Reusability Analysis for Lunar Landers

by

Ryan de Freitas Bart

Submitted to the Department of Aeronautics and Astronautics in partial fulfillment of the requirements for the degree of Master of Science in Aerospace Engineering

at the

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Abstract

As the world prepares to return to the lunar surface with the Artemis program, NASA recommended developing a reusable lunar lander to enable a sustainable exploration program. It is logical to think that reusing a lunar lander instead of building a new one for each mission is more cost-effective, but this statement only holds for some mission architectures. This work explores the principal effects of adding reusability and quantifying its impact on manned lunar lander designs. A multidisciplinary model is presented that explores the lunar lander design space to understand how reusability impacts the chosen objectives of IMLEO, life-cycle cost, and safety. The choice of reusability and the use of a cislunar space station are found to have the greatest impact on the chosen objectives. The effect of changing the refurbishment cost per mission is also analyzed. An optimization framework is then applied to the resulting model to determine the optimal lunar lander design for a given mission. For the Artemis program, an expendable design was found to be more cost-effective than implementing reusability as only five lunar surface missions are currently planned. However, if additional missions are added a hybrid implementation of reusability, which entails reusing the transfer and ascent elements multiple times but changing out line-replacement units after each mission, becomes most cost-effective. Furthermore, a fully reusable design, where the transfer and ascent elements only require consumables to be replenished between missions, was found to be less cost-effective than a hybrid design for the range of values studied. The results of this work are intended to guide the development of the next generation of lunar landers and future space systems as the model and optimization framework can be adapted to explore the benefit of reusability for a wide range of space systems.

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Chapter 1

Introduction

Reusability is the only means to have a sustainable manned presence in space. In 1981, the Space Shuttle became the first reusable spacecraft. Over thirty years the Shuttle demonstrated spacecraft reusability, but it did not produce the main benefit associated with reusability: affordability. In recent years, several companies have utilized reusability to achieve cost savings for space systems, mainly launch vehicles and capsules. Now NASA is looking to develop a reusable lunar lander sustainable program to land humans on the surface of the Moon by 2024. Before reusability is incorporated into lunar lander designs, it is imperative to understand when reusability is more cost-effective than expendable systems.

1.1 The Rise of Reusability in Space Systems

1.1.1 The First Reusable Space System

At the beginning of the space age, every vehicle sent in orbit was expendable. When NASA sent astronauts to the Moon in the 1960s, none of the mission elements were reused; this required every element to be manufactured, assembled, tested, and launched for each mission. This approach was not sustainable, so NASA looked to reusability for their next spacecraft, the Space Shuttle.

The shuttle was envisioned as a low-cost method for transportation to space sta-
tions in low Earth orbit. Initial plans called for the Shuttle to fly an average of 43 missions per year, yet the highest launch cadence achieved over the life of the program with a fleet of four Shuttles was eight flights per year during the 1990s. Over its thirty-year history, the space shuttle flew 135 missions, with Discovery being the most reused orbiter at 39 flights.

Figure 1-1: Atlantis landing after the final flight of the Space Shuttle Program

The average turnaround time for a Shuttle, from landing to next launch, was 126 days which included 93 days for orbiter turn-around. Repair tasks performed usually included tile repair, payload bay reconfiguration, inspections and cleaning, system testing, calibrations, and refueling [1]. After every flight, each shuttle engine was removed for inspection and any malfunctioning electronics boxes were replaced. In addition to regular maintenance after each flight, orbiters would also undergo a depot maintenance period after several years where major components were removed and the structure was inspected. The extensive maintenance required between flights resulted in high costs per flight.

**Space Shuttle Reusability Challenges**

One of the main reasons the Shuttle had a high cost per launch was due to a lack of developmental funding. This dearth of funding was a consequence of the Office of Management and Budget’s desire to drive down cost in the post-Apollo era. When NASA proposed the Shuttle program, the Office of Management and Budget required
NASA to get an outside economic analysis of the space shuttle, which had not been required for earlier space programs [2]. NASA contracted Mathematica to perform an economic analysis of the space shuttle program which concluded the Shuttle could be economically viable if it could capture most of the existing launch market. The report estimated initial non-recurring development costs of $7.5 billion with a cost per flight of approximately $10 million if 500 missions were flown over the lifetime of the Shuttle [3]. Although the results of the Mathematica report seemed promising at the time, it forced NASA to justify the Shuttle on economic grounds instead of highlighting its ability to advance reusable space systems. Additionally, to justify the cost of developing the Shuttle, NASA designed the shuttle to fulfill a variety of roles. These additional requirements, along with the requirements for a low development cost, led to a system with high operational costs [2].

The shuttle’s operating environment was another barrier to affordability. During launch and reentry, the Shuttle experienced extreme heating and loading thus necessitating costly repair to many systems after each flight. Reusable systems which operate solely in-space will not experience these environmental extremes, thus alleviating some maintenance costs. However, the space shuttle had the advantage of being serviced on the ground while in-space reusable systems will have to be serviced in-situ. The Shuttle was the first reusable launch vehicle and provided insight into reusing space systems, but it was unable to deliver cost savings compared to expendable vehicles causing many to doubt the utility of reusability.

**Other Twentieth Century Attempts at Reusability**

In addition to the space shuttle, several other reusable or repairable space systems have been developed by NASA and other governments for technology development and operational programs. The main areas explored were reusable propulsion systems, in-space servicing of spacecraft, and reusable launch systems.

Reusable propulsion systems were developed for the Space Shuttle, International Space Station (ISS), and Mir space station [1]. The Soviet Union first demonstrated in-space refueling with Progress resupply ships that transferred hypergolic propellants
to Mir modules. The ISS uses the same refueling technology to transfer propellant from Progress M1 resupply ships. The first American in-orbit propellant transfer occurred during STS-41G when an EVA crew transferred hydrazine between a simulated tanker and a Landsat-type propulsion system interface.

In-space servicing with astronauts has also been demonstrated. During the Shuttle program, several missions were carried out to repair and upgrade the Hubble Space Telescope (HST). Astronauts living aboard the ISS also continuously repair and upgrade the space station. Both of these activities have provided valuable knowledge on maintaining systems in space, but widespread in-space servicing will not be possible only with astronauts. Instead, servicing must be performed either remotely by astronauts or autonomously using a servicing spacecraft. Orbital Express first demonstrated autonomous spacecraft servicing in 2007 when hydrazine propellant was transferred between two spacecraft.

For reusable launch vehicles in the 1990s, McDonnell Douglas developed a single stage to orbit reusable launch vehicle named DC-X that used vertical takeoff and landing. A one-third scale DC-X prototype was built and tested with several successful takeoffs and landings before the program was canceled due to lack of funding. NASA then continued the program and built the DC-XA, an upgraded version of the DC-X, which achieved a turn around time of only twenty-six hours during flight testing [4]. Whilst nor the DC-X or DC-XA became operational, they provided valuable experience on reusing launch vehicles with vertical take-off and landing, expertise private companies have used to develop the next generation of reusable launch vehicles.

1.1.2 Reusability in the Twenty-First Century

A New Generation of Reusability

In the early years of the twenty-first century, NASA decided to retire the reusable Space Shuttle and refocus on a new expendable launch vehicle, the Space Launch System (SLS). During the same time, a range of private companies began experimenting with reusable launch vehicles. It would be a decade before the first of these vehicles
demonstrated the benefit of reusability.

In November 2015, Blue Origin became the first entity to successfully vertically land a vehicle returning from space when their New Shepard rocket landed safely after a suborbital flight.

![New Shepard landing after a suborbital test flight](image)

Only a month later another private company, Space Exploration Technologies Corporation (SpaceX), successfully landed the first stage of their orbital booster, the Falcon 9. In January 2016, Blue Origin reused their recovered booster in another successful flight, a feat SpaceX would also achieve the following year. These new reusable launch vehicles have reduced the cost per launch. This reduction is realized with the Falcon 9 launch vehicle which costs $62 million per launch when expended and only $50 million when reused [5]

**Overcoming the Challenges of Reusability**

These companies are facing many of the same problems with reusability NASA did with the Space Shuttle. After each Falcon 9 flight, SpaceX performs most refurbishment work on the engines and the thermal protection system [6]. This refurbishment is costly and has resulted in a long turn around time between Falcon 9 launches. Blue Origin has encountered similar problems with its New Shepard vehicle. Commercial companies have also had recent success in the area of in-space servicing. In February
2020, Northrop Grumman’s Mission Extension Vehicle 1 (MEV-1) performed the first robotic spacecraft servicing mission when it docked with and performed maneuvers for Intelsat-901 in geosynchronous orbit.

To have a successful reusable system, an entity must be able to balance between retiring risks and focusing on economic costs [7]. Private companies have been successful with reusability as they have been able to accept a higher level of risk, which has allowed them to drive down costs. Although the New Shepard and Falcon 9 vehicles both are designed to one day carry humans, with the prior intending to carry tourists on suborbital flights and the latter preparing to launch astronauts into orbit, they have only performed uncrewed flights for the first decade of their use. This period of uncrewed flights has allowed both companies to slowly incorporate reusability into their vehicles by upgrading systems and adding technologies specifically for reusability which increases risk. Although the Shuttle underwent improvements during its lifetime, these improvements went through a rigorous risk reduction process as the Shuttle was a manned vehicle. When the day finally does come for New Shepard and Falcon 9 to carry humans, both vehicles will already have extensive experience with reusability.

1.2 Exploration in the Age of Reusability

In 1961 before a joint session of Congress, President John F. Kennedy proclaimed the United States would land a man on the Moon and return him safely to Earth by the end of that decade. Less than nine years later, NASA successfully landed the first humans on the Moon. The program was canceled soon after and it has been almost fifty years since humans have set foot upon the Moon. The program was canceled soon after and it has been almost fifty years since humans have set foot upon the Moon. The program was canceled soon after and it has been almost fifty years since humans have set foot upon the Moon. The program was canceled soon after and it has been almost fifty years since humans have set foot upon the Moon. Although the Apollo program was a resounding success, it did not provide the foundation for a sustained program of exploration. All of the systems used for the Apollo program, from the Saturn V to the Lunar Module, were expendable. As such, these systems were used once and then discarded. As humans venture further into deep space towards Mars and Near-Earth objects, it will become increasing difficult to use expendable systems
that cannot be repaired or provide routine resupply from Earth [8]. Reusable systems will play several roles in exploration programs, from enabling sustainable missions to the lunar surface to serving as the building blocks for infrastructure in deep space [9]. Reusability will be instrumental to allow for a sustained exploration program; however, reusability only is beneficial under certain circumstances.

1.2.1 NASA’s Current Exploration Plans

NASA is planning to return to the Moon by 2024 to prepare for a manned Mars landing around 2040 [10]. Unlike Apollo, the Artemis program will follow a sustainable approach centered around Gateway, a cislunar space station from which a reusable lunar landing system will take astronauts to the lunar surface. The proposed lunar landing system has three main components: the transfer element, the descent element, and the ascent element. The transfer and ascent elements are both reusable. As the Artemis program demonstrates, future exploration plans have already incorporated reusability into their design; yet the impact of reusability is still poorly understood.

1.3 Thesis Organization and Approach

This thesis explores the implementation of reusability in a lunar mission architecture to determine the optimal design for NASA’s next-generation lunar lander. Chapter 2 provides an overview of the areas of reusability analysis, previous lunar lander design studies, and design optimization. Several previous lunar lander designs are discussed to explore the different architectures that have been proposed over the years. Chapter 3 focuses on the main architectural decisions for a lunar lander while exploring the impact of each decision on reusability. Chapter 4 then delves into the development of a lunar lander model to determine when reusability should be used. Lastly, Chapter 5 summarizes the contributions of this work and provides areas for future study. The main goal of this work is to provide a framework to quantify the reusability of a system and determine when reusability should be implemented.
Chapter 2

Background and Research Gap

An overview of the main topics used in this work including reusability analysis, lunar lander design analysis, and Multidisciplinary Design Optimization (MDO) are presented. Several previous lunar lander architectures are also discussed including NASA’s current lunar lander reference design.

2.1 State of Reusability Analysis

The main goal of reusability is to develop systems that can perform the stated mission several times without major refurbishment or repair. As it will take many years to reach total reusability, there are several levels of reusability more applicable to near-term space systems. The next level down from full reusability is reusability with module replacement. A system in this category may be reused several times but modules must be changed out after every mission. Lastly, there are expendable systems. Although reusability usually produces a more cost-effective architecture, this is not true for all systems especially if the mission will only be performed a limited number of times. Additionally, many space systems have several elements or stages where it may be more desirable to only reuse some modules.
2.1.1 Quantifying Reusability with Reliability

Unlike other design characteristics such as mass and cost, the reusability of a system cannot be easily quantified. However, the reliability of a system can be quantified and is regularly calculated for space systems. As a system cannot be reused unless it can reliably perform its mission several times, reusability can be estimated with reliability.

