OPTIMIZING STAKEHOLDER OBJECTIVES OF SPACE EXPLORATION ARCHITECTURES USING PORTFOLIO OPTIMIZATION

A Dissertation

Submitted to the Faculty

of

Purdue University

by

William J. O'Neill

In Partial Fulfillment of the

Requirements for the Degree

of

Doctor of Philosophy

December 2020

Purdue University

West Lafayette, Indiana

THE PURDUE UNIVERSITY GRADUATE SCHOOL
STATEMENT OF DISSERTATION APPROVAL

Dr. Dan Delaurentis, Chair

School of Aeronautics and Astronautics

Dr. David Spencer

School of Aeronautics and Astronautics

Dr. Karen Marais

School of Aeronautics and Astronautics

Dr. Robert Kenley

School of Industrial Engineering

Approved by:

Dr. Gregory Blaisdell

Head of the School Graduate Program

To my parents, sister, and brother.

To my girlfriend, Megan.

ACKNOWLEDGMENTS

First, I would like to thank my family for supporting my efforts. This would not be possible without the advice and encouragement from my father, mother, brother Brian, sister Bridget, and girlfriend Megan. My support structure of friends at Purdue helped me get by and make West Lafayette feel more like home especially David Mirkhaef, Matt Gripe, Nick Sinopoli, Shanleigh Thomson, Matt Steiner, Kelly Mann, Devon Parkos, Cesare Guariniello, Sam Rotella, and Marat Kulakhmetov.

Colleagues in the System of Systems group were a great source of advice and sounding board for different ideas. I'm thankful for the patience of Navin Davendralingam and Cesare Guariniello for answering countless questions related to SoS, optimization, and space architectures. Being a part of the Purdue-Marshall Space Flight Center team with Ashwati Das, Shubham Singh, Kshitij Mall, Steven Zusack, and Cesare Guariniello was a great personal experience and test-bed for many of the early versions of Programmatic Portfolio Optimization. Colleagues at Johnson Space Center that supported this work including Clark Esty and Ryan Whitley were great resources in helping apply the methodology. I'd like to especially thank Julie Kramer White and Dan Dumbacher for helping me connect with people within NASA and opening doors for me. The support in editing and revising done by Megan Smith, Madelyn Vandewalle, Mike Vandewalle, Molly Bannon, Cesare Guariniello, Professor Kenley, Don Tandy, and my father cannot be understated.

Lastly I would like to thank Professor Daniel Delaurentis and my committee, Professor Robert Kenley, Professor Karen Marais, Professor Daniel Dumbacher, and Professor David Spencer. Their guidance, advice, and encouragement provided a substantial help.

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ABSTRACT

O'Neill, William Ph.D., Purdue University, December 2020. Optimizing Stakeholder Objectives of Space Exploration Architectures Using Portfolio Optimization . Major Professor: Daniel D. Delaurentis.

The large number and significant variety of systems available for space exploration missions produce countless potential architecture combinations. Compounding this are the scheduling intricacies of system life-cycle phases, time dependent operational dependencies, as well as the uncertainty associated with each system and technology in terms of cost, schedule, and performance. Traditional architecting emphasizes the individual design of component systems over the wide-ranging and robust assessment of architecture options early in mission design. A top down method that can assess the capabilities, requirements, and risks associated with the diversity of available space systems and form optimal portfolios of interdependent systems is necessary. This dissertation describes and demonstrates a portfolio optimization technique that can design and assess Lunar space exploration architectures by optimizing on programmatic objectives such as cost, performance, schedule, and robustness while simultaneously accounting for system operational interdependencies and schedule dependencies of the selected systems. Several specific enhancements to the Robust Portfolio Optimization method are produced, resulting in the the novel Progarmamtic Portfolio Optimization (PPO) approach: including life-cycle phase modeling, variable capability sizing of systems, and multi-domain constraints to model time dependent objectives.

1. INTRODUCTION

Nations of Earth have long desired to explore and settle the distant planets and moons of our solar system but have been limited by technological and managerial challenges. Many of the challenges are common to System of Systems(SoS) type problems including the evolutionary nature, the vast range of distances over which systems interact, and the emergent behavior resulting from the complex interactions of constituent interdependent systems. Further challenging is the extensive number of potential systems and technologies and the resulting inability to properly assess trade-offs due to the exponential number of system combinations, complex interaction of systems, scheduling dependencies, and the diversity of system type, performance, cost, and schedule.

The crux of NASA's proposed future in human space exploration is a cisLunar orbital platform enabling human lunar surface missions with the horizon goal of landing humans on Mars. NASA has proposed avenues to accomplish these goals with a myriad of subsequent technical and strategic decisions that would need to be made. Further, private companies are developing their own vehicles and mission objectives for cislunar activities. While a bespoke, clean sheet architecture could be designed to accomplish mission objectives, it may be advantageous to integrate a portfolio of existing and future systems. This modular and reusable approach can aid in achieving overarching stakeholder objectives, minimizing both development cost and schedule, reducing risk through flight tested hardware, and encouraging commercial and international involvement. An abundance of potential systems exist or are in near-term developmental stages that could compose an optimal architecture. NASA's stakeholders and many of the companies tasked with developing Deep Space Habitats for NASA's NextSTEP-2 public-private partnership have chosen this modular strategy [1].

The questions that remain are which systems and technologies to select, when to develop, produce and operate them, and how they interact. Given the large number of potential choices for various systems and the differences in capabilities, cost, schedule, and robustness of each, the resulting combinatorial problem becomes difficult to evaluate in terms of overall architecture cost, performance, schedule and robustness. Compounding this is the scheduling dependencies that exist between systems, the cost of Design Development Test and Evaluation(DDT&E) of new technologies, and the impact of evolving stakeholder objectives.

NASA, the Department of Defense and commercial entities have built numerous tools to design and size space systems for specific roles within an architecture. Many of these tools have been crafted from extensive experience and literature review to accurately design that specific type of system. While these tools are adept for their use scenario, the ability to synthesize at the architecture level is often lacking. A new method, like the Programmatic Portfolio Optimization method presented in this paper, that can leverage this wealth of knowledge and tools and apply them at the architecture level is necessary.

Several methods exist that optimize space exploration architectures as networks of interacting systems including: the interplanetary logistics model [2], the Exploration Architecture Model for In-space and Earth-to-orbit(EXAMINE) [3], and the Generalized MultiCommodity Network Flow(GMCNF) model [4]. These methods often focus on minimizing the Initial Mass in Low Earth Orbit(IMLEO) by varying the parameters of the interacting systems as a means of estimating cost. Improving on this topic are methods that incorporate time-based modeling of system operations within the optimization model such as the time-expanded GMCNF [5, 6] which enables operational scheduling of elements. Chen et al further improved the method by using fully periodic Time-Expanded Networks(TENs) to make the computational model more scalable [7] and applicable to larger human space exploration architectures.

While these methods are effective at designing an architecture where every element is optimized for the reduction of mass, our approach emphasises treatment of the architecture programmatics related to the previously discussed modular architecture composed of systems of mixed development. Future space architectures will be collaborative in nature, a 'System of Systems' where each system is independently designed and managed. Additionally, impacts of architecture choices are often most apparent in the budget and schedule impacts due to the system development status. A portfolio optimization that enables the stakeholder to investigate the trade-space of both future and existing systems as well as their impact on budget and schedule metrics is necessary.

Space system development organizations have devised numerous techniques and approaches to evaluate which technologies to prioritize and invest [8, 9]. While these are powerful tools, they often do not completely assess all of the interdependencies within the architecture. These techniques either focus on how technologies correlate with stakeholder objectives or how they affect directly dependent systems and thus lack a direct connection to the stakeholder objectives at the architecture level.

The methodology demonstrated here addresses the aforementioned deficiencies by utilizing a robust portfolio approach with three enhancements. The basis is derived from Markowitz's modern portfolio theory [10, 11] which is widely used to compare risk and reward of selected options. It has since been applied to engineering problems by allowing constraints to be enforced within the optimization that represent system to system interactions. Several studies using Robust Portfolio Optimization(RPO) have been conducted involving different naval warfare scenarios by Davendralingam et al where mission performance was compared to mission cost and several forms of architecture robustness [12, 13]. Studies by Walton and Mehr have examined the uncertainty and development risk of space systems architectures using portfolio theory [14, 15]. The key enhancements differentiating our work are: 1) The addition of lifecycle phase scheduling constraints within the optimization while still accounting for the inter-dependencies between selected systems. This allows the user to compare various technologies and systems of varying levels of maturity or readiness and their impact on budget and schedule metrics. 2) The ability to optimize system sizing within the optimization process. Previous versions of RPO required fixed capabilities and requirements within the optimization and thus limit further optimization of the architecture. 3) The ability to optimize specific objectives for specific periods of time to optimize an architecture for evolving stakeholder objectives.

This dissertation is organized in chapters as follows: Chapter 2 details an overview of potential application areas with stakeholder analysis needs. Chapter 3 is a review of other tools and methodologies and identification of limitations. Chapter 4 is a detailed overview of the Programmatic Portfolio Optimization methodology. Chapter 5 is an example application of the methodology enhancements with simple scenarios. Chapter 6 is an example application of the methodology to a human Lunar orbit mission architecture. Chapter 7 is an example application of the methodology to a human Lunar lander mission architecture. Chapter 8 is an application of the methodology to a human Mars surface mission architecture. Chapter 9 is an application of the methodology to a stepping-stones style Moon then Mars architecture. Chapter 10 is an overview of the work and recommendations for future research. The appendix details the inputs used within the example applications as well as some additional figures and tables.

2. MISSION APPLICATIONS

Three different areas of application, or scope, have been identified where Robust Portfolio Optimization, with some enhancements, would be beneficial including: 1) a human lunar surface exploration mission,2) a human Mars surface exploration mission, and 3) a stepping-stones approach that combines the first two scope areas into an evolution of one to the other. These scope areas represent unique portfolio problems with unique objectives, requirements, time scales, and potential systems that require different analysis methods to obtain solutions for each unique portfolio problem. The aim of this section is to highlight key traits, major decisions, and systems for each of the respective scope areas.

The first scope area represents a human lunar surface exploration mission. Currently, NASA's goal of landing humans on the moon by the year 2024 is a combination of government contracts and several public-private partnerships called NextSTEP-2 [1]. A central theme of this approach is the proposal and selection process where different commercial partners propose their own plan for building an orbital platform and designing a landing architecture. The result of this phase of the NextSTEP program will be a number of unique modular architectures in which NASA can use, modify, or combine to fit their own cis-lunar architecture.

The second scope area represents a human surface exploration mission of the planet Mars. The concept of landing humans on Mars dates back to Wernher von Braun's "Das Marsprojekt" [16] and many studies have examined different approaches including NASA's Design Reference Architectures 1-5 [17–20], Dr. Robert Zubrin's Mars Direct concept [21], and several studies by the European Space Agency [22] and the Russia space agency Roscosmos [22]. While each study is unique, they all share the same difficulties of long transit times with high exposure to radiation and zero gravity, large propulsion system requirements, complex logistics problems, and the development of long-lead technologies. Many proposals solve some of these challenges with common traits including pre-deploying assets to the Mars surface and in-situ resource utilization by making propellant, oxygen, and water from the martian environment.

The third scope area covers the timespan from the current day, through the cislunar architecture and eventually to the execution of a Martian mission. A major theme of this scope area is the adoption of technologies that benefit both a lunar and Mars mission. Key here is that while some technologies may benefit both scope areas, they may be sub-optimal choices for a single scope area but provide an overall benefit to the combined scope area.

2.1 S1 - NASA Lunar Human Exploration Architecture

After the Apollo program ended, the United States focused its efforts on Low Earth Orbit [23] but have been plotting a return to the moon since the early 90s, including various proposals ranging from short duration lunar landings to more rooted missions with lunar surface bases or habitable rovers [24]. International organizations have proposed multilateral plans including the Lunar Village concept [25]. A mission architect has many potential operational concepts at hand to try and maximize stakeholder objectives and achieve requirements. Several key concept-level decisions define the architecture trade space and are listed below. Further questions remain including the use of In-Situ Resource Utilization(ISRU) for propellant production, propellant type, crew size, and landing location.

1. **Staging Orbit** - Whether staging operations should be held in a Low Earth Orbit, a more distant lunar orbit like a Near Rectilinear Halo Orbit (Current plan), or a Low Lunar Orbit.

2. **Number of Ascent and Descent Elements** - The number of systems or elements required to bring crew members from a staging point to the Lunar surface and back. This ranges from single stage architectures to as many as four propulsive elements.

3. <u>**Duration of surface stay**</u> - Length of stay on lunar surface ranging from short single day missions to multi-week expeditions.

In 2019, the Trump administration called for NASA to have "boots on the moon" by the year 2024 [26]. Recent policy directives have heavily shaped the potential solutions to the above questions. A hotly debated trait of the administration's plan is the use of an orbital platform in a Near Rectilinear Halo Orbit(NRHO) that could enable assembly and refueling of lander elements as well as the use of less expensive commercial launch vehicles for delivering lander elements [27,28]. Given that funding and development has already begun, it firmly answers the question of staging orbit. This platform represents a departure from the Apollo program where every piece of the mission was launched on a single massive rocket to Low Lunar Orbit(LLO). Its necessity results from decisions made two decades earlier during the Constellation program. The Orion spacecraft does not have the performance to complete both the capture and the departure burn at a LLO orbit. During the Constellation program the Altair lunar lander was responsible for the capture burn, however the launch performance of the Space Launch System rocket is not sufficient to launch both Orion and a propulsive element and therefore Orion by itself is not suitable for operations at LLO. However it does have the performance to get to and from the Near Rectilinear Halo Orbit where the "Gateway" would be located. Shown in Figure 2.1 is a trajectory transfer map of the main orbit options being compared for a lunar mission [29] that details the approximate required change of velocity(ΔV) between each staging point.

While the NRHO requires more propulsive capability of the lander element, the trade-off is the ability to send larger cargo to Lunar orbit via low-cost commercial heavy lift vehicles on less demanding Ballistic Lunar Transfer(BLT) trajectories than a LLO transfer which would require much larger rockets or smaller payloads. A third potential option is a High Lunar Orbit which is a compromise between the two options. A fourth option includes the use of Low Earth Orbit as a staging point using commercial heavy lift vehicles to separately launch and rendezvous the Orion crew vehicle and a propulsive stage. This could negate the requirement for expensive

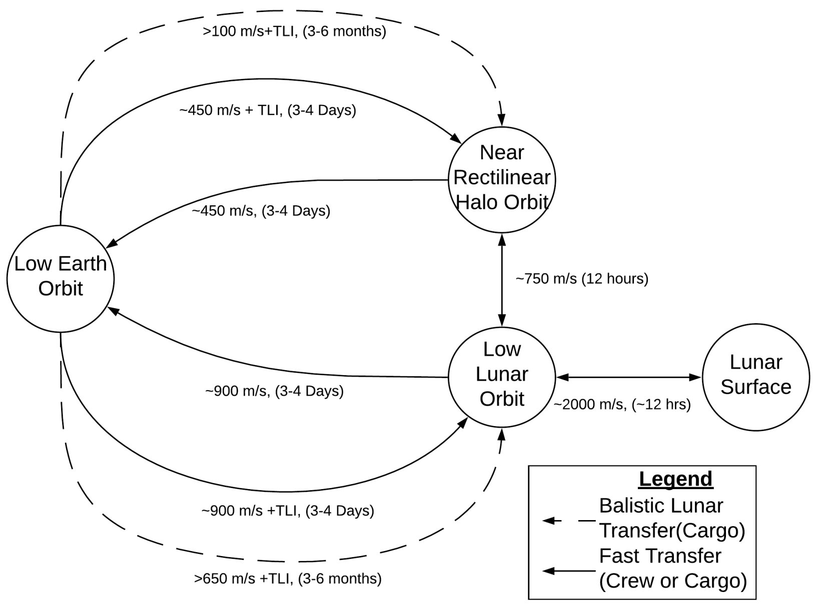


Fig. 2.1.: Low Lunar Orbit and Near Rectilinear Halo Orbit staging options as and transfer cost as detailed in [29]. This diagram details the approximate velocity change required to transition from one orbit to another.

government super heavy lift vehicles but would require substantial changes to Orion as well as the development of a new propulsive stage capable of orbit loiter, rendezvous, and docking. Listed in Table 2.1 are some of the benefits and drawbacks of the NRHO and LLO staging option.

Another major decision facing stakeholders is the number of lander elements to descend from the staging point to the lunar surface and then ascend from the surface back to the staging point. This decision is tightly correlated with the staging point decision. Several options exist for the number of elements with many varieties in how those elements are utilized. Architectures with elements ranging from a single stage vehicle to those requiring three or more have been proposed. Due to the estimated 15t delivery to NRHO limit of commercial heavy lift launch vehicles, the 1 and 2 stage lander architectures become difficult to infeasible based on propulsive sizing. A

Table 2.1.: Benefits and drawbacks of different lunar architecture orbit options

	NRHO Approach	Apollo Approach
Benefits	• Ability to send lander elements to NRHO with low-cost commercial launch vehicles • No propulsive cost of reaching high science value sites at poles • Stability of halo orbits and low-cost delivery enables low cost re-usability of lander elements	• Fewer lander elements required • Less complex architecture • Fewer launches
Drawbacks	• Additional 750 m/s of velocity change required for both transit from NRHO to LLO and LLO to NRHO	• Requires redesign of Orion crew vehicle or new design • May require an upgraded Space Launch System or other super heavy launch vehicle • For polar landing sites, 3100 m/s additional velocity change for plane change maneuver • Station-keeping in LLO or HLO poses difficulty for re-usability

3 element lander architecture is feasible but may require refueling prior to a mission given certain propulsive technologies.

The length of stay on the surface is a major decision that is tied to stakeholder value. Short missions on the surface can accomplish a relatively large amount of science but longer durations allow for more frequent and longer duration experiments on the surface. However, longer duration surface stays require more crew consumables in terms of oxygen, water, food and other soft goods. Long duration stays also require a surface habitat as well as power and surface systems. The potential for using propellant derived from ice on the lunar surface could eliminate the need for refueling from earth and enable massive cost savings. For missions using NRHO as a staging ground, the length of stay is dictated by the period of the orbit leading to either single day stays or stays of increments of approximately seven days [29].

Key to NASA's acquisition plan is the NextSTEP-2 public-private partnership [30]. This platform will leverage existing systems and technology from the commercial sector. As part of the "Next Space Technologies for Exploration Partnerships-2" [31], NASA has solicited designs from 6 competing companies for a cis-lunar deep space habitat and several proposals for technologies including specific habitat systems, in-space manufacturing, power and propulsion systems, and In-Situ Resource Utilization (ISRU) technology. The focus of the partnership is to develop deep space habitat designs for orbit around the moon. An effective habitat comprises a pressurized volume plus an integrated array of complex systems and components that include Environmental Control and Life Support Systems (ECLSS), Power and Thermal management systems, Command Data Handling(CDH), logistics management, radiation mitigation and monitoring, fire safety technologies, crew health systems, and docking capabilities. From these proposed architectures, NASA plans to advance promising plans for further development and prototyping.

One of the stated objectives of the Artemis Program is to use the lunar environment to develop and test technologies that benefit a future Mars mission [30]. This environment is ideal for the testing of several key technologies such as deep space habitat systems and advanced propulsion concepts like Solar Electric Propulsion(SEP) or Nuclear Thermal Propulsion(NTP). It's unclear how much surface habitation will be conducted but there is the possibility of surface habitats and In-Situ Resource Utilization(ISRU). ISRU has the potential to take resources on the lunar surface and turn them into oxygen for breathing, water for drinking and propellant for propulsion.

Any lunar architecture will require a plethora of new systems and added complexity. The NASA plan will need the orbital habitat as well as the systems to launch all of the components to space, send them to lunar orbit, and assemble them. Additionally, logistic missions to resupply and refuel the Deep Space Gateway(DSG) and lander elements will be required. While the DSG will be involved in testing of crucial technology for future missions, it will require less internal space, power, and dry mass than the ISS and will be comparatively small in cost. NASA plan or other, the two or three element lander architecture will need to be designed, integrated, and launched as well. If a lunar surface base is attempted, then the habitats, supporting infrastructure, cargo landers and surface mobility will also need to be designed and operational. For each of these potential systems there are many different choices for each.

A non-exhaustive list of the general classes of systems is as follows:

- Launch vehicles
- Launch facilities
- In space propulsion systems
- Orbital propellant depots
- Crew vehicles
- Deep space habitat modules
- Power and propulsion modules
- Lunar lander elements
- Lunar surface power
- Lunar habitat systems
- Lunar mobility systems
- In-Situ Resource Utilization systems
- Deep Space Network and communication systems
- Logistics and supply modules
- Mission Control Centers

Across these classes, some systems exist, while others are conceptual (yet to be designed). Each of the potential systems is differentiated in cost, performance, schedule, operational uncertainty, financial uncertainty, and compatibility. A selection, or portfolio, of these unique systems could be combined to create a working architecture that meets overall mission goals and satisfies requirements. The difficulty from a stakeholders perspective is that the assembly of an optimal set of systems requires more than a simple interaction of parts, and it needs to account for the scheduling of systems as well as the assessment of total cost and annual budget. For instance, a stakeholder may have to choose between a system that could be deployed sooner or a system that may offer long-term cost savings but would delay the schedule of the first lunar landing. The standard design tools and systems engineering practices break down when trying to solve this problem. Because of the large number of potential systems, there exist near infinite potential portfolios with different stakeholder value. Due to the intractability of the number combinations, conducting trade-off analysis becomes difficult to impossible.

2.2 S2 - Mars Human Exploration Architecture

Mars is substantially harder to reach than the moon requiring more delta-v, longer travel times, more difficult Entry Descent and Landing(EDL), and more difficult ascent from the surface. The nominal mission duration is nearly 2 years and thus requires redundant systems and adequate logistics and supplies for the entire mission. The crew during this long mission is subjected to deep space radiation and reduced gravity for much of this time. Many of the proposed systems and technologies are of a low Technology Readiness Level(TRL) and require significant development and impose operational and financial uncertainty on the mission. Some of the key concept level decisions stakeholders must contest are:

1. **Transfer Trajectory** - Either a conjunction class or opposition class trajectory

2. **Deployment of Mars Surface Systems** - Whether to pre-deploy cargo or arrive with it

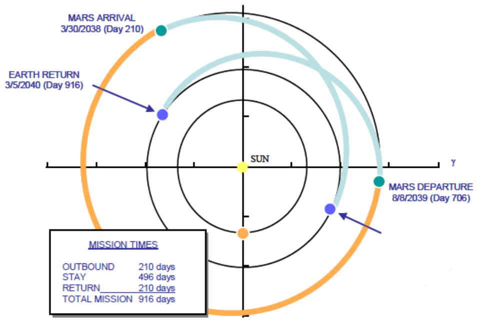
3. **Mars Capture Method** - Whether to use aerocapture around Mars or use a propulsive method

4. **Mars Ascent Propellant** - Whether to produce propellant on the martian surface for the return journey

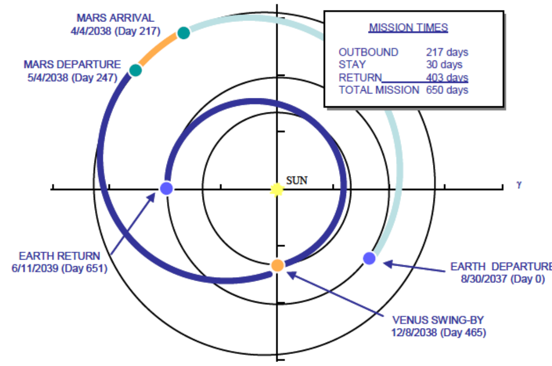
5. **Interplanetary Propulsion Type** - The propulsion type for in space transit

The difference in the orbital period of the two planets results in a closest approach every 2.1 years. A mission manager must plan to launch crew or cargo during these launch windows. A short response resupply or rescue is thus impossible. There are two resulting types of trajectories that result in two different classes of missions: opposition and conjunction class missions. The conjunction class mission incorporates a long stay(500+ days) on the martian surface whereas the opposition class mission incorporates a short stay(30-90 days). While the long stay mission does incur a longer overall mission time, the crew spends a substantially longer time on the surface with almost the same amount of time spent in deep space. A majority of mission proposals select the long stay mission type. Two other subcategories of long stay trajectories include a “fast transit” which reduces the time spent in space with a higher energy transfer and a “low energy” transit used for cargo which requires less energy but takes longer.

