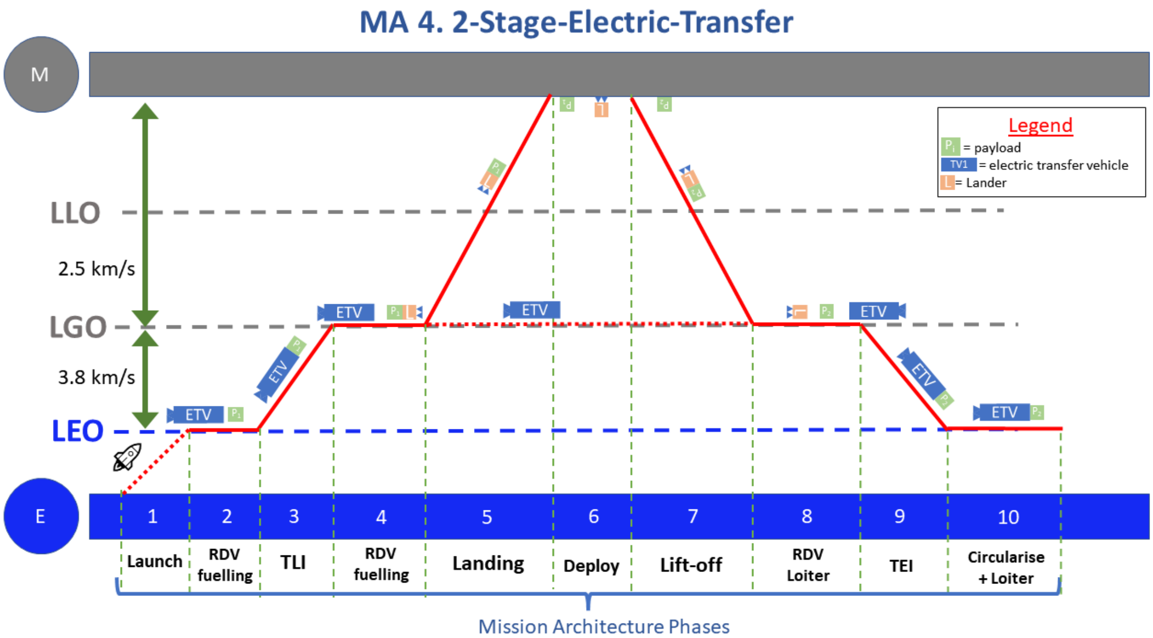

Figure 6: The Electric Space Tug Mission Architecture

Construction: The Electric Space Tug is constructed the same as the MA 2. Space Tug architecture.

1. Payload and fuel launched to LEO
2. RDV and transfer of the fuel and payload to TV1
3. ETV performs TLI
4. ETV RDVs with Lander at LOP-G and refuels
5. Landing
6. Payload deployment and pickup
7. Take-off
8. Lander RDVs with ETV at LOP-G and passes payload
9. ETV takes payload to TEI
10. ETV circularises to LEO, delivers payload and loiters for next mission.

2.4.5 MA 5: The Old Fashioned Way

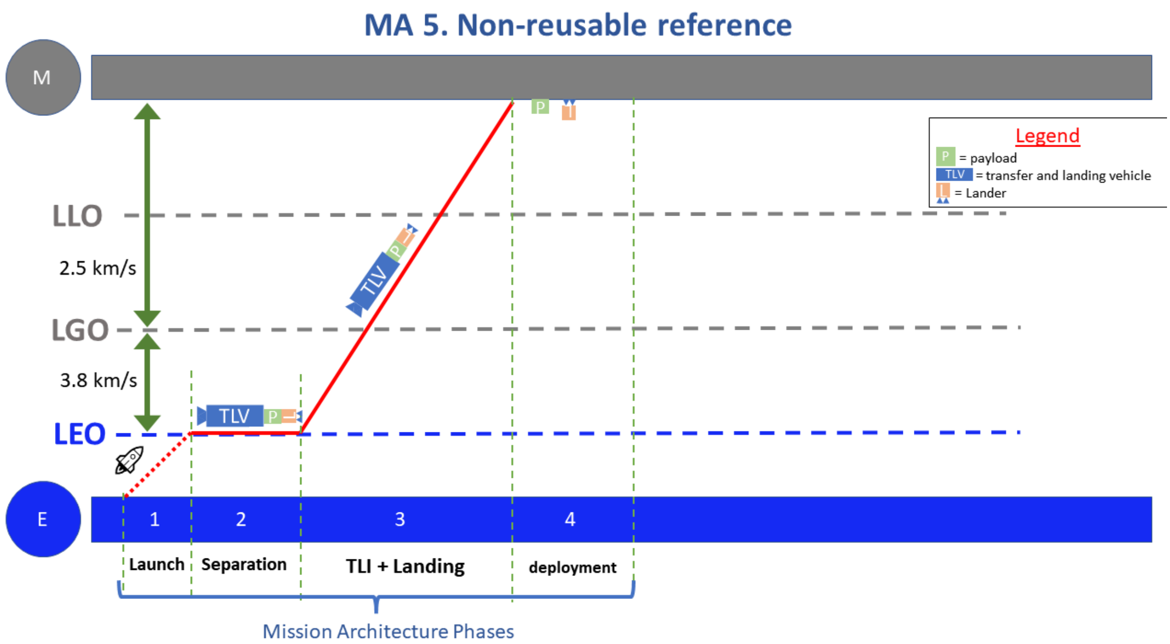

Figure 7: The Old Fashioned Way

Construction: No construction necessary. All systems are launched fully fuelled on the first launch. *Operation:* The Old Fashioned Way as the name implies, has a similar mission architecture to almost all unmanned lunar landers in the past.