The reliability of a system can be defined as the probability the system will function without failure that impairs the mission under stated conditions for a stated period. The probability a system will operate for the specified amount of time without failure follows an exponential decay and can be described as

\[ R = e^{-\lambda t} \] (2.1)

where \( \lambda \) is the failure rate and \( t \) is the time since the operation began. Failure rates can be determined either using historical data, extensive testing or fault analysis of a system [11, 12, 13]. Based on historical spacecraft failures, the systems with the highest failure rates are the communication and attitude control subsystems.

In more complex systems, several subsystems usually work together or sometimes independently to achieve a task. To determine the probability of failure for a system that has \( n \) non-redundant elements, all of which are essential for operation, the system reliability is defined as

\[ R_s = \prod_{i=1}^{n} R_i = e^{\sum \lambda_i t} \] (2.2)

where \( R_i \) is the reliability and \( \lambda_i \) is the failure rate of each of the individual elements.

For systems which consist of \( n \) elements where only one of these elements is needed for successful operation, the reliability is given by

\[ R_p = 1 - \prod_{i=1}^{n} (1 - R_i) \] (2.3)

To achieve high reliability in a system, adequate testing, redundancy, and flexibil-
ity are needed. Adequate margins should also be added and manufacturing processes should be controlled. Reliability can be introduced with software by allowing a system to be repogramed through a software update to work even with hardware failures. However, engineers should be careful when using this method as software failures generally occur with rare events such as exception handling and hardware failure management [13].

2.1.2 Historical Manned Spacecraft Reliability

The reliability of manned spacecraft can be estimated using the Loss of Crew (LOC) metric, which captures the probability of death or permanent disability to an astronaut, and the Loss of Mission (LOM) metric, which captures the probability the objective of a mission will not be completed. It is important to note, that it is not possible to determine the true reliability of a spacecraft especially when human decision making is involved and thus these metrics only serve as estimates. At the end of the Space Shuttle program, the LOC probability was 1 in 65. To improve safety for the Constellation program, NASA had a goal of having a 1 in 1000 probability for loss of crew and a 1 in 200 probability for loss of mission[14]. For comparison, the LOC probability for a commercial aircraft flight is less than 1 in 1,000,000. However, the Constellation requirements were deemed impracticable, and more risk was allowed for future manned spacecraft as demonstrated by the Commercial Crew program. For Commercial Crew, the Loss of Crew (LOC) requirement is 1 in 270 while the Loss of Mission (LOM) requirement is 1 in 55 [15].

2.1.3 Methods for Reusability Analysis

Several methods have been developed to estimate the reusability of systems or to guide development with reusability in mind [16, 17, 8, 18]. Previous work has determined the level of reusability which can be incorporated into a space system with current technologies and suggested areas to focus reusable technology development [1].

The space shuttle was the first attempt to quantify the impact of reusability on
a manned spacecraft with high mission flexibility [19]. When designing the Space Shuttle, engineers did not have a systematic method to create a reusable system. Instead, they worked to increase the reliability of each system to allow it to be used for multiple missions. For some systems, the necessary reliability was achieved by adding redundancy and margins, but other systems required new technologies.

For example, the Space Shuttle requirements called for a lightweight thermal protection system (TPS) which could be reused 100 times during reentry with temperature exceeding 2000 degrees F with minimum maintenance or refurbishment [20]. Before the Shuttle, all spacecraft used ablative heat shields which could not be reused as the material would erode during reentry. To make a reusable heat shield, new materials and methods had to be developed. The Space Shuttle’s reusable nature also provided unique operating challenges for the TPS such as the need to waterproof the silica tiles each flight. The need to waterproof the tiles was not in the original design and was only discovered during STS-4 when a thunderstorm occurred right before the scheduled launch. The silica tiles absorbed about 500 pounds of water from the thunderstorm due to their low-density design. When STS-4 entered orbit, the water began to expel from the tiles producing a small change in the Shuttle’s orbit. After STS-4, each silica tile had to be injected with ScotchGuard, a waterproofing agent, to prevent water absorption. This process had to be repeated before every flight as the temperatures experienced during reentry would evaporate the waterproofing agent [21]. Experience from the Space Shuttle program demonstrates reusability for space systems requires a range of new technologies and cannot be achieved just by changing current operating methods.

2.2 Previous Lunar Lander Design Analysis

Most of the previous work regarding lunar lander design has been done through developing mission concepts for particular objectives [22, 23, 24]. These studies give several important insights. First, lunar surface refueling is preferred to in-space refueling but more difficult to implement as it requires in-situ propellant production. Additionally,
a lander should be designed with commonality among the crew and cargo versions and both the ascent and descent stage should be reused. Lastly, adding a Lunar Orbit Insertion (LOI) stage decreases overall lander mass [25].

Studies focused on NASA’s current lunar mission design have determined Gateway provides several advantages over a traditional Lunar Orbit Rendezvous mission mode including as a logistics node for lunar and Mars missions and a platform to demonstrate telerobotic operation [22]. Previous work has also reasoned that utilizing a cislunar space station with a reusable landing system will give the lowest life cycle cost for a long-term exploration program while enabling commercialization of cislunar space [8]. Other work has shown the value of designing Lunar exploration systems to be extensible to Mars architectures.

Regarding lunar lander design trades, landers that do not require the use of cranes for offloading are preferred for infrastructure development [26]. Previous work has also pointed to oxygen/methane as the preferred propellant for lunar landers. Lastly, unless propellants can be made on the lunar surface, a fully reusable single-stage lander is not more cost-effective than an expendable lander [27].

2.3 Methods for Spacecraft Design Analysis

Complex engineering problems are traditionally designed using Single Discipline Optimization (SDO) where one discipline is focused on at a time. That discipline is then optimized with certain constraints. Afterward, the same process is performed for all other disciplines. This method can be simple and effective for systems with only a few disciplines. For complex systems with several disciplines, this method will likely lead to a suboptimal design. Even though each subsystem is optimized by itself, the combination of these optimal designs will lead to imbalances.

For example, when designing an airplane, the propulsion team may first optimize engine fuel efficiency by adding a new injector system which is heavier than the old one. Now the design moves to the structural team who must add additional structural elements to support the heavier engines. If the extra structural mass incurs a fuel
efficiency loss greater than the efficiency gained by the new engines, this design change should not be done. With an SDO process, this issue will not be discovered until later in the design cycle when it will be costly to change. Therefore, the complex nature of many systems necessitates a multidisciplinary approach to optimize designs.

2.3.1 Multidisciplinary Design Optimization

Multidisciplinary Design Optimization (MDO) allows designers to consider a large range of possible decisions and compare their influence on the overall design, thus performing system-level optimization instead of just optimizing each subsystem individually. Due to its resource-intensive nature, MDO has traditionally not been performed until a system has reached a detailed design stage. This leads to organizations adopting ad-hoc design processes early in the process where only a few architectures are evaluated. Such an approach leads to a wide range of optimal designs never being considered by MDO when it is applied to the system in later stages. To alleviate this issue, there is a need to incorporate lower fidelity, less resource-intensive, MDO into early design stages. The work presented in this thesis is an example of an early-stage application of MDO.

The first step in any MDO process is to identify the design variables, which are the decisions that must be made when designing the system. When designing an Earth-observing spacecraft, the design variables may be the aspect ratio, the number of cameras, transmit power, orbital altitude, and thrust level. Next, the objectives are chosen to reflect the important characteristics of the system which need to be minimized or maximized. For an Earth-observing spacecraft, some objectives may be cost, weight, and surface coverage. Next system parameters and constraints are identified. Some parameters for an Earth-observing spacecraft may be launch date and design life, while some constraints may be the maximum mass and cost. The next step is to gather individual models for each discipline within the overall system. The interconnections between the disciplines are then identified. This allows the individual models to be linked together to form a system evaluation framework that calculates the system objectives for any input combination of the design variables.
With an evaluation function, the design problem may be formulated as a Nonlinear Programming (NLP) problem as shown in Equation 2.4.

\[
\begin{align*}
\min & \quad J(x, p) \\
\text{subject to} & \quad g(x, p) \leq 0 \\
& \quad h(x, p) = 0 \\
\text{where} & \quad J = [J_1(x) \ldots J_z(x)]^T \\
& \quad x = [x_1 \ldots x_n]^T \\
& \quad g = [g_1(x) \ldots g_m(x)]^T \\
& \quad h = [h_1(x) \ldots h_m(x)]^T \\
& \quad x_{i, LB} \leq x_i \leq x_{i, UB} \quad (i = 1, \ldots, n)
\end{align*}
\]

Although it would be desirable to evaluate every design to find the global optimum, this is not possible for systems with a large number of design choices. As such, the evaluation function is used in a general optimization framework, such as Gradient-Based, Genetic Algorithms, or Simulated Annealing to determine an optimized design without needing to compute every design possibility. Once the system evaluation function and optimizer are in place, changes can be made to the underlying assumptions or models with almost no change on the higher levels.

### 2.3.2 Applications of MDO

MDO has already been applied to a range of architectures including aircraft, launch vehicles, and mission profiles [28, 29, 30]. Within the area of lunar lander design, MDO has been used to calculate the optimal path to the lunar surface from cis-lunar orbit [31] and to optimize the design of lunar cargo landers [32, 33].
2.4 Lunar Lander Architectures

Although the Apollo Lunar Module (LM) is the only lunar lander that has landed humans on the Moon, several designs have been proposed and advanced over the years. This section will provide an overview of each of these previous designs and the architectural decisions made. Table 2.1 provides an overview of the key characteristics of each of these designs.

Table 2.1: Characteristics of previous lunar lander designs

|                     | Apollo | Russian LK | Altair | Artemis                  |
|---------------------|--------|------------|--------|--------------------------|
| **Mission Mode**    | LOR    | LOR        | LOR    | LOR with Gateway         |
| **Landing Location**| Equatorial | Equatorial | Lunar South Pole | Lunar South Pole         |
| **Crew Size**       | 2      | 1          | 4      | 2                        |
| **Stay Duration**   | 1-3 days | 1-2 days  | 14 days | 14 days                  |
| **Staging Method**  | Surface Staging | Surface Staging | Surface Staging | Descent/Surface         |
| **Propulsion**      | Aerozine-50/NTO | N202/UDMH | Ascent: Hypergolics | Descent: LOX/LH2        |
|                     |        |            | Descent: LOX/LH2 | LOX/Methane             |
| **Reusable**        | No     | No         | No     | Yes                      |
| **Dry Mass**        | 10850  | Unknown    | 14500  | Unknown                  |
| **Launch Mass**     | 36200  | 5500       | 45864  | Unknown                  |

2.4.1 Previous Lunar Lander Architectures

Apollo Lunar Module

The Apollo Lunar Module, developed by NASA and Grumman Aircraft, remains the only vehicle to carry humans to the surface of the Moon. The LM carried a crew of two astronauts from Low-Lunar Orbit (LLO) to equatorial sites on the lunar surface where the crew could stay for up to three days before needing to return to orbit. The Apollo mission architecture used a Command Module (CM) for the flight with
a Service Module (SM) for Lunar Orbit Insertion and a dedicated lander to descend from LLO to the surface. Lunar Orbit Rendezvous was used as the mission mode, where two astronauts traveled to the lunar surface in the LM while a third astronaut remained in LLO in the command module. The LM used surface staging where the ascent stage would separate from the descent stage on the surface and return to orbit. Both the descent and ascent stages used hypergolic propellants.

As the Apollo LM is the only lunar lander to reach the surface of the Moon, several important lessons have been learned from its design and operation. The biggest challenge in designing the LM was mass, as the LM had to be light enough to fly on the Saturn V rocket which was smaller than the originally planned Nova launch vehicle. Another challenge was dust mitigation as Apollo 12 and 15 had sufficient surface dust blown up by the descent engine exhaust plume to obscure the crew’s vision [34]. The Apollo missions also gave operational delta-v values for traveling to the lunar surface with Apollo 11 recording a descent delta-v of 2.14km/s and an ascent delta-v of 1.85km/s [35].

![Figure 2-1: Apollo 16 Orion LM on lunar surface with Lunar Roving Vehicle](image)

**Russian LK Lunar Lander**

Although not known at the time, during the 1960s the Soviet Union was also planning to land a man on the Moon. The Soviets planned for a 10-12 day mission with four days in lunar orbit, of which up to 48 hours would be spent on the lunar surface [36].
To accomplish this task, the Soviets developed several spacecraft including a lunar lander.