Unlike the moon, Mars has a thin but still substantial atmosphere composed of 95% CO_2 at about 1% the pressure of Earth’s atmosphere. While thin, it still poses a challenge to Entry Descent and Landing in that the entry velocity requires a heat shield and parachutes are not very effective below Mach 1. Thus propulsive landings with rocket engines are required. The CO_2 atmosphere and hydrogen from water ice permits the production of methane rocket fuel and oxygen for Mars ascent and return to earth. Small scale ISRU testbeds have demonstrated this ability on earth, but not to the required scale and not on the martian surface.



(a) Conjunction class mission



(b) Opposition class mission

Fig. 2.2.: Trajectory design of opposition and conjunction class Mars missions from NASA Design Reference Architecture 5 [32]

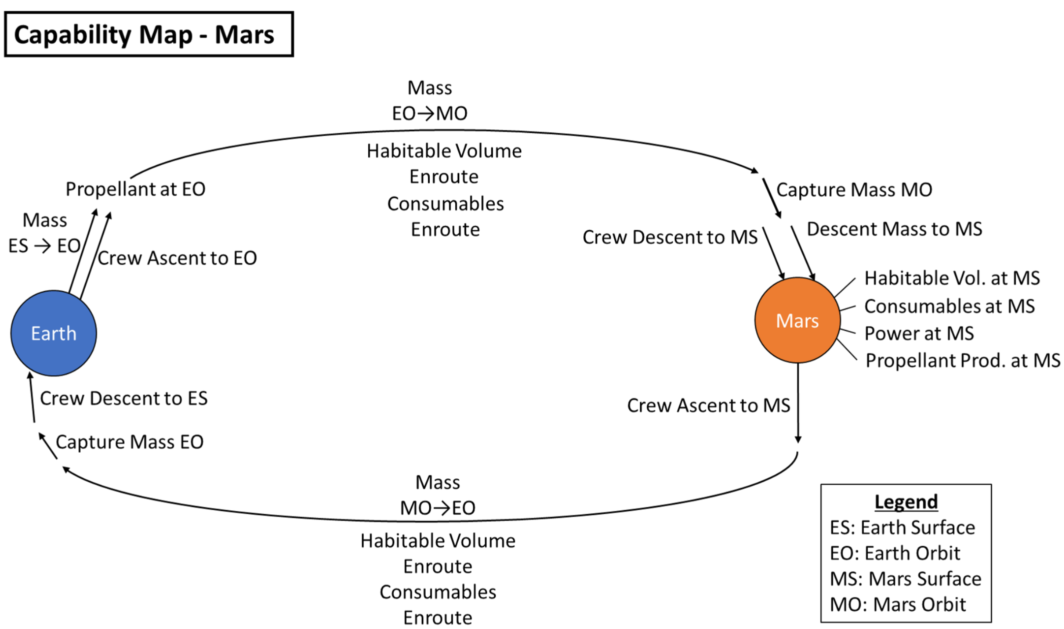


Fig. 2.3.: High level mission Con-ops detailing types of capability transfer within architecture

Like the lunar mission, the mission architect must make several major mission level decisions and system trades for many of the different system types. The challenge for the architect is finding the right combination of systems that achieve or optimize stakeholder objectives. Demonstrated in Figure 2.3 is a capability mapping of a potential Mars mission that demonstrates at a very high level the integration challenge of the various types of capabilities. Many of these capabilities are time dependent and thus have strict scheduling constraints to meet launch windows.

- Launch vehicles
- Launch facilities
- In space propulsion systems
- Orbital fuel depots
- Crew vehicles
- Deep space habitat modules
- Power and propulsion modules
- Mars Capture systems elements
- Mars crew landers
- Mars cargo landers
- Mars surface power
- Mars surface habitat systems
- Mars mobility systems
- Mars Ascent Vehicles(MAV)
- Earth Re-entry systems
- Mars ISRU plant
- Deep Space Network and communication systems
- Logistics and supply modules
- Mission Control Centers

2.3 S3 - Multi-Epoch Multi-Destination Mission

The third scenario focuses on the NASA stepping-stones approach for exploring the moon and sequentially Mars [33]. Proponents of this approach have lauded the benefits as it allows technologies and systems required for a martian mission to be tested in a less demanding environment. The most likely sequence is a campaign utilizing a Deep Space Habitat and Lunar landings followed by a campaign leveraging the experience gained from deep space orbital habitats, landing vehicles, and surface utilization to land on Mars. This scope area is fundamentally the fusion of two different architectures, from which one evolves to become the other.

A key piece of NASA's proposed lunar mission is an orbital station that that acts as a crew habitat, docking station for lunar lander elements, and has the potential to provide power and propellant refueling. Many technologies can be tested in this environment that would be useful for a Mars mission. Some of these key technologies include advanced propulsion systems such as Solar Electric Propulsion (SEP) and Nuclear Thermal Propulsion (NTP), as well as long term cryogenic storage of propellants and the use of propellant depots. However, there are technologies required for Mars that cannot be tested in-depth in the lunar environment such as some of the In-Situ Resource Utilization(ISRU) technologies and the Entry Descent and Landing(EDL) technologies.

Unlike the ISS, any trip to Mars does not allow for timely resupply or an emergency return to Earth. Required supplies must either be predeployed to the destination or brought with the crew. A high value is thus placed on efficient life support systems that can recycle and filter air and water efficiently. These are systems that are proposed being evaluated and tested in the cislunar environment. Developing and testing these technologies in the cislunar environment effectively buys-down some of the risk and uncertainty in terms of the reliability for use in a Mars mission.

The systems required for the stepping-stones scenario are those of the lunar and martian scenarios with one significant distinction. There may be systems that were not optimal for either individual scope area but are optimal for the stepping-stones architecture. In a sense these systems are globally optimal for the stepping-stones architecture but not locally optimal for the individual architectures.

Development of PPO is driven by the need to address portfolios of systems at each time point and how carrying systems and technologies forward to the next mission time point affects decision-making. Thus, PPO would help find the best combination of systems to develop for early portions of the mission that in turn benefit the later events of the mission. Systems that are required for Mars missions may be adequate but not optimal for cislunar missions and thus may not be immediately obvious. This approach allows mission managers to see the impact of technology investment across an entire architecture. In another manner, this approach could help identify cislunar architectures that are more flexible to possible changes in later mission goals.

In terms of analysis tools, many of the same needs and requirements of the previous scope areas are true of the stepping-stones architecture with one additional requirement. When planning this type of architecture the stakeholder must account for objectives that may change over time. For instance, in the first half of the architecture, the stakeholder will value a lunar mission and may have little desire for a Mars mission and for the second half would have high value for a Mars mission but little value for a lunar mission. Optimizing for only a single mission type or each mission individually does not satisfy the stakeholder needs of an evolving architecture.

2.4 Summary

While there are several common themes between the three scope areas, they are differentiated by the overall mission goals, required systems, the time frame surrounding the decisions being made, as well the level of abstraction of the systems. Table 2.2 summarizes, compares, and contrasts the different scope areas.

In a way, each scope area can be thought of as a necessary step that leads to the next scope area in that they overlap each other in time and breadth. For instance, the NextSTEP program is a phase in NASA's plan for a cislunar mission and the cislunar mission is a phase of the stepping-stones program. Each scope area differs in terms of technology development. The first scope area does not require large expenditures to develop new technologies whereas the later part of the stepping-stones architecture requires the research and development of new technologies. Developing and testing these new technologies in the cislunar environment may be suboptimal in terms of lunar exploration, but is a necessary step in order to develop the required systems for a Mars mission. The effects of the decisions that NASA makes in the first scope area ripple through the later scope areas.

The three different scope areas identified form opportunistic test cases for the application of a portfolio theory methodology such as Robust Portfolio Optimization. RPO is a technique that helps SoS managers identify not only optimal systems, but

Table 2.2.: Summary of different scope areas including traits and analysis methodology needs

Scope Area	Key Traits	Requirement on Architecture Analysis Methods
Lunar Human Exploration Architecture	• Missions duration measured in weeks • Delta-V budget of 20 km/s	• Portfolio Selection • Schedule Analysis • Financial robustness • Operational robustness • System sizing
Mars Human Exploration Architecture	• Missions measured in years • Delta-V budget of 30 kms • Launch windows every 2 years • Pre-deploy of assets and ISRU highly beneficial	• All Above+ • Technology Development
Multi-Epoch Multi-Destination Mission	• Multiple missions • Multiple destinations • Changing objective with time	• All Above+ • Time dependent objectives

optimal systems that are robust in terms of uncertainty and risk. The application of the RPO methodology alone would lend well to these three scope areas. However, with some critical enhancements to the methodology, a greater impact can be made by assessing the time dependent impact of specific decisions.

3. LITERATURE REVIEW AND GAP ANALYSIS

Guariniello et al. [34] have called for a change in how space mission architectures are traditionally designed and planned. Their paper makes the case that current space exploration planning utilizes a “bottom-up” approach that relies on a team of technical experts designing a single-threaded architecture. Such methods struggle to properly account for the size and complexity of space system architectures and often limit the scope and design space of such studies. Additionally, the number of decisions to be made and systems involved create a massive combination space that makes it difficult to assess all possible decision choices and can lead to overlooking of optimal solutions.

A top-down approach is needed to account for the complexity of interactions, variability of systems, evolutionary nature of the SoS, and both the managerial and operational independence of the constituent systems. This approach is not meant to replace the traditional approaches but to complement and align them. The Robust Portfolio Optimization approach is one method identified that could link the bottom up approach to the top level view of stakeholder objectives, budgets, and architecture entities. The following subsections detail some of the current techniques related to RPO and identify literature gaps.

3.1 System of Systems Research

While the definition of a System of Systems(SoS) is not uniquely defined, the definition by Maier constitutes one of the most agreed upon: “a SoS is a special system, often complex, consisting of a collection of systems that collaborate for a unique purpose but also retains operational and managerial independence.” The five heuristics highlighted by Maier are useful for making this distinction and are paraphrased below [35]. These traits are useful in understanding the difference between a monolithic system and a System of Systems and why certain tools and methodologies that are valid for a system are not valid for the SoS.

Table 3.1.: Traits of System of Systems defined by Maier [35]

Property	**Description**
Operational Independence	Elements have their own useful purposes outside the SoS
Managerial Independence	Elements operate independently and are provided unique purposes by owners and operators
Geographical Distribution	Elements are physically distributed and can be linked by information (not mass, energy)
Evolutionary Development	The SoS is constantly changing; elements are being added and removed
Emergent Behavior	Properties of whole emerge from the assembly and interaction of elements

A human space exploration architecture demonstrates many of the traits highlighted by Maier and is an excellent example of an SoS. Many of the systems have purposes outside of the space exploration architecture including communication systems and launch services. Many of the systems are operated by different government, commercial, and foreign entities. Many of the required systems are well beyond geographically co-located and operate throughout space. Historically, space exploration architectures have evolved in terms of the progression of their capabilities. The early days of the American space program is a demonstrative example which began with suborbital flights, evolving to multi-day missions, to lunar flyby missions, and finally to landing a man on the moon. The International Space Station is another example in that it began with the Russian Zarya module and through the collaboration of many different nations evolved to the football field sized station of today.

NASA and industry have decades of experience, knowledge, and tools for designing space systems. The majority of these tools focus on specific systems or subsystems. While these tools are adept for their use-case, there often is a lack of ability to synthesize them at the architecture level. For instance, countless analysis and tools exist for assessing spacecraft propulsion, surface habitats, and landing systems but few methods exist to optimize the outputs of this analysis. A method that can leverage this wealth of knowledge and tools and apply them at the architecture level is necessary.

Delaurentis has proposed a three-phase methodology to model and analyze transportation systems that can be applied to other SoS [36]. The three phase model includes the definition phase, the abstraction phase, and the implementation phase. During the definition phase, the user frames the problem and identifies the operational context, status quo, barriers, and stakeholder goals. Key to the definition phase is identifying system levels of abstraction and system hierarchy. Many governing ground rules and assumptions are identified in the definition phase as well. During the abstraction phase, the user identifies the governing physics of the problem, identifies the inner workings of the architecture, and estimates the properties of potential systems. Lastly, in the implementation phase, the user performs the actual modeling and analysis of the whole SoS, exploring the behavior, dynamics, and features of the systems involved.

Many Systems of Systems can be visualized and analyzed as graphs or networks. Examples include air transportation networks, the internet, power grids, disease spread, and military command and control networks. Analyzing SoS using network theory can be split into assessing pre-configured networks and designing or optimizing a network that represents the SoS.

Many techniques to quantitatively assess network properties have been developed. Newman and others have identified many properties of networks including degree correlation, network growth models, and network centrality amongst many others [37]. These properties can be useful for understanding the inner workings of an SoS and how it can be improved. Some properties are especially useful for space based SoS, such as the work by Albert in identifying weak points to improve the network resilience against loss of nodes [38]. While these are important properties, their application favors analysis of existing networks but nonetheless are still educational when designing an architecture.

System Operational Dependency Analysis(SODA) is a methodology devised to investigate the impact and propagation of failures or partial failures of systems in an interconnected SoS. SODA models an SoS as a network and uses: A) a set of parameters to characterize the relationship between connected nodes as well as accounting for B) the Self Effectiveness, or health, of each system. While SODA can be used to compare different configurations of an SoS and the impact on robustness, it requires that the topology and relations between those systems are known a priori. SODA has been applied to both human lunar and martian architectures to assess robustness and resiliency of the architecture [39, 40].

Several methods exist that optimize space exploration architectures as networks of interacting systems including: the interplanetary logistics model [2], the Exploration Architecture Model for In-Space and Earth-to-Orb(EXAMINE) [3] and the Generalized MultiCommodity Network Flow(GMCNF) model [4]. These methods often focus on minimizing the Initial Mass in Low Earth Orbit(IMLEO), as a means of estimating cost, by varying the parameters of the interacting systems.

Some of these methods can be categorized as "Space Logistics," which is defined by the American Institute of Aeronautics and Astronautics(AIAA) Space Logistics Committee as "the theory and practice of driving space systems design for operability and managing the flow of material, services and information needed throughout the system life-cycle." In practice this relates to the sizing and operational concept of a space exploration architecture while accounting for consumable resources like system mass, propellant, and consumables. Metrics like initial launch mass or delivered payload can be optimized for specific missions. Massachusetts Institute of Technology pioneered some of the early space logistics research when they launched the Space

Logistics Project in coordination with NASA's Constellation Program. Central to this project was network modeling of space architectures by Taylor et al. where terrestrial logistics modeling tools were adapted for space transportation [2]. The central theme of this method applies network theory in modeling the architecture as nodes and arcs where resources are transferred between locations and systems. Improving on this topic are methods that incorporate time based modeling of system operations within the optimization model such as the time-expanded GMCNF [5,6]. Chen et a. further improved the method by using fully periodic Time-Expanded Networks(TENs) to make the computational model more scalable [7]. Numerous tools, including work by Shull, have been developed to help model these logistics problems both in terms of design and assessment of existing networks [41].

While these methods are effective at designing an architecture where every element is optimized, our approach emphasises treatment of the architecture programmatics related to the previously discussed modular architecture composed of systems of mixed development. Future space architectures will be collaborative in nature, where each system is independently designed and managed. Additionally, much of the effects of these architecture choices are apparent in the budget and schedule impacts due to the system development status. A portfolio optimization that enables the stakeholder to investigate the trade-space of both future and existing systems as well as their impact on budget and schedule metrics is necessary.

3.1.1 Multi Stakeholder Dynamic Optimization

Multi Stakeholder Dynamic Optimization(MUSTDO), is a technique used for assessing multistage SoS system acquisition with resource constraints, uncertainty and competing stakeholders [42]. MUSTDO is applicable to an acknowledged SoS problem where there is some authority or manager that has partial control of the multiple stakeholders that may have conflicting goals. The goal of the SoS manager is to maximize a capability over time while managing resources amongst the SoS participants who may have competing objectives. MUSTDO employs a transfer contract mechanism to model how SoS participants share resources to maximize SoS capability at the global level. MUSTDO applies Approximate Dynamic Programming(ADP) to relate approximate future values of capability with the value of transfer contract pricing. ADP is a well recognized method for addressing complex multistage decision making problems with uncertainty in operations research. In essence, MUSTDO selects optimal systems at the participant perspective that mirror the optimal selection of systems at the SoS managers perspective through the use of the transfer contract mechanism. While MUSTDO solves a portfolio problem that evolves over time, it does not determine how systems interact with each other and does not account for the development and production scheduling of said systems.

3.1.2 Robust Optimization and Portfolio Based Methodologies

Portfolio theory approaches were designed in order to assemble the most promising investment portfolio given uncertainty by Harry Markowitz [11]. The problem statement focused on how to predict which stocks would grant the highest return given the predicted performance and the inherent correlated uncertainty. This technique allowed the user to gauge risk aversion in terms of how the portfolio of stocks is selected. There are some obvious parallels between the selection of stocks and the selection of future technologies, but the application of a financial engineering tool to a systems engineering problem requires a bit of modification. Much attention has been paid to robust optimization of portfolio selection problems. The term robust optimization has taken many forms but generally is concerned with the protection of a decision against uncertainty. In this sense, the robustness or protection guaranteed is evaluated for the worst case scenario of possible uncertainty. Gabrel identifies two forms of potential uncertainty: 1) uncertainty on feasibility and 2) uncertainty on an objective value [43]. Uncertainty in feasibility aims to provide a feasible solution that is optimal but provides a specific probabilistic guarantee against constraint violation.

Several methods and formulations exist for obtaining optimal robust solutions in regards to feasibility including work by Ben-Tal and Nemirovski [44]. The difficulty associated with this approach is that robust solutions often severely deteriorate the objective value. The formulation Provided by Bertsimas and Sim provides a method of guaranteeing the feasibility of a solution, within a degree of confidence, without adversely affecting the objective function [45]. This formulation is one of several used in the upcoming analysis. Uncertainty in solution parameters can also impact the objective function directly. Many formulations exist to determine robust and optimal solutions for the worst case effect on the objective function. Several techniques exist for robust optimization of a single objective including Thiele who investigates cost uncertainty and Gancarova and Todd who investigate an inaccurate measure as the objective function [46, 47]. Work by Tutuncu applies Semi Definite Programming to solving the mean variance problem to a multi objective robust optimization problem [48]. By selecting a specific uncertainty set, the solution space becomes bounded and is able to be solved. This is one of the formulations used in the upcoming analysis.

Robust Portfolio Optimization (RPO) is a portfolio theory based optimization tool for comparing different selections, or portfolios, of systems that combine to meet overall Systems-of-Systems requirements and effectively accomplish an overall goal. These systems are affected by constraints that come from technological, operational or budgetary concerns as well as system to system integration. This method has its roots in financial engineering where it is used to maximize expected profit while minimizing the risk of a collection of investments. As a result, it is well suited for comparing risk and reward of selected options. It has since grown to apply to engineering problems by allowing constraints to be enforced on the solutions. In the engineering sense, this can be used to help managers choose which technologies to invest in given uncertain capabilities. What differentiates RPO from other forms of Multidisciplinary Design Optimization methods, is its basis in network theory. Each system is highly integrated into the larger SoS by its respective capabilities and requirements. These requirements are satisfied by other nodes in the network allowing for a collaborative operation.

The interactions between nodes are modelled by the following five rules grounded in network theory: capability, requirements, compatibility, relay, and bandwidth which are defined below:

- **Capability** - Each node has an upper bound on the capabilities that it can provide.
- **Requirements** Each node has requirements that must be met in order for it to function. These requirements can be met by capabilities of other connected nodes.
- **Compatibility** - Certain nodes can only connect to other nodes based on pre-established rules.
- **Relay** - A node can transfer a finite amount of capability between nodes.
- **Bandwidth** - A finite number of connections can be made to other nodes.

The approach to solving a problem of this class is similar to traditional process for the engineering design of a system [49]. First, mission objectives and requirements are defined. A functional decomposition is completed in order to determine classes of necessary systems of which unique systems can be associated to each class of system. Next, a library of unique systems with values for each of the five network constraints as well as cost and uncertainty are defined. In regards to space exploration architectures, these systems can range from different launch vehicles, habitat systems, power systems, propulsion systems, and crew return vehicles. Each system has a different associated cost, performance, and requirements to function. This lends well to the current status of the space industry, where there are often several providers of systems with similar functionality. These systems combine to form a library of possible choices that are used in the optimization. A mixed integer optimization scheme is applied to find a portfolio of systems that maximizes key mission objectives given the network constraints of the individual systems.

3.2 Technology Roadmaps and Developmental Dependencies

Many of the necessary technologies required for a human Mars exploration mission lack the technological maturity for flight application on such a mission. NASA needs to invest in these technologies in order to enable such a mission. Fixed budgets, competing technologies, and uncertainty in performance and development make the decision on how to invest difficult.

NASA and other organizations have come up with numerous techniques and approaches to prioritize which technologies to invest and prioritize [9, 50]. While powerful tools, these techniques do not completely evaluate the system to system interactions within the architecture. These techniques either focus on how these technologies correlate with stakeholder objectives or how they affect directly dependent systems. However, these techniques rarely assess how decisions affect the entire architecture. For instance, Solar Electric Propulsion technologies meet stakeholder goals of improving propulsive efficiency and may directly affect launch vehicle design and selection, but impacts such as the potential benefit of additional cargo capacity and what that can mean for redundancy or supply margins are not examined. It also becomes difficult to measure the impact of these decisions while other decisions are being made, i.e. when concurrent decisions are being made. This makes ranking the importance of multiple decisions or groups of decisions difficult.

In any architecture or program, there will be a schedule for how systems are designed and built. This reduces to a set of tasks that depend on each other and must be completed in a specific order. Knowing the order of tasks and what are the most critical tasks to accomplish is of high value to program managers. There have been several attempts at creating methodologies to analyze and even optimize possible schedules. One of the first attempts was a combination of Program Evaluation and Review Technique(PERT) [51] and Critical Path Method(CPM) [52]. This method combined technique examined the order of which tasks must be fully completed and derived the optimal schedule or the schedule with least possible delays.

Individual system development can be thought of as a task in these algorithms. In system development there exist systems whose developmental progress depends on the development of another system. These have been characterized as developmental dependencies [53]. These dependencies arise from a number of reasons including technology dependence, system sizing or manufacturing processes.

In PERT and CPM, these dependencies are complete in that each task must be fully completed before the next can begin. In reality, there exist many examples where full completion of a predecessor system is not necessary. Instead these dependencies can range from being complete dependencies where development of the dependent system cannot begin untill the development of the preceding system has completed or partial dependencies where the dependent system needs only partial completion of the preceding system.

Guariniello and Delaurentis demonstrates the modeling of partial developmental method using System Developmental Dependency Analysis(SDDA) [53]. In this method, dependencies are modeled using a two parameter relationship that accounts for the partial development requirement as well as how the development schedule responds to delays encountered by predecessor systems. This method has been applied to a number of architectures including naval warfare and space exploration missions [53].

NASA and the DoD have developed the Technology Readiness Level metric for evaluating the developmental maturity of a technology. This scale ranges from the conceptual level(Level 1) to operating systems and technologies(Level 9) [54]. Kenley et al. have developed models for evaluating the cost and schedule time required for transitioning between TRL levels [55]. While these metrics and methods do not track the developmental dependencies between systems in an architecture, they do provide methods for evaluating the individual development of each system. These methods could potentially be combined with a dependency analysis like PERT of SDDA to more accurately evaluate both cost and schedule impacts of a specific architecture. Through the use of workshops composed of technical experts NASA has developed a tool and database called the Technology Cost and Schedule Estimation tool (TCASE) [56, 57]. However, this database is not available for academic use.

3.3 Literature Gap

While each of these methodologies give decision makers manager more insight into architecture design problems and their possible solutions, there is room for improvement. The combinatorial complexity of space architectures and the interactions between the inherent systems create issues that the RPO methodology, to some degree, can alleviate. It is postulated in the coming sections that RPO, with some modifications, can be an effective tool at comparing architecture choices in terms of cost, performance, risk, schedule, and/or other stakeholder metrics.