1. All Systems launched into LEO
2. TV-Lander-Payload stack seperate from launcher
3. TV performs TLI onto selenographic trajectory. TV seperates from lander. Landing.
4. The lander deploys its payload.

2.5 Mission Architecture Evaluation

The mission architectures were evaluated on seven figures of merit (FOMs). These figures of merit were chosen based on the project interest in cost, pragmatic concerns for deliverable payload mass, and integration with other lunar infrastructure projects. The FOMs are shown in Table 6

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 6: Figures of Merit for the Mission Architectures

2.6 Sizing Methodology

Ground Rules and Assumptions

- The Lunar Gateway, set to be operation in the 2030s at the same time as the mission architectures of this project, 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 orbiat fuel depots around the moon and the Esprit module on the lunar gateway [30], [31], [32]
- There will be no propellant manufacturing infrastructure on the lunar surface in the time frame of the mission architectures (launch in 2030s) so any reusable lunar lander must have enough ΔV to land and return again.
- The down payload capacity and the up payload capacity are the same for all architectures. This is necessary because these mission architectures are designed to operate when a lunar outpost is already constructed and operating. To avoid the eventuality of a lunar landfill next to lunar outposts, and to allow for the eventual export of manufactured propellant to lunar orbit the capacity for payload delivery must be matched by the payload lift capacity.
- All of the mission architectures are assumed to begin development in 2024, launch in 2032 and operate until disposal in 2047

The mass based FOMs 3 and 4 are evaluated using sizing rules extracted from the data bases and parametric sizing rules from literature. The sizing procedure for a mission architecture went as follows:

1. Choose a mission architecture to size.
2. Use the following ΔV map to evaluate the required ΔV for each system.

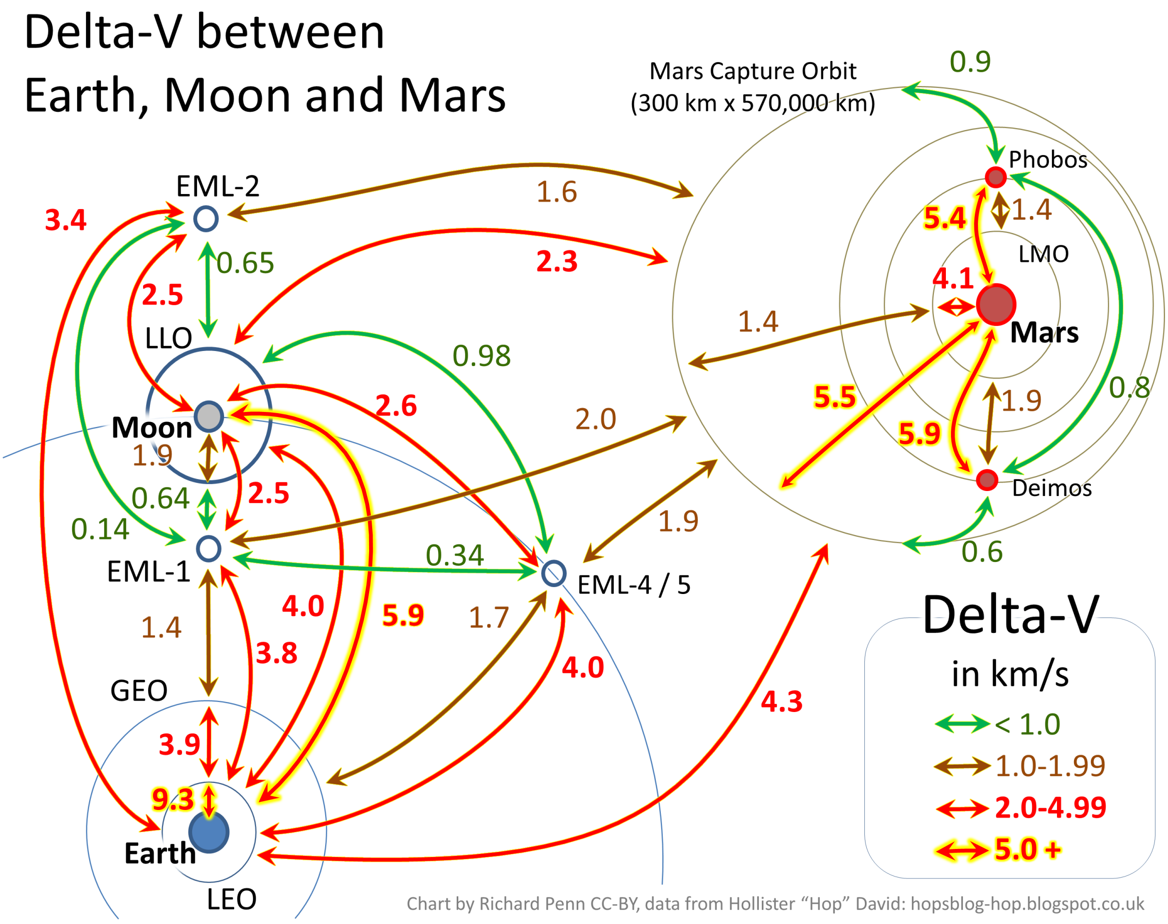

Figure 8: ΔV map for lunar and martian transfers. [33]

3. Set landed payload mass to desired amount.
4. Size the dry mass of the lander using the relevant sizing rule.

$$m_{dry} = \frac{m_p}{\alpha_1} \tag{6}$$

where α_1 is a coefficient found from the Lander Database.

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.

$$m_{prop} = \alpha_2 \cdot (e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p)) \tag{7}$$

Where α_2 is a coefficient found from the Lander Database.