For missions to the lunar surface, the Soviets developed the L2 (Lunar Orbit Module) and the “Lunniy korabl” (LK or L3). The L2 was an enhanced version of the Soyuz spacecraft that served as a command and service module to carry two cosmonauts to LLO. The L2 went through several years of development, yet it never flew in space. The L3 was a small piloted lander that could carry one cosmonaut to the lunar surface. The L3 had a primary engine with a single-nozzle and a two-nozzle backup engine with four verniers all of which used N2O4 and UDMH propellants. All of the engines on the L3 used turbopumps, unlike the Apollo LEM which used pressure-fed engines.

Figure 2-2: Soviet LK-3 engineering test article at the Science Museum London

The Soviet lunar mission profile used the Block D rocket stage to perform a mid-course correction and lunar orbit insertion. After LOI, a cosmonaut would spacewalk from the L2 to the L3 as the two modules did not have a pressurized tunnel between them. The Block D stage would then be used for the descent to 1-3 km above the lunar surface after which the L3 engines would take over [36]. The landing was performed manually by the cosmonaut using control sticks while looking out of a
downward-facing window. The L3 used the same engine for final descent and ascent so only staging of the landing legs structure was performed on the lunar surface [37]. After the ascent, the L3 docked with a honeycomb drogue fixture on the L2 using a spring-loaded probe thus alleviating the need for precise alignment between the two craft. The cosmonaut would then spacewalk back to the L2 for the return trip to Earth.

From 1970-1972, the Soviet Union successfully tested the LK three times in Earth orbit, certifying it for use on a manned mission. Due to issues with the N1 rocket, which was supposed to carry the astronauts and LK to the Moon, the program was canceled in 1974 before any crewed missions were flown.

In addition to planning for a manned lunar landing, the Soviets also planned to carry out a circumlunar mission. For this mission, the Soviets developed the L1 (Zond), a simplified version of the Soyuz spacecraft, which would carry one or two cosmonauts around the Moon and back to Earth. The L1 completed several unmanned circumlunar missions, but never flew a manned mission.

**Altair Lunar Lander**

From 2005 to 2009, NASA developed the Altair lunar lander for the Constellation program. The Altair lander was derived from the Exploration System Architecture Study (ESAS) which provided the reference Lunar Surface Access Module (LSAM) concept. Only conceptual designs were made for the Altair lander. Altair was designed to carry a crew of four astronauts to the lunar south pole where NASA planned to build a lunar base. Altair used surface staging where the descent and ascents stages were separated by an airlock to allow the crew to perform EVAs without the need to depressurize the entire lander. The descent stage also included a habitable space for the crew. This design also allowed the descent stage, which would be left behind on the surface, to be used as part of a future lunar outpost. The Constellation program was canceled in 2010 and the Altair lunar lander did not proceed past initial design studies.
Robotic Lunar Landers

In addition to manned lunar landers, several robotics landers have been designed over the years. During the 1960s both the Soviet Union, with the Luna program, and the United States, with the Surveyor program, soft-landed several vehicles on the Moon to characterize the surface for future manned landings. The Soviet Union performed the first soft landing on the lunar surface in February 1966 with Luna 9 which took the first pictures from the surface of the Moon; the United States achieved its first soft landing on the lunar surface in June of the same year with Surveyor 1. To land on the lunar surface, the Luna landers used retrorockets for descent and airbags for final landing, while the Surveyor landers followed a direct trajectory to the Moon and used retrorockets to decelerate until 3.4 meters above the surface where the lander would freefall to the surface. The Luna program also performed the first robotic sample return, with Luna 16 returning a lunar soil sample in September 1970. Luna 17 delivered the first rover to the surface of another astronomical body carrying the Lunokhod 1 rover to the lunar surface in November 1970.

In recent years, NASA has maintained a reference design for a robotic lunar lander which can take a 300 kg payload to anywhere on the lunar surface including the poles [38]. This lander is designed to launch on a commercial launch vehicle and fit within a five-meter fairing. The braking into lunar orbit is accomplished with a solid rocket motor that separates before the landing approach phase. The lander uses a descent
control system along with Terrain-Relative Navigation to complete the final descent and landing phases. The launch mass of the lander is approximately 4,250 kg with a dry mass of 1,400 kg.

2.4.2 Current Lunar Lander Reference Design

To return humans to the surface of the Moon by 2024 as part of the Artemis program, NASA is developing a new lunar lander known as the Human Landing System (HLS). The baseline HLS design consists of three elements: a transfer stage, a descent stage, and an ascent stage. The transfer and ascent stages are planned to be reusable. The transfer stage performs maneuvers to take the HLS from Gateway to LLO orbit whereas the descent stage will take astronauts down to the lunar surface and the ascent stage will return them from the surface to Gateway. Due to the short timeline to return to the Moon, NASA does not plan to incorporate sustainability into the first Artemis missions, instead opting to wait until at least 2026 to fly the first reusable lander. The first human lunar missions may also not use Gateway to simplify the missions, however Gateway is critical for a reusable lander.

The Artemis program plans to launch one mission each year to the Moon. The first mission, Artemis 1, will be an unmanned test of the Space Launch System (SLS) and Orion Multi-Purpose Crew Vehicle (MPCV) which will take Orion on a circumlunar trajectory. Next, Artemis 2, will take the first humans back to cislunar space as Apollo 8 did in the Apollo program. The next mission, Artemis 3, is set to be the first manned lunar landing in over fifty years.

For missions where Gateway is used, before humans can land on the Moon several support missions will deliver Gateway modules to cislunar space. With Gateway ready to accommodate astronauts, Artemis 3 will launch with four crew members and head for Gateway. Once at Gateway, two crew members will use the Human Landing System to descend to the lunar south pole while the other crew members stay in orbit at Gateway. Artemis 3 will consist of a 6.5-day surface mission to wait until Gateway is in the optimal position for a return to orbit from the lunar surface.

In September 2019, NASA issued a call for proposals for the Artemis Human Land-
ing System. In April 2020, NASA selected three companies, Blue Origin, Dynetics, and SpaceX, to proceed with development. Blue Origin is developing a three-stage lander called the Integrated Landing Vehicle (ILS). For the ILS, Blue Origin will build the descent element, Lockheed Martin will develop the ascent element using technologies from the Orion capsule, Northrop Grumman will develop the transfer element based on its Cygnus spacecraft, and Draper will contribute the guidance, navigation, and control, avionics. The second company, Dynetics, is developing a single element system that uses modular propellant vehicles to refuel before descending to the surface. Lastly, SpaceX plans to use their Starship vehicle with refueling in LEO for missions to the lunar surface.

Figure 2-4: Rendering of the Integrated Landing System on the lunar surface

2.5 Research Gap for Lunar Lander Design Analysis

As NASA prepares to return to the lunar surface with a reusable lunar lander, it is critical to understand when a reusable design is more beneficial than an expendable one. Specifically, there is a need to determine the threshold, in terms of the number of missions, at which a reusable lunar lander is more cost-effective than an expendable one. This work achieves this objective with a comprehensive lunar lander model to analyze the impact of reusability. The ultimate aim of this work is to determine impactful considerations for lunar lander reusability and to create an analysis tool that will advise decision-makers on the feasibility and cost-benefit trade-offs of a reusable lunar lander.
Chapter 3

Key Architecture Decisions

This chapter examines the significant architectural decisions which define a lunar lander design. The impact of each of these decisions on the reusability of a design is also discussed.

3.1 Lunar Mission Mode

The mission mode is one of the first decisions made for a lunar mission. The most common options are Lunar Orbit Rendezvous (LOR), Earth Orbit Rendezvous (EOR), and direct flight [39]. LOR consists of rendezvous performed in both Earth orbit and lunar orbit. In Earth orbit, the Command and Service Modules dock with the lunar lander although all of these elements may be launched on the same vehicle. In lunar orbit, the lunar lander undocks to descend to the surface and then redocks after ascent from the surface. The Apollo missions used LOR as it only required one launch of the Saturn V, despite the increased risk as it required a rendezvous event in lunar orbit. Unlike LOR, EOR only requires one rendezvous in Earth orbit where the main stack, consisting of the command module, service module, lunar touchdown module, and translunar injection stage, docks with a tanker to refuel. For this mission mode, the command and service modules land on the lunar surface in addition to the lunar touchdown module. Although EOR has less risk, it was not chosen for the Apollo program as it would have required two Saturn V launches and in-orbit refueling tech-
nology which was deemed too complex and expensive. The final mission mode is direct flight. Direct flight is similar to the EOR mission mode except it uses a single heavy-lift launch vehicle and no refueling in Earth orbit; however, these differences greatly reduce the payload delivered to the lunar surface. For the Apollo program, to use the direct flight option with the same payload mass to the lunar surface as LOR, a launch vehicle almost twice as large as the Saturn V would have been required.

In addition to being used by the Apollo program, LOR is used for every other major lunar mission architecture, although in varied forms. For this analysis, the mission mode is fixed as Lunar Orbit Rendezvous as the launch vehicles needed to achieve the same payload to the lunar surface with the other mission modes will likely not be available within the next decade; however, there are two options within this mode, traditional LOR, and LOR with Gateway.

### 3.1.1 Gateway Mission Profile

NASA’s current lunar mission design uses LOR with Gateway. Gateway is a proposed space station in cislunar orbit where crews will wait before going to the lunar surface in a reusable lander. Gateway will enable a manned presence in deep space and serve as a stepping off point for missions to the Moon and Mars. Gateway is composed of a series of interlocking pressurized elements provided by the United States and international partners. The baseline Gateway configuration includes systems for power, propulsion, communication, habitation, airlocks, and a robotic arm [40]. The use of Gateway provides operational flexibility by allowing a larger range of mission types to be carried out from a central starting point. Gateway also enables short-term expeditionary missions to explore any site on the moon [24]. Gateway will be in a Near-Rectilinear Halo Orbit (NRHO) and will cycle near the moon every 6-7 days [41]. The optimal departure time for lunar surface missions conducted from Gateway is before periapsis passage, which allows for a return to Gateway in 6.5 days.

Gateway serves as a maintenance base to perform repairs and refuel after each mission. To use Gateway for maintenance, the lander must travel from LLO to Gateway’s NRHO orbit, requiring a delta-v of 775 m/s each way. This incurs an
extra delta-v requirement of about 1600 m/s over traditional Lunar Orbit Rendezvous missions. For comparison, descent to the surface from a 100 km, LLO requires a relatively consistent delta-v of 1700 m/s [41]. It is possible for a reusable architecture to not use Gateway, but such an architecture would require servicing the lander in LLO presumably without support from external elements.

The Artemis baseline Human Landing System uses Gateway for surface missions with the following sequence of events. First, the entire stack, which includes the Transfer Element (TE), Descent Element (DE), and Ascent Element (AE), undocks from Gateway. Three hours later, the TE initiates the NRHO departure burn which is followed by a 12-hour coast. After coasting, the TE executes the Lunar Orbit Insertion burn to put the stack into a 100 km LLO circular orbit. The TE then separates and heads back to Gateway for reuse. Meanwhile, the Descent and Ascent Elements loiter in LLO for three revolutions to allow for sufficient navigation and orbit determination. The Descent Element then executes the Descent Orbit Initiation (DOI) burn to place the stack on an elliptical transfer orbit with an apoapsis of 100 km and a periapsis of 15.24 km. After coasting to the periapsis region of transfer orbit, the DE executes the descent burn to bring the lander to the surface. When the crew is ready to depart from the surface, the AE launches back into a 100 x 15.24 km elliptical orbit. The AE then executes a burn to circularize its orbit and loiters for three revolutions similar to the descent sequence. Lastly, the AE executes the Lunar Orbit Departure (LOD)
burn followed by a 12-hour coast to return to Gateway.

3.2 Staging

The number of stages used is another key decision that greatly impacts the overall architecture. A designer may wish to use a single stage to eliminate hazardous separation and docking events, yet this will likely lead to a more massive system. Multiple stages will make the system more complex, but the system will be more efficient and adaptable. From the standpoint of reusability, both staging concepts have merit. With a single-stage concept, the entire vehicle can be reused. For a multistage concept, reusability may only be incorporated into the stages where it will provide the greatest benefits. However, single-stage lunar lander concepts are not considered in this analysis, as previous work found such concepts to only be viable with the use of In-Site Resource Utilization for refueling on the lunar surface [22].

3.2.1 Staging Options

The main staging options for lunar landers are surface staging, transfer staging, and descent staging. The most common staging type, surface staging, has a staging event on the lunar surface. The Apollo Lunar Module used surface staging where the descent stage performed powered descent from LLO to the surface and the ascent stage returned the crew to orbit to rendezvous with the Command and Service Module. Surface staging also provides redundancy during descent as the staging event can be initiated to separate the ascent stage and return the crew to orbit.