Some of the identified literature gaps are listed below:

- Many technology road maps focus on how a technology correlates to a perceived benefit without examining system interactions and thus fail to assess the impact across an entire architecture. Therefore, comparing investment between technologies is difficult or inaccurate at the architecture level.

- The bottom up approach that is prevalent in most architecture studies limits SoS managers from effectively comparing possible combinations of systems and technologies in terms of high level stakeholder objectives like cost, schedule, robustness, and performance.

- The current version of RPO lacks the ability to assess schedule impacts of different technology choices and the combined effects of multiple technology choices. For space systems where there is a wide range of system maturity, this is problematic.

- The current version of RPO lacks the ability to optimally size systems within the architecture and requires knowledge of system capabilities prior to optimization.

For architectures with cascading inter-dependencies like space systems this is problematic.

- The current version of RPO lacks the ability to optimize stakeholder objectives for specific time domains, which is helpful in planning stepping-stones space architectures.

4. METHODOLOGY

An enhanced version of Robust Portfolio Optimization(RPO) is the proposed methodology for selecting candidate systems that form feasible architectures that accomplish overarching mission goals while assessing cumulative cost, schedule, performance, and risk [58]. These systems are collectively affected by imposed constraints due to technological, operational, or budgetary concerns. The method demonstrated here extends the Robust Portfolio Optimization formulation detailed by Davendralingam [58] via the addition of three enhancements: system life-cycle phase modeling(E1), variable capability sizing(E2), and time dependent objective domains(E3). This enhanced version of Robust Portfolio Optimization will henceforth be called Programmatic Portfolio Optimization(PPO) given that it can select optimal portfolios of interconnected systems in terms of stakeholder programmatic objectives such as cost, performance, schedule, and robustness.

4.1 Robust Portfolio Optimization Methodology Formulation

The basis of Programmatic Portfolio Optimization is formed by Robust Portfolio Optimization which is distinguished by its use of network theory to delineate possible portfolios. Each system is treated as a potential discrete node in an interacting network that can potentially contribute to the capabilities of the overall architecture. The interactions between systems are modeled by five different aspects: Capability, Requirements, Compatibility, Relay, and Bandwidth and are defined below.

- **Capability** Each system has an upper bound on the capabilities that it can provide to other systems or the architecture.

- **Requirements** Each system has requirements that must be met in order for it to function. These requirements can be met by capabilities of other connected systems.
- **Compatibility** Certain systems can only connect to other systems based on pre-established rules.
- **Relay** A system can transfer a finite amount of capability between systems.
- **Bandwidth** A finite number of connections can be made to other systems.

Each system contributes specific capabilities to the SoS but may also have requirements that must be met by other systems in the network in order to function. Furthermore, there may be unique circumstances where systems have compatibility, bandwidth, or relay constraints that impact the potential connectivity of systems. For instance, a specific module on the space station may be able to produce electrical power if its thermal requirements are met with the help of another system. It can then relay a finite amount of power to other systems provided that bandwidth and compatibility constraints are met. Many other examples can be found in thermal control networks, command and data handling networks, and Environmental Control and Life Support System (ECLSS) onboard a crewed spacecraft.

The overall problem formulation is composed of a selected objective function and many constraints enforced on the optimization. Equation 4.1 represents an example multi-objective function in terms of the overall SoS capabilities that are to be summed across the entire network as well as a summation of the unit cost of selected systems. The individual system capabilities S_{ci} represent a systems performance in each form of capability c. $C_{cost,i}$ represents the unit cost of each system i. The Allocation, A_i^B, is a vector that represents whether each potential system within the architecture is selected and is a decision variable manipulated in the optimization routine. This vector can include multiple instances, or copies, of a specific type of system such that multiple units of a system may be present in an architecture. Constants C_{Cap} and

C_{Cost} are used to normalize the objective function and constants W_1 and W_2 are used to weight the objectives against each other. Various different objective functions are used in this study to maximize or minimize certain properties of a space architecture.

$$Objective = max\left(W_1 * \sum_i \left(\frac{S_{Ci} * A_i^B}{C_{Cap}}\right) - W_2 * \sum_i \left(\frac{C_{cost,i} * A_i^B}{C_{Cost}}\right)\right) \tag{4.1}$$

This optimization routine is subject to the network constraint aspects listed previously. Key to these constraints is the decision variables A_i^B and X_{cij}. The capability transfer X_{cij} represents any potential transfer of capability from any system i to any system j for every potential type of capability c. Some examples include: payload mass launched to orbit, electrical power, and crew consumables(food, oxygen, and water). This decision variable is of dimension $[n,n,c]$ where n is the number of potential systems and c is the number of capabilities. $X_{cij,bin}$ is the binary version of X_{cij} that relates whether a system is connected to another regardless of the amount of capability transfer. Each system has a finite capability that it can provide to other systems in the network and is modeled by Equation 4.2, applied to every column j of X_{cij}:

$$\sum_j X_{cij} \leq A_i^B S_{ci} \tag{4.2}$$

Where the sum of X_{cij} represents the total amount of a specific capability c leaving system i going to all other systems by summing over j. This represents the fact that the summation of all outgoing capability from a system must be less than the maximum possible capability that the system can produce. Similarly, requirements of each system are enforced with the following constraint, applied to every row i of X_{cij}. S_{rj} represents the requirement that system j needs to function and is a vector of user specified constants.

$$\sum_i X_{cij} \geq A_j^B S_{rj} \tag{4.3}$$

Each system has a finite number of connections, or bandwidth, that can be made with other systems. Shown here is the form of constraint used to enforce this where $X_{ij,bin}$ represents all of the systems j that system i connects with and $Limit_i$ represents the finite number of connections that can be made. This constraint can be enforced on either the inflow or outflow of connections with a change of index.

$$\sum_j X_{ij,bin} \leq Limit_i \tag{4.4}$$

To account for the ability to relay capability between systems, the relay property can be enforced upon the architecture through Equation 4.5 and applied for every system and for each capability c. The capability being produced by a system as well as the capability being received is balanced with the amount of capability that is required as well as the capability leaving the system to all other nodes. This constraint is not used in this analysis but is included for the benefit of the reader.

$$A_j^B S_{cj} + \sum_i X_{cij} - A_j^B S_{rj} - \sum_j X_{cij} = 0 \tag{4.5}$$

Compatibility constraints are addressed through a matrix of binary constants K_{ij} in which each cell value represents whether a specific system i can connect to a system j as shown below. Where $K_{ij} = 1$ if system i is compatible with system j and $K_{ij} = 0$ if system i is incompatible with system j. This constraint can be enforced for any potential connection between a system i and a system j. An example use case for this would be preventing low thrust propulsion systems with large transit times being used for crew vehicles like the Orion space capsule.

$$X_{ij,bin} <= K_{ij}, [0, 1] \tag{4.6}$$

A number of other constraints may be imposed on the architecture by the SoS manager. Mandatory systems can be specified via the simple constraint shown in Equation 4.7 with a a value of 1 constraining the system to be selected.

$$A_k^B = l, [0, 1] \tag{4.7}$$

In general, uncertainty is represented in robust optimization in two ways: 1) uncertainty in feasibility (constraint satisfaction) and 2) uncertainty in the objective function [43]. Uncertainty in feasibility takes an operational view of robustness by examining the degree to which system to system dependencies, manifested as constraints, are met. The Bertsimas-Sim formulation addresses this as parametric uncertainty in the constraints of a linear optimization problem without impacting the objective function [45]. This formulation formed the basis of Robust Portfolio Optimization for System of Systems problems by Davendralingam [13]. The Bertsimas-Sim formulation solves the traditional linear programming problem shown in Equation 4.8 (traditionally $AX \leq b$). Here C replaces the traditional A to avoid confusion with previous variables. The matrix C_{ij} contains uncertain elements that exist within symmetric intervals that belong to set J_i and are bounded by $[C_{ij} - \hat{C}_{ij}, C_{ij} + \hat{C}_{ij}]$. This uncertainty interval represents a symmetric and bounded random variable which obeys a symmetric but potentially unknown distribution. These bounds can be estimated through designer intuition, simulation outcomes, parametric analysis, or confidence intervals derived from test data.

A conservatism parameter Γ_i is used to adjust the degree to which feasibility (constraint satisfaction) is enforced within Equation 4.9. However, the introduction of uncertainty into the coefficients of C_{ij} results in a nonlinear problem formulation. The Bertsimas-Sim formulation as highlighted in Equations 4.9-4.12 converts the nonlinear problem into a linear form. Several new variables are necessary including the auxiliary variables p_{ij},y_{ij} and z_{ij}. Further documentation and the full proof can be found in Ref [45]. A weakness of the Bertsimas-Sim formulation is that it does not account for correlation and assumes independent uncertainties between the coefficients and thus departs from the traditional Markowitz formulation of which correlated uncertainties was a central theme.

$$CX \leq b \tag{4.8}$$

$$\sum_j C_{ij}X_j + z_i\Gamma_i + \sum_{i \in J_i} p_{ij} \leq b_i \tag{4.9}$$

$$z_i + p_{ij} \geq \hat{C}_{ij}y_i \tag{4.10}$$

$$-y_i \leq x_j \leq y_i \tag{4.11}$$

$$p_{ij}, y_{ij}, z_{ij} \geq 0 \tag{4.12}$$

The adapted linear formation with uncertainty $\hat{C}_{ij}$ in this study is applied to the capability and requirement constraints, Equations 4.2 and 4.3, to model operational uncertainty in terms of mission success. This formulation incorporates the parameter Γ as a measure of stakeholder conservatism that provides a degree of protection from uncertain linear constraint infeasibility.

The second form of uncertainty representation, that in the objective function, is implemented via a penalty function within the optimization objective function. While there exist more advanced formulations like the Markowitz Formulation [11] and work by Ttnc and Koenig [48] that are highly adept at solving portfolio problems with correlated uncertainty, the systems with appreciable uncertainties in the current work were found to be uncorrelated and thus a simpler linear formulation was used. An example objective of this form is shown in Equation 4.13 where the goal is to maximize C, but also minimize uncertainty $\hat{C}$ relative to stakeholder conservatism Γ. Here C can be a performance parameter, cost factor, or some measure of stakeholder value. The uncertainty $\hat{C}$ represents a standard deviation of that objective parameter C. As with the operational uncertainties, $\hat{C}$ can be estimated with the same methods. As the value of Γ increases, the stakeholder tolerance of uncertainty decreases and lower performing but more certain systems may become optimal selections.

$$Objective = max\left(\sum_i C_iA_i - \Gamma\sum_i \hat{C}_iA_i\right) \tag{4.13}$$

4.2 Scheduling Enhancement to Robust Portfolio Optimization

To incorporate life-cycle phase scheduling into the mathematical model, additional decision variables and constraints were added to the formulation. These include decision variables for each type of system and unit of a system representing the beginning and end of the development phase of system type s ($t_{DB,s}$,$t_{DE,s}$), the beginning and end of the production phase of each system unit i ($t_{PB,i}$,$t_{PE,i}$), and the beginning and end of the operational phase of each system unit i ($t_{OB,i}$,$t_{OE,i}$). Begin and end times were constrained for both development and production phases given their respective phase durations($T_{D,s}$,$T_{P,i}$) for each type of system as shown in Equations 4.14 and 4.15. Phase durations are estimated either by published data or by parametric estimation using NASAs Advanced Missions Cost Model. For systems which have already completed development, a zero development duration was assumed. Further details of the estimation of phase length will be discussed in Subsection 4.5.1. The development phase and production phase are related through Equation 4.16. It was assumed that the first unit could begin production during the development phase as a prototype but could not be completed before the end of the development phase as shown in Equation 4.16. The beginning of production of additional units was constrained by the end of production of the preceding unit as shown in Equation 4.17. The operational time of a system was constrained to be no earlier than the end of production as shown in Equation 4.18.

$$t_{DE,s} \geq t_{DB,s} + T_{D,s} \tag{4.14}$$

$$t_{PE,i} \geq t_{PB,i} + T_{P,i} \tag{4.15}$$

$$t_{PE,i} \geq t_{DE,s} \tag{4.16}$$

$$t_{PB,i+1} \geq t_{PE,i} \tag{4.17}$$

$$t_{OB,i} \geq t_{PE,i} \tag{4.18}$$

In order to associate the time dependent system-to-system operational dependencies to the life-cycle schedule constraints, an application of the Big-M formulation was

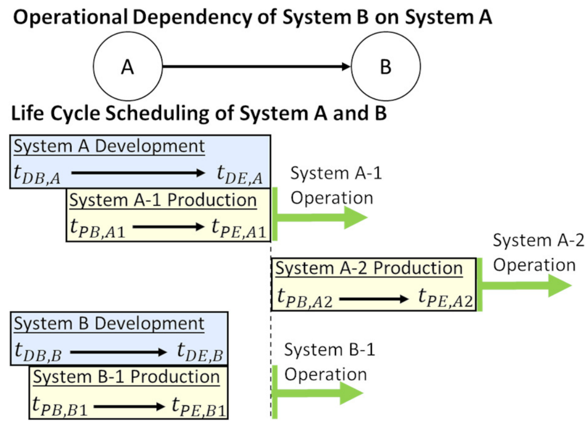


Fig. 4.1.: Scheduling of life-cycle phases and operational dependencies

used to link the time of first operation (t_{OB}) to the operational dependencies $X_{cij,bin}$ as shown in Equation 4.19. Simply put, a system j cannot become operational unless the system(s) i that support it are also operational at that time. The Big-M formulation was required to enforce the constraint between two systems where a dependency exists but avoid affecting any systems that do not share a dependency. The value of M was chosen to be sufficiently large such that the constraint is enforced but does not adversely hinder the speed of optimization convergence. This constraint is enforced for every non-diagonal index of the matrix $X_{cij,bin}$.

$$t_{OB,j} - t_{OB,i} \geq -(1 - X_{cij,bin}) * M \tag{4.19}$$

Figure 4.1 graphically displays a potential solution with the previously mentioned variables and constraints for a simple scenario. The linking of the operational constraints with the scheduling constraints is demonstrated in that system B has an operational dependency on system A and therefore cannot begin operation until System A is also operational, even if System B is manufactured before system A. This example obeys both the scheduling and operational constraints.

A final constraint can be applied on an as needed basis and is used to incorporate technology dependencies within the scheduling. This is applied for systems that have a dependence on a technology with a low Technology Readiness level($TRL < 6$). The constraint actualizes that the technology development must be complete before the system's Design, Development, Test, and Evaluation(DDT&E) moves past the initial design phase. For a system i that has a technological dependency on technology d, system i's development cannot advance past D development until that technology development is complete. Value $D\%$ can be tailored to the field of study. For space systems D is set to a value of 10-20% to represent the end of NASA system life-cycle Phase A "Technology Concept and Technology Development" [59].

$$t_{DE,d} \leq t_{DB,i} + T_{Di} * (D/100) \tag{4.20}$$

4.3 Variable Capability Enhancement to RPO

The second enhancement that separates PPO from RPO is the ability to account for variable sizing of systems, thus allowing the performance of certain systems to be determined within the optimization. In certain portfolio problems with a large number of different candidate systems that require varying levels of a capability, it becomes cumbersome or impossible to model every size of supporting system with a fixed amount of capability. By allowing the capability of that supporting system to be controlled within the optimization, the number of candidate systems and the number of constraints within the problem for each potential supporting system is greatly reduced. This is especially significant for sizing of lander systems where there may be multiple propulsive elements staged together with cascading dependencies. This also improves architecture cohesion, in that systems can be sized for exactly their role. Variable sizing can be applied to any system as long as the relationship between the system capabilities and the resulting requirements is known. For each variable system, the type of the variable capability, the resulting type of requirements and the mathematical relationship between the two are defined a priori. The mathematical relationship between the variable capability and variable requirement is implemented through optimization constraints.

Several new variables are needed for implementation of the variable capability enhancement: two optimization floating point decision variables $Var_C(i)$ and $Var_R(i)$ representing variable capability and variable requirements for each variable system i. In this analysis the relationship between $Var_C(i)$ and $Var_R(i)$ was assumed to be linear, thus Constant values $m_{Req}(i)$ and $b_{Req}(i)$ represent the slope and intercept of the relationship between $Var_C(i)$ and $Var_R(i)$. Additional constants specifying the upper and lower bounds of the capability ($UB_C(i)$,$LB_C(i)$) and requirements ($UB_R(i)$,$LB_R(i)$) are defined. The following equations are applied to only the systems that have been identified as having variable capability. Equation 4.21 defines the linear relationship between a systems variable capability $Var_C(i)$ and its resulting variable requirement $Var_R(i)$ using the slope and intercept values $m_{Req}(i)$ and $b_{Req}(i)$. These values can be determined through the literature review, historical relations, parametric analysis, or software tools like the Beyond LEO Architecture Sizing Tool(BLAST) [60]. A linear relationship between the variable capability to the variable requirement is assumed in this study. If the desired relationship is not linear, a mixture of systems with a piece-wise relationship can be used.

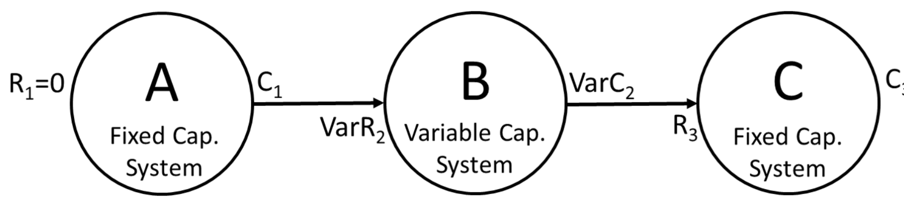


Fig. 4.2.: System to system interactions with variable capabilities and requirements

$$Var_R(i,n) \geq Var_C(i) * m_{Req}(i,n) + A_i^B * b_{Req}(i,n) \tag{4.21}$$

The decision variable representing the variable capability Var_C is allowed to vary within some upper UB_C and lower bounds LB_C if it is selected ($A(i) = 1$), but constrained to have a zero value if the system is not selected ($A(i) = 0$) as shown below in equations 4.24 , 4.23, 4.24 and 4.25

$$A_i^B * UB_C(i) \geq Var_C(i) \tag{4.22}$$

$$LB_C(i) \leq Var_C(i) + LB_C(i) * (1 - A_i^B) \tag{4.23}$$

$$Var_C(i) \leq UB_C(i) \tag{4.24}$$

$$0 \leq Var_C(i) \tag{4.25}$$

Similar to Equation 4.2, the total capability leaving system i is constrained to be less than the total capability Var_C of system i as defined by Equation 4.26.

$$Var_C(i) \geq \sum_j X_{cij} \tag{4.26}$$

A similar set of equations is necessary for the variable requirements $Var_R(i,n)$. Equations 4.27-4.29 constrain $Var_R(i)$ to be less than $UB_R(i)$ and greater than $LB_R(i)$ if system i is selected or zero if it is not. If system i has multiple requirements($n \geq 1$) that are a function of $Var_C(i)$, then Equations 4.27-4.29 can be repeated n times.

$$LB_R(i,n) \leq Var_R(i,n) + m * LB_R(i,n) * (1 - A_i^B) \tag{4.27}$$

$$Var_R(i,n) \leq UB_R(i,n) * A_i^B \tag{4.28}$$

$$Var_R(i,n) \geq 0 \tag{4.29}$$

Equation 4.30 confirms that sufficient capability is being provided to the variable system by other systems to meet the variable requirement and supersedes the fixed requirement constraint in Equation 4.3.

$$\sum_i X_{cij} \geq Var_R(i,n) \tag{4.30}$$

The above equations can be applied to any system that is deemed to have a variable capability. When applying these constraints, the user will need to determine the relationship between the capability and the requirements for the system prior to optimization. This can be accomplished in several ways, including parametric sizing tools, general physics constraints, expert opinion, and general rules of thumb.

4.4 Multi-Domain Objectives

The third enhancement is the addition of multistage decision making to the PPO method using optimization to influence the objective function. As identified in Chapter 2, many decisions must be made at different points in time and the value a stakeholder gains from a certain decision varies with time. An example use case for this enhancement is to select systems that enable lunar missions for a period of time then to evolve to selecting systems that enable a mars mission for a later period of time. This does not necessarily mean that the stakeholder may not have value for one of the missions in the off-period, but that the optimization will prioritize missions that align with the stakeholder value at the time.

Different portions of a mission timeline are split into domains. Each system or technology selected in an early domain can potentially be carried forward to the next domain. Constraints on budget, facilities, and manpower, as well as changing stakeholder objectives may control how many systems may be produced or operated at a time.

Multi-domain optimization is accomplished using a modification of the scheduling constraints from the E1 enhancement to control when desired systems are operational. Using these constraints, multiple domains may be created to tailor stakeholder value. We introduce two constant variables $T_{MDB,d}$ and $T_{MDE,d}$ that define the beginning and end of a stakeholder value of time domain d. Equation 4.31 constrains the operation start time of a set of systems d that have stakeholder value within time domain d. This effectively constrains the start time to that specific domain meaning that if the start time of that system cannot fit in that period of time due to operational or scheduling constraints, it cannot be selected for incorporation into the architecture. The system(s) i can be included into an objective function as shown in Equation 4.32 where those system(s) have value V_i. Multiple time domains can be created to suit changing stakeholder needs as shown in Figure 4.3.

$$T_{MDB,d} \leq t_{OB,d} \leq T_{MDE,d} \tag{4.31}$$

$$Obj = \sum iV_i * A_i \tag{4.32}$$

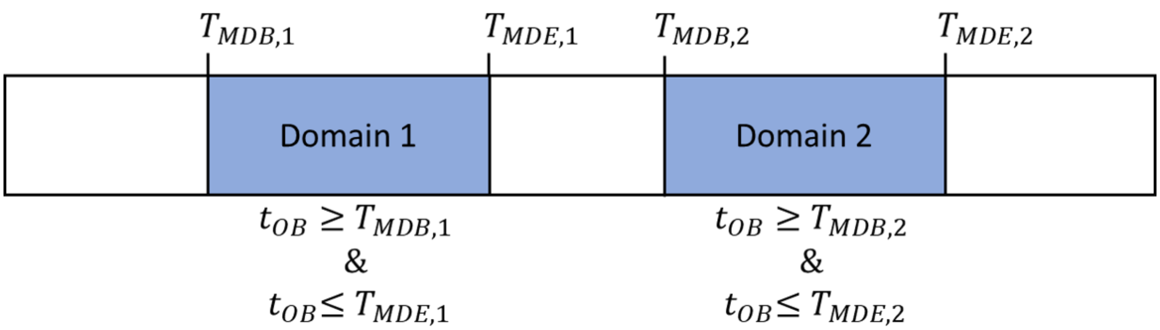


Fig. 4.3.: Graphical representation of the relationship between time domains and governing equations

Multiple time domains can be created using multiple iterations of Equation 4.31. Different formulations were examined to achieve multi-domain optimization. Several techniques create a new binary vector that mirrors the allocation vector A for controlling when systems are operational. However the formulation as described is numerically simpler for the solver which leads to substantially quicker solve times.

4.5 Application of Methodology to Human Space Exploration Architectures

Figure 4.4 presents the process flow for executing the PPO methodology for a space exploration setting, but can be applied to other scenarios as well. The three phase process consists of decomposing the problem(Phase 1), assembling a library of candidate systems(Phase 2), and converting the problem to a Mixed Integer Linear Programming (MILP) optimization and solving the resulting problem(Phase 3).

The first phase of this process is a functional decomposition to identify the dependencies and inner workings of a human space exploration architecture and begins with determining the stakeholder goals and requirements. These high level objectives are decomposed into measurable functions necessary to satisfy those objectives. The problem is further decomposed by defining an interdependent network of functions that satisfy the goal functions. This step also identifies the types of capability transfer that might be present in the architecture. Next, general systems are identified that fulfill the functions. There may be many unique systems that can be represented by a general class that can fulfill a given objective. A list of unique systems is formed that eventually become the Candidate System Library.

The second Phase of this process is the assembly of a Candidate System Library(CSL), from which systems can be selected to form a functional architecture. Unique systems are identified for each of the general classes of systems identified in the first phase. Key to the formation of the CSL is to identify and quantify system requirements and capabilities, system uncertainties and risks, connectivity and incompatibility issues, as well as life-cycle cost and schedule components for development, production, and operation. Each of these characteristics has been estimated through the help of Subject Matter Experts, available literature and historical relations. The specific candidate systems form the candidate system library from which combinations of systems can then be formed into feasible architectures.