6. Use the lander dry mass $m_{dry_{Lander}}$ and sizing rules from the database to size the Structure, Power, Propulsion, Avionics, and Thermal subsytems. Note for avionics. Since it does not scale linearly with dry mass (according to cost engineering experts at PRICE Systems) the avionics mass is taken to be the same as the mass fraction for the Telespazio lander in [10]. ie. 3.1% of dry mass instead of 8% implied by the database sizing rule.

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

$$T_{req} = 1.78 \cdot m_{total} \cdot g_{moon} \tag{8}$$

This come directly from the value used for Apollo missions [34]

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. Use the characteristic energy charts of known transfer vehicle to identify a reference TV. The characteristic energy of a transfer vehicle is defined as:

$$E_k = \frac{1}{2} \cdot m_{inert} \cdot \Delta V^2 \tag{9}$$

In figure 9 the characteristic energy curves of the transfer vehicles are shifted left by an amount equal to their dry mass, such that the x-axis is equivalent to the payload mass. The values of ΔV for reference TVs comes from the database.

$$\Delta V = \sqrt{\frac{2 \cdot E_k}{m_p + m_{dry}}} \tag{10}$$

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.

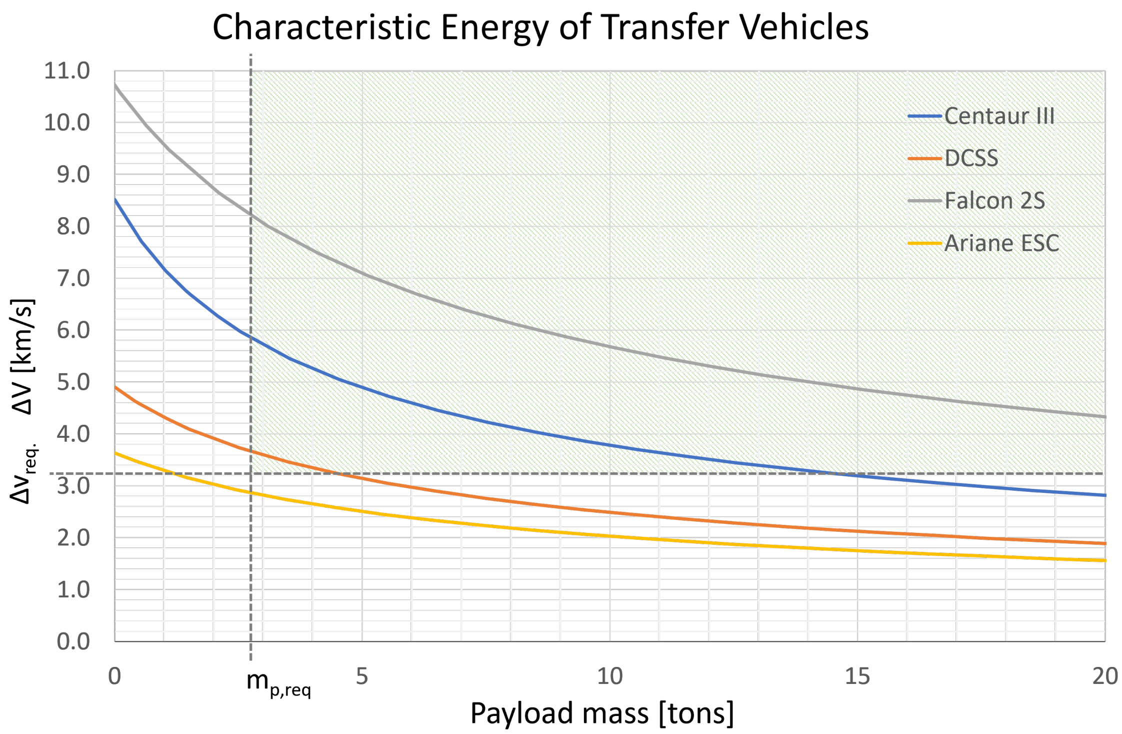

Figure 9: Generalised constraint diagram for Transfer vehicles

Set the dry mass of the TV to be equal to that of the TV with the characteristic energy curve closest to the bottom left corner of the green zone.

$$m_{dry_{TV}} = m_{dry_{TV_{ref}}} \tag{11}$$

Anything in the green zone would be feasible, but it is best to chose a vehicle which is close to your design point such that there is as little scaling to do as possible. [TODO falcon 2s scaling method. see sizing calculator sheet 1 MA 3]

9. Add the the TV dry mass and the TV payload mass and use Tsiolkovsky's equation to find the propellant requirement. (find the ΔV from figure 8 and set the I_{sp} to that of the reference TV)

$$m_{prop} = e^{\frac{\Delta V}{I_{sp} \cdot g_0}} \cdot (m_{dry} + m_p) - (m_{dry} + m_p) \tag{12}$$

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.

2.7 Costing Methodology

FOM1 and FOM2 are evaluated by dividing the total life cycle cost [$] by the total payload carried over the lifetime of the architecture [kg].
Ground rules and assumptions.

- Since the most re-used space vehicles, the space shuttle, and Falcon 9 boosters have 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 15years, that makes 60 cycles of the mission architectures with 3 flight models of all the systems.

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

- During development of a lunar lander, about 2.4 equivalent production units of protoypes will be needed. This comes from a similar costing exercise done by PRICE systems for Thales Alenia.

The life cycle cost is broken down into 4 sections:

- Design and development costs

- Architecture production costs
- Operating costs
- disposal costs

Design and development costs of the Systems are evaluated using TruePlanning. an industry standard cost estimation tool used by the author in the Space Station Design Workshop at University of Stuttgart in 2022. It works by using statistical and parametric cost estimation relationships to estimate the costs of components given their mass and some details on their purpose. An example Product Breakdown structure is shown in figure 10.