The next staging option is transfer staging. With transfer staging, a separate stage is used to transport the lander to LLO, either from Gateway or by performing the LOI on a trajectory from Earth. This offloads the extra delta-v from the descent stage, thus decreasing its total mass. Reducing the mass of the descent stage has the added benefit of alleviating the need for deep throttling of the descent engine. For architectures without transfer staging, deep throttling is generally needed due to the large difference in mass of the descent stage at the beginning versus the end of descent.
As the transfer stage is only used for orbital maneuvers, it can use an engine with a larger nozzle that increases its \( I_{sp} \) and thus efficiency. Larger nozzles are not used for the descent stage as the added mass from the larger nozzle and the additional clearance required under the vehicle for landing generally outweigh the performance benefits. Additionally, the transfer stage engine would not require throttling and could be similar to an existing launch vehicle upper stage, thus reducing the need for extensive development. NASA’s current Artemis architecture uses a transfer stage to transfer the lander from Gateway in NRHO to LLO. A transfer stage was also used for the Surveyor robotic missions to the lunar surface in the 1960s.

The final staging type is descent staging, which introduces a staging event during the descent from LLO to the lunar surface to reduce the mass of the lander for the final descent. This action reduces the propellant needed for the descent stage, thus increasing its payload capability [42]. With descent staging, either the entire descent element or empty propellant tanks are staged after a majority of the descent burn has been performed. Staging the entire descent element provides a greater benefit over only staging empty propellant tanks but requires a time-critical engine restart during the landing sequence. This issue is alleviated if descent staging is used with surface staging, but this requires an extra stage to perform only a small amount of delta-v, a trade-off that is presumably less efficient.

3.2.2 Reusability and Staging

The staging options which lend themselves most to reusability are the options that allow an efficient path back to the starting point for future reuse. If transfer staging is used, the transfer stage can return to its starting point for the next mission after it moves the lander into LLO. The excess fuel needed to return the transfer stage does not place a burden on the other stages. This makes transfer staging a favorable option for a reusable architecture. Both surface staging and descent staging are neutral towards reusability as it will be infeasible to transport the descent stage back to orbit without refueling on the surface. Regardless of the staging option chosen, the ascent stage should be reused as it will always be brought back to the starting point.
3.3 Landing Accuracy

For a lunar lander, the Guidance, Navigation, and Control (GNC) system is responsible for ensuring safe and precise landings without a priori knowledge of hazard-free landing locations. To perform a safe landing, the GNC system must have two main characteristics. The first is the ability to detect hazards large enough to endanger the vehicle but too small to be detected from orbit; the second is the ability to work in real-time throughout the entire descent and landing to allow the lander time to maneuver if a hazard is detected. The design of the GNC system is driven by the required landing accuracy which determines the acceptable domain of landing locations.

There are several methods to achieve higher landing accuracy including radar assisted visual landing, inertial navigation, and Terrain Relative Navigation (TRN). If a base is being developed, higher landing accuracy can be obtained by placing radio beacons on the surface to guide a lander during descent. Without surface beacons, TRN is state of the art and can achieve landing precision on the order of 10-100 meters [43]. TRN technology is still being developed for spacecraft and will first be used by the Mars 2020 mission to land the Perseverance rover at Jezero Crater. Sensors for TRN can include radar, LIDAR, or cameras although each of these sensors has drawbacks that prevent them from being used exclusively. The resolution of a radar system is too low to resolve small hazards, but it can cover large swaths of terrain. LIDAR systems have high resolution but are limited in terms of range. Cameras are less expensive and require fewer resources than radar and LIDAR, but are constrained by lighting conditions. When used together, these sensing techniques offer more flexibility for mission timing and lighting requirements. One way to combine these systems would be to use radar for sensing during the approach phase, then use LIDAR and cameras to determine an exact landing location when the lander is close to the surface.
3.3.1 Landing Accuracy for Apollo

The Apollo Lunar Module mainly used crew visual sighting for landing. Before each Apollo mission, a landing site was chosen using orbital photos with a one-meter resolution at best which was inadequate to identify all landing hazards. Therefore, during the approach phase, the LM would be oriented sideways to allow the crew to see the desired landing area from the window and choose a landing site. This required strict limits to be levied on the approach trajectory regarding the approach angle, sunlight location, and view angle to ensure shadows from hazards would be visible to the crew. The Apollo Lunar Module touchdown limitations were 10 ft/s of vertical velocity and about 3-4 ft/s of horizontal velocity. The LM could handle hazards up to 0.5 m and a slope of 12 degrees. The Apollo 11 landing ellipse was 20 km long and 5 km wide with the actual landing site about 7 km from the center of the ellipse. Landing precision improved with Apollo 12, which had a predicted landing ellipse of 13 by 5 km. In actuality, Apollo 12 landed only 163 m from its target landing site.

3.3.2 Landing Accuracy for Reusability

Landing accuracy greatly impacts the effectiveness of a reusable system. With high landing accuracy, a reusable lander may land in more hazardous terrains and return to the same landing location to buildup infrastructure. High landing accuracy will also be necessary to utilize refueling from In-Situ Resource Utilization (ISRU) on the lunar surface.

3.4 Propulsion

For the propulsion system, the two main trades considered are the propellant type and the feed mechanism. The first trade for the propulsion system lies between hypergolic and cryogenic propellants.
3.4.1 Propellant Trade

Hypergolic Propellants

Hypergolic propellants are well developed and have been used on several manned spacecraft including the Apollo Lunar Lander, Space Shuttle, and ISS. Hypergolic propellants spontaneously ignite when they come in contact, thus they do not require ignition and are commonly used in simple pressure-fed engines. These combinations are stable at ambient temperature and pressure, although they can still freeze or boil in other environments. The main disadvantages of hypergolic propellants are their toxicity, a fact that has increased their cost in recent years [44], and lower performance than cryogenic propellants relative to their mass. However, hypergolic propellants have a long history of use in manned spacecraft and non-toxic hypergolics are being developed. The Apollo Lunar Module descent/ascent engines and Reaction Control System (RCS) both used hypergolic propellants. Specifically, the Lunar Module Descent Engine (LMDE) developed by TRW used Aerozine 50, a mixture of half UDMH and half hydrazine, and nitrogen tetroxide (NTO) to achieve an $I_{sp}$ of 311 seconds. Another popular hypergolic propellant combination, used mostly for Reaction Control Systems (RCS), is monomethylhydrazine (MMH) and nitrogen tetroxide (NTO). This combination was used for the Apollo Command Module RCS and the Space Shuttle OMS which used AJ10-190 engines with an $I_{sp}$ of 316 seconds.

Cryogenic Propellants

Cryogenic propellants usually have higher performance per unit mass than hypergolic propellants, but they must be kept at cryogenic temperatures. Additionally, unlike hypergolics, cryogenic propellants require an ignition source, which increases the complexity of the engine and the start-up time. Minimizing the start-up time is critical if the ascent stage is used as an abort mode during descent to the lunar surface. Two common cryogenic propellant combinations are liquid oxygen and hydrogen (LOX/LH2) as well as liquid oxygen and methane (LOX/MH4).

The main advantage of LOX/hydrogen is its high $I_{sp}$ of about 450 seconds. How-
ever, the low density of hydrogen increases the required tank mass and mass of the helium pressurant necessary which largely eliminates this advantage. The deep cryogenic nature of liquid hydrogen also makes it difficult to control boiling off [41]. The large temperature difference between liquid oxygen and liquid hydrogen requires separate cooling systems for each propellant. The large tanks required to store liquid hydrogen might also be an issue for many lander designs that have limited space.

LOX/methane has a lower $I_{sp}$ than LOX/hydrogen at approximately 350 seconds, but the higher density of methane compared to hydrogen allows for smaller propellant tanks that save mass. Although LOX and methane are cryogenic propellants, their similar temperatures allow them to insulate together so only one cryocooler is required. Previous studies have shown a LOX/methane ascent stage has the lowest system mass compared to other cryogenic and hypergolic propellants even when the excess mass from insulation and refrigeration systems and the heat rejection system is included [44].

Liquid oxygen and hydrogen engines have been used extensively including in the Saturn V second and third stages as well as the Space Shuttle main engines. Liquid oxygen and methane engines have never been flown, although two operational engines are currently being developed, Blue Origin’s BE-4 and SpaceX’s Raptor engine.

### 3.4.2 Feed System Trade

Another important trade-off for the propulsion system is the method used to feed the propellants into the combustion chamber. Pressure fed engines are generally simpler and more reliable as they have fewer moving parts; however, pressure-fed engines have lower combustion chamber pressures. This necessitates the need for a larger thrust chamber than a turbopump engine, which increases mass. Additionally, thicker tank walls are needed for propellant and pressurization system tanks as the system is highly pressurized while firing. The Apollo Lunar Module used a pressure fed system with helium as the pressurant for both the descent and ascent engines.

Turbopump engines are generally heavier due to the pumps, turbines, and other equipment necessary to move the propellants; however, turbopump engines have
higher combustion chamber pressures which allow for smaller throat diameters and thus a smaller thrust chambers. Shrinking the thrust chamber also allows for a higher nozzle expansion ratio than afforded to the same pressure-fed system and thus a higher $I_{sp}$. However, this advantage may be diminished by the power required to operate the pumps. Turbopump engines also have lower inlet pressure, so the propellant tank walls can be thinner to reduce mass. Turbopump engines are not frequently used for low thrust engines as the mass of the pumps is difficult to scale down for low propellant flow.

The trade-off between pressure-fed and turbopump feed systems depends on each specific case and should be made when specific requirements are known for the system. During the Apollo program, pressure fed was chosen due to the available pump technology and the simplicity of the design for reliability of the ascent stage.

### 3.4.3 Considerations for Reusability

A reusable lunar lander is expected to have long periods of inactivity either on the surface during a mission or in-orbit between missions. These long waiting periods favor hypergolic propellants over cryogenic propellants as hypergolic propellants are stable at ambient temperature [45]. However, cryogenic propellants may still be feasible as boil-off rates for cryogenic propellants have been shown to be as low as 2% per month in flight tanks using passive methods including high-performance multi-layer insulation (MLI), low conduction pathways and vapor cooled shields [44].

Another consideration for propulsion system reusability is if the propellant can be made in-situ on the lunar surface. Without in-situ propellant production, all propellant must be delivered from Earth, requiring the lander to be refueled in orbit. Out of the propellants discussed, only oxygen and hydrogen are available on both the Moon and Mars, while methane can only be made on Mars as carbon is not readily available on the Moon. This makes oxygen and methane a good choice for extensibility to Mars. A propellant system that uses oxygen is also favorable as oxygen is needed for several other on-board systems.

Lastly, for a propulsion system to be reusable, it must have the ability to be
restarted several times without excessive degradation. Hypergolic engines can easily be restarted several times by opening and closing the propellant values. This has been demonstrated on small hypergolic thrusters which are commonly fired many times over the life of a mission. Cryogenic engines have also been restarted, including the RL-10 on the Centaur upper stage and the J-2 on the Saturn V second and third stages. The J-2 on the Saturn V third stage could be restarted up to three times with the number of restarts only limited by the finite amount of gaseous helium on-board the stage [46]. The minimum time to restart cryogenic engines is on the order of a few minutes while the maximum time is limited by the ability of the system to maintain the cryogenic propellants at the required temperatures. Regenerative cooling should be used on reusable engines to prevent damage to the nozzle and main combustion chamber during firing. This technology has been used on the Saturn V first stage F1 engines as well as the reusable RS-25 Space Shuttle Main Engines.

3.5 Redundancy

As discussed earlier, reusable systems must be reliable. One way to increase the reliability of a system is through redundancy. Although fault avoidance processes, such as design reviews and system testing, are valuable, redundancy introduces fault tolerance which protects against a wider range of failure modes. Simply put, redundancy is used as a method of fault tolerance to ensure a component failure does not cause mission failure. There are three main ways to achieve redundancy, which are described below.

**Same Design Redundancy** uses two or more identical instances of a component. This type of redundancy can be used to keep systems in reserve where only a subset of the total instances is active at a time or to institute a voting system for fault detection where multiple instances are always active so their outputs can be compared. Same Design Redundancy is effective against random failures, yet it is not effective against design deficiencies as the same failure will likely occur in both components.
Diverse Design Redundancy is when components of different designs are used to produce the necessary system response. This method is resistant to design deficiencies and the back-up unit may just be an older flight-proven model of the primary unit; however, using diverse design redundancy may add cost and delay schedules as two different systems must be built. One example of a system with diverse design redundancy is the oxygen generation system on the ISS which consists of the American Oxygen Generating System (OGS) and the Russian Elektron system.

Functional redundancy works by having different means to perform the desired action, such as using an inertial measurement unit and a star tracker to determine orientation. This method protects against random failures and design deficiencies. However, this method may be infeasible for many applications due to the cost and time required to design and build two distinct systems.