The third phase of the process assembles the systems, constraints, and objectives into a Mixed Integer Linear Programming(MILP) problem and solves the resulting problem. This includes the operational constraints, schedule constraints, annual budget constraint(if implemented), and any space architecture specific constraints. An objective function is defined that details the stakeholders goals and any mission specific requirements are defined as constraints. The resulting problem is a Mixed Integer Linear Programming problem, of which there are several available solvers. The constrained problem was converted to a mathematical programming problem via decision variables using a MATLAB package called YALMIP [61] and was solved using the GUROBI commercial solver [62]. The resulting solution can be examined by the user and/or stakeholders and the decision can be made to either A) re-evaluate some of the initial conditions such as system values within the CSL or requirements that were placed on the architecture or B) evaluate the architecture with other tools and methods.

4.5.1 Characterization of Individual Potential System Cost and Schedule Impacts

For each potential system, both the Design, Development, Test, and Evaluation (DDT&E) phase as well as the Production phase were estimated and represent non-recurring and recurring costs. A mixture of literature and parametric modeling was used to estimate the cost components for each system. For systems currently offered by a commercial partner, market pricing was used for the production cost with a zero DDT&E cost. An updated version of NASA's Advanced Missions Cost Model(AMCM) was used to estimate the cost of future systems not currently in use or offered by a commercial partner. AMCM is a parametric technique that estimates system life-cyle cost and schedule components using the following variables to relate to historical systems: number of systems to produce, dry mass (kg), system type, initial operating year, system generation, and technical complexity and program complex-

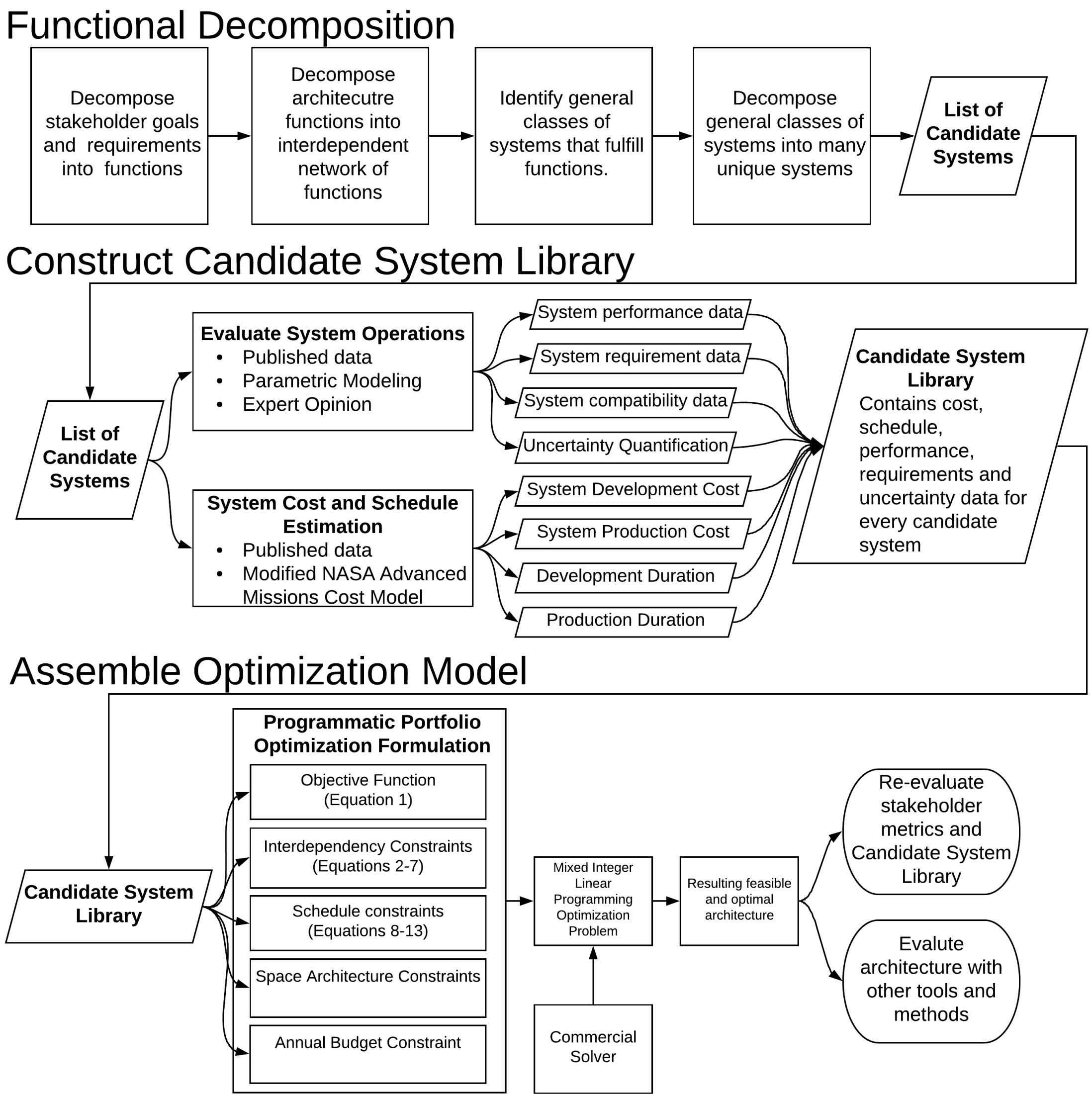


Fig. 4.4.: pPO methodology process

ity. A modified version of AMCM by Rolley et al. modernizes and updates the parameters to represent lunar and Martian systems [63]. One of the key differences in the propulsion systems demonstrated in this study was the Technology Readiness Level(TRL) of those systems and its impact on the cost and schedule components; in that lower TRL systems exhibited relatively longer development times and larger associated costs.

The life-cycle cost of each system ultimately affects the total architecture cost, which can be assessed as either an objective or enforced as a constraint within the optimization. Equation 6.1 represents how the total architecture cost, including life cycle cost components, can be mathematically modeled within RPO. Where A_i^B represents a binary decision variable representing each potential system i and $A_{Dev,s}^B$ is a binary decision variable representing the development of system of type s. $Cost_{NR}$ and $Cost_R$ represent the non-recurring DDT&E cost and recurring production cost of each system respectively. An important note here is that constraints were added such that a system cannot be selected unless its associated DDT&E component is selected.

$$TotalCost = \sum_i A_i^B * Cost_{R,i} + \sum_s A_{Dev,s}^B * Cost_{NR,s} \tag{4.33}$$

5. DEMONSTRATION OF METHODS - FUNDAMENTAL EXAMPLES

5.1 Overview

This section demonstrates the application of the three previously described enhancements to the Robust Portfolio Optimization methodology with simplified cases composed of fictitious example systems. While these are not complex or difficult problems to solve by hand, they serve as a bridge between the methodology and the more complex space exploration architecture trade studies. Each example demonstrates one of the enhancements and has a corresponding unique Candidate System Library.

5.2 Life-Cycle Phase Schedule Modeling example(E1)

This example demonstrates the scheduling enhancement(E1) made to PPO, specifically Equations 4.14-4.19. A fictitious scenario is imagined where a stakeholder wants to maximize a specific capability of an architecture within a specific period of time and has a number of candidate systems in which to do so. The fictitious stakeholder wants to maximize capability C_3 while minimizing cost within some given time period. A library of candidate systems exists with unique systems that have specific capabilities, requirements and life-cycle phase components. While this is a simple example, it contains some complexity to demonstrate the operational and scheduling constraints. The objective function below balances minimizing total cost and maximizing capability C_3. An end time constraint for this architecture was set such that all development, production, and operation is complete by 10 days. The Candidate System Library(CSL) for this example is shown in Table 5.1 and is composed of three types of systems: A, B and C with capabilities C_1, C_2 and C_3. Each system has specific capabilities and requirements as well as unique cost and schedule life-cycle components. Lifecycle cost and schedule components are listed in days and a generic cost unit.

Table 5.1.: Candidate System Library for E1 example with system capabilities, requirements, schedule components, and cost components.

System Type	Capabilities	Requirements	Dev. Time (T_D)	Prod. Time (T_P)	Min Oper. Time ($T_{O,min}$)	Max Oper. Time ($T_{O,max}$)	Dev. Cost ($Cost_{NR}$)	Prod. Cost ($Cost_R$)	Oper. Cost ($Cost_{Oper}$)	Num. of Potential Units
A	2 C_1	-	2	3	1	2	2	1	2.00	8
B	1 C_2	1 C_1	2	2	1	2	1	2	1.00	8
C	1 C_3	2 C_2	3	3	1	2	3	1	3.00	8

$$Obj_{Cost} = \sum_i A_i^B * Cost_{R,i} + \sum_s A_{Dev,s}^B * Cost_{NR,s} + \sum_s (t_{OE,s} - t_{OB,s}) * Cost_{O,s} \quad (5.1)$$

$$Obj_{Cap,C3} = \sum_i C_{3,i} A_i^B \quad (5.2)$$

$$Objective = Obj_{Cap,C3} + Obj_{Cost} \quad (5.3)$$

An optimization problem composed of the constraints in Equations 4.2 - 4.19 and the objective function in Equation 5.3 with decision variables A_i^B, $A_{Dev,s}^B$, X_{cij}, $t_{DB,s}$, $t_{DE,s}$ $t_{PB,i}$, $t_{PE,i}$,$t_{OB,i}$, and $t_{OE,i}$ is formulated as previously described. Parameters within the constraints are populated from the Candidate System Library shown in Table 5.1. The Candidate System Library has more specific units of each potential system than feasible within the time period. In this example System C provides the capability C_3 of stakeholder value and must be supported by other systems. The optimization determines the selection of systems, timing of life-cycle components and transfer of capability between systems. The optimization problem is formulated with the YALMIP [61] package within MATLAB and solved with the GUROBI solver [62]

The overall results of the optimization are shown in Table 5.2 with a max architecture value of 3 C_3 capability and total architecture cost of 45 units. The value of the decision variables including the Allocation vector as well as start/end times of the each life cycle phase are shown in Table 5.3. The graphical representation of the adjacency matrix X_{cij} shown in Figure 5.1 represents the system to system interactions in terms of the connectivity, value, and type of capability transfer. The interactions obey the operational constraints in Equations 4.2 - 4.7. As can be seen in Figure 5.1, for each system of type C, 2 units of system type B provide 1 unit of C_2 capability each. Systems of type B are supported by a system of of type A which provides 1 unit of C_1 capability to each B system. Systems of type C in this example provide the C_3 capability valued by the stakeholder. Overall, 3 groupings of these systems is feasible within the time period.

The results of the life-cycle phase scheduling can be visually seen in Figure 5.2. The life-cycle phase scheduling obeys the scheduling constraints in Equations 4.14-4.19. Due to the lifecycle scheduling, only 3 units of system C can be completed in the constrained 10 day period.

Table 5.2.: High level Results of optimization including total architecture cost, optimal number of systems selected, and relevant details of the optimization mechanics

Result	Value
Total Architecture Cost	45
Number of Type A	3
Number of Type B	6
Number of Type C	3
Operation start time of first C system	4.5
Operation end time of last c system	9.5
Optimization Setup Time	1.4381 s
Optimization Run Time	0.7549 s
Best Objective	-8.27E-01
Best Bound	-8.27E-01

Table 5.3.: Optimal decision variable values of solution resulting from optimization of the E1 problem. Values include system allocation as well as the begin and end time of each lifecycle phase(development, production, and operation).

System	A	t_{DB}	t_{DE}	t_{PB}	t_{PE}	t_{OB}	t_{OE}
Sys A - 1	1	0	2	1.5	4.5	4.5	5.5
Sys A - 2	1	0	2	3.5	6.5	6.5	7.5
Sys A - 3	1	0	2	5	8	8.5	9.5
Sys A - 4	0	0	2	8	8	8.5	8.5
Sys A - 5	0	0	2	10	10	10	10
Sys A - 6	0	0	2	10	10	10	10
Sys A - 7	0	0	2	10	10	10	10
Sys A - 8	0	0	2	10	10	10	10
Sys B - 1	1	0	2	1.5	3.5	4.5	5.5
Sys B - 2	1	0	2	2.5	4.5	4.5	5.5
Sys B - 3	1	0	2	3.5	5.5	6.5	7.5
Sys B - 4	1	0	2	4.5	6.5	6.5	7.5
Sys B - 5	1	0	2	5.5	7.5	8.5	9.5
Sys B - 6	1	0	2	6.5	8.5	8.5	9.5
Sys B - 7	0	0	2	8.5	8.5	8.5	8.5
Sys B - 8	0	0	2	10	10	10	10
Sys C - 1	1	0	3	1.5	4.5	4.5	5.5
Sys C - 2	1	0	3	3	6	6.5	7.5
Sys C - 3	1	0	3	5.5	8.5	8.5	9.5
Sys C - 4	0	0	3	10	10	10	10
Sys C - 5	0	0	3	10	10	10	10
Sys C - 6	0	0	3	10	10	10	10
Sys C - 7	0	0	3	10	10	10	10
Sys C - 8	0	0	3	10	10	10	10

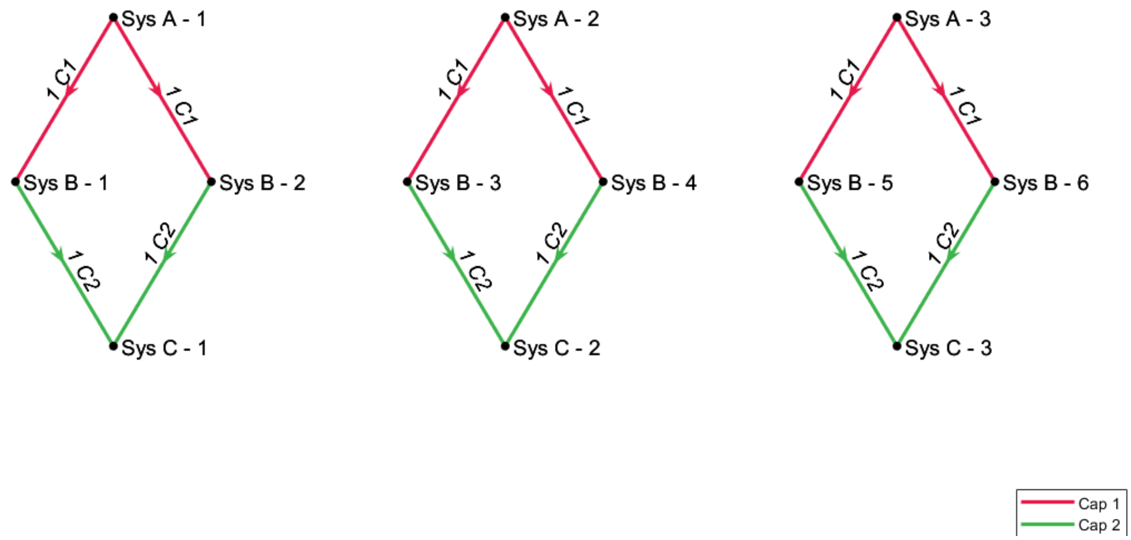


Fig. 5.1.: System to System interactions defined by the variable x_{cij} from the solution of the optimization problem with capability transfer type and value labeled in the graph. For example, system C-1 is dependent systems B-1 and B-2 and receives 2 C_2 capability from those two systems. System B-2 is dependent on system A-1 and receives 1 C_1 capability from that system

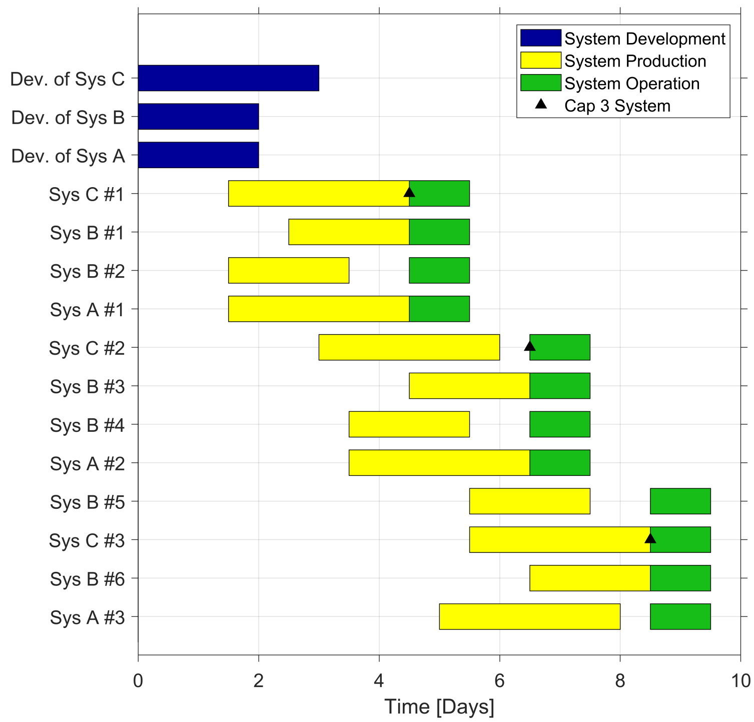


Fig. 5.2.: Gantt chart of system life-cycle phases depicting system scheduling dependencies resulting from optimization solution.

5.3 Variable Capability Example (E2)

This example demonstrates the variable capability enhancement to PPO, specifically, the addition of Equations 4.21-4.30 as constraints on the optimization. In a similar fashion to the previous example, this demonstration is based on a simplistic scenario consisting of systems A,B,and C with the addition of a new system D. Systems of type D have a variable capability of type C_2 whose value is determined by the optimization. System D is proposed as an optimally sized replacement for the use of 2 units of system B. The development and production schedule components of system D are given fixed values but the cost components are allowed to scale with its variable capability. The relationship between capability C_2 and the requirements is of linear form with slope of 1 and intercept of zero and is enabled within the optimization through Equation 4.21.

Hand calculations will prove that the system type D is the better choice over system B, but this simple example demonstrates that the PPO method can both select and size an optimal architecture. An optimization problem composed of the operational constraints, scheduling constraints, and the variable capability constraints as previously described is formulated and solved. The Candidate System Library is shown in Figure 5.4 including the new system D. The Candidate System Library has more specific units of each potential system then necessary.

Table 5.4.: Candidate System Library for E2 example with system capabilities, requirements, schedule components, cost components, as well as variable sizing parameters. Parameters governing the variable sizing include the upper and lower bounds of the variable capability and variable requirement as well as the relationship between the two variables.

System Type	Cap.	Req.	Dev. Time (T_D)	Prod. Time (T_P)	Min Oper. Time ($T_{O,min}$)	Max Oper. Time ($T_{O,max}$)	Dev. Cost	Prod. Cost	Oper. Cost	Cap U Bnd	Cap L Bnd	Req U Bnd	Req L Bnd	Req. Slope	Req. Int	Cost Slope	Cost Int
A	2 C_1	-	2	3	1	2	2	1	2	-	-	-	-	-	-	-	-
B	1 C_2	1 C_1	2	2	1	2	1	2	1	-	-	-	-	-	-	-	-
C	1 C_3	2 C_2	3	3	1	2	3	1	3	-	-	-	-	-	-	-	-
D	Var C_2	Var C_1	2	3	1	2	Var	Var	1	2	0	3	1	1	0	2 C_2	1

$$Obj_{Cost} = Obj_{FixedCost} + Obj_{VarCost} \tag{5.4}$$

$$Obj_{FixedCost} = \sum_i A_i^B * Cost_{R,i} + \sum_s A_{Dev,s}^B * Cost_{NR,s} + \sum_s (t_{OE,s} - t_{OB,s}) * Cost_{Oper,s}) \tag{5.5}$$

$$Obj_{VarCost} = \sum_i Var_{C,FS(s)} * Cost_{VNRm,s} + \sum_i Var_{C,i}B * Cost_{VRm,i} \tag{5.6}$$

$$Obj_{Cap,C3} = \sum_i C_i A_i \tag{5.7}$$

$$Objective = Obj_{Cap,C3} + Obj_{Cost} \tag{5.8}$$

The overall results of the optimization are shown in Table 5.5 and the value of the decision variables including the Allocation as well as start/end times of the each life cycle phase are shown in Table 5.6. As can be seen, the result of the optimization includes the selection and sizing of system D without system B being selected. In this example the variable enhancement has been demonstrated. The graphical representation of the adjacency matrix X_{cij} shown in Figure 5.4 represents the system to system interactions in terms of the connectivity, value, and type of capability transfer. The interactions obey the operational constraints. The results of the life-cycle phase scheduling can be visually seen in Figure 5.3. The life-cycle phase scheduling obeys the scheduling constraints of the optimization. The results of the system sizing are, as predicted, a capability of 2 C_2 and a requirement of 2 C_1. The resulting capability is sized such that the capability of system D is 2 C_2 with a corresponding requirement of 2 C_1.

Table 5.5.: High level Results of E2 results from the optimization including total architecture cost, optimal number of systems selected, total C_4 capability of architecture, and relevant details of the optimization mechanics

Result	Value
Total ArchitectureCost	52
Number of Type A	4
Number of Type B	0
Number of Type C	4
Number of Type D	4
Capability 3	4
Operation start time of first C System	4.5
Operation last time of C System	9
Optimization Setup Time(s)	1.1300964
Optimzation Run Time(s)	0.3719036
Best Objective	-3.89E+00
Best Bound	-3.89E+00

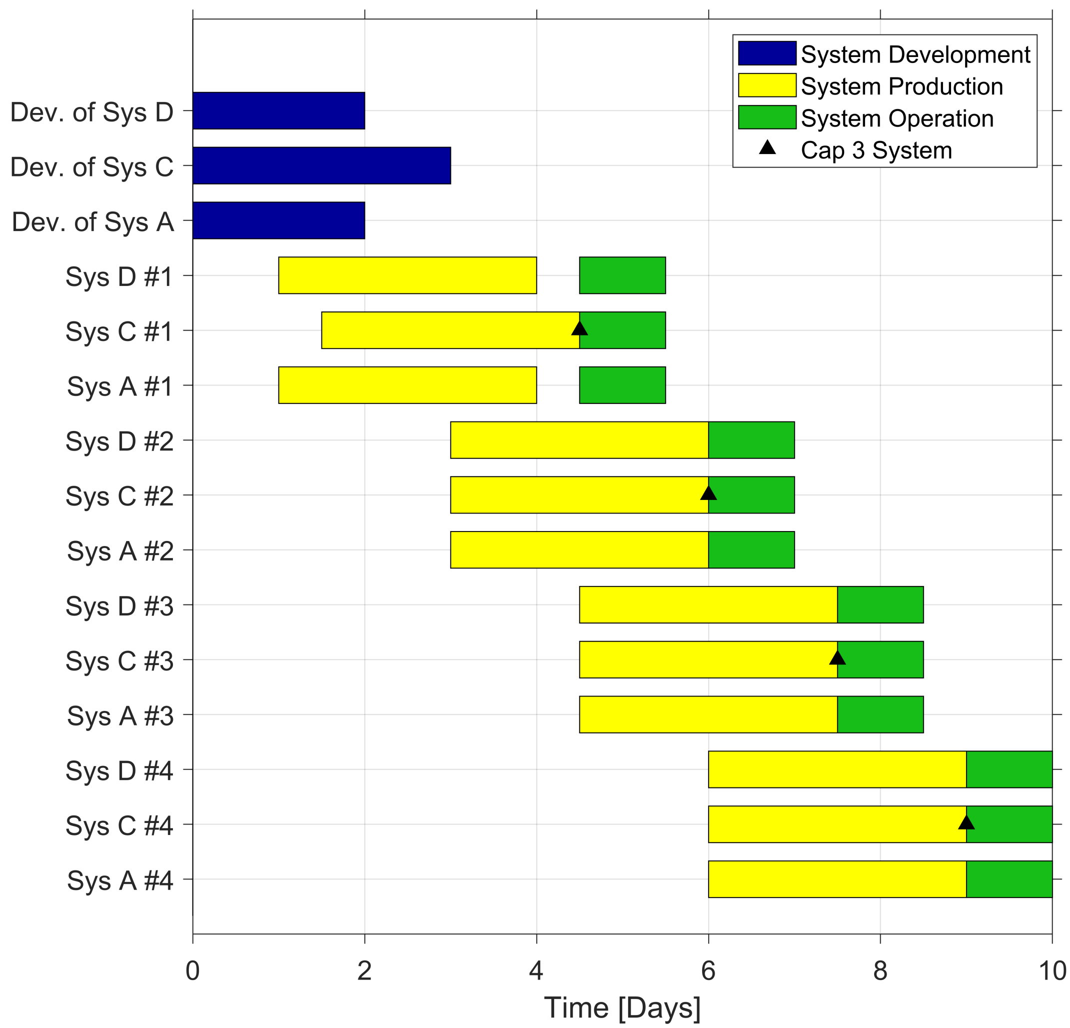


Fig. 5.3.: Gantt chart of system life-cycle phases depicting system scheduling resulting from optimization solution of the E2 example

Table 5.6.: Optimal decision variable values of solution resulting from optimization of the E2 problem. Values include system allocation as well as the begin and end time of each lifecycle phase(development, production, and operation).