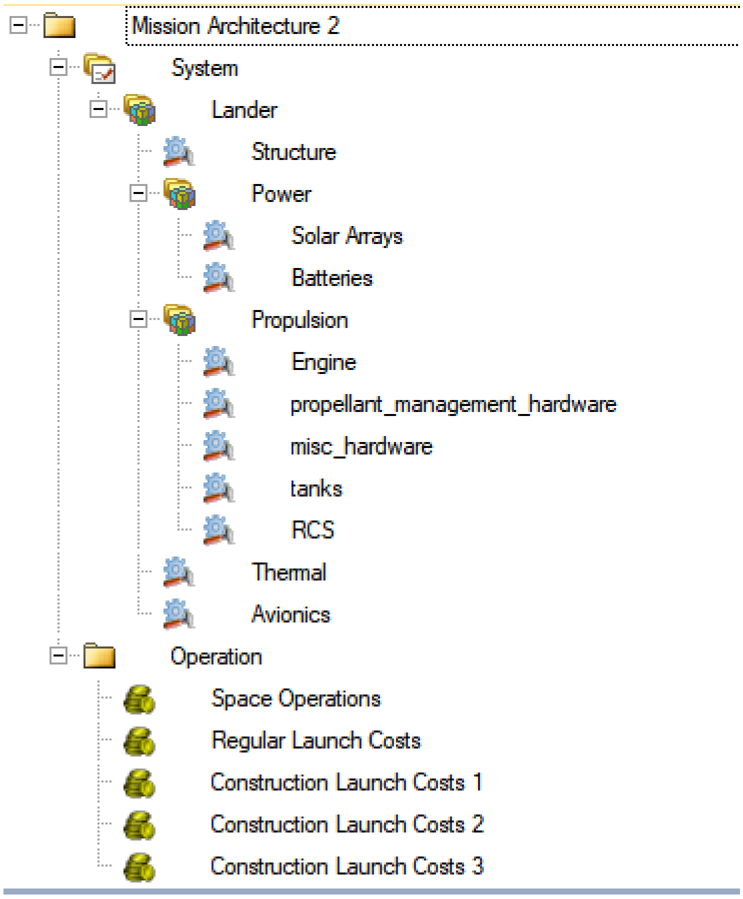

Figure 10: Product Breakdown Structure of mission architecture in True Planning

Each subsystem/component in the PBS of the System as an input sheet where the parameters for calculating the cost are placed.

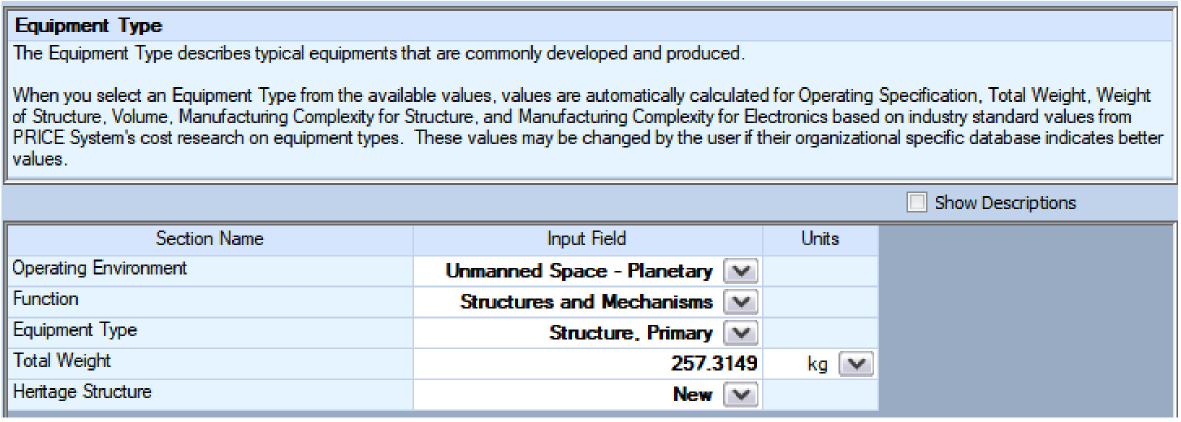

Figure 11: Equipment type specifications for a structural component

All the subsystems/components, were classified in equipment type as "Unmanned Space - Planetary"

The construction and operating costs of each mission architecture consist almost entirely of launch costs. They are input as Other Cost objects in TruePlanning. The cost of construction C_c is given by:

$$C_c = \frac{m_c}{C_{L,sp}}[\$] \tag{13}$$

Where m_c is the construction mass, consisting of the total dry mass of of the Systems to be launched and the propellant mass required to place them in the correct orbits as described in their mission architecture. $C_{L,sp}$ is the specific launch cost of the construction launcher. This is found from the database. Note that the construction launcher must have a payload to LEO capability to carry the system with the greatest dry mass to orbit.

The operational costs for each cycle of a mission architecture is given by

$$C_o = \frac{m_{prop} + m_p}{C_{L,sp}} + C_G[\$] \tag{14}$$

Where m_{prop} is the total fuel requirement which needs to be launched from Earth according to the mission architecture,m_p is the payload mass being delivered from Earth to the lunar surface, and C_G is the ground segment cost. C_G is estimated to be the cost of 7-10 full time engineers operating the mission architecture. This estimate is based on the personnel requirement for the LEOP phase of an unmanned satellite which is approximately 20 people according to an expert from OneWeb. However, C_G is usually negligable in comparrison to the launch costs so it is neglected.

The disposal cost C_D is considered to be the cost of bringing all of the systems in the mission architecture back from their their orbits around the moon and in cis-lunar space plus putting them on a re-entry trajectory into Earth's atmosphere.

2.8 Trade-off Analyses

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 12.

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 12: Pairwise comparison of the seven FOMs

This analysis, shows clearly that the cost related FOMS are the most important in this project. This is consistent with the mission objective because it specifically mentions cost effectiveness. Due to it's very low score, system flexibility and extensibility is excluded from further analysis.