3.5.1 Levels of Redundancy

Most aerospace systems use up to three levels of redundancy, as excess levels of redundancy require additional resources without large gains in reliability. For systems with short design lives, redundancy can be designed into the vehicle to last the entire system life. This approach becomes less feasible as the design life increases since the resources required to implement the redundancy will overcome the design. For a reusable system, units that are expected to fail and for which it is unreasonable to build in redundancy should be designed to be replaced in-situ. Additionally for reusable systems, components with rapidly advancing technology, such as avionics, should also be designed to be upgraded in-situ.

3.5.2 Redundancy in the Space Shuttle

For a reusable system to be successful, it must include a way for the crew and mission controllers to assess the state of all systems after each flight. This was the area in which the Space Shuttle fell short. As the sensors built into the Shuttle did not give
all the desired information about the state of systems, many systems had to be taken apart and inspected after each flight. Although the process of disassembling systems after each Shuttle flight was costly, it was valuable to understand system performance and diagnose any possible failures. To alleviate costly inspections, a wide array of sensors should be incorporated into each system to give engineers the information they need to assess its readiness for the next flight. The knowledge gained from the post-flight analysis can be used to make important design improvements [47]. Extensive testing should be conducted to characterize an entire reusable system and determine the components which will wear the fastest [21]. This process will allow for the creation of a maintenance plan to dictate what systems should be inspected after each mission and what spare components should be included.

3.6 Reusability

3.6.1 Lessons From Aircraft Reusability

Since the first powered flight over 100 years ago, airplanes have become the ideal reusable aerospace system. From the initial airplane only being able to fly hundreds of feet with one person on-board, commercial airliners now carry hundreds of people on flights across the globe repeatedly with only replacing consumables. While the performance of airplanes has increased over the years, the number of failures has decreased. When a failure does occur, it has become easier to determine its cause as modern-day airplanes include a wide array of sensors that cover all major systems. Airplane inspection programs have also evolved to ensure safe and reliable operation. This has led to a 95% decrease in aviation fatalities in the United States from 1998 to 2008 and made flying the safest form of travel.

The success of aircraft reusability provides several important lessons for spacecraft reusability. First, the state of every system must be known before each flight and any issues must be remedied. Next, any failures which occur should be investigated and learned from. Lastly, periodic inspections are needed to find issues before they occur.
3.6.2 Subsystem Reusability

To achieve system-level reusability for a lunar lander, the reusability of each subsystem must be evaluated as every subsystem is not equally suited for reuse. Complex systems that experience heavy use such as propulsion and life support systems may not be cost-effective choices for full reusability. Instead of making these subsystems fully reusable, it may be beneficial to make them repairable or replaceable in-situ, while making all other systems fully reusable. The Lunar Gateway would provide an ideal place for astronaut servicing of a reusable lunar lander. The capability for astronauts to repair and replace modules in-space has already been demonstrated by the Space Shuttle program that performed five servicing missions to the Hubble Space Telescope in LEO between 1993 and 2009.

Figure 3-2: Astronauts repairing the Hubble Space Telescope during the first of five servicing missions
3.6.3 Reusability Penalty

There is a penalty for making a system reusable which usually comes in the form of reduced performance. This penalty is due to the additional systems and consumables which must be added to support reuse. For the Falcon 9 launch vehicle, the reusability penalty comes from the additional propellant needed for landing and the landing systems, which include the grid fins and landing legs. When a Falcon 9 performs a landing near its launch site for reuse its payload capacity is reduced by about 30%. This translates to a payload reduction from 22,800 kg to 15,600 kg for a 1,100 km orbit with an inclination of 28.5 degrees. The payload reduction for launch to a 27-degree inclination geosynchronous transfer orbit has a similar reduction from 8,300 kg to 5,500 kg. However, if a Falcon 9 lands on a barge at sea, the performance penalty is reduced to about 15%.

For a lunar lander, the reusability penalty will likely result in a similar reduction in performance. Although reusable landers will incur a mass penalty, reusability only requires propellant and a new descent stage to be launched each mission instead of an entire lander. Furthermore, if ISRU is available then the penalty from reusability will be dwarfed by the benefit provided as a reusable system will not require a new descent stage or propellant resupply for each mission.

3.6.4 Certifying A Manned Spacecraft for Reuse

Before every mission, NASA requires that manned spacecraft receive a Certificate of Flight Readiness. For lunar lander reusability, this requires a method to certify the lander for flight while in cislunar space. The safety of a manned spacecraft can be summarized by the Loss of Crew (LOC) and Loss of Mission (LOM) metrics. Therefore, if these values can be determined for a lander in cislunar space, then the lander can be certified for flight. One method to determine these values in cislunar space is a Probabilistic Risk Assessment (PRA). In a PRA, the risk of an event is quantified by the severity of possible adverse consequences and the likelihood of adverse consequences occurring [48]. To obtain these values, the state of each subsystem after the
previous flight must be known. This can be accomplished through a combination of manual inspections and integrated sensors.

### 3.7 Extensibility to Mars

As NASA’s long term goal is a mission to Mars, it is important to consider how near-term lunar mission decisions impact future Mars missions. By making decisions with future Mars objectives in mind, it is likely these decisions will be non-optimal and more expensive in the short-term; however, they will be an investment that will pay dividends in the long term [9, 49]. However, technologies only relevant to Mars missions should not be considered when developing lunar missions.

Lunar missions are ideal for testing new technologies in an analogous environment that provides a fast return to Earth. As such, the Moon will act as a proving ground for new spacecraft, equipment, and operational techniques [50]. Extensive testing of Mars hardware can be carried out during lunar missions. Lunar technologies that are desirable to be extensible to Mars include long term in-space and surface habitats, deep-space maneuvers, and landing systems. Lunar surface missions can also provide experience with cryogenic propulsion and fluid management, terminal landing dynamics, and deep space operations [51]. Gateway provides in-space transportation commonality by demonstrating high power solar electric propulsion, deep-space operations, and mitigation of risks associated with human health.

The type of lunar campaign carried out will also impact the extensibility of lunar systems to Mars. Four main lunar mission classes are ordered from least extensible to most extensible: Gateway only, Sortie-Class, Global Exploration, Lunar Base. Previous work has shown global exploration missions that have long duration surface stays with pre-deployed assets at multiple locations are the best option for cost-effective extensibility. Such missions provide benefits to Mars missions in areas of reusable lander development, power generation, and surface exploration experience [51]. Although lunar base missions that have six-month surface stays at a fixed location provide the most extensibility to Mars.
Chapter 4

Lunar Lander Model

A computational model to simulate a lunar lander was developed to determine the threshold at which a reusable lunar lander is more cost-effective than an expendable one. First, the model is used to explore the lunar lander design space and compare design choices for each key architectural decision. Then, the creation of an optimization framework to determine the best lunar lander design for the stated objectives is discussed. Lastly, the optimal level of reusability depending on the number of missions is examined and the optimization framework is applied to NASA’s current Artemis architecture.

4.1 Model Overview

The lunar lander model created for this work is composed of a series of models that combine to form an evaluation function to calculate the objective values for any input design. A central module calls each disciplinary module and controls the data flow between modules. The model for each discipline is modular and can be replaced or updated as long as the interface with the central module is maintained. To integrate the disciplinary models, the design variables, parameters, constraints, and objectives were first identified. The design variables chosen for the model reflect the key architecture decisions discussed in the previous chapter. Constraints are also applied to the model so only feasible designs that can be developed in the next ten years are
generated. Three sustainability-focused objectives are used to quantify the value of each design: Initial Mass in Low-Earth Orbit (IMLEO), Life-Cycle Cost (LCC), and Safety.

4.2 Subsystem Modules

The disciplines involved were chosen to cover the key architecture decisions for a lunar lander and to estimate important values for the system such as total mass, power, and propellant need. An independent module contains each discipline, each of which will be discussed in this section.

4.2.1 Performance

The Performance module uses a system’s level of reusability, extensibility, and design lifetime to calculate multipliers for each subsystem. These multipliers allow for the values of other subsystems to be scaled based on the level of reusability and extensibility required. Extensibility measures the capabilities added to a subsystem to allow the design to be easily adapted for future missions. Systems with high extensibility can receive a multiple up to thirty percent over the baseline of a system with no extensibility. For reusability, multipliers are assigned using the required subsystem reliability, where expendable systems are the baseline and reusable systems can receive multipliers up to fifty percent. The Performance module also assigns a Technology Readiness Level (TRL) based on the level of reusability desired with higher levels of reusability receiving lower TRL values as they require additional technology development. Lastly, the multipliers are combined and then assigned to individual subsystems. The system lifetime is also determined based on the level of reusability required. Expendable systems have system lifetimes on the order of the mission duration, while reusable systems must last the length of the program if they are to be reused for all missions.
4.2.2 Avionics

The avionics system is sized in two components, the Guidance, Navigation, and Control (GNC) system and the communications system. First, a multiplier is calculated for the delta-v required to safely land depending on the landing accuracy of the design. Designs with lower landing accuracy values require less delta-v to land as they can identify a safe landing site from high altitudes and need fewer control actions close to the surface. Next, the control system actuators and sensors are determined based on the landing accuracy and control delta-v. Actuators modeled include thrusters, reaction wheels, and control moment gyroscopes and sensors include star trackers, sun sensors, horizons sensors, and inertial measurement units. A baseline GNC system is then formulated depending on the desired landing accuracy. Designs with a landing accuracy of greater than 500 meters require only radar while designs with a landing accuracy between 100 and 500 meters need both radar and cameras. To achieve a landing accuracy of less than 100 meters, Terrain Relative Navigation (TRN) must be used with a LIDAR sensor. Once the GNC sensors are chosen, a multiplier is calculated for the system based on the required landing accuracy. Previous work has shown there is a relationship between GNC system size and landing accuracy, with a smaller landing ellipse necessitating larger GNC systems [43]. Then, the communication system is sized by estimating the mass and average power values for the radio and antenna. Next, the entire avionics system is scaled by the reliability and extensibility multiple generated by the Performance module. Lastly, the thermal output of the avionics module is calculated as 90% of its power usage.

4.2.3 Power

The design of the power system is broken down into two areas, power generation, and power storage. The mass for each of these systems is calculated using the total power required by the other subsystems and the system lifetime. There are two options for the power generation system, multi-junction solar arrays or hydrogen/oxygen fuel cells. Solar arrays can only be used for missions to the lunar poles as everywhere else
on the moon experiences long periods of darkness during the lunar night. On the other hand, fuel cells can be used anywhere on the Moon, but they require consumables to generate power. Once a power generation option is chosen, the system is sized using simple physics-based equations. The mass of the power storage system is then calculated based on the number of discharge cycles and the power storage density of the battery type selected. The mass of the regulation and distribution equipment is estimated as 20% of the power generation and storage masses. The total thermal output of the power system is the combination of the thermal loads from the power generation and storage systems. If solar arrays are used, the thermal output is set to zero. If fuel cells are used, the thermal output is calculated based on the power output. The thermal load of the batteries is determined using the mass of the batteries and the type of batteries used, where each battery type has a specified thermal output per unit mass. The entire system is sized to end-of-life values to account for degradation; thus more massive power systems are required for reusable systems that have long design lives.

4.2.4 Propulsion

The Propulsion module is responsible for estimating the masses of the remaining subsystems and determining the IMLEO of a design. Mass estimating relationships (MERs), which take into account the mass of the subsystems that have already been sized, are used to estimate the masses of other subsystems [24, 52]. Then, the mass of the transfer stage is estimated from the mass of a Cygnus spacecraft, which is currently being studied by Northrop Grumman for use as a transfer vehicle for lunar missions. To obtain the final masses of each of the stages, the payload mass is added to the mass of the descent stage while the return mass is added to the ascent stage mass. Once all of the stages have been sized, the Tsiolkovsky rocket equation is used to calculate the amount of propellant required for each stage. The delta-v values used for each stage in the rocket equation depend on the staging choice of the design as well as if the design uses Gateway or not. For reusable architectures, a tanker element is also sized based on the propellant required by the transfer and ascent stages. To
calculate the total IMLEO of an architecture, each of the elements is assumed to be transported to either LLO or Gateway by a Centaur upper stage. Once the IMLEO of each stage is calculated, the IMLEO’s are compared to determine if any of the stages can co-manifest. If so, those stages are combined into one launch vehicle and the IMLEO calculations are redone using a single upper stage for both elements. The number of launches needed to place the entire lander system into Low-Earth orbit is then calculated. For reusable architectures, the number of launches required per mission to deliver a new descent stage, and propellant for the reused stages to cislunar orbit is also calculated.