System	A	t_{DB}	t_{DE}	t_{PB}	t_{PE}	t_{OB}	t_{OE}
Sys A-1	1	0	2	1	4	4	6
Sys A-2	1	0	2	2.5	5.5	5.5	7.5
Sys A-3	1	0	2	4	7	7	8.5
Sys A-4	1	0	2	5.5	8.5	8.5	10
Sys A-5	0	0	2	10	10	10	10
Sys A-6	0	0	2	10	10	10	10
Sys A-7	0	0	2	10	10	10	10
Sys A-8	0	0	2	10	10	10	10
Sys B-1	0	0	0	0	0	0	0
Sys B-2	0	0	0	0	0	0	0
Sys B-3	0	0	0	0	0	0	0
Sys B-4	0	0	0	7	7	0	0
Sys B-5	0	0	0	10	10	0	0
Sys B-6	0	0	0	10	10	0	0
Sys B-7	0	0	0	10	10	0	0
Sys B-8	0	0	0	10	10	10	10
Sys C-1	1	0	3	1.5	4.5	4.5	5.5
Sys C-2	1	0	3	3	6	6	7
Sys C-3	1	0	3	4.5	7.5	7.5	8.5
Sys C-4	1	0	3	6	9	9	10
Sys C-5	0	0	3	10	10	9	9
Sys C-6	0	0	3	10	10	9	9
Sys C-7	0	0	3	10	10	9	9
Sys C-8	0	0	3	10	10	9	9
Sys D-1	1	0	2	1	4	4	5.5
Sys D-2	1	0	2	2.5	5.5	5.5	7
Sys D-3	1	0	2	4	7	7	8.5
Sys D-4	1	0	2	5.5	8.5	9	10
Sys D-5	0	0	2	10	10	10	10
Sys D-6	0	0	2	10	10	10	10
Sys D-7	0	0	2	10	10	10	10
Sys D-8	0	0	2	10	10	10	10

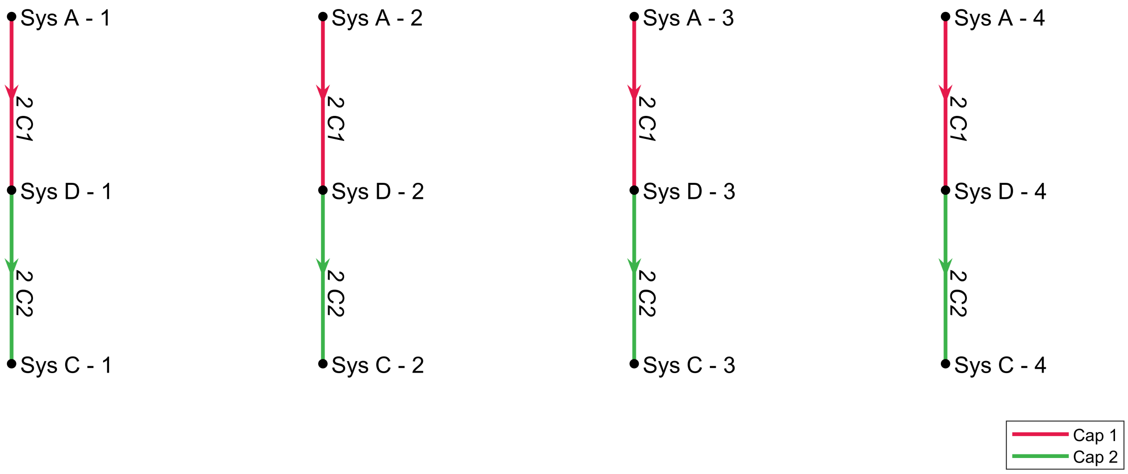


Fig. 5.4.: System to System interactions defined by the variable x_{cij} from the solution resulting from optimization with capability transfer type and value labeled in the graph. Systems of type D is optimally sized to provide 2 C_2 capability to systems of type C and requires 2 C_1 provided by systems of type A.

5.4 Multistage Decision making example(E3)

The third simple example demonstrates the multi-domain constraints (enhancement 3). This example follows a scenario such that during the first half of a timeline the stakeholders value one specific capability and then after a midway point switch to valuing only another capability. A new set of candidate systems is introduced and detailed in Table 5.7. Given the Candidate System Library, the following system combinations are possible and shown in Figure 5.5.

Table 5.7.: Candidate System Library for E3 example with system capabilities, requirements, schedule components, cost components, as well as variable sizing parameters. Parameters governing the variable sizing include the upper and lower bounds of the variable capability and variable requirement as well as the relationship between the two variables.

System Type	Capabilities	Requirements	Dev. Time (T_D)	Prod. Time (T_P)	Min Oper. Time ($T_{O,min}$)	Max Oper. Time ($T_{O,max}$)	Dev. Cost	Prod. Cost	Oper. Cost
Sys F	1 C_1	-	2	2	1	10	4	1	1
Sys G	1 C_2	1 C_1	2	1	1	10	4	1	1
Sys H	1 C_4	2 C_1	3	2	0.5	10	4	1	1
Sys I	1 C_2 & 1 C_3	1 C_1	4	2	1	10	5	1	1
Sys J	1 C_3	-	2	2	1	10	4	1	1
Sys K	1 C_5	1 C_3	3	2	1	10	4	1	1
Sys L	1 C_6	1 C_5	3	2	0.5	10	4	1	1

In this scenario, stakeholders value maximizing the C_4 capability in the first 6 days(0-6) and maximizing the C_6 capability in the last 4 days(6-10) and don't value those capabilities otherwise. Additionally, the fictitious stakeholders also wishes to minimize total cost and will not tolerate excess systems that do not contribute value. A suitable objective function is shown below. Weighting between total cost and maximizing capability is set to 50-50.

$$Obj = Cost + \sum_i X_i * S_{i,C4} + \sum_i X_i * S_{i,C6} \tag{5.9}$$

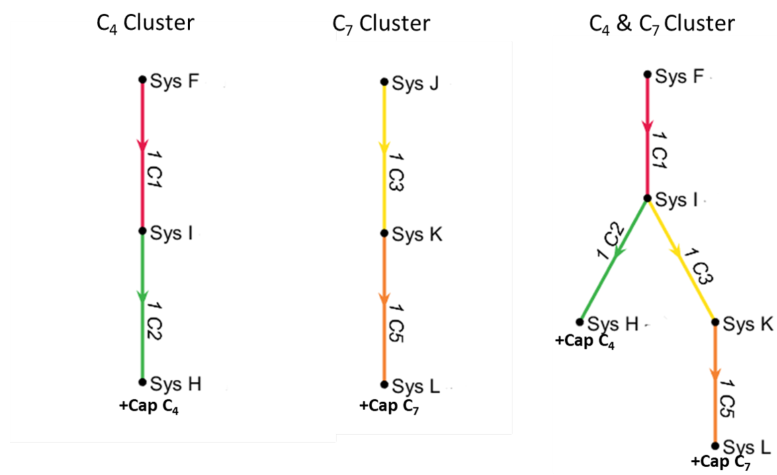


Fig. 5.5.: Example potential system to system interactions that can be formed within this scenario.

When applying the optimization without the multi domain constraints the resulting architecture maximizes capability regardless of the time domains as shown in the scheduling chart in Figure 5.6. Thus, selecting systems and having them operational when the stakeholder gains no value from them results in less capability and wasted resources.

However, when the multi domain constraints are enacted, the systems become operational only when the stakeholder gains value from them as shown in Figure 5.8. The resulting system selection has also changed in that different systems are selected to maximize the capability being produced in the different domains. The resulting system allocation and values of decision variables is detailed in Table 5.9.The system to system interactions are shown in Figure 5.9.

The architecture without the multi-domain constraints expends resources(cost units) on producing and operating systems outside of their respective domains. The resulting total cost of 84.5 units is higher than the total cost of the architecture with the multi-domain constraints at 65 units. The stakeholder true value of the architecture can be calculated by summing the C_4 and C_6 of active systems within their respective time domains. For C_4 systems this is a system that is active between 0 and

6 days and for C_6 systems this is a system that is active between 6 and 10 days. The stakeholder true value of the architecture without domain constraints is 7 and the stakeholder true value of the architecture with domain constraints is 8. The architecture with domain constraints results in higher stakeholder true value and a lower total cost.

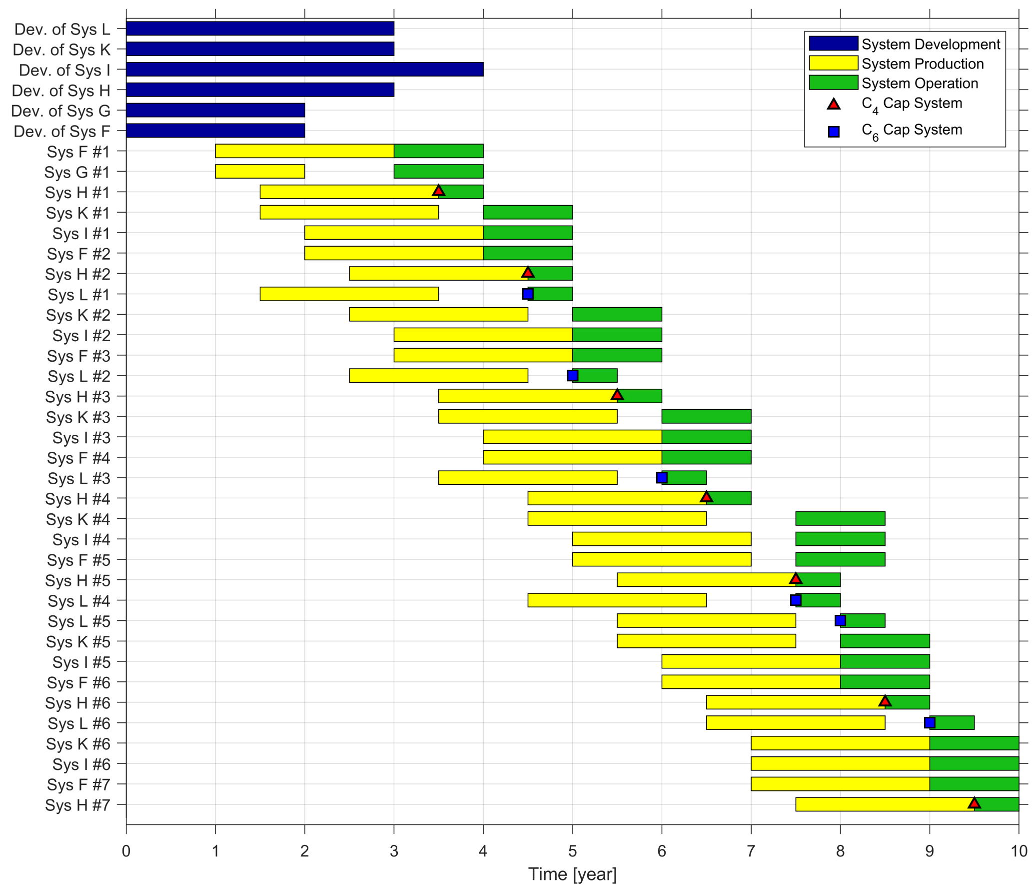


Fig. 5.6.: Gantt chart of system life-cycle phases depicting system scheduling resulting from optimization solution

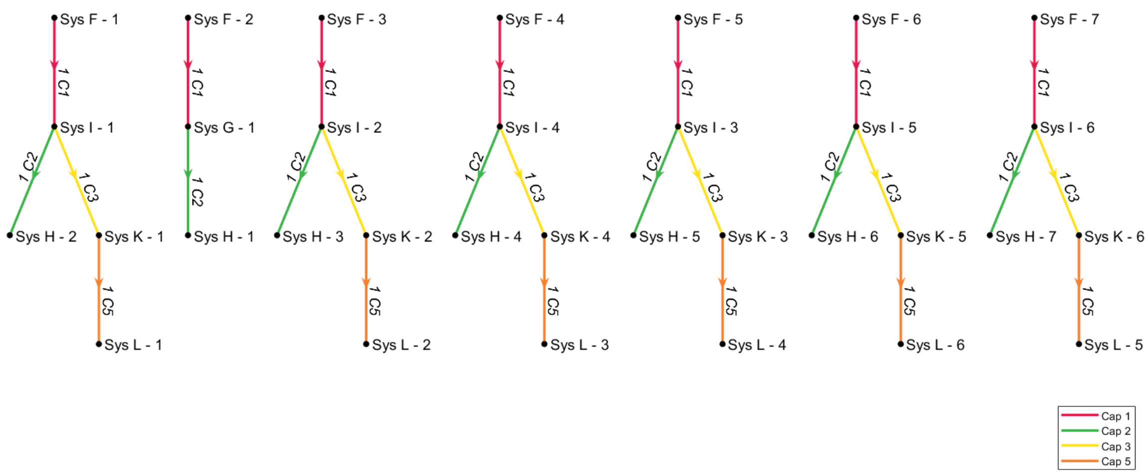


Fig. 5.7.: System to System interactions defined by the variable x_{cij} from the solution resulting from optimization with capability transfer type and value labeled in the graph

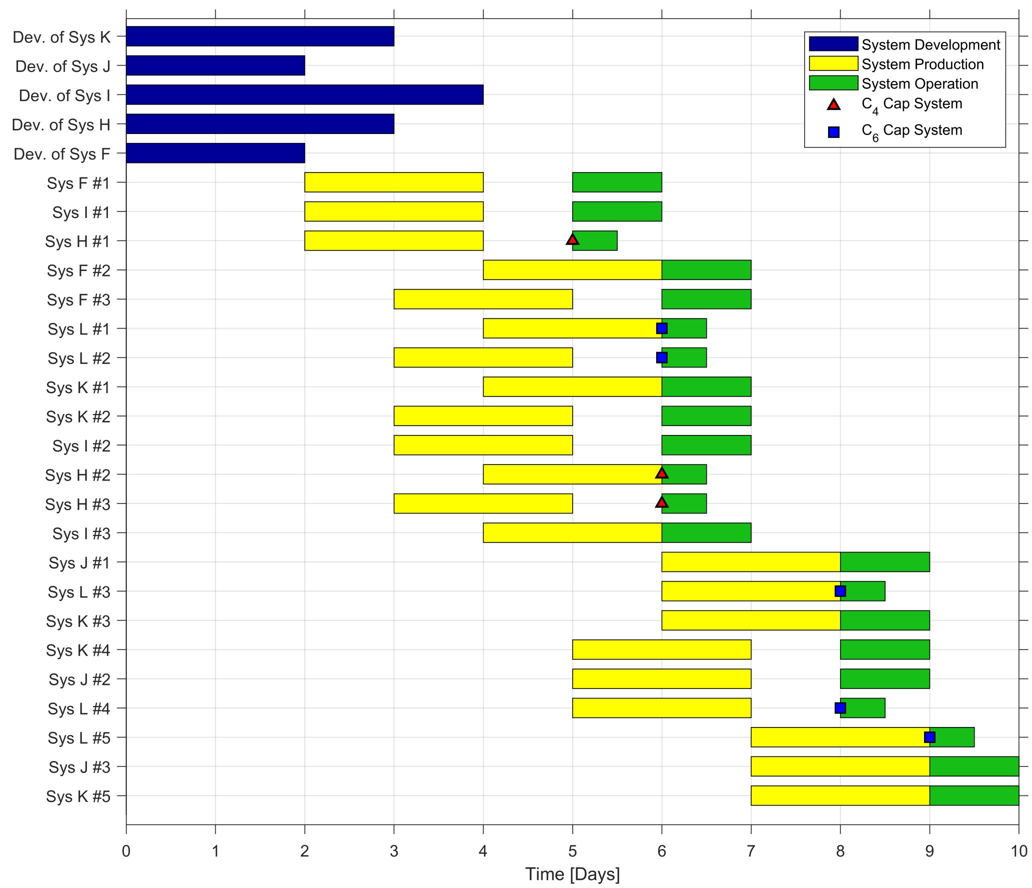


Fig. 5.8.: Gantt chart of system life-cycle phases depicting system scheduling resulting from optimization solution of the E3 example

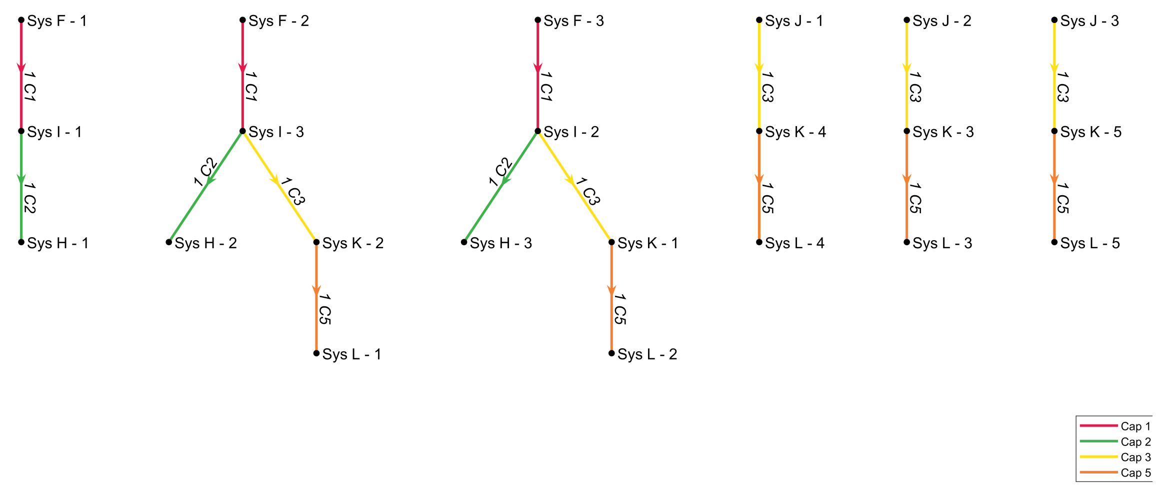


Fig. 5.9.: System to System interactions defined by the variable x_{cij} from the solution resulting from optimization with capability transfer type and value labeled in the graph. These results come from the Multi Domain Example E3

Table 5.8.: Optimal decision variable values of solution resulting from optimization of the E3 problem. Values include system allocation as well as the begin and end time of each lifecycle phase(development, production, and operation).

System	A	t_{DB}	t_{DE}	t_{PB}	t_{PE}	t_{OB}	t_{OE}
Sys F - 1	1	0	2	2	4	5	6
Sys F - 2	1	0	2	3	5	6	7
Sys F - 3	1	0	2	4	6	6	7
Sys F - 4	0	0	2	6	6	6	6
Sys F - 5	0	0	2	6	6	6	6
Sys G - 1	0	0	0	0	0	0	0
Sys G - 2	0	0	0	0	0	0	0
Sys G - 3	0	0	0	0	0	0	0
Sys G - 4	0	0	0	0	0	0	0
Sys G - 5	0	0	0	0	0	0	0
Sys H - 1	1	0	3	2	4	5	5.5
Sys H - 2	1	0	3	3	5	6	6.5
Sys H - 3	1	0	3	4	6	6	6.5
Sys H - 4	0	0	3	8	8	6	6
Sys H - 5	0	0	3	10	10	6	6
Sys I - 1	1	0	4	2	4	5	6
Sys I - 2	1	0	4	3	5	6	7
Sys I - 3	1	0	4	4	6	6	7
Sys I - 4	0	0	4	6	6	6	6
Sys I - 5	0	0	4	6	6	6	6
Sys J - 1	1	0	2	5	7	8	9
Sys J - 2	1	0	2	6	8	8	9
Sys J - 3	1	0	2	7	9	9	10
Sys J - 4	0	0	2	9	9	9	9
Sys J - 5	0	0	2	9	9	9	9
Sys K - 1	1	0	3	3	5	6	7
Sys K - 2	1	0	3	4	6	6	7
Sys K - 3	1	0	3	5	7	8	9
Sys K - 4	1	0	3	6	8	8	9
Sys K - 5	1	0	3	7	9	9	10
Sys L - 1	1	0	3	3	5	6	6.5
Sys L - 2	1	0	3	4	6	6	6.5
Sys L - 3	1	0	3	5	7	8	8.5
Sys L - 4	1	0	3	6	8	8	8.5
Sys L - 5	1	0	3	7	9	9	9.5

Table 5.9.: High level Results of optimization resulting from with and without the multi-domain constraint formulation including total architecture cost, optimal number of systems selected, and relevant details of the optimization mechanics. True stakeholder value is is calculated by summing the capability obtained within their prospective time domains.

Result	Mixed Objective w/o Multi Domain Constraints	Mixed Objective with Multi Domain Constraints
Total ArchitectureCost	84.50	65.00
Number of Type F	7	3
Number of Type G	1	0
Number of Type H	7	3
Number of Type I	6	3
Number of Type J	0	3
Number of Type K	6	5
Number of Type L	6	5
Architecture Total C_4 Cap	7	3
Architecture Total C_6 Cap	6	5
Optimization Setup Time(s)	1.5077	1.6504
Optimzation Run Time(s)	1.0963	2.5146
True Stakeholder Value	7	8

6. DEMONSTRATION OF METHODS - LUNAR ORBIT EXAMPLE SCENARIO

The method as previously described is applied to a cislunar human space exploration mission consisting of a Deep Space Habitat as well as robotic lunar landers deployed to the lunar surface with particular focus on demonstrating the RPO method with the PPO scheduling enhancement(E1). This specific mission is examined at a high level in terms of cost, schedule, performance, and robustness. The Candidate System Library for this study, shown in Tables A.1, A.2, and A.3 is non-exhaustive but includes several options for many of the systems required for a cislunar mission. Since the data used for many systems is representative, but not authoritative, the results are examined from the the perspective of evaluating efficacy of the method. However, based on the preliminary results, some interesting findings are available and presented here.

Figure 4.4 presents the process flow for executing the enhanced RPO methodology for a space exploration setting. In particular, this scenario examines a scenario consisting of a Deep Space Habitat as well as robotic landers deployed to the lunar surface. The three phase process consists of decomposing the problem(Phase 1), assembling a library of candidate systems(Phase 2), and converting the problem to a Mixed Integer Linear Programming (MILP) optimization and solving the resulting problem(Phase 3).

The first phase of this process is a functional decomposition to identify the dependencies and the inner workings of a human space exploration architecture. The first step of this phase begins with determining the overall stakeholder goals and requirements. These high level objectives are decomposed into measurable goal functions necessary to satisfy those objectives. The problem is further decomposed by defining an interdependent network of functions that satisfy the goal functions. This step also identifies the types of capability transfer that might be present in the architecture. The list and description of the types of capability present in this scenario is detailed in Table 6.1. Next, general systems are identified that fulfill the functions. There may be many unique systems that can be represented by a general class that fulfills a given objective. A list of unique systems is formed that eventually become the Candidate System Library.