3 Results and Discussion

In this section all of the results of the project are presented and discussed. That is:

- All of the sizing rules which were gathered
- The sized mission architectures.
- The performance of the mission architectures with respect to the figures of merit.
- A subsystem level mass breakdown of the lunar lander designed for mission architecture 2 (the mission architecture which which won the trade-off)

3.1 Collated Sizing Rules

3.1.1 Lander MERs

For sizing rules between the major mass parameters of lunar landers see Table 7

Major mass estimation relationships		
Lander Class	eqaution	R^2
Small	$m_p = m_p$	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	$m_p = m_p$	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	$m_p = m_p$	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 were used to size the lander of all the mission architectures. The other systems in the mission architectures were sized and costed according to the methods describe in section 2.5. Each mission architecture had the following mass breakdown.

Mission Architecture Sizing Results									
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 five mission architectures

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

FOMs raw values						
MA	Cost [$/kg_landed]	Cost [$/kg_returned]	Landed_Payload [kg/launch]	Returned_payload [kg/launch]	Delivery_Time [days]	System_complexity [#rdvs]
MA1	63,020.28	63,020.28	2,000.00	2,000.00	5.13	3
MA2	26,341.12	26,341.12	1,962.71	1,962.71	5.90	8
MA3	39,804.96	39,804.96	1,005.53	1,005.53	6.19	11
MA4	10,512.35	10,512.35	1,997.63	1,997.63	270.68	8
MA5	64,449.44	NA	1,999.57	NA	3.31	1

Table 9: results of all the evaluations of the Figures of Merit

FOMs normalised scores							
MA	Cost [$/kg_landed]	Cost [$/kg_returned]	Landed_Payload [kg/launch]	Returned_payload [kg/launch]	Delivery_Time [days]	System_complexity [#rdvs]	Total Scores
weightings	0.25	0.14	0.18	0.14	0.11	0.11	0.93
MA1	0.55	0.00	17.76	14.21	10.51	7.79	50.82
MA2	14.78	8.31	17.43	13.94	10.48	2.92	67.87
MA3	13.19	7.38	17.86	14.29	10.47	0.00	63.18
MA4	16.52	9.33	17.74	14.19	0.00	2.92	60.70
MA5	0.00	NA	17.76	0.00	10.58	9.74	38.08

Table 10

From this trade-off analysis. the MA 2 is the clear winner. Therefore, it is decided 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 the cost of the mission architecture five, the non-reusable Apollo style benchmark is the worst performing in terms of cost 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 figure 13. 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.

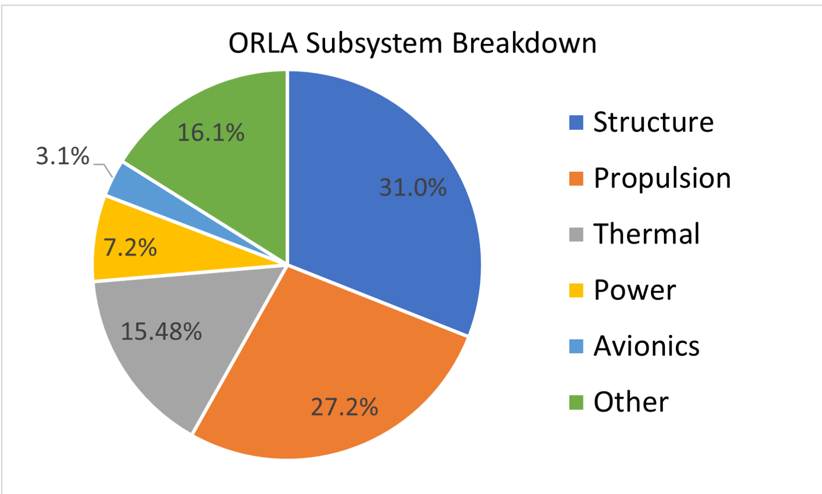

Figure 13: Mass breakdown of the lunar lander in mission architecture 2

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]	23212
I_{sp}	[s]	450
ΔV	[m/s]	5000
Thrust	[kN]	66.9
Power	[kW]	10.8
Developent cost	[$]	731,000,000
Unit Production Cost (3)	[$]	256,000,000
Delivery Cost w/ MA 2	[$/kg]	38,054

Table 11: Design Summary of ORLA Lander

	Costs : Mission Architecture 2 - [System Folder] Currency in USD ($) (as spent)	Total
1	Development	731,805,052
2	Production	2,084,555,778
3	Operation & Support	4,404,564,455
4	Total	7,220,925,285