The Propulsion module estimates the safety of a design which depends on the number of staging events, the number of docking events, the propellant used, and the landing accuracy. The safety of the propellant used is quantified through known risk values where hypergolic propellants have the highest risk and cryogenic propellants have a substantially lower risk. Landing accuracy is also used to calculate safety as better landing accuracy is considered to be safer because hazards can be avoided more easily.

4.2.5 Thermal

The thermal subsystem is simulated using three main parts: a passive cooling system, an active cooling system, and a heat rejection system. First, the steady-state temperature of the design is calculated using the projected and radiating areas, solar flux, system thermal output, and emissivity values. Then, the steady-state temperature and the desired propellant boil-off rate are used to size the passive cooling system, which consists of multi-layer insulation surrounding the propellant tanks. Next, the active cooling system is sized, which consists of cryocoolers to reduce propellant boil-off. Hypergolic propellants require the least thermal control as they are stable at ambient temperature and only use passive thermal insulation. The other propellant options are both cryogenic and involve both passive and active cooling. For a LOX/CH4 system, the active cooling system consists of only one 90 K cryocooler while a LOX/LH2 system requires two cryocoolers, one at 90 K and one at
Lastly, the radiators for the heat rejection system are sized using physics-based equations, where the temperature of the radiator surface is assumed to be 250 K, and space is 2.7 K. The radiators are sized to reject the heat output from the other subsystems as well as from the cryocoolers.

### 4.2.6 Cost

The Cost module calculates the life-cycle cost of a design. First, Cost Estimating Relationships (CERs) are used to calculate the Research, Development, Test, and Evaluation (RDT&E) as well as the production costs for the descent and ascent stages. The CERs used are a combination of CERs for manned and unmanned spacecraft [52, 53]. A cost multiplier adjusts between unmanned and manned CERs [54]. The cost of the transfer stage is calculated using its dry mass multiplied by the cost per kg of a Cygnus spacecraft. Then, the RDT&E costs for each stage are summed and a cost multiplier is applied based on the current TRL needed for the stage [55]. The TRL cost multiplier equals $5.449e^{(-0.2632T)}$ where T is the TRL level [55]. The production costs for each stage are calculated similarly and then summed to calculate the Theoretical First Unit (TFU) cost. The integration, program level, ground station, and operations cost are then estimated as fractions of the TFU cost. Next, the total non-recurring cost is calculated as the sum of RDT&E, program level, and ground station costs. The cost process now splits into two tracks, one for expendable designs and one for hybrid or reusable designs.

For expendable designs, a learning curve is applied to the TFU cost to determine the production cost for each mission. Next, the mission launch cost is calculated by multiplying the number of launches needed for the lander by the cost of a New Glenn launch vehicle. The New Glenn vehicle was chosen to launch the lander over other commercial heavy-lift launch options, especially the Delta IV Heavy and Falcon Heavy, due to its seven-meter diameter payload fairing size. The recurring costs are then calculated as the sum of the production cost, operations cost, and launch costs.

For hybrid or reusable design, the production cost is equal to the cost of the TFU. The launch cost is the number of launches for one lander multiplied by the cost of
a New Glenn launch vehicle. Unlike expendable design, the hybrid and reusable designs require propellant resupply and thus one extra launch for each mission carried out. Propellant resupply launches use the Falcon Heavy due to its lower cost and higher mass to LEO than the New Glenn. Next, the mission repair costs are estimated relative to the TFU cost using a predetermined parameter which is set before program execution. Then, the RDT&E cost, the production cost of the first lander, and the lander launch costs are summed to calculate the total non-recurring cost. Subsequently, the recurring cost for hybrid and reusable designs is calculated as the sum of the propellant launch costs and the repair cost of each mission. Lastly, the present value of the total program cost is calculated using a predefined discount rate and converted to Fiscal Year 2020 dollars to get the life-cycle cost.

4.3 Model Formulation

4.3.1 Design Variables

The design variables encompass the key architecture decisions which most affect the design of a lunar lander. The design variables used in the model are summarized below in Table 4.1 for the continuous variables and Table 4.2 for the discrete variables.

Continuous Design Variables

**Landing Accuracy** defines the maximum distance the lander can be from its target location at touchdown. The model accepts landing accuracy values between 10 and 1000 meters with a nominal landing accuracy of 250 meters, which is the approximate landing accuracy achieved by Apollo 16 and Apollo 17. Landing accuracy determines the set of possible landing locations and drives the hardware sizing for the GNC system.

**Program Duration** is the length of the program from the first launch until the final return to Earth. The minimum program duration is one year, corresponding to only one mission being carried out, and the maximum duration is 20 years, as
after such time it is unlikely the same system will still be in use. The nominal program duration value is set to five years, which is the proposed duration for NASA’s Artemis program.

**Mission Rate** is the number of missions started per year, which has a minimum value of one mission per year and a maximum of ten missions per year. The nominal mission rate is set to the planned Artemis mission rate of one mission per year, although the Apollo program achieved a mission rate of about 2.25 missions per year.

| Variable | Description         | Lower Bound | Nominal Value | Upper Bound | Units       |
|----------|---------------------|-------------|---------------|-------------|-------------|
| A        | Landing Accuracy    | 10          | 250           | 1000        | meters      |
| T        | Program Duration    | 1           | 5             | 20          | years       |
| M        | Mission Rate        | 1           | 1             | 5           | missions/year |

**Discrete Design Variables**

**Gateway** defines whether or not the design uses the Lunar Gateway in cislunar orbit.

The Lunar Gateway will serve as a staging point and maintenance base for lunar landers before their descents to the surface. Designs that use Gateway must travel from the NRHO orbit where Gateway is located to LLO before starting the descent to the lunar surface, thus imposing a delta-v penalty. Designs that do not use the Lunar Gateway go directly to LLO.

**Staging** determines how many stages the design will use. The Surface staging option uses only two stages, an ascent stage, and a descent stage, while the other two options require an additional transfer stage. The Transfer/Surface option performs one staging event in LLO and one on the surface. The Transfer/Descent option also performs one staging event in LLO but performs its second staging event during descent instead of on the surface.
**Propellant** defines the propellant combination used. There are three possible propellant combinations: Aerozine 50/Nitrogen tetroxide, Liquid oxygen/Liquid hydrogen, and Liquid oxygen/Liquid methane.

**Reusability** describes the overall level of reusability in the design. Only the ascent and transfer stages can be reused as it is assumed infeasible to reuse the descent stage without surface refueling. Expendable designs can only be used once while a hybrid system may be used several times with units changed out after each mission. The Space Shuttle falls into the hybrid category as units were changed out after each flight. A reusable system can be used several times while only requiring replenishment of consumables and minor maintenance between uses. With this ranking system, the Falcon 9 launch vehicle is between a hybrid and reusable vehicle as a single booster can be reused several times, but extensive inspections and maintenance are required between each flight.

**Extensibility** quantifies the number of additional capabilities incorporated in the lander design. Designs with no extensibility are built only to satisfy the requirements for the current mission. When low extensibility is used, certain systems such as the life support or propulsion system may be designed for use with future missions. If high extensibility is used, the entire system can be easily adapted to future missions.

### Table 4.2: Discrete Design Variables

| Variable | Description | Option 1 | Option 2 | Option 3 |
|----------|-------------|----------|----------|----------|
| G        | Gateway     | Yes      | No       | -        |
| S        | Staging     | Surface  | Transfer/Surface | Transfer/Descent |
| P        | Propellant  | Hypergolics | LOX/Methane | LOX/LH2 |
| R        | Reusability | Expendable | Hybrid | Reusable |
| E        | Extensibility | None | Low | High |
4.3.2 Parameters

Several parameters are used in the model to define the values needed for the analysis. The parameter values were chosen and tuned to align with NASA’s Artemis program.

Landing Location is set as the Lunar South Pole, which is the proposed landing location for Artemis missions.

Mission Mode is specified as Lunar Orbit Rendezvous, which is proposed for use by the Artemis program.

Descent Staging determines when the staging events occur during descent to the surface for designs that use descent staging. The Descent Staging parameter is defined as the amount of delta-v remaining in the descent when the staging event occurs. Larger values will offload more of the descent to the ascent stage while lower values will make the staging event more similar to surface staging.

Payload to Lunar Surface is the mass of the payload which must be delivered to the surface.

Return Payload is the mass of the payload which must be returned with the crew either to LLO or to the Lunar Gateway.

Discount Rate is the investment rate of return used for the present value calculations. A higher discount rate will reduce the design’s life-cycle cost while a lower discount rate will increase the life-cycle cost.

Learning Curve Slope defines the percent reduction in the average cost per unit each time the number of units produced doubles. A slower learning curve slope will reduce the cost savings per each additional lander produced while a faster curve will give additional cost savings. The nominal value for the learning curve is 95% which is slower than the values suggested by the literature. This slower value was chosen as the cost of design changes during the program is assumed to reduce the efficiency gained for each additional unit [54].
**Technology Readiness Level** represents the design maturity of the least developed component in the lander system. This value is then used by the Cost module to adjust the development costs [55]. For the model, expendable designs are considered to be TRL 7, hybrid designs are TRL 5, and reusable designs are TRL 4.

**Unmanned to Manned Cost Factor** enables cost estimates made with CERs for unmanned spacecraft to be converted to the correct values for manned spacecraft. The literature shows that manned spacecraft generally cost three times as much as unmanned spacecraft per unit mass [54].

**Hybrid Repair Cost** defines the fraction of the TFU cost required to repair a hybrid reusability lander after each flight. The value for the Hybrid Repair Cost was determined by looking at the repair costs for previous hybrid reusable space systems, specifically the Space Shuttle and the Falcon 9. When the Shuttle contract was awarded in 1972, each Shuttle cost about $1.7 billion to build which amounts to $10.5 billion today. In 2010, the estimated repair cost per Shuttle flight was between $775 million and $1.2 billion which is between $920 million and $1.5 billion today [56]. These values give a Hybrid Repair Cost for the Shuttle between 7.38% to 14.29% of the first unit cost. For the Falcon 9 rocket, SpaceX has claimed reusing a first stage booster results in a 30% price reduction, which lowers the price of a Falcon 9 from $61.2 million to $42.8 million for a savings of $18.36 million [57]. It is estimated that the cost to build a Falcon 9 first stage is $27.5 million, thus the cost per flight to refurbish the Falcon 9 is $9.14 million. This gives the Falcon 9 a Hybrid Repair Cost of 33.24%. However, as a lunar lander will be refurbished in cislunar orbit instead of on Earth, the repair cost will be greater. With these values, the cost to repair a hybrid lunar lander in cislunar orbit is estimated with a lower bound of 20% and an upper bound of 60% of the TFU cost with a nominal value of 40%. A large range of values is allowed as the true value cannot be determined since there is insufficient flight data on the cost to refurbish space systems for reuse.
**Reusable Repair Cost** is similar to Hybrid Repair Cost except it applies to fully reusable lander designs. As no space system has ever achieved full reusability, the Reusable Repair Cost could not be estimated based on historical values and was instead estimated to be 25% of the Hybrid Repair Cost. The nominal value for the Reusable Repair Cost is estimated to be 10% with a lower bound of 5% and an upper bound of 20%.

**New Glenn Cost** is the cost to launch a Blue Origin New Glenn vehicle for a mission where the first stage is reused.

**Falcon Heavy Cost** is the cost to launch a SpaceX Falcon Heavy vehicle where the side-boosters and center core are reused.

Table 4.3: Model Parameters

| Parameter | Description                        | Value     | Units  |
|-----------|-----------------------------------|-----------|--------|
| LOC       | Landing Location                   | Lunar South Pole | -      |
| MM        | Mission Mode                       | LOR       | -      |
| DS        | Descent Staging                    | 500       | m/s    |
| PTS       | Payload to Lunar Surface           | 1.5       | metric tons |
| RP        | Return Payload                     | 0.5       | metric tons |
| DR        | Discount Rate                      | 5         | %      |
| LCS       | Learning Curve Slope               | 95        | %      |
| TRL       | Technology Readiness Level         | 7         | -      |
| CR        | Unmanned to Manned Cost Factor     | 3         | -      |
| HRC       | Hybrid Repair Cost                 | 0.4       | -      |
| RRC       | Reusable Repair Cost               | 0.1       | -      |
| NG        | New Glenn Cost                     | 150M      | USD    |
| FH        | Falcon Heavy Cost                  | 90M       | USD    |
4.3.3 Constraints

The constraints used by the model ensure that all of the designs generated are feasible and comply with current guidelines for manned lunar missions.

**Initial Mass in Low Earth Orbit** restricts the total mass of all of the elements in LEO including propellant. The nominal value for this constraint is 135 metric tons which is equal to the payload mass of three New Glenn launches.

**Loss of Mission Probability** is the maximum probability that a mission will not complete its objectives. This constraint ensures that no design has a Loss of Mission probability above 1 in 85, which is the value used by NASA for Commercial Crew missions to the ISS [15].