Table 6.1.: Lunar orbit architecture capability descriptions

Capability	Description
Exploration	A top level capability equal to the crew duration spent in space (days)
Crew Earth Surface to Lunar Orbit	Transfer and habitability of crew between Earth surface and Lunar Orbit (n)
Habitable Volume at Lunar Orbit	Habitable Crew Volume in Lunar Orbit (m^3)
Electric Power	The amount of electrical power supplied or required (kWe)
Deliver Mass to LEO	Ability to deliver systems to Low Earth Orbit (kg)
Deliver Mass to LO	Ability to deliver systems to Lunar Orbit (kg)
Return Crew to Earth	Ability to return crew members to Earth (n)
Consumables	Breathable air, water, food, and supplies (kg)
Science Airlock Access	Ability to transfer lunar samples to within Deep Space Habitat
Lander Ability	Ability to land and collect samples on Lunar Surface (kg)
Mission Control	Ability to command and control space systems
Return Samples to Earth	Ability of a vehicle to return samples back to Earth from Lunar orbit (kg)

The network of interacting capabilities provides a useful visual for understanding how systems transition between state locations such as Earths surface, Earth orbit, lunar orbit, and the lunar surface. The network diagram in Figure 6.1 illustrates how some of the transitive capabilities interact. A key assumption in this analysis was that rendezvous in Low Earth Orbit is possible for any system. For example, an in-space propulsion system could be launched on a different launch vehicle and rendezvous with another space system that needs to be sent to lunar orbit. This assumption was found to heavily favor the use of more economical but less capable commercial rockets as opposed to the more expensive but also more capable government super heavy lift rockets. While this assumption was upheld for this analysis, it is possible to enforce that an in-space propulsion system must launch on the same rocket as its associated cargo.

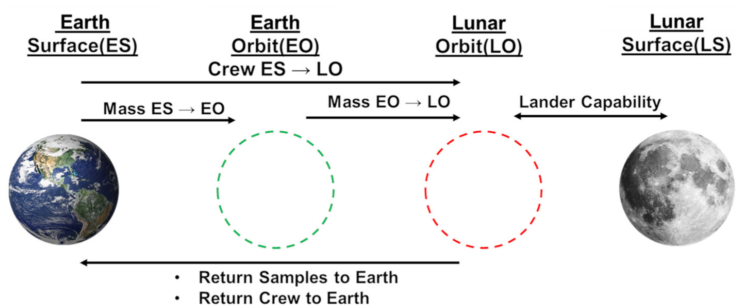


Fig. 6.1.: State/Location transition network

The second Phase of this process is the assembly of a Candidate System Library(CSL), from which systems can be selected to form a functional architecture. Unique systems are identified for each of the general classes of systems identified in the first phase. Key to the formation of the CSL is the identification of systems and the quantification of system capabilities and requirements, system uncertainties and risks, connectivity and incompatibility issues, as well as life-cycle cost and life-cycle schedule components for development, production, and operation. Each of these characteristics has been estimated through the help of Subject Matter Experts, available literature, and historical relations. The specific candidate systems form the Candidate System Library from which combinations of systems can be formed into feasible architectures. The CSL for this study is represented in Table A.2 and A.3. While this CSL is non-exhaustive, it is sufficient to demonstrate the suggested method.

The third phase of the process assembles the systems, constraints, and objectives into a Mixed Integer Linear Programming(MILP) problem and solves the resulting problem. This includes the operational constraints, schedule constraints, annual budget constraint(if implemented), and any space architecture specific constraints. An objective function is defined that details the stakeholders goals and any mission specific requirements are defined as constraints. The resulting problem is a Mixed Integer Linear Programming problem, of which there are several available solvers. The constrained problem was converted to a mathematical programming problem via decision variables using a MATLAB package called YALMIP [61] and was solved using the GUROBI commercial solver [62]. The resulting solution can be examined by the user and/or stakeholders, and the decision can be made to A) re-evaluate some of the initial conditions such as system values within the CSL or requirements that were placed on the architecture or B) to further evaluate the resulting architecture with other tools and methods.

One of the initial goals of our work is to demonstrate a method of selecting systems based on trade-offs between cost, schedule, risk, and performance. Both the launch vehicle class and in-space propulsion class of systems were used as the barometer in this preliminary effort, and as such, a greater variety of these systems are present in the Candidate System Library. Furthermore, the chemical in-space propulsion system was assessed as less expensive and more certain in terms of performance and cost. The Nuclear Thermal Rocket(NTR) in-space propulsion system was more expensive and more uncertain in terms of performance and cost. In this study, for the same capability, the NTR system was estimated to have a lower launch mass than the comparably sized chemical propulsion system. Additionally, several choices for launch vehicles ranging in cost and performance were available and include both commercial and government systems.

A number of domain specific constraints were enforced on the architecture to more accurately model a space exploration architecture. For instance, a payload that needs to be placed into orbit cannot be launched by a combination of two or more rockets and must be launched by a single rocket or instead split up into smaller systems. An assumption was made that rendezvous in earth orbit with an in-space propulsion system that can send payloads to the moon was feasible. This allowed the utilization of medium to heavy lift rockets instead of super heavy lift rockets in some scenarios. However, this assumption can be relaxed and the requirement for launching the in-space propulsion system and cargo on the same rocket can be enforced within the model. The technical challenge of rendezvous in Earth orbit and the necessary R&D is represented in the development and production cost and schedule components of such systems. While a chemical or nuclear thermal rocket can send a crew capsule quickly to a lunar orbit, a pure solar electric propulsion system was deemed to be too slow for crewed missions without some small impulse stage to reduce transit time. More propulsion system choices, including hybrid electric systems, as well as additional propulsion sizes will be the focus of further study. Additionally, several constraints relating crew operations and human factors were enforced within the model. Habitable volume constraints relative to crew size and crew duration were chosen based on the findings of the NASA 2011 Habitable Volume Workshop [64]. Similarly, consumable requirements for crew were based on open loop Environmental Control and Life Support Systems with estimates from several studies [65, 66].

There were a number of systems that were deemed necessary to any deep space habitat and include: a crew airlock, a science airlock, a robotic arm, and a propellant storage system, and several robotic lander missions. These systems were often associated with stakeholder objectives and were deemed required for the architecture and constrained using Equation 4.7. Each required system has its own cost, schedule, and physical requirements that must be met by the architecture.

Additionally, other constraints defined by the physics of space exploration are modeled. One such constraint is the rocket equation, in that there is a propellant requirement for any translational maneuver used to change the spacecraft's orbit that depends on the vehicles performance, propellant, mass, and the associated payload masses. There are at least two methods of accounting for the rocket equation. The simplest approach is to assign capabilities to systems that represent the payload translational capability, which bypasses the addition of the rocket equation as a nonlinear constraint. This would require calculation prior to optimization to determine the resulting requirements, but would reduce the computational complexity of the problem being solved. The more complex, but potentially more accurate, method would be to include the rocket equation as a constraint combined with specific rocket engine performance and the treatment of propellant as a commodity. The results demonstrated here apply the first method of accounting for translational capability before optimization but may utilize the second method in future work.

For each potential system, both the Design Development Test and Evaluation (DDT&E) phase as well as the Production phase were estimated and represent non-recurring and recurring costs. A mixture of literature and parametric modeling was used to estimate the cost components for each system. For systems currently offered by a commercial partner, market pricing was used for the production cost with a zero DDT&E cost. An updated version of NASA's Advanced Missions Cost Model(AMCM) was used to estimate the cost of future systems not currently in use or offered by a commercial partner. AMCM is a parametric technique that estimates system life-cyle cost and schedule components using the following variables to relate to historical systems: number of systems to produce, dry mass (kg), system type, initial operating year, system generation, technical complexity and program complexity. A modified version of AMCM by Rolley et al modernizes and updates the parameters to represent lunar and Martian systems [63]. One of the key differences in the propulsion systems demonstrated in this study was the Technology Readiness Level(TRL) of those systems and its impact on the cost and schedule components in that lower TRL systems exhibited longer development times and larger associated costs.

The life-cycle cost of each system ultimately affects the total architecture cost, which can be assessed as either an objective or enforced as a constraint within the optimization. Equation 6.1 represents how the total architecture cost, including life cycle cost components, can be mathematically modeled within RPO. Where A_i^B represents a binary decision variable representing each potential system i and $A_{Dev,s}^B$ is a binary decision variable representing the development of each system type s. $Cost_{NR}$ and $Cost_R$ represent the non-recurring DDT&E cost and recurring production cost of each system respectively. An important note here is that constraints were added such that a system cannot be selected unless its associated DDT&E component is selected. Documented in Table A.1 are the life-cycle cost components for each system as well as the parameters used to estimate those values.

$$TotalCost = \sum_i A_i^B * Cost_{R,i} + \sum_s A_{Dev,s}^B * Cost_{NR,s} \tag{6.1}$$

The results in this section are as follows: A) a demonstration of the methodology applied to portfolio selection of an architecture in terms of a single objective metric, B) a demonstration of a Pareto trade-off analysis between cost and a performance metric, C) a demonstration of methods of accounting for financial and operational robustness, D) a Pareto trade-off between architecture readiness and total architecture cost, E) and the impact of an annual budget limit on system selection and resulting schedule.

6.1 Demonstration of Single Objective Optimization

The first demonstration of this method details the optimization of a space exploration architecture for minimum total architecture cost with some associated constraints in terms of mission capabilities as well as the Candidate System Library as previously detailed. Requirements are imposed on the architecture in that a total of 4 crewed missions must be sent to a Deep Space Habitat, which is composed of a Power Propulsion Element, a habitation module as well as several required systems(Robotic Arm, Crew Airlock, Propellant Storage Module, and Science Airlock).

The resulting architecture is both feasible and optimal in that it was the minimum cost solution while still obeying all of the imposed constraints. The topology of the architecture is detailed in Figure 6.2 and represents the capability transfer between systems to meet system to system requirements. For clarity, the connections representing the mission control capability were removed from the network diagram. A sand chart is shown in Figure 6.3a and demonstrates the calculated annual budget for each year. Detailed in Table 6.3b is the number and type of each system that was selected for the architecture. A Gantt chart of the phase scheduling is shown in Figure 6.4 and demonstrates the outcome of the scheduling constraints imposed on the architecture. The resulting architecture utilizes Nuclear Thermal Rocket in-space propulsion as well as commercial heavy lift launch vehicles. While this is the minimum cost architecture, its first crewed mission is delayed by the development of the NTR system compared to other architectures. The two year cadence of crewed missions is a result of the two year duration of NTR production. The production of an Orion vehicle and the required launch vehicles the year of a crewed mission results in the non-uniform annual spending as seen in the sand chart. This single point optimization serves as an example of the applied methodology and the available output for every single point along the Pareto frontiers found in the following analyses.

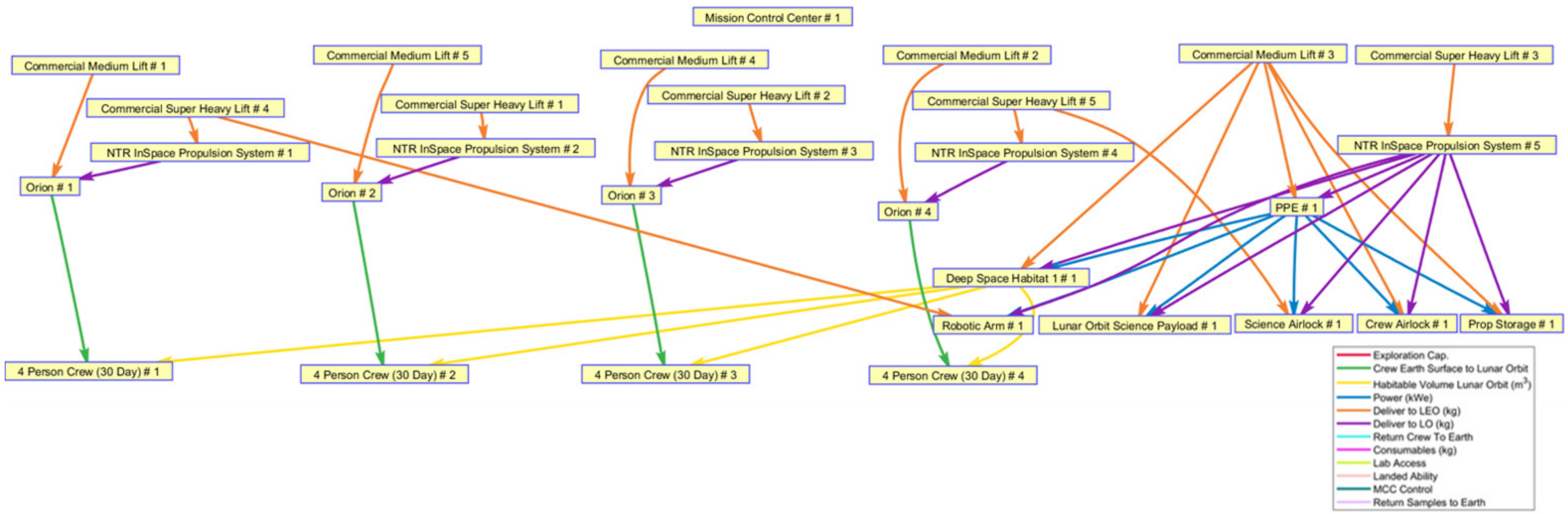
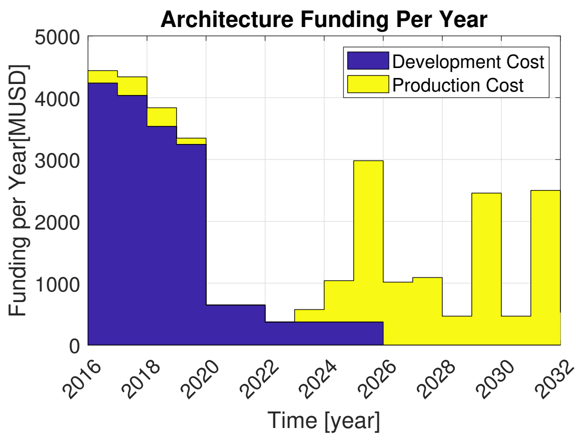


Fig. 6.2.: System to System interactions defined by the variable x_{cij} from the solution of the optimization problem with capability transfer type labeled in the graph. For example each lunar orbit mission is supported by a crew who is supported by Orion and a Deep Space Habitat. These support systems are launched into space on rockets and sent to NRHO with a propulsion stage.



(a)

Systems Selected	Quantity
4 Person Crew, 30 Day	4
Commercial Super Heavy Lift	5
Commercial Medium Lift	5
Orion Vehicle	4
Deep Space Habitat	1
Power Propulsion Element	1
Science Airlock	1
Crew Airlock	1
Robotic Arm	1
Propellan t Storage	1
Lunar Orbit Science Payload	1
NTR In-Space Propulsion System	5
Robotic Landers	4
Mission Control Center	1
Total Architecture Cost(2016 BUSD)	33.265

(b)

Fig. 6.3.: Total development and production budget per year(a) and table of architecture system selection and cost(b)

6.2 Architecture Total Cost vs Architecture Performance

A powerful application of Programmatic Portfolio Optimization is the ability to explore the design space and examine the impact of different stakeholder metrics and how they influence system selection. In this demonstration, two metrics, a performance objective and total architecture cost, compete against each other and form a Pareto frontier of optimal portfolios of systems. The performance objective in this demonstration is crew duration spent at a deep space habitat measured in crew-days, where a longer duration stay would require more consumables, more supplies, and a higher volume habitat. Options for varying sizes of habitat, crew stay durations and logistics vehicles for resupply were modeled in the the Candidate System Library. Several other systems were deemed highly desired space systems including robotic landers, a crew airlock, a science airlock, and a robotic arm and were constrained to be mandatory systems of any portfolio.

The objective function was set to minimize the total architecture cost as shown below, and constraints were placed on the architecture such that a minimum number

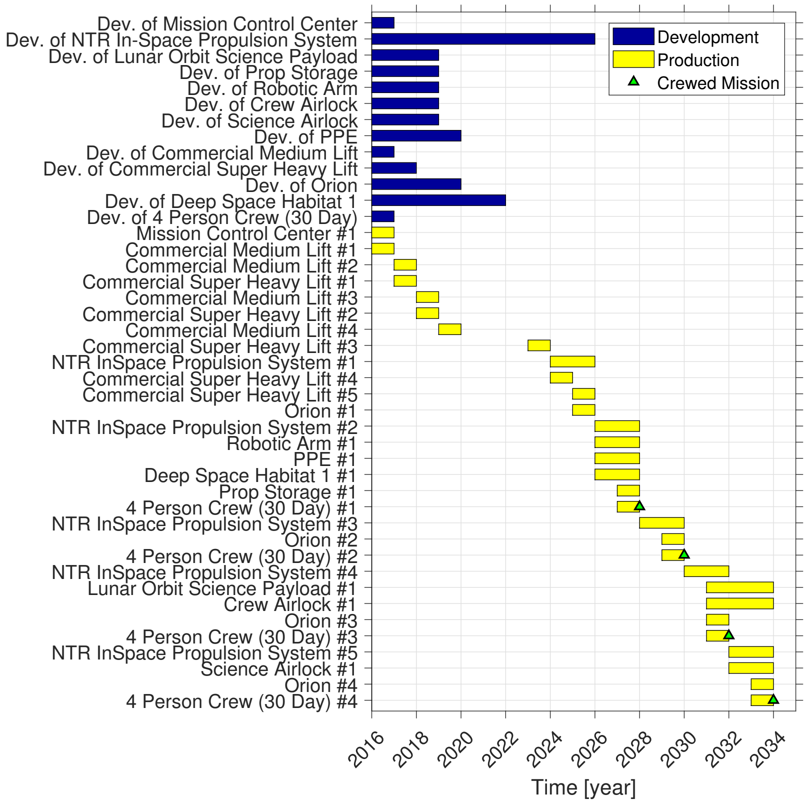


Fig. 6.4.: Life-cycle phase scheduling of single objective optimal architecture

of crew-days spent at the Deep Space Habitat were required. The Pareto frontier was evaluated by gradually increasing the required crew duration and determining the optimal architecture at each step.

$$Objective = \sum_i A_i^B * Cost_{R,i} + \sum_s A_{Dev,s}^B * Cost_{NR,s} \tag{6.2}$$

Here, A_i^B represents a binary decision variable representing each unit of a potential system choice and A_s^B is a binary decision variable representing the development of any potential system. $Cost_{R,i}$ and $Cost_{NR,s}$ represent the non-recurring DDT&E cost and recurring production cost of each system respectively.

Some constraints were used to provide reasonable bounds on the problem. In order to provide a nontrivial solution, a minimum of a single crewed mission is required. To provide an upper bound on the problem, a maximum of four crewed mission were

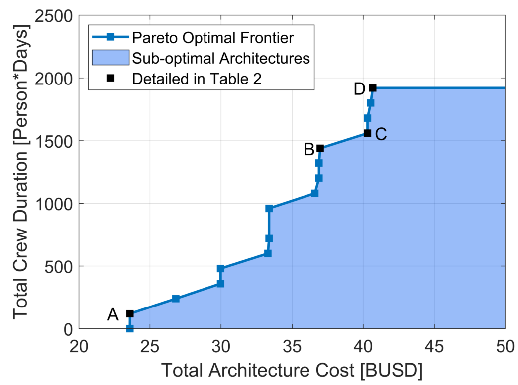


Fig. 6.5.: Pareto frontier of total architecture cost versus total crew duration

allowed to the deep space habitat. While the deep space habitat could accommodate many more missions, this method of bounding allowed for demonstration of the capability trade-off.

The resulting Pareto frontier and table of selected architectures can be found in Figure 6.5 and Table 6.2. The saw tooth appearance of this Pareto frontier is a result of technologies that enable longer duration stays but require the expenditure of budget for development and production of these systems. An increased number of crewed missions and longer duration missions require larger habitats, more supporting systems, and thus a higher associated total cost. Table 6.2 does not detail all architectures along the Pareto frontier but does illustrate how additional logistics flights and different deep space habitats enable both additional missions and longer duration missions. For instance, architecture B employs a larger Deep Space Habitat and logistics fights and is thus able to sustain a 90 day mission. These additional systems and their required supporting systems increase the total cost but allow for a higher crew-days performance metric. Similar behavior exists in architectures C and D where even higher crew-days metrics are achieved. While this is a simple example with some expected results, it demonstrates the ability to compare competing metrics and to select optimal portfolios along the Pareto frontier for those metrics.

Table 6.2.: Associated total architecture cost and total crew duration of select portfolios on the Pareto frontier with associated allocation of systems

Portfolio	A	B	C	D
Total Crew Duration [Crew*Days]	120	1440	1560	1920
Total Archtiecture Cost [2016 BUSD]	23.59	36.97	40.30	40.65
4 Person Crew (30 Day)	1	0	1	0
4 Person Crew (60 Day	0	0	0	0
4 Person Crew (90 Day)	0	0	0	0
4 Person Crew (120 Day)	0	3	3	4
Deep Space Habitat 1	1	0	0	0
Deep Space Habitat 2	0	1	1	1
Deep Space Habitat 3	0	0	0	0
Orion	1	3	4	4
Commercial Heavy Lift	0	0	0	0
Commercial Super Heavy Lift	2	6	7	7
Commercial Medium Lift	1	2	3	3
Government Super Heavy Lift 1B	0	0	0	0
Government Super Heavy Lift 1A	0	0	0	0
Logistics Module 1	0	4	4	0
Logistics Module 2	0	0	0	4
Logistics Module 3	0	0	0	0
Logistics Module 4	0	0	0	0
PPE	1	1	1	1
Science Airlock	1	1	1	1
Crew Airlock	1	1	1	1
Robotic Arm	1	1	1	1
Prop Storage	1	1	1	1
Lunar Orbit Science Payload	1	1	1	1
SEP In-space Propulsion System	0	0	0	0
NTR In-Space Propulsion System	2	5	6	6
Chemical In-Space Propulsion System	0	0	0	0
Mission Control Center	1	1	1	1

6.3 Operational and Financial Robustness

Two methods of examining robustness are investigated including uncertainty of architecture feasibility(operational) and uncertainty in architecture total objective value(cost). Some assumptions were made about the capabilities, costs, and uncertainty associated with the various propulsion systems. The uncertainty bounds used in the operational robustness formulation and the standard deviations used in the cost robustness formulation were determined through literature review and subject matter expert estimates. These estimates were heavily dependent on the Technology Readiness level(TRL) and complexity of the system. The uncertainty set of the performance and the standard deviations of the cost components is represented by a percentage of the expected value as shown in Equations 6.3 to 6.5. The U values used in this study are $U_{NTR} = 1.0$, $U_{SEP} = 0.5$, $U_{Chem} = 0.25$, $U_{GoVSHLV} = 0.25$, and $U_{CommHLV} = 0.5$ with a standard value of 0.05 for other system. The operational uncertainty set for each system can be found in Table A.2 and the standard deviations of the cost components can be found in Table A.1.

$$\sigma_{Dev,s} = U_s * Cost_{NR,s} \tag{6.3}$$

$$\sigma_{Prod,i} = U_i * Cost_{R,i} \tag{6.4}$$

$$\hat{S}_{ci} = U_i * S_{C,i} \tag{6.5}$$

The first method utilizes the Bertsimas-Sim formulation to evaluate the robustness of feasibility [45] specifically applied to the in-space transit capability where systems are sent from Earth orbit to Lunar orbit. This effectively addresses the concern that a propulsion system, even if not operating nominally, will be able to fulfill the mission requirements of launching a payload into space. Weighting coefficient Γ represents the stakeholder risk aversion to the likelihood of failure of mission akin to a confidence level for the mission. Simply put, the product of Γ and statistical uncertainty must be less than the margin of system performance required.

Architectures with minimum total cost were determined for various values of risk aversion Γ. The results of this assessment of stakeholder operational risk aversion, or operational conservatism, can be found in Figure 6.6. When risk aversion is low, Nuclear Thermal Rocket propulsion and commercial heavy lift launch vehicles are selected. With increasing risk aversion Γ, additional NTR systems are selected to provide sufficient margin. With further increasing risk aversion together with the higher uncertainty of the NTR performance, chemical rocket propulsion systems are selected.