Figure 14: Headline cost results of the ORLA Lander

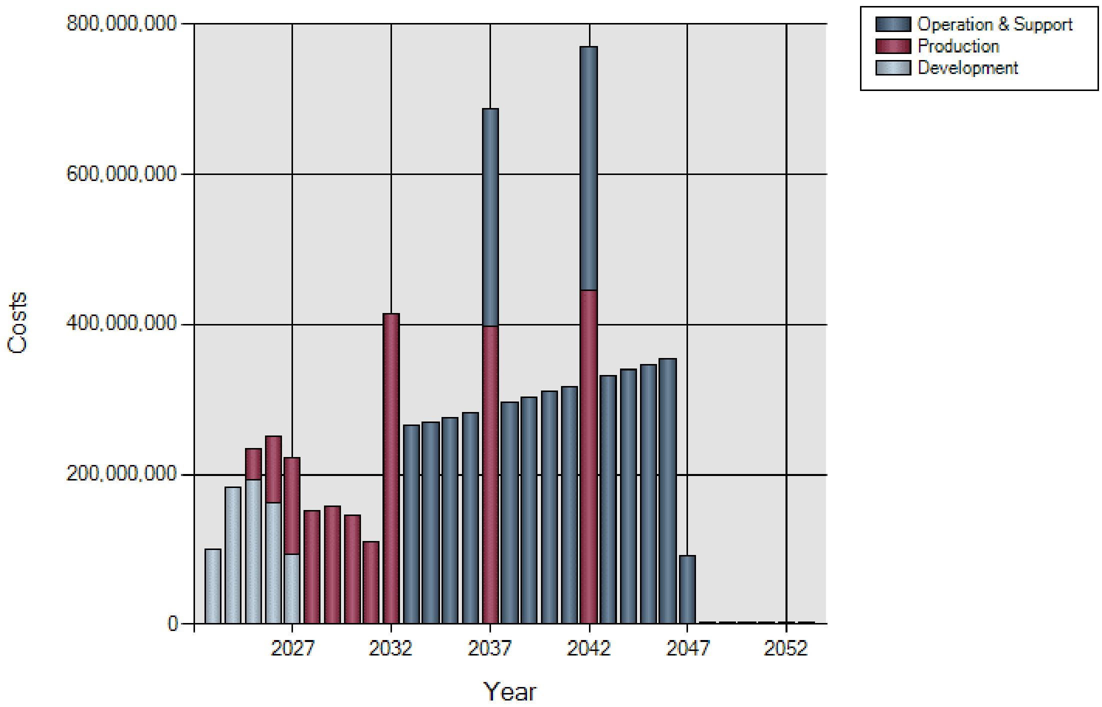

Figure 15: Costs over the lifetime of the project lasting from 2023 to 2047 (25 years) (as spent)

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 costs 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 Systems after they have been re-used 20 times each. These spikes can be attenuated if the operator is willing to accept an interruption of cargo delivery service for one year. For the TruePlanning project file is available on the project github.

In comparison to other lunar lander designs, the project cost. The Telespazio Lunar Lander costed in [10] has a combined development and unit production cost of \$784 million [10]. 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 Telespazio lander is in fact three time more expensive (\$776,000/kg for Telespazio versus \$230,000/kg for ORLA). The Gryphon lander in [15] has a combined development and unit production cost of \$6500 million. 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, generally speaking, are not absurd.

Conclusion

In conclusion, this project has conceived and evaluated five mission architectures for unmanned cargo delivery from Earth to the lunar surface using a systems engineering approach. The five architectures were assessed based on seven figures of merit, and a trade-off analysis was conducted to identify the most cost-effective and suitable architecture for delivering a 2-ton payload to the lunar surface.

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, as various factors such as the choice of launcher and trajectory impose size and performance limits on the lander.

The winning mission architecture identified in this project involves a reusable space tug, and a lander design capable of carrying 2-tons of cargo down to the lunar surface and back up again. The lander design was sized using statistical sizing rules derived from a database of lunar landers and transfer vehicles compiled from an extensive literature review. Its development, production, and operation costs were evaluated using TruePlanning software. After subsystem sizing the lander was named ORLA.

Further development of this project would do well to focus on automating the sizing of lunar landers and mission architectures and use more advanced iterative methods of re-sizing the dry mass based on the propellant mass estimation. A TRL study of the subsystems of the mission architectures would be very interesting. In terms of cost engineering, further work with TruePlanning to estimate the cost of technology development and also finding cheaper ways to build lunar landers and transfer vehicles would be an interesting line of investigation.