**Mission Duration** constrains the number of days which the crew can spend in space. This constraint ensures that long-duration trajectories with lower delta-v values are not considered. The nominal mission duration is set at sixty days to align with the current longest proposed Artemis mission.

**Number of Missions** limits the total number of missions which can be carried out. This constraint is similar to putting an upper bound on life-cycle cost and ensures that only a reasonable number of missions that could be carried out by the Artemis program are considered.

**Life Cycle Cost** places a minimum bound on the total cost for a lunar lander program, including development and operations costs, to guarantee all architectures are feasible.

**Element IMLEO** is the maximum mass that any single element including propellant can have in Low Earth Orbit. This constraint comes from the maximum payload mass the New Glenn vehicle can deliver to LEO.
Table 4.4: Model Constraints

| Constraint | Description                               | Type     | Bound | Units      |
|------------|-------------------------------------------|----------|-------|------------|
| IMLEO      | Initial Mass in Low Earth Orbit           | Less than| 135   | metric tons|
| MFR        | Loss of Mission Probability               | Less than| 1.176 | percent    |
| MD         | Mission Duration                          | Less than| 60    | days       |
| NM         | Number of Missions                        | Less than| 25    | missions   |
| LCC        | Life Cycle Cost                           | Greater than| 500M | USD        |
| PAY        | Element IMLEO                             | Less than| 45,000| kg         |

4.3.4 Objectives

Three sustainability-focused objectives were chosen to evaluate the benefit of each design and allow a simple comparison between a large number of designs.

**Initial Mass in Low Earth Orbit** is the sum of the masses of all of the elements including propellant after launch to Low Earth Orbit.

**Life-Cycle Cost** denotes the total cost of the entire program for a specified design. This objective allows the most cost-effective option to be determined.

**Safety** is determined as the level of risk in the design. The risk for a design is quantified as the number of hazardous events that occur, which include staging and docking events, along with perceived risk values for the landing accuracy and propellant chosen.

Table 4.5: Model Objectives

| Objective | Description                               | Units             |
|-----------|-------------------------------------------|-------------------|
| IMLEO     | Initial Mass in Low Earth Orbit           | metric tons       |
| LCC       | Life-Cycle Cost                           | USD               |
| SAF       | Safety                                    | # Hazardous Events|
4.3.5 Model Design

The lander model integrates each of the disciplinary modules through a central evaluation program as shown in Figure 4.6. The inputs to each of the modules are shown in the first row and the outputs of the model are shown in the last column. The model evaluates a design using the following process. First, the Performance module calculates multipliers to scale each subsystem for the level of reusability and extensibility desired. The Performance module also determines the system lifetime and TRL based on the reusability of the design. Then, the model runs the Avionics module which uses the design’s landing accuracy and the avionics multiplier to size the GNC and communications systems. The values determined for these systems include their mass, average power required, thermal output, and control delta-v. Next, the model moves on to the Power module, which sizes the power generation and storage components to determine their mass and thermal output. The program duration greatly influences these systems for hybrid and reusable landers as these values are sized for end-of-life use.

The Propulsion module is then executed. The Propulsion module first calculates the dry mass of each stage and then uses the rocket equation to compute the propellant needed for each stage. With these values, the total IMLEO and safety of the design are computed. Next, the Thermal module sizes the passive and active cooling systems. The values for these systems are outputted back to the Power and Propulsion modules. As the Power, Propulsion, and Thermal modules are interdependent, the evaluation function loops through them until their output values converge. Then, the Cost module calculates the life-cycle cost based on the choice of reusability, the TRL of the design, the program duration, the mission rate, and the masses of the stages. Lastly, the model outputs the three objectives: IMLEO, Safety, and Life-Cycle Cost.

The model was optimized for efficiency to allow for a large number of designs to be quickly evaluated. A single evaluation takes less than 45 milliseconds. This efficiency scales as one-hundred evaluations take less than 4.2 seconds and one-thousand evaluations take less than 45 seconds.
### Table 4.6: Dependency matrix describing the interactions between modules

| Reusability, Extensibility | Landing Accuracy | Program Duration, Mission Rate | Propellant, Extensibility, Staging, Gateway | Program Duration, Mission Rate, Reusability |
|---------------------------|-----------------|--------------------------------|---------------------------------------------|---------------------------------------------|
| **Performance**           | Multiplier      | Multiplier                      | Multiplier, System Lifetime                 | Multiplier, System Lifetime                 |
| **Avionics**              | Avg. Power      | Mass, Control Delta-v           | Thermal Output                              |                                             |
| **Power**                 | Avg. Power      | Mass                            | Thermal Output                              | System Masses                              |
|                           | Avg. Power      | Propulsion                      | Thermal Output                              | IMLEO, Safety                              |
|                           | Avg. Power      | Mass                            | Thermal Output                              |                                             |

**4.3.6 Model Validation**

The model was validated against the Apollo Lunar Module by comparing actual LM values against predicted values generated by the model [58]. The LM was chosen for comparison as it is one of the only two manned lunar landers that have flown in space. The other lander that has been flown is the Russian LK lander. The LK was not used to validate the model as its design is not comparable to the baseline lander architecture assumed for this work. The design variable values used for the Apollo LM are listed in Table 4.7 and the results from the validation run are shown in Table 4.8.
Table 4.7: Design variables values used for validation

| Design Variable    | Apollo |
|--------------------|--------|
| Landing Accuracy   | 250    |
| Program Duration   | 4      |
| Mission Rate       | 2.25   |
| Gateway            | No     |
| Staging            | Surface|
| Propellant         | Hypergolics |
| Reusability        | None   |
| Extensibility      | None   |

Table 4.8: Apollo LM subsystem masses versus predicted subsystem masses

|                      | Actual [kg] | Predicted [kg] | Error [%] |
|----------------------|-------------|----------------|-----------|
| Ascent Dry Mass      | 2553        | 2873           | 12.53     |
| Ascent Wet Mass      | 4785        | 5674           | 18.58     |
| Descent Dry Mass     | 3240        | 3302           | 1.91      |
| Descent Wet Mass     | 11500       | 11817          | 2.76      |
| Power, Control, & Data | 516      | 713.32         | 38.24     |
| Thermal Mass         | 170         | 244.65         | 43.91     |
| Structure Mass       | 459         | 547.41         | 19.26     |
| Propulsion Mass      | 510         | 574.48         | 12.64     |
| ECLSS mass           | 288         | 656.89         | 128.09    |

The model predicts the overall dry masses of the ascent and descent vehicles within 13% and 2% of the actual values, respectively. However, the model over-predicts the ascent wet mass by almost 20%, likely because the return payload value
used in the model is greater than the actual return payload for the Apollo LM. For the individual subsystem masses, the values generated by the model differ by over 35% from the actual values for the power, thermal, and ECLSS subsystems. This error is acceptable as it does not affect the output of the model since the objectives only depend on the overall stage masses. Additionally, the model accurately estimates the life-cycle cost of the Apollo LM to be $20.553 billion in FY2020 dollars, which is only 8.17% percent off the actual cost of about $19 billion in FY2020 dollars [59].

4.4 Design Space Exploration

A full-factorial enumeration of the discrete design variables was performed to understand how the reusability, Gateway, and staging decisions impact the three objectives. The continuous design variables were not modified during this process and were set to reflect the current Artemis lunar lander design where landing accuracy is one-hundred meters, program duration is five years, and one mission is flown per year. The full-factorial enumeration produced 162 architectures, which were evaluated and the results are presented in this section.

4.4.1 Reusability Trades

The trade-offs for the different reusability options were investigated for the IMLEO and LCC objectives. The results of the architecture enumeration with the reusability value of each design highlighted are shown in Figure 4-1. The x-axis is the total IMLEO for the entire program, which is calculated as the IMLEO required for one mission multiplied by the number of missions which for this experiment was five missions. The y-axis is the life-cycle cost for the program. There is a clear divide between the expendable and reusable architectures as expendable architectures have low cost and high total IMLEO while reusable architectures are the opposite. This relationship is expected as reusable designs only require propellant to be launched each mission instead of the entire lander thus reducing the total IMLEO for the program. Between the two types of reusable designs, the hybrid designs have lower
total IMLEO and life-cycle costs than the fully reusable designs. This experiment also revealed that for a program with only five missions, an expendable design is more cost-effective than a reusable one.

![Comparison of Reusability Options](image)

Figure 4-1: Effect of reusability on life-cycle cost

Figure 4-2 shows the effect of reusability on the safety objective. There is no significant difference in the level of safety between the different reusability options. This is consistent with the expected result as the choice of reusability does not affect any of the variables which contribute to the safety objective.
4.4.2 Utility of Gateway

The Artemis program includes extensive use of the Lunar Gateway as every mission to the lunar surface will first stop at the cislunar space station. Traveling to Gateway provides missions with the benefit of a base in cislunar orbit, but at a penalty. The results of the full factorial enumeration were also used to quantify the penalty of utilizing the Lunar Gateway as shown in Figure 4-3 where the Gateway and staging options for each design are highlighted.

Figure 4-2: Effect of reusability on safety objective
Architectures which include Gateway have higher IMLEO values on average due to the extra propellant which is necessary to travel from Gateway to LLO and back. The average IMLEO for all the architectures that use Gateway is 101.55 metric tons versus only 77.12 metric tons for architectures without Gateway. However, the IMLEO shown here is for only the first mission and does not represent the IMLEO reduction gained with a hybrid or reusable lander. As such, it does not make sense to use Gateway for expendable designs where the full IMLEO penalty is applied every mission. For hybrid or reusable designs, it may still be beneficial to use Gateway as the large increase in IMLEO is only applied to the first mission when the lander is launched. The propellant delivery used for each mission would see a decrease in IMLEO as less delta-v is needed to travel to Gateway than LLO from LEO.

The different options for the staging design variables were also explored to understand if surface staging is more advantageous than using transfer staging as proposed by Artemis. Regarding the staging design variable, designs without a transfer stage
require less IMLEO on average than designs with them. A transfer stage is beneficial though as it offloads delta-v from the descent stage and can easily be reused since it always stays in orbit.

4.5 Multi-Objective Optimization

For any design, an Analysis of Alternatives study is necessary to evaluate, analyze, and trade the many courses of action before down selection to one or a few chosen alternatives. With Analysis of Alternatives, a small number of architectures are chosen by the mission designers to evaluate in detail. To ameliorate this issue for future lunar lander designs, a multi-objective optimization framework was created with the reusable lunar lander model which allows a user to determine the optimal design for any set of design variables and parameters. This capability will allow mission designers to explore the entire design space before narrowing in on the desired solution.

The optimization framework starts with a heuristic algorithm to explore the large design space and determines the optimal values for the discrete design variables that cannot be handled by gradient-based algorithms. Then, the optimal design from the heuristic algorithm is used as the starting point for a gradient-based algorithm that finds the optimal values for the continuous design variables.

Simulated Annealing was chosen for the heuristic algorithm due to its random search behavior in early iterations which transitions to a gradient-like behavior in later iterations. Figure 4-4 shows this behavior with the objective value during each iteration of a Simulated Annealing run plotted and the current optimal value highlighted. During the early iterations, the search is almost random as the objective values vary widely, but as the number of iterations grows, the algorithm converges towards a set of optimal designs in a gradient type search. This behavior allows the Simulated Annealing algorithm to explore the entire design space while being able to narrow in on and refine an optimal design.
However, Simulated Annealing is unable to truly perform gradient descent; instead, it randomly perturbs the design vector to find a better solution. This method does not follow the gradient and thus may find a point close to the optimal solution. The algorithm will not continue down to the global optimum, instead opting to continue to another random point. This issue was mitigated by using a gradient-based method on the optimal solution found by Simulated Annealing. With this method, the optimal values for the discrete design variables are found by the Simulated Annealing algorithm and the continuous design variables are optimized by the gradient-based algorithm. The optimizer was designed to be easily adaptable for a range of mission designs by modifying the input design variables, parameters, constraints, and objectives. The disciplinary modules can also be independently modified to adapt the model to other systems from launch vehicles to airplanes.

4.6 Sensitivity Analysis

A sensitivity analysis was performed to understand which design variables, parameters, and constraints drive the optimum solution and how changes in these variables will change the objective values. The objective used when finding the optimal so-
olution for this sensitivity analysis was the cost per unit mass in IMLEO. Figure 4-5 shows the normalized sensitivity values for each of the continuous design variables which were calculated using a forward Euler approximation with a small step. None of the continuous design variables affect IMLEO, but both mission rate and program duration have a large effect on life-cycle cost. This is expected as if either of these variables increased, the total number of missions performed rises and the life-cycle cost increases. Landing accuracy has a small effect on the safety of a design, as when the landing ellipse is increased, the lander will have a reduced requirement to detect surface hazards.