The second form of uncertainty representation, that in the objective function, examined uncertainty in architecture cost and represents a measure of financial robustness. The minimum architecture cost is evaluated using an objective where a 1-sigma financial uncertainty is included as a penalty function multiplied by associated risk aversion factor Γ. Optimal portfolios were found for increasing values

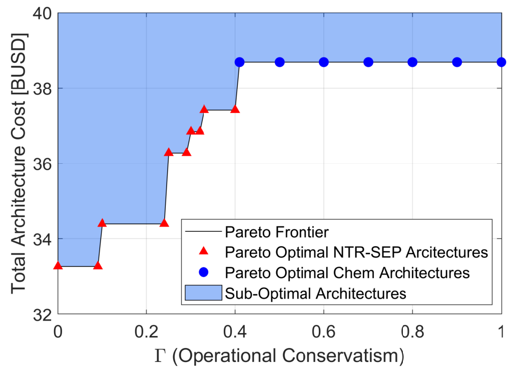


Fig. 6.6.: Results from minimum cost optimization using Bertsimas-Sim methodology. Each plot point represents a single architecture.Note: The same chemical architecture is pareto optimum for the various levels of conservatism; this cost is the same for all blue points.

of financial conservatism. Several portfolios dominate over certain ranges of Γ and can be found on the resulting Pareto frontier of risk aversion and total cost as shown Figure 6.7. When stakeholders are risk tolerant($\Gamma < 1.1$) architectures including commercial heavy launch vehicles and Nuclear Thermal Rockets dominate. When risk aversion is high(Γ¿1.1) and thus tolerance of cost overruns low, government super heavy launch vehicles and in space chemical propulsion are selected. This simple example demonstrates the ability to conduct a trade-off analysis with total architecture cost and financial uncertainty.

$$Objective = TotalCost + \Gamma * \left(\sum_s A^B_{Dev,s} * \sigma_{Dev,s} + \sum_i A^B_i * \sigma_{Prod,i} \right) \tag{6.6}$$

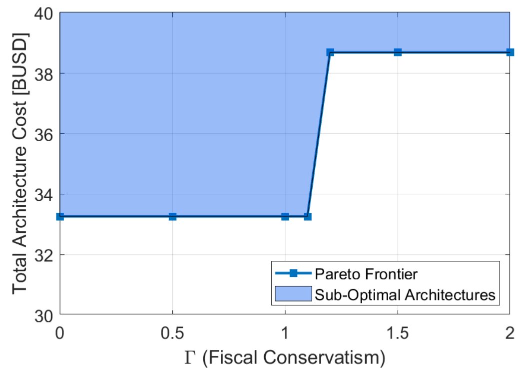


Fig. 6.7.: Results from minimum cost optimization using the uncertainty of objective method applied to the evaluation of financial risk. Each plot point represents a single architecture.

6.4 Architecture Readiness vs Architecture Total Cost

In most exploration missions, architecture cost and schedule compete against each other. For single system comparisons, cost and schedule can easily be compared, however in a larger interconnected architecture these comparisons become more complex and often intractable. The enhanced version of Robust Portfolio Optimization with the scheduling adaptation allows for the trade-off of different objectives, including schedule, and the exploration of a pareto frontier composed of different portfolio combinations.

In a similar fashion to the comparison of crew duration and total architecture cost, a comparison between architecture readiness time and architecture total cost is demonstrated. Here, architecture readiness time represents the year of first possible crewed mission. Shown in Figure 6.8 is a trade-off comparison of a multi-objective optimization of the year of first crewed launch and total architecture cost with the associated system allocation seen in Table 6.3. The absolute minimum cost architecture was composed of only nuclear in-space propulsion systems and commercial launch vehicles whereas the absolute minimum schedule architecture was composed of chemical in-space propulsion systems and government heavy launch vehicles. The impact of how an architect values the immediacy to fly the first crewed mission and how that affects the total cost of the entire architecture is demonstrated by the selection of different portfolios. Since this optimization carries a requirement of four missions, certain systems may be advantageous to flying a crewed mission sooner, but have negative impacts to the resulting total architecture cost even after the first mission has been flown. This is primarily due to the large Design Development Test and Evaluation (DDT&E) cost of certain space systems. For instance, Nuclear Thermal Rockets and Solar Electric propulsion require longer development times due to their lower Technology Readiness Level but indirectly enable significant cost savings over chemical in space propulsion. These advanced propulsion systems negate the need for government super heavy launch vehicles and allow commercial launch vehicles to be used, however because of the lower TRL of these advanced in-space propulsion systems require longer development and production time and thus delay the readiness time of the architecture. The middle point represents a compromise between cost and schedule in which Solar Electric Propulsion is used for DSH elements and Nuclear Thermal in-space propulsion is used for crewed missions.

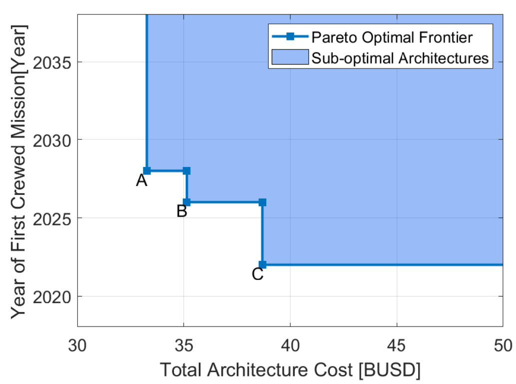


Fig. 6.8.: Pareto frontier of total architecture cost versus architecture readiness time

Table 6.3.: Results of architecture readiness time versus total architecture cost

Portfolio	A	B	C
Year of First Crewed Flight	2028	2026	2023
Total Archtiecture Cost [2016 BUSD]	33.26	35.14	38.69
4 Person Crew (30 Day)	4	4	4
4 Person Crew (60 Day	0	0	0
4 Person Crew (90 Day)	0	0	0
4 Person Crew (120 Day)	0	0	0
Deep Space Habitat 1	1	1	1
Deep Space Habitat 2	0	0	0
Deep Space Habitat 3	0	0	0
Orion	4	4	4
Commercial Heavy Lift	0	0	0
Commercial Super Heavy Lift	5	6	0
Commercial Medium Lift	4	2	0
Government Super Heavy Lift 1B	0	0	0
Government Super Heavy Lift 1A	0	0	5
Logistics Module 1	0	0	0
Logistics Module 2	0	0	0
Logistics Module 3	0	0	0
Logistics Module 4	0	0	0
PPE	1	1	1
Science Airlock	1	1	1
Crew Airlock	1	1	1
Robotic Arm	1	1	1
Prop Storage	1	1	1
Lunar Orbit Science Payload	1	1	1
SEP In-space Propulsion System	0	2	0
NTR In-Space Propulsion System	5	4	0
Chemical In-Space Propulsion System	0	0	5
Mission Control Center	1	1	1

6.5 Impact of Annual Budget Limit

Space agencies must operate within their financial bounds and thus the annual budget constrains the scope of missions as well as the resulting schedule. With the enhancement of scheduling to the Robust Portfolio Optimization, limits on annual funding are implemented by constraining the sum of all of the active development, production, and operational phases of each selected system for each year.

Application of the annual budget constraint is demonstrated here with the previously described architecture of 4 crewed missions and the required DSH elements, optimized for a combination of minimum architecture completion time at varying budget limits. In this example, completion time is defined as the time at which a Deep Space Habitat with the required elements is operational and 4 missions have been flown to the DSH. Figures 6.10 and 6.11 demonstrate how the annual budget impacts the scheduling of life cycle phases. The difference in system completion time is clearly seen in that the 4.4 billion dollar annual budget case has a completion time of 10 years whereas the 2.4 billion dollar annual budget has a completion time of 14 years.

Shown in Figure 6.9 is the Pareto frontier of the completion time of a specific architecture given an enforced budget constraint. The impact of an annual budget has a large effect on the completion time of the architecture. As budget increases, the architecture reaches a point where additional funding does not improve the schedule. Additionally, as annual funding decreases, the architecture completion time increases until the architecture becomes infeasible. The curve has several inflection points resulting in large changes in completion time with minimal change in funding. These points are a result of the ability to afford multiple development projects at the same time given the specific budget. For architectures with annual funding greater than 2.5 billion dollars, chemical propulsion systems are selected, but with less than 2.5 billion dollars Nuclear Thermal Rocket propulsion systems are selected. While the Nuclear Thermal Rocket has a higher production cost than the comparable chemical system, the sum of government heavy launch vehicles and chemical in space propulsion becomes infeasible when the annual budget is reduced.

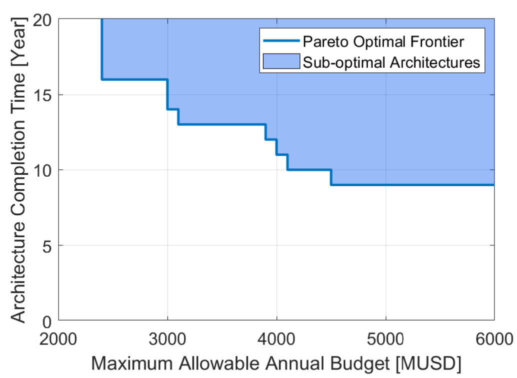
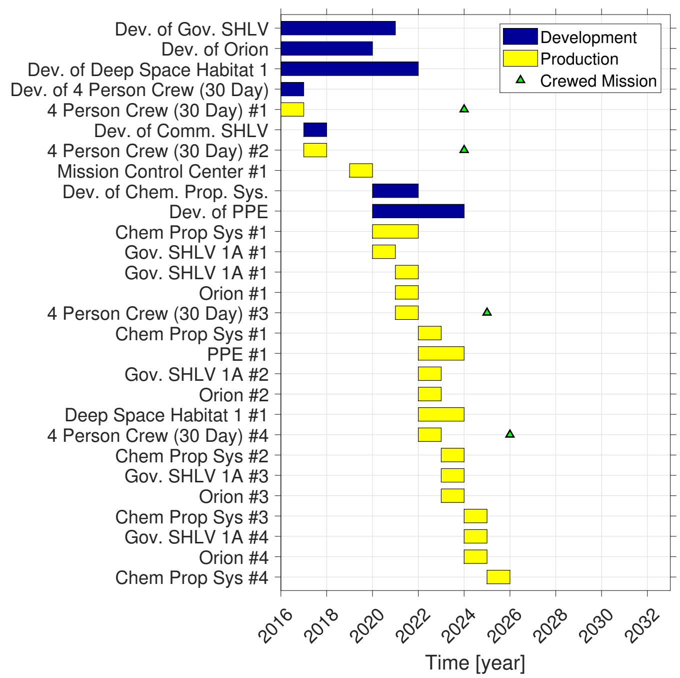
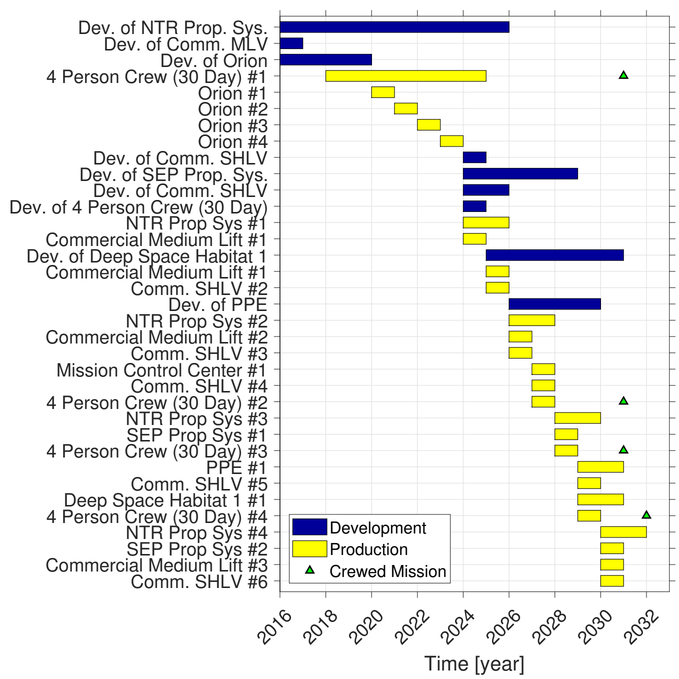


Fig. 6.9.: Max annual budget versus year of architecture completion

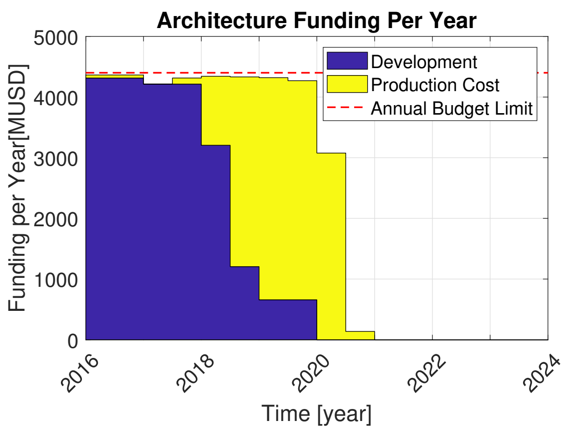


(a) 4.4 Billion dollar annual budget limit

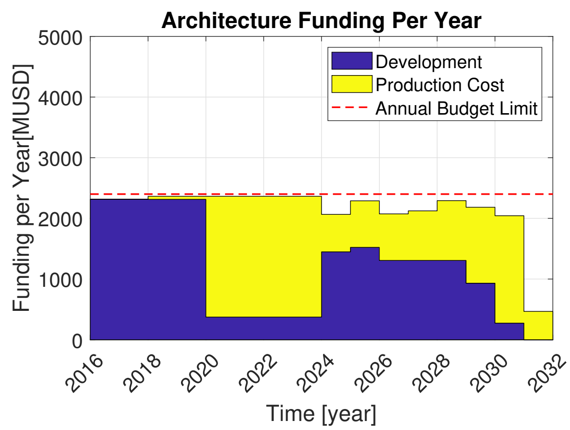


(b) 2.4 Billion dollar annual budget limit

Fig. 6.10.: Comparison of sequencing of development, production, and operation phases of systems given different annual budget limits



(a) 4.4 Billion dollar annual budget limit



(b) 2.4 Billion dollar annual budget limit

Fig. 6.11.: Comparison of annual development and production spending given different annual budget limits

6.6 Limitations and Future Improvements

Several limitations were discovered in this application of this enhanced RPO method as applied to a space exploration architecture and are planned to be addressed in future studies. First, the use of fixed values for the system capabilities and requirements is a restrictive assumption. While some currently available systems may have fixed values for their capabilities and requirements, a mission architect has some control over how other systems are designed and can size a system for the architecture. A future methodology improvement addresses this limitation using parametric relations to size systems within the optimization. These parametric relations could be based on historical data, modeling, or physics and would be used to scale the capabilities and requirements of a system. The suggested enhancement could be implemented with a combination of piece-wise linear relations and a Big-M formulation that relates capabilities and requirements. The resulting enhancement would remain a linear problem but would increase solving time due to the additional Big-M formulation [67].

The Candidate System Library for this study was non-exhaustive for the purpose of demonstration but includes several options for many of the systems required for a cislunar mission. In reality, many more potential systems are available resulting in a much larger design space and resulting combination space. For instance, the mission architect has many choices and sizes for propulsion systems that each have their own unique capabilities, cost, and associated constraints. A more in-depth study used in decision making would use a much wider breadth of potential systems.

7. DEMONSTRATION OF METHODS - LUNAR SURFACE EXAMPLE SCENARIO

This demonstration applies the Programmatic Portfolio Optimization method to the Lunar surface scope area and specifically highlights the scheduling enhancement(E1) and variable capability enhancement(E2). The scenario posed in this study mirrors NASA's proposed Artemis program where an orbital station placed in a Near Rectilinear Halo Orbit(NRHO) around the moon supports a multi-element lander approach to send crew to the surface. A key trait of this approach is the use of low cost commercial heavy lift launch vehicles to economically send cargo to the NRHO with the drawback of requiring a 3 element landing architecture to fit within the mass and volume requirements of the rocket. The concept of operations is shown in Figure 7.1. In this scenario a Transfer Element(TE) is used to move a Descent Module(DM) and Ascent Module(AM) from NRHO to a Low Lunar Orbit(LLO). The TE is temporarily left in LLO but is eventually placed in a safe disposal trajectory. From there the DM moves the AM to the surface of the moon. Because of the launch window to return to NRHO the crew has a surface duration time of either less than 48 hours or 7 day increments. The AM then returns the crew back to the station/Orion at NRHO. A second study examines lunar surface operations in the context of longer duration surface stays. The longer duration necessitates a surface habitat, power generation system, logistics resupply modules and the possibility of In-Situ Resource Utilization for propellant production. A trade explored in the long duration study is adding a staging point at the lunar surface by fueling of the AM at the lunar surface with propellant produced through ISRU. This could eliminate the need for refueling elements for the AM as well as smaller DM and TE elements resulting from an empty AM.

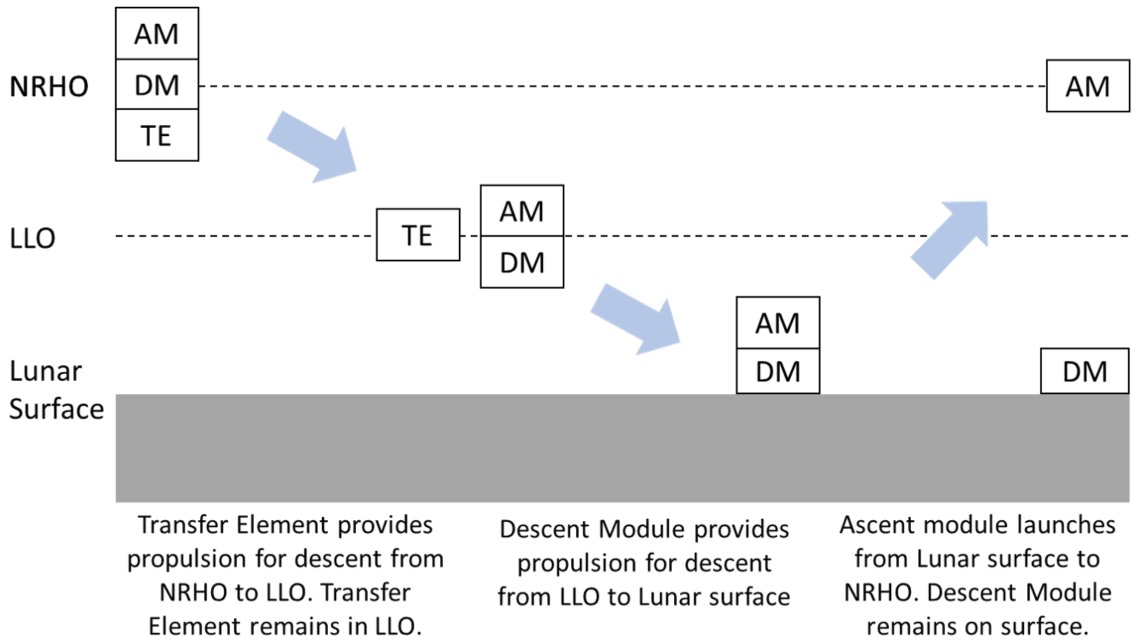


Fig. 7.1.: Three element Lunar Lander concept of operations

7.1 Capabilities

A functional decomposition of the architecture was completed that determined the inner workings of the architecture and highlighted the types of capabilities to examine with the PPO method. A list and description of the capabilities is found in Table 7.1. A compromise is made between capturing the highest level of detail within the architecture and simplifying the model to allow for reasonable optimization solve times. For instance, logistics at LS encompasses water, oxygen, food, clothes, hygiene products, and more but is summed as one quantity.

Table 7.1.: Lunar surface architecture capability descriptions

Capability	Description
Exploration	A top level capability equal to the crew duration spent on the surface of the Moon (days)
Mass ES to NRHO	Mass delivered to Near Rectilinear Halo Orbit from Earth Surface (t)
Crew Capability	Ability of crew to complete mission objectives
Deliver Crew to-from NRHO	Transfer and habitability of crew between Earth surface and Near Rectilinear Halo Orbit and vice versa(n)
Habitable Volume at NRHO	Habitable Volume at Near Rectilinear Halo Orbit (m^3)
Electrical Power and Docking at NRHO	Electrical power and Docking at Near Rectilinear to support other vehicles
Mass Transfer NRHO to LLO	Mass delivered to Low Lunar Orbit from Near Rectilinear Halo Orbit (t)
Mass Transfer LLO to LS	Mass delivered to Lunar Surface from Low Lunar Orbit (t)
LH2 Propellant at NRHO	LH2 & LOX Propellant available in Near Rectilinear Halo Orbit (kg)
CH4 Propellant at NRHO	CH4 and LOX Propellant available in Near Rectilinear Halo Orbit (kg)
Storable Propellant at NRHO	Storable Propellant available in Near Rectilinear Halo Orbit (kg)
LH2 Propellant at LS	LH2 and LOX Propellant created on Lunar Surface (kg)
Consumables at LS	Consumables required for crew duration at Lunar Surface
Habitable Volume at LS	Habitable volume on Lunar Surface (m^3)
Electrical Power at LS	Electrical Power at Lunar Surface

7.2 Candidate System Library

A list of general classes of systems were identified that fulfill the functions of the architecture. Unique system choices for each of these general classes were identified for addition to the Candidate System Library (CSL) for use in the Programmatic Portfolio Optimization that apply to both a short stay architecture and a long stay architecture. These options heavily mirror NASA's Artemis Missions. While there exist many alternative concepts as identified in the scope section, intentionally limiting the scope to parallel the Artemis mission was A) timely and B) fit within the bounds of a limited analysis team without the knowledge and software tools of a government agency like NASA. Additional resources and access to proprietary data and tools would allow for the assessment of additional concepts.

Key to the formation of the CSL is to identify and quantify each systems capabilities, requirements, risks, connectivity/compatibility issues, as well as life-cycle cost and schedule components for development, production and operation. Each of these characteristics has been estimated through the use of either parametric sizing tools, Subject Matter Experts, available literature, or historical relations. This list of systems with their attributes form the Candidate System Library from which combinations of systems can then be formed into feasible architectures. An overview of the candidate systems used in both the short stay and long stay missions is shown in Table 7.2 with their respective capability type, requirement type, and technology dependency. More detailed attribute values for capability type, requirements, cost and schedule components are found in the appendix. Many systems capabilities and requirements are known prior to optimization and have fixed values. Other systems, like the Transfer Element and Descent Module are allowed to scale in terms of their capabilities and requirements and are sized within the optimization. The sizing of these is discussed in Subsection 7.2.1. This CSL contains systems with both fixed capabilities and variable capabilities.

Table 7.2.: Overview of Candidate System Library for Lunar Surface Missions

Short Stay Systems				
General Class	**Unique System**	**Capability**	**Requirement**	**Technology Dependence**
Transfer Element (TE)	• LH2 TE • CH4 TE • Press. fed Storable TE • Pump fed Storable TE	• Transfer mass from NRHO to LLO	• Mass delivered to NRHO • Propellant (Off-Load)	• CFM (CH4 and LH2)
Descent Module (DM)	• LH2 DM • CH4 DM • Press. fed Storable DM • Pump fed Storable DM	• Transfer Mass from LLO to Lunar Surface	• Mass delivered to NRHO • Transfer mass from NRHO to LLO • Propellant (Off-Load)	• CFM (CH4 and LH2)
Ascent Module (AM)	• LH2 AM • CH4 AM • Press fed Storable AM • Pump. fed Storable AM	• Crew Ascent/Descent (NRHO to LS to NRHO)	• Mass delivered to NRHO • Transfer mass from NRHO to LLO • Transfer Mass from LLO to Lunar Surface • Propellant (Re-use)	• CFM (CH4 and LH2)
Launch Vehicle (LB)	• Government SHLV • Commercial Heavy LV	• Mass delivered to NRHO		
Refueling Element (RE)	• LH2 RE • CH4 RE • Storable RE	• LH2 Propellant • CH4 Propellant • Storable Propellant	• Mass delivered to NRHO	• CFM (CH4 and LH2)
Gateway Element	• Gateway Habitat • Power Propulsion Element	• Power • Docking • Habitable Volume	• Mass delivered to NRHO	
Crew transfer vehicle	• Orion	• Deliver/Return Crew To/From NRHO	• Mass delivered to NRHO • Docking	
Crew	• 4 Person Crew	• Crew capability	• Deliver/Return Crew To/From NRHO • Habitable Volume	
Long Stay Systems				
General Class	**Unique System**	**Capability**	**Requirement**	
Surface Systems	• ISRU	• Propellant at LS	• Mass delivered to NRHO • Power at LS • Transfer mass from NRHO to LLO • Transfer Mass from LLO to Lunar Surface	• ISRU • Precision Landing
	• Surface Habitat	• Habitable Volume LS	• Mass delivered to NRHO • Power at LS • Transfer mass from NRHO to LLO • Transfer Mass from LLO to Lunar Surface	• Precision Landing
	• Power System	• Power at LS	• Mass delivered to NRHO • Transfer mass from NRHO to LLO • Transfer Mass from LLO to Lunar Surface	• Nuclear Reactor • Precision Landing

In this study, 4 distinct propellant choices for the lander elements were examined: A) Hydrogen based elements B) Methane based elements 3) storable propellant (Monomethylhydrazine and Nitrogen Tetroxide) pump-fed cycle based elements and 4)storable propellant pressure-fed cycle based elements. These choices are differentiated by their performance, technological complexity, and program complexity. For instance, methane and hydrogen have higher performance in terms of their specific impulse but require technology investment in Cryogenic Fluid Management and additional power and inert mass on the lander element. The storable options are less complex but have lower performance in terms of their specific impulse. An assumption for this study was made such that for each trade the propellant for all 3 elements is the same. The assumption is relaxed in a later example.