A Appendix

A.1 Complete Database of Lunar Landers

Lunar Landers Database											
Year	Organisation	Project name	total mass [kg]	dry mass [kg]	bloc payload mass [kg]	prop mass [kg]	propellant	1st stage Isp [s]	dV [m/s]	type	sources
1966	USSR	Ye-6M Luna 9	1538	847	99.8	591.2	HNO3/Amine	287	2630	pod	[13] [5]
1969	USSR	Ye-8-5M Luna 16	5750	1880	520	3350	HNO3/UDMH	314	1880	2stage	[5][6] [8]
1969	NASA	Apollo 12	15065	2034	4819	8212	N2O4/AZ49	311	2273	2stage	[3]
1971	NASA	Apollo 15	16447	2626	4795	9026	N2O4/AZ46	311	2250	2stage	[3]
2007	NASA	Gryphon	43501	8500	18634	16367	LOX/LH2	451	2117	1stage	[15]
2021	CNSA	Chang' 5	3800	1200	800	1800	N2O4/UDMH	333	1930	2stage	[7][35]
1966	NASA	Surveyor 1	995.2	294.3	33	667.9	MON-10/MMH	230		2stage	[36]
1967	NASA	Surveyor 3	1026	296	33	697	MON-10/MMH	230		2stage	[37]
1967	NASA	Surveyor 5	1006	303	33	670	MON-10/MMH	230		2stage	[38]
1967	NASA	Surveyor 6	1006	299.6	33	673.4	MON-10/MMH	230		2stage	[39]
1968	NASA	Surveyor 7	1039	306	33	700	MON-10/MMH	230		2stage	[40]
1969	NASA	Apollo 11	15103	2034	4821	8248	N2O4/AZ50	311	2261	2stage	[3]
1970	NASA	Apollo 13	14916	2109	4489	8318	N2O4/AZ44	311	2263	2stage	[3]
1971	NASA	Apollo 14	15034	2134	4700	8200	N2O4/AZ45	311	2263	2stage	[3]
1972	NASA	Apollo 16	16447	2626	4795	9026	N2O4/AZ47	311	2267	2stage	[3]
1972	NASA	Apollo 17	16447	2626	4795	9026	N2O4/AZ48	311	2265	2stage	[3]
1988	NASA	8801-EE-1 Crew	48218	9823	6000	32395	LOX/LH2	451	4380	1stage	[3]
1988	NASA	8801-EE-1 Cargo (D+A) LH2	54463	9823	14000	30640	LOX/LH2	451	4380	1stage	[3]
1988	NASA	8801-EE-1 Cargo (D+A) MMH	67326	7899	14000	45427	N2O2/MMH	333	4380	1stage	[3]
1988	NASA	8801-EE-1 Cargo (D) LH2	60075	9823	25000	25252	LOX/LH2	451	2100	1stage	[3]
1988	NASA	8801-EE-1 Cargo (D) MMH	69298	7899	25000	36399	N2O2/MMH	333	2100	1stage	[3]
1992	NASA	9205-FLO-1 Cargo	93037	12992	35894	44151	LOX/LH2	466		2stage	[3]
1992	NASA	9205-FLO-1 Crew	93038	12472	36384	44182	LOX/LH2	465.5		2stage	[3]
1994	US Naval Academy	Low-Cost Unmanned Lunar Lander	3775	530	200	3045	Hydrazine	320		2stage	[41]
1995	NASA	9508-HLR-1 Crew	5038.6	1673.7	473.3	2891.6	LOX/RP1	300	3732	1stage	[3]
2005	NASA	0507-ESAS-C Crew	37494	10814	980	25700	LOX/CH4	362	3772	1stage	[3]
2005	NASA	0507-ESAS-A Crew	45862	7665	13102	25095	LOX/CH4	451	1900	2stage	[3]
2005	NASA	0507-ESAS-B Crew	81911	9726	42472	29713	LOX/CH4	350	2841	2stage	[3]
2006	NASA	0605-LLPS-MSFC-6 Cargo	49972	10426	11502	28044	LOX/LH2	450		2stage	[3]
2010	NASA	COMPASS	462	171.1	14.9	276	Solid Motor	160		2stage	[42]
2013	CNSA	Chang'e 3	3780	1200	170	2410	N2O4/UDMH	333	1880	1stage	[43], [44]
2018	NASA	Robotic Lunar Lander Concept	15387.2	4687.2	1000	9700	LOX/CH4	350	3006	1stage	[45]
2019	NASA	Pallet Lander	4250	1372	300	2578	MON25/MMH	333		2stage	[46]
2019	SpaceIL	Beresheet	585	150	0	435	MON/MMH	318	1930	1stage	[47], [9]
2019	ISRO	Vikram	1471	626	27	818	unknown		2000	1stage	[48]
2019	CNSA	Chang'e 4	3640	1200	170	2270	N2O4/UDMH	333	1880	1stage	[49], [44]
2022	JAXA	OMOTENASHI	14.6	0.7	0	13.9	solid motor		2500	pod	[50], [51]
2022	iSpace	Hakuto-R M1	1000	340	30	630	N2O4/MMH	333		1stage	[52]
2022	Mahajan, Condon	Optimally staged Lander	42267	9861	9121	23285	unknown	340	2520	2stage	[53]

Table 12: The complete database of lunar landers

References

[1] J. R. Wertz, W. J. Larson, D. Kirkpatrick, and D. Klungle, *Space mission analysis and design.* Springer, 1999, vol. 8.

[2] R. A. R. J. de Paor, Conall, "An integrated design platform to analyse and size planetary exploration systems applied to lunar lava tube exploration," in *International Astronautical Congress.* Paris: IAF, 2022, pp. First Page–Last Page.

[3] J. Connolly and L. After, "Nasa lunar lander concepts beyond apollo," *National Aeronautics and Space Administration, Washington DC*, 2009.

[4] M. Isaji, I. Maynard, and B. Chudoba, "Surface access architecture modeling: Trend analysis and classification from a lander database," in *2018 AIAA SPACE and Astronautics Forum and Exposition*, 2018, p. 5131.

[5] W. T. Huntress, M. I. Marov *et al.*, *Soviet Robots in the Solar System: Mission Technologies and Discoveries.* Springer, 2011.

[6] B. Harvey, *Soviet and Russian lunar exploration.* Springer Science & Business Media, 2006.

[7] Spaceflight101. (2020) Chang'e 5. [Online]. Available: https://spaceflight101.com/change/change-5/

[8] M. Wade, "Luna ye 8-5," http://www.astronautix.com/l/lunaye-8-5.html, accessed: March 8, 2023.

[9] NSSDC. Beresheet. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=2019-009B

[10] N. M. F. Eilingsfeld, "Building block element life cycle costs," PRICE Systems, Frankfurt, Germany, Report SD-TN-PR-D19, 2012.

[11] U. Global. (2023) Trueplanning. [Online]. Available: https://www.unisonglobal.com/product-suites/cost-engineering/trueplanning/

[12] B. Harvey, *Soviet and Russian lunar exploration.* Springer Science & Business Media, 2006.

[13] M. Wade. Luna e-6. [Online]. Available: http://www.astronautix.com/l/lunae-6.html

[14] B. Malzberg, *Beyond Apollo.* RosettaBooks, 2010.

[15] D. Arney, J. Hickman, P. Tanner, J. Wagner, M. Wilson, and A. Wilhite, "Gryphon: A flexible lunar lander design to support a semi-permanent lunar outpost," in *AIAA SPACE 2007 Conference & Exposition*, 2007, p. 6169.

[16] P. Blau. Chang'e 5 spacecraft overview. [Online]. Available: https://spaceflight101.com/change/change-5/

[17] Wikipédia. Chang'e 5. [Online]. Available: https://fr.wikipedia.org/wiki/Chang%27e_5

[18] ULA, "Delta iv payload planners guide," Jun 2007.

[19] ——, "Atlas v launch services user's guide," March 2010.