![Continuous Design Variables Normalized Sensitivities](image)

Figure 4-5: Sensitivity analysis for continuous design variables

The sensitivities of the discrete design variables are shown in Figure 4-6 which were calculated by moving to the next higher choice for each variable. All of the discrete design variables increase IMLEO when perturbed, with the reusability and Gateway variables having the largest effect. Reusability has a large effect on IMLEO as increasing its value increases the mass of the vehicle. Gateway also greatly increases IMLEO due to the extra delta-v required to travel to Gateway and back each trip. Regarding life-cycle cost, the reusability and Gateway design variables are once again
the main drivers. An increase in reusability leads to a decrease in life-cycle cost as for the number of missions used in this analysis a hybrid architecture has a lower life-cycle cost than an expendable one even though the expendable mission has a lower cost per unit mass in IMLEO. On the other hand, the cost greatly increases when Gateway is added to an architecture. This effect is due to the inherent correlation between mass and cost for the CERs that are used to estimate the life-cycle cost. Traveling to Gateway requires additional mass over an architecture that travels directly to LLO; thus, the cost for an architecture that uses Gateway will be greater. Regarding the safety objective, the propellant, staging, and Gateway choices have the largest effects. Both the staging and Gateway design variables impact the number of staging and docking events that must be performed, thus decreasing the safety value. The propellant choice also affects the safety value as hypergolic propellants are considered more dangerous than cryogenic ones.

![Figure 4-6: Sensitivity analysis for discrete design variables](image)

Discrete Design Variables Normalized Sensitivities

| Parameter      | IMLEO          | Life Cycle Cost | Safety          |
|----------------|----------------|-----------------|-----------------|
| Extensibility  | ![Graph](image) | ![Graph](image) | ![Graph](image) |
| Reusability    | ![Graph](image) | ![Graph](image) | ![Graph](image) |
| Propellant     | ![Graph](image) | ![Graph](image) | ![Graph](image) |
| Staging        | ![Graph](image) | ![Graph](image) | ![Graph](image) |
| Gateway        | ![Graph](image) | ![Graph](image) | ![Graph](image) |

The normalized sensitivity values for each of the numerical parameters are shown in Figure 4-7. Only three parameters affect the IMLEO of a design: return payload, payload to surface, and descent staging. When either the return payload or payload...
to the surface is increased, the IMLEO also grows. However, higher values for the
descent staging parameter give lower IMLEO values. This denotes that separating
the descent stage from the ascent stage earlier in the landing process so only the
ascent stage performs the final landing is advantageous.

Figure 4-7: Sensitivity analysis for model parameters

The element IMLEO and learning curve slope parameters have the greatest in-
fluence on life-cycle cost. The element IMLEO parameter defines the largest mass
any individual element can have in LEO and is introduced by the lifting capability of
the launch vehicle used. As such, higher values of element IMLEO will reduce launch
costs if the lander can be placed in LEO with fewer launches. The learning curve slope
defines the reduction in cost achieved for each successive lander produced. When this
value is increased, there is a similar increase in life-cycle cost as the cost savings for
each subsequent lander produced decreases. On the other hand, an increase in the
discount rate reduces the life-cycle cost as the present value of money is assumed to
be greater than its future value. Lastly, reusable and hybrid repair costs both affect
the life-cycle cost. These values increase the life-cycle cost as they increase the repair
cost for each mission.
4.7 Results

4.7.1 Impact of Reusability

The model described in the previous sections was applied to the baseline Artemis architecture to determine the optimal reusability level based on the number of missions carried out. The number of missions was varied by changing the mission rate while leaving the program duration constant at five years. The results of this analysis are shown in Figure 4-8. This trade-off depends heavily on the Hybrid Repair Cost (HRC) and the Reusable Repair Cost (RRC) parameters. The plot on the left uses the estimated lower bounds for HRC and RRC while the plot on the right uses the estimated upper bounds. These bounds were estimated using the HRC values for existing hybrid reusable space systems as described earlier in this chapter. As shown by the left plot, when the lower bound values are used for the repair costs an expendable lander is the most cost-effective design for ten missions or less. For programs with greater than 10 missions, a hybrid design becomes the most cost-effective option. However, if the repair parameters are raised to their upper bounds, hybrid reusability does not prove more cost-effective until 16 missions. In both cases, hybrid reusability is always more cost-effective than full reusability for the number of missions studied.

Figure 4-8: Comparison of different values for Hybrid and Reusable Repair Costs

To understand why the cost savings from reusable designs are not present until several missions have been performed, it is helpful to look at the cost breakdown
shown in Figure 4-9. An expendable lander has a relatively low non-recurring cost of about $3.5 billion with a high recurring cost of about $0.85 billion. The main driver of this high recurring cost is the need to produce a new lander and launch it for every mission. The non-recurring cost for a hybrid lander is about $7.5 billion, which is over twice as much as the same cost for an expendable lander. Furthermore, the non-recurring costs for a reusable lander are almost $12 billion, which is over three times more than the non-recurring costs for an expendable lander. However, by accepting the large development costs, recurring costs of $0.4 billion for a hybrid lander and $0.35 billion for a reusable design can be achieved.

As only five missions are planned for the Artemis mission, these findings suggest that an expendable landing system, where the transfer and ascent stages are expendable in addition to the descent stage which is always assumed to be expendable in this work, should be used. However, if NASA’s ultimate objective is to create a sustainable mission architecture that can one day be used by other entities, hybrid reusability should be chosen. Fully reusable designs are unlikely to produce cost savings due to the amount of technology development required and the small number of missions that are likely to be carried out during the system’s lifetime.
4.7.2 Optimal Lunar Lander Design for Artemis

The lunar lander optimization framework was used to determine the optimal lunar lander design starting from the current Artemis mission architecture. The reusability option was fixed as hybrid and the objective used for the optimization was the cost per unit mass in low-earth orbit which is calculated by dividing the life-cycle cost by the total IMLEO for the program. The optimal design is shown along with the initial Artemis design in Table 4.9. The optimal design is similar to the Artemis design as both designs use Gateway and a Transfer/Surface staging method. However, the designs differ for all of the continuous design variables, the propellant, and extensibility choices.

Table 4.9: Optimal design for Artemis lunar lander

| Design Variable     | Artemis | Optimal Value |
|---------------------|---------|---------------|
| Landing Accuracy    | 100     | 688.89        |
| Program Duration    | 5       | 5.15          |
| Mission Rate        | 1       | 2.33          |
| Gateway             | Yes     | Yes           |
| Staging             | Transfer/Surface | Transfer/Surface |
| Propellant          | LOX/Methane | Hypergolics   |
| Reusability         | Hybrid  | Hybrid        |
| Extensibility       | Low     | None          |

As shown in Table 4.10, the optimal design has a lower IMLEO and lower life-cycle cost but a higher safety value than the Artemis design. The reduction in IMLEO is achieved through the optimal design’s lower extensibility level and the use of hypergolic propellants. The optimization framework chose the lowest level of extensibility as the lander model currently does not apply a discount to the life-cycle cost for the future cost savings provided by higher levels of extensibility. Therefore, it is possible that a low level of extensibility, as chosen for the Artemis design, is optimal but
additional work is needed to make a final determination. The choice of hypergolic propellants reduces IMLEO by eliminating the need for active cooling systems but at a penalty for the safety of the design. The optimizer chose the higher-risk option as only the IMLEO and life-cycle cost objectives were used to determine the benefit of an architecture while the safety objective was constrained to be below a maximum value. Additionally, the optimal design has a similar program duration and higher mission rate than the Artemis design, thus the optimal program has a greater number of missions. A greater number of missions decreases the cost per mission as the fixed program costs are amortized over more missions. However, if the program duration is increased past the optimal rate the program can experience parts obsolescence and if the mission rate is increased further additional resources will be required to process missions in parallel.

Table 4.10: Optimal lander architecture compared to Artemis design

| Objective               | Artemis | Optimal | Units                |
|------------------------|---------|---------|----------------------|
| IMLEO                  | 102.97  | 96.56   | metric tons          |
| Life-Cycle Cost        | 21.77   | 18.03   | FY2020 Billion USD   |
| Safety                 | 7.26    | 7.76    | # Hazardous Events   |
| Cost per Unit Mass     | 26.38   | 15.51   | FY2020 KUSD per kg   |
Chapter 5

Conclusion

The contributions of this work are summarized and the expected impact of the results is discussed. Limitations of the work are presented along with areas for future work. Lastly, a recommendation is presented for the implementation of reusability in future space systems.

5.1 Contributions

The key architecture decisions for a lunar lander and the impact of reusability on each decision were discussed. A lunar lander model that incorporated reusability was developed to investigate the trade-off between the three levels of reusability identified and to understand the driving design variables. Next, the model was combined with an optimization framework to determine the optimal lunar lander design for a given set of inputs. This framework was used to determine the optimal lunar lander design for the Artemis mission architecture.

The results presented here will inform mission designers about the important effects of reusability on lunar lander design and the level of reusability that is warranted for a given mission. The framework can be leveraged to understand the impact of reusability on a wide range of systems by replacing the disciplinary modules.
5.2 Limitations

This work has several limitations that can be addressed by future research. The current model accurately predicts the overall mass for each lunar lander stage but does not accurately predict the subsystem masses. This issue did not affect the results of this work as the analysis was based upon the overall stage masses. However, to increase the fidelity of the model, reusability should be implemented on a subsystem level which would require accurate values for each subsystem. Additionally, this work used multipliers based on a subsystem’s required reliability to assess the impact of adding reusability to a design. This method allows for efficient estimation of the impact of reusability by sacrificing fidelity. To accurately determine the impact of reusability, the modifications required to make each subsystem reusable should be examined and their effects summed to give the total impact of reusability.

The refurbishment cost for each mission with a hybrid or reusable lander, where the transfer and ascent stages are reused, was calculated using a factor of the theoretical first unit cost. For hybrid landers, this factor was estimated using values from previous semi-reusable space systems. As all of the previous semi-reusable space systems were repaired on Earth and of vastly different designs than a lunar lander, the repair factor estimated from these values was likely inaccurate. As the repair cost drives where the trade-off between each level of reusability occurs, additional work should be performed to determine these values at a higher fidelity.

Furthermore, the life-cycle cost for each design is estimated using cost estimating relationships. This approach enables the cost of each element to be determined with limited information but can lead to errors greater than 25% in the final cost. To generate better estimates, costs should be evaluated for individual modules or components using data from previous missions or current hardware.
5.3 Future Work

There are several areas of this work that can be expanded upon. One possible focus for future efforts is the impact of ISRU on the tradeoff between the different levels of reusability. This topic is perhaps the most significant as the introduction of ISRU to an architecture will likely reduce the threshold at which a reusable lander should be used.

Future studies may develop a more in-depth level of cost estimating for each subsystem with reusability. One potential method to accomplish this is to examine the additional work that must be carried out to make a subsystem reusable. The ability to adapt current expendable space system designs to be reusable is another interesting area for future research.

The impact of NASA’s budget on each architecture is another important area for future research as reusable landers require high development cost, which may be infeasible given NASA’s budget. Another avenue for future work is investigating the economic benefit of an architecture.

5.4 Recommendation for Implementing Reusability

An expendable lunar lander design was found to be the most cost-effective option for the currently planned Artemis program. However, this approach is not sustainable thus other factors in addition to cost must be included to determine the optimal level of reusability. Primarily, the ability of a design to support future missions makes reusable designs more favorable as it increases the number of missions performed. Additionally, the number of missions at which the trade-off occurs between the different levels of reusability depends on the refurbishment and repair costs required for each mission. For the best-case refurbishment and repair costs used in this analysis, an expendable system is more cost-effective than a reusable one for fewer than ten missions. As the Artemis program has only five surface missions planned, an expendable design is the most cost-effective. However, if the number of proposed missions
is increased to more than ten, hybrid reusability, where line-replaceable units are re-
placed in-situ after each mission, becomes the most cost-effective. A hybrid approach
allows for lower non-recurring development costs than those of a fully reusable lander
and technologies to be incrementally advanced as long as they adhere to the unit
interfaces. Subsystems with high reliabilities should be made reusable as they will re-
quire only modest modifications from their existing designs while subsystems with low
reliabilities should be made modular and thus easily replaceable. However, there is
a threshold where a fully reusable design becomes more cost-effective and thus more
sustainable than a hybrid one, but this threshold is above the number of missions
examined for this analysis.

Therefore, it is recommended the Artemis program uses a hybrid reusable lander
as it will enable sustainable exploration of the lunar surface. Missions back to the
lunar surface will only be the first step as humans voyage further into the solar system
and reusability will be there every step of the way.
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