For each potential system, the life cycle cost and schedule component values of the Design, Development, Test, and Evaluation (DDT&E) phase, the production phase, and the operation phase were estimated. A mixture of literature and parametric modelling was used to estimate the cost components for each system. For systems currently offered by a commercial partner, market pricing was used for the production cost with a zero DDT&E cost. A modified version of NASAs Advanced Missions Cost Model (AMCM) was used to estimate the cost of future systems not currently in use or offered by a commercial partner. The modified version of AMCM by Rolley et al. adjusts the parameters to represent lunar and Martian systems [63]. AMCM is a parametric technique that estimates system life-cycle cost and life-cycle phase durations using the following variables to relate to historical systems: dry mass (kg), system type, initial operating year, system generation, and difficulty.

Each system was examined in order to determine if it required any technologies that were below NASA Technology Readiness Level(TRL) 6 [68]. For each technology the cost and schedule required to raise the technology to TRL 6 was either estimated or determined through literature. For short duration stay missions Cryogenic Fluid Management was identified as a technology needed for the Methane and Hydrogen based landers. For the long duration stay missions several technologies were identified including precision automated landing, nuclear power and In-Situ Resource utilization for hydrogen based landers.

7.2.1 Element Parametric Sizing

Key to the application of the variable capability sizing is the relationship between system capabilities and requirements. The lander elements in this study were the primary variable systems and were assessed using a NASA proprietary tool called the Beyond Low Earth Orbit (LEO) Architecture Sizing Tool (BLAST). The values used in this study are well documented in an export controlled conference paper that applies the BLAST estimates with the PPO method [69].

The software allows the user to select from a range of Mass Estimating Relations (MERs) for each of the subsystem options: avionics, consumables, crew, EVA, environment, power, propulsion, and structures. After selecting the desired MERs for each element, the user builds the mission sequence a list of mission phases such as joining elements together, performing burns, and separating elements and the software then iterates to a closed solution from a mass perspective and provides the user with a breakdown of the system masses over time. Additionally, BLAST offers the ability to perform data sweeps, parametric sweeps, or distributions for any of the user inputs within the program. This allows the user to gain an understanding of the sensitivities of various aspects of the elements (such as specific impulse, duration, propellant choice, growth allowances, etc.) for the given mission profile (usually with regard to the number and size of burns). In this study, the payload mass is varied to determine the resulting gross mass and propellant mass.

Ascent element sizing was performed for each of the four propellant options: pump-fed LOX/LH2, pump-fed LOX/CH4, pressure-fed storable, and pump-fed storable. For the descent and transfer elements, parametric sizing and sensitivities were performed to generate gross mass curves for each element with varying payloads. For the descent element, the payload sweep was designed to reflect potential ascent gross masses, while the values for the transfer element were selected to reflect both the descent and ascent element mass.

The assumptions for the amount of Mass Growth Allowance (MGA) used for each subsystem in the sizing for each of the three elements can be seen in Table 7.3, with the propellant specific assumptions in Table 7.4. The avionics, consumables, and EVA subsystems have 0 percent MGA since they are based on historical flight data. The crew subsystem represents the weight of the crew, which is approximated as 80 kg each. The other subsystems MGA values are based on information in the AIAA standard for mass properties control for space systems based on the estimated design maturity [70].

Mass and power value estimates for active Cryogenic Fluid Management (CFM) for in-space and surface elements were estimated, as BLAST does not currently have the capability

Table 7.3.: Mass Growth Allowance assumed percentage for each subsystem.

Subsystem	MGA(%)
Avionics, consumables, Crew, EVA	0
Environment	23
Power	30
Propulsion, Structures	25

to size systems with active CFM. Active CFM was selected for each of the elements since the mission architecture in this dissertation has each of the elements transferring out to NRHO on a BLT. Additionally, the ascent and decent element structural masses were assumed to have the same inert mass fraction as the respective Apollo elements [71].

Table 7.4.: Assumptions for each of the four propellant options examined: pump-fed LOX/LH2, pump-fed LOX/CH4, pressure-fed storable, and pump-fed storable.

Propellant	Isp (s)	Thrust(N)	Residuals(%)	FPR (%)	Reference Engine
LOX/LH2	450	101,800	2	0	RL 10C-1
LOX/CH4	360	40,000	2	0	New Engine
Pressure-fed storable	326	33,600	2	0	AJ10
Pump-fed storable	340	55,000	2	0	RS-72

Inert mass breakdowns for each of the ascent elements for the four different propellant systems can be seen in Table 7.5. Independent of the propellant selection, the avionics, consumables, crew, environment, and EVA subsystems all are sized to the same mass. This is a direct result of the mission profiles assuming an 8 day duration(2 surface days, 6 orbit days) with 2 crew members and a total of 5 EVAs. The power numbers vary due to the increased power load required by the hydrogen and methane systems due to active CFM, which also impacts the propulsion subsystem. For the purposes of this analysis, it is assumed that the ascent module is capable of returning to Gateway with 100 kg of samples from the lunar surface.

Table 7.5.: Subsystem and propellant mass breakdown for the ascent elements. Note that all numbers are in kg.

Subsystem	**LOX/ LH2 (kg)**	**LOX/ CH4 (kg)**	**Press-Fed Storable (kg)**	**Pump-Fed Storable (kg)**
Avionics	311	311	311	311
Consumables	147	147	147	147
Crew	160	160	160	160
Environment	851	851	851	851
EVA	503	503	503	503
Power	922	871	848	848
Propulsion	1,812	1,110	1,162	1,123
Structures	2,403	2,019	2,033	2,013
Payload	100	100	100	100
Inert Mass	7,210	6,073	6,116	6,057
Propellant Mass	6,159	7,067	8,227	7,658
Gross Mass	13,369	13140	14,343	13,715

The size of the Ascent Module will have a direct impact on the size of the Descent Module and Transfer Element. Plots showing this relationship can be seen in Figure 7.2 for each of the four propellant options. Additionally, the size of the ascent and descent element directly impacts the size of the transfer element responsible for pushing the stack from NRHO to LLO. Plots detailing the transfer element grows as a function of the combined descent and ascent element mass can be seen in Figure 7.3. For both of these plots, solid lines represent the gross mass and dotted lines represent the propellant mass. The blue curve represents LOX/LH2, orange is LOX/CH4, grey is pressure-fed storable, and green is pump-fed storable.

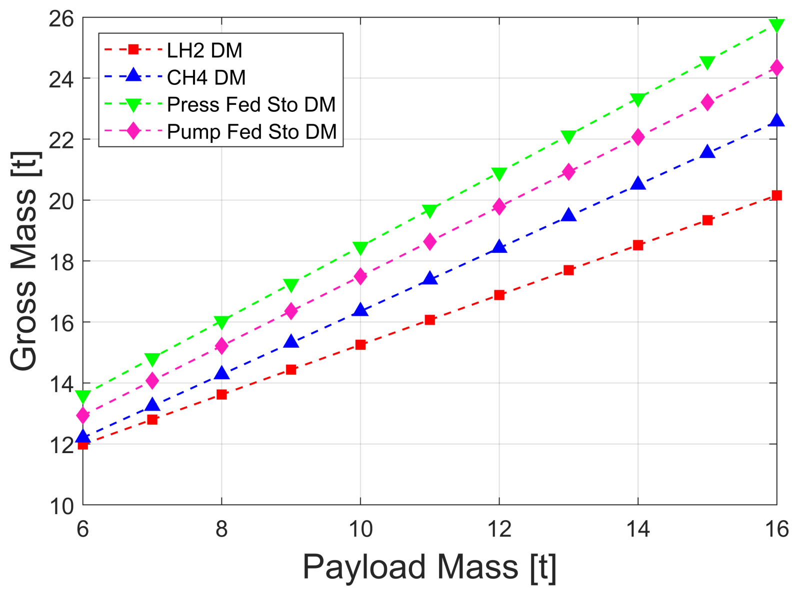


Fig. 7.2.: Relationship between payload(or ascent element mass) mass delivered to Lunar surface from LLO and system gross mass for lunar Descent Modules for various propellant choices.

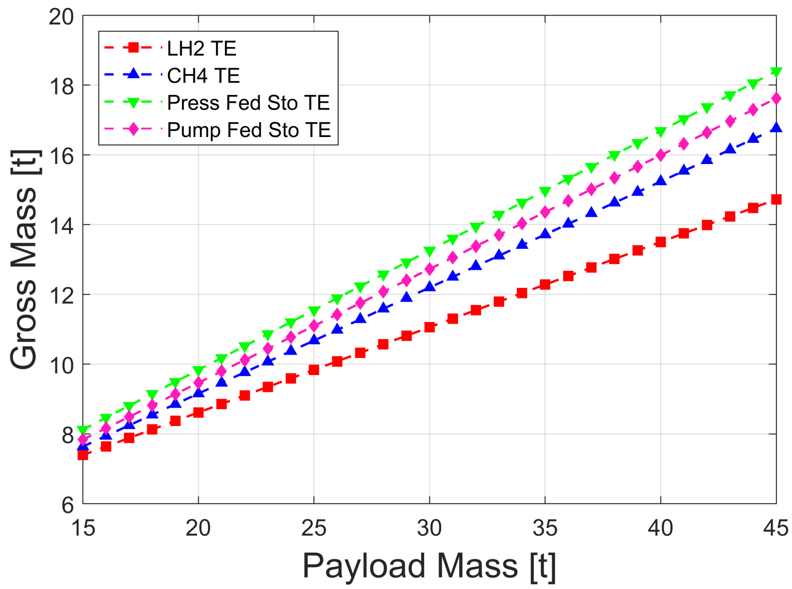


Fig. 7.3.: Relationship between payload mass delivered to LLO from NRHO and system gross mass for lunar Descent Modules for various propellant choices.

A majority of the descent element masses and some of transfer element masses for larger payloads are larger than the 15 t CLV limit discussed earlier. For all of these cases, the respective element offloads propellant to get under the 15 t limit and thus requires a refueling element to be launched to top off the necessary elements in order to achieve a successful landing mission. Within the analysis, it is assumed that the refueling elements have a lifetime of 3 years and any leftover propellant in the refueling element can be used on subsequent missions.

7.3 Short Stay Surface Missions

The PPO method as previously described is applied to a short stay human lunar lander space exploration mission and is examined at a high level in terms of cost and schedule. The results include: A) Demonstration of the methodology applied to portfolio selection of a single lunar lander mission for four specific propellant choices. Namely a Hydrogen based architecture, a Methane based architecture, a pressure fed storable based architecture and a pump fed storable architecture. B) Application of the methodology to a campaign of several lunar landing missions in order to evaluate long term trade-offs with cost and schedule.

7.3.1 Single Landing Study

The PPO methodology is applied to a simple example scenario with a single required mission and with a multi-objective function to reduce both cost and schedule. This effectively represents the cost of a single mission. The Candidate System Library for this study is non-exhaustive, but includes several options for many of the systems required for a cislunar mission. However, based on the preliminary results, some interesting findings are available and presented here. To compare the various propellant options, 4 cases are evaluated with constraints to only use specific propellant options. The schedule and operational inner workings of the architecture can be seen and demonstrated at this simple level with only 1 lunar surface mission.

The high level results of the optimization in terms of the overal cost, year of first mission and system selection are shown in Table 7.6. Due to its low technical complexity, the pressure fed storable dominates in terms of having an earliest mission of 2025 and a minimum total cost of 24 billion USD(2018) for the entire DDT&E, production, and operation of Gateway elements, landing elements, Orion, and all of the necessary launch vehicles. It should be noted that since the architecture start date is 2018, most of the development expenditures for SLS and Orion have already occurred and are not fully included in this figure. The more technologically advanced architectures have later dates for first missions and larger initial costs. However, as demonstrated in later sections, some of the more efficient architectures become cost competitive around 4-5 missions as the pressure fed storable requires several more refueling elements for subsequent missions.

Table 7.6.: Stakeholder metrics and system selection(allocation) of single mission study resulting from optimization. Total architecture cost is represents the entire cost of flying a single mission including development, production, and operation.

	Architecture	Hydrogen	Methane	Press Fed Storable	Pump Fed Storable
	Total Cost of First Mission[BUSD]	26.72	25.68	24.31	24.42
	Year of First Mission	2027.35	2027.35	2025.58	2025.66
	# of Surface Missions	1	1	1	1
System Allocation	Orion	1	1	1	1
	SLS	1	1	1	1
	Commercial HLV	6	6	6	6
	PPE	1	1	1	1
	Gateway MiniHab	1	1	1	1
	4 Person Crew	1	1	1	1
	LH2 TE	1	0	0	0
	LH2 TE Prop Off Load	0	0	0	0
	Meth TE	0	1	0	0
	Meth TE Prop Off Load	0	0	0	0
	Pres Fed Sto TE	0	0	0	0
	Pres Fed Sto TE Prop Off Load	0	0	1	0
	Pump Fed Sto TE	0	0	0	1
	Pump Fed Sto TE Prop Off Load	0	0	0	0
	LH2 DM	0	0	0	0
	LH2 DM Prop Off Load	1	0	0	0
	Meth DM	0	0	0	0
	Meth DM Prop Off Load	0	1	0	0
	Pres Fed Sto DM	0	0	0	0
	Pres Fed Sto DM Prop Off Load	0	0	1	0
	Pump Sto DM	0	0	0	0
	Pump Fed Sto DM Prop Off Load	0	0	0	1
	LH2 AM	1	0	0	0
	Meth AM	0	1	0	0
	Pres Fed Sto AM	0	0	1	0
	Pump Fed Sto AM	0	0	0	1
	LH2 Refuel Tanker	1	0	0	0
	Meth Refuel Tanker	0	1	0	0
	Sto Refuel Tanker	0	0	1	1

In this study the capability of the lander elements was determined within the optimization. As can be seen in Table 7.7, feasible values for the capabilities were found by the optimization and the resulting requirements scaled accordingly. The values for the variable capability are the exact minimum required to support other systems and to not contribute extraneous costs.

As noted before, two options were given for each Transfer Element and Descent Element: An option that scaled capability with a required gross mass delivered to NRHO up to the 15 mt commercial limit and a second option with a 15 mt gross mass delivered to NRHO but with the requirement that it be fully fueled before use. Several of the lander elements

Table 7.7.: Variable sizing values of Transfer Element and Descent Module determined by optimization. These values represent the optimum values for the variable capability and requirements of each system.

Propellant Architecture	Sys.	Requires Propellant at NRHO?	Variable Capability	Variable Requirement 1	Variable Requirement 2	Fixed Requirement
LH2	TE	No	NRHO to LLO Mass 31,376 kg	Deliver Mass to NHRO 11,396 kg	n/a	n/a
	DM	Yes	LLO to LS Mass 13,370 kg	LH2 Prop. at NRHO 3,006 kg	NRHO to LLO Mass 18,006 kg	Deliver Mass to NHRO 15,000 kg
Methane	TE	No	NRHO to LLO Mass 32,749 kg	Deliver Mass to NHRO 13,031 kg	n/a	n/a
	DM	Yes	LLO to LS Mass 13,140 kg	CH4 Prop. at NRHO 4,609 kg	NRHO to LLO Mass 19,609 kg	Deliver Mass to NHRO 15,000 kg
Press-fed Storable	TE	Yes	NRHO to LLO Mass 38,103 kg	Sto. Prop. at NRHO 1,037 kg	Deliver Mass to NHRO 15,000 kg	
	DM	Yes	LLO to LS Mass 14,342 kg	Sto. Prop. at NRHO 8,761 kg	NRHO to LLO Mass 23,761 kg	Deliver Mass to NHRO 15,000 kg
Pump-fed Storable	TE	No	NRHO to LLO Mass 36,799 kg	Deliver Mass to NHRO 14,945 kg	n/a	n/a
	DM	Yes	LLO to LS Mass 14,342 kg	Sto. Prop. at NRHO 7,457 kg	NRHO to LLO Mass 22,457 kg	Deliver Mass to NHRO 15,000 kg

required prop offloading. This demonstrates the novel capability of variable sizing of systems within the portfolio optimization.

With single propellant choice architectures, the variable scaling is not strictly necessary and can be avoided by specifying known fixed capabilities and requirements prior to optimization. If an architecture has a mix of propellant choices, such as Methane AM and Hydrogen DM and Tug, the variable scaling becomes much more beneficial to the user as the number of potential combinations becomes intractable. However, an assumption was made such that refueling vehicles were only single propellant type and thus a mixed propellant architecture would require the development, production and operation of multiple types of refueling elements. However, a cost optimal architecture was found that combined pump fed and pressure fed storable propellants as will be demonstrated in the next section.

The life-cycle phase scheduling of each propellant architecture is shown in the gantt charts in Figures 7.4 - 7.7. The difference in the year of first mission for the various propellant choices results from the development phase of the lander elements. The development of the Ascent Module is the critical path system for each architecture and the required development time dictates the year of first mission. As can be seen, the combination of

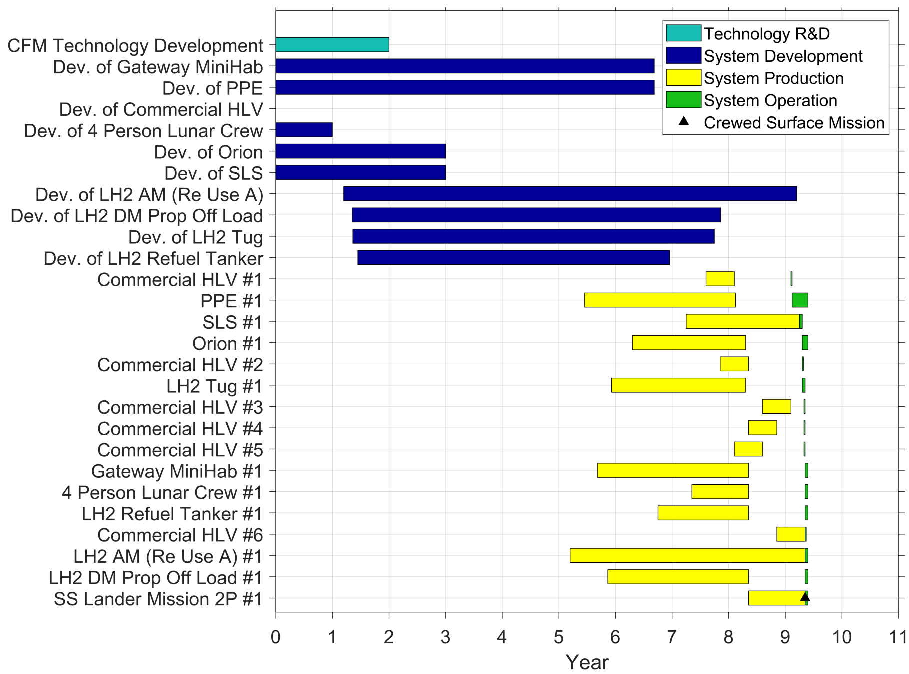


Fig. 7.4.: Hydrogen propulsion based architecture single mission scheduling

longer system development time and the required technology development lead to a later first mission.

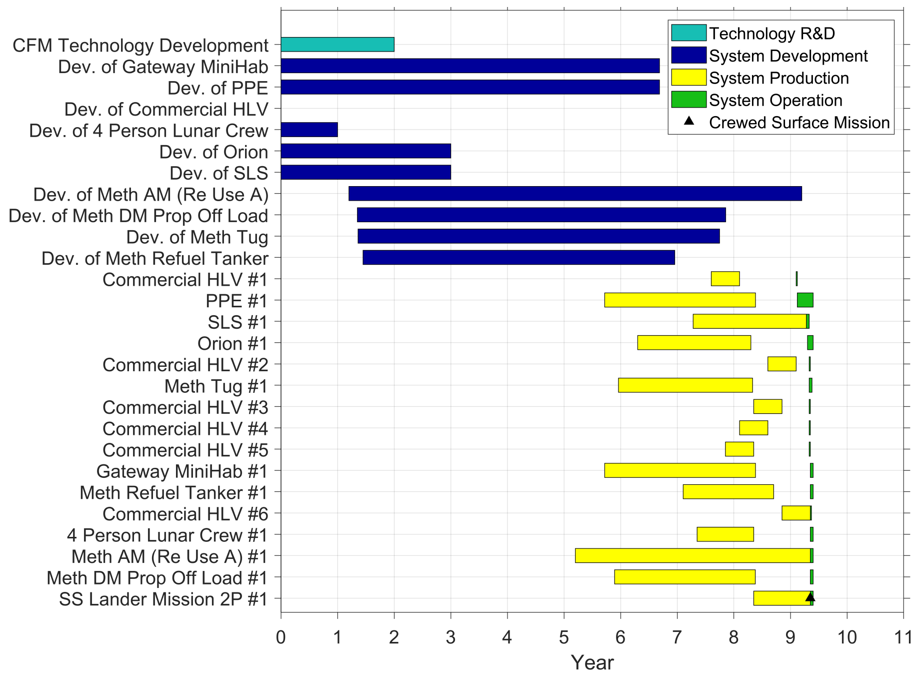


Fig. 7.5.: Methane propulsion based architecture single mission scheduling.

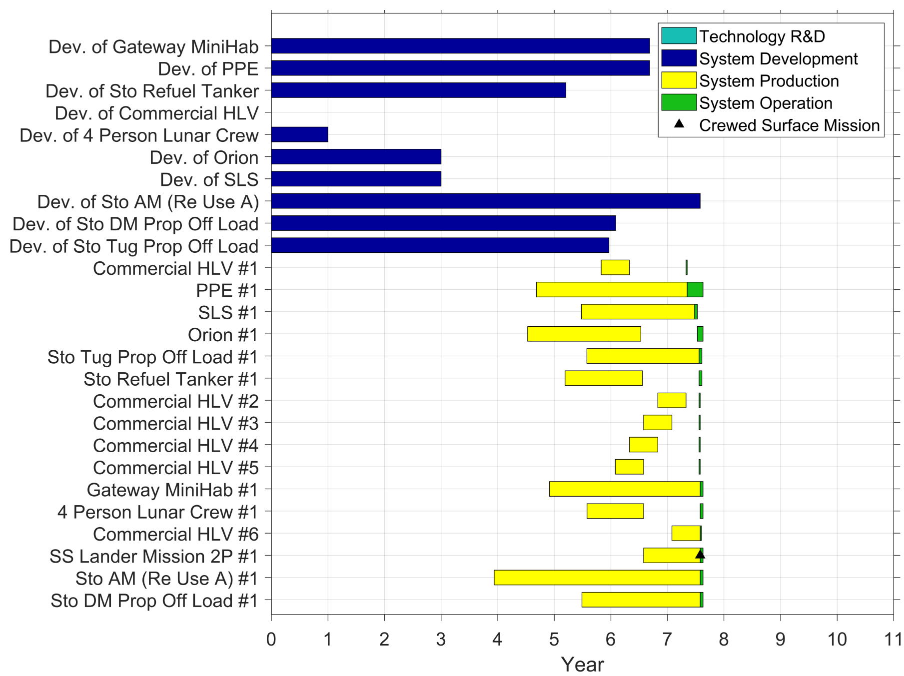


Fig. 7.6.: Storable pressure fed propulsion based architecture single mission scheduling