[20] SpaceX. Starlink launch. [Online]. Available: https://www.youtube.com/watch?v=pIDuv0Ta0XQ

[21] S. Insider. Falcon 9. [Online]. Available: https://www.spaceflightinsider.com/hangar/falcon-9/

[22] Arianespace. Flight va237 – viasat-2 / eutelsat 172b | ariane 5 launch | arianespace. [Online]. Available: https://www.youtube.com/watch?v=XkIr2fhWC_s&t=2544s

[23] ——, "Ariane 6 user manual, issue 2, revision 0," February 2021.

[24] SpaceX, "Capabilities & services," 2021, accessed on: March 6, 2023. [Online]. Available: https://www.spacex.com/media/Capabilities&Services.pdf

[25] ULA, "Rocket builder," 2021, accessed on: March 6, 2023. [Online]. Available: https://www.rocketbuilder.com/start/configure

[26] L. Monde, “Les européens s’apprêtent à mettre ariane 6 en chantier,” 2014, accessed on: March 6, 2023. [Online]. Available: https://www.lemonde.fr/economie/article/2014/12/01/les-europeens-s-appretent-a-mettre-ariane-6-en-chantier_4532259_3234.html

[27] T. W. House, “Text of a letter from the president to the speaker of the house of representatives and the president of the senate,” 2019, accessed on: March 6, 2023. [Online]. Available: https://www.whitehouse.gov/wp-content/uploads/2019/10/shelby-mega-approps-10-23-19.pdf

[28] Wikipedia, “Vega (rocket),” [Online], 2023, accessed: Mar. 6, 2023. [Online]. Available: https://en.wikipedia.org/wiki/Vega_(rocket)

[29] R. L. USA, “Rocket lab usa poised to change the space industry,” n.d., accessed: Mar. 6, 2023. [Online]. Available: https://www.rocketlabusa.com/updates/rocket-lab-usa-poised-to-change-the-space-industry/

[30] A. Petro, A. Muscatello, J. O. Keller, and G. G. Skinner, “A cis-lunar propellant infrastructure for flexible path exploration and space commerce,” National Aeronautics and Space Administration, Glenn Research Center, Cleveland, Ohio 44135, NASA Technical Memorandum NASA/TM-2012-217235, April 2012, paper presented at the Space 2011 Conference and Exposition, sponsored by the American Institute of Aeronautics and Astronautics, Long Beach, California, September 27–29, 2011. [Online]. Available: https://ntrs.nasa.gov/api/citations/20120016774/downloads/20120016774.pdf

[31] M. Baine, G. Grush, and E. Hurlbert, “An open exploration architecture using an l-1 space propellant depot,” in *Proceedings of the SpaceOps 2010 Conference: Delivering on the Dream*, ser. AIAA paper, no. 2010-2158, NASA Mars. Huntsville, Alabama: American Institute of Aeronautics and Astronautics, April 2010, pp. 1–9, this material is declared a work of the U.S. Government and is not subject to copyright protection in the United States.

[32] S. Fuller, E. Lehnhardt, C. Zaid, and K. Halloran, “Gateway program status and overview,” in *72nd International Astronautical Congress*, no. IAC-21, A3, 2B, 13, x6624, 2021.

[33] R. Penn, “Delta v earth moon mars,” 2006, accessed: March 7, 2023. [Online]. Available: https://en.wikipedia.org/wiki/Lunar_orbit#/media/File:Delta_V_Earth_Moon_Mars.png

[34] E. M. Jones, “Lm tapemeters, associated switches, and thrust-to-weight indicator,” 2007, accessed: March 7, 2023. [Online]. Available: https://www.hq.nasa.gov/alsj/alsj-Tapemeters.html

[35] Wikipedia. (2022) Chang’e 5. [Online]. Available: https://fr.wikipedia.org/wiki/Chang%27e_5

[36] NSSDC. Surveyor 1. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=1966-045A

[37] ——. Surveyor 3. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=1967-035A

[38] ——. Surveyor 5. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=1967-084A

[39] ——. Surveyor 6. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=1967-112A

[40] ——. Surveyor 7. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=1968-001A

[41] U. S. N. A. A. E. Department, "Low-cost unmanned lunar lander," 1996, assistant Professor: Walter IC Daniel, Annapolis, Maryland.

[42] M. L. McGuire and S. R. Oleson, "Compass final report: Low cost robotic lunar lander," Tech. Rep., 2010.

[43] NSSDC. Chang'e 3. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=2013-070A

[44] Q. Wang and J. Liu, "A chang'e-4 mission concept and vision of future chinese lunar exploration activities," *Acta Astronautica*, vol. 127, pp. 678–683, 2016. [Online]. Available: https://www.sciencedirect.com/science/article/pii/S0094576515302770

[45] R. Alexander, G. Chavers, and T. Percy, "Robotic lunar lander concept," in *International Space Development Conference*, May 2018.

[46] L. Kennedy, "Nasa lunar lander reference design," Tech. Rep., 2019.

[47] NSSDC. Beresheet. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=2019-009B

[48] C. Kumar. Chandrayaan-2: All you need to know about india's 2nd moon mission. [Online]. Available: https://timesofindia.indiatimes.com/india/chandrayaan-2-all-you-need-to-know-about-indias-2nd-moon-mission/articleshow/70207662.cms

[49] NSSDC. Chang'e 4. [Online]. Available: https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=2018-103A

[50] "Omotenashi," https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=OMOTENASH, accessed: March 8, 2023.

[51] J. A. E. Agency, "Omotenashi lunar lander mission: Overview," Tech. Rep., 2013. [Online]. Available: https://www.isas.jaxa.jp/home/omotenashi/img/OMOTENASHItoutline13.pdf

[52] National Space Science Data Center (NSSDC), "2022-168A," https://nssdc.gsfc.nasa.gov/nmc/spacecraft/display.action?id=2022-168A, 2022, [Online; accessed 8-March-2023].

[53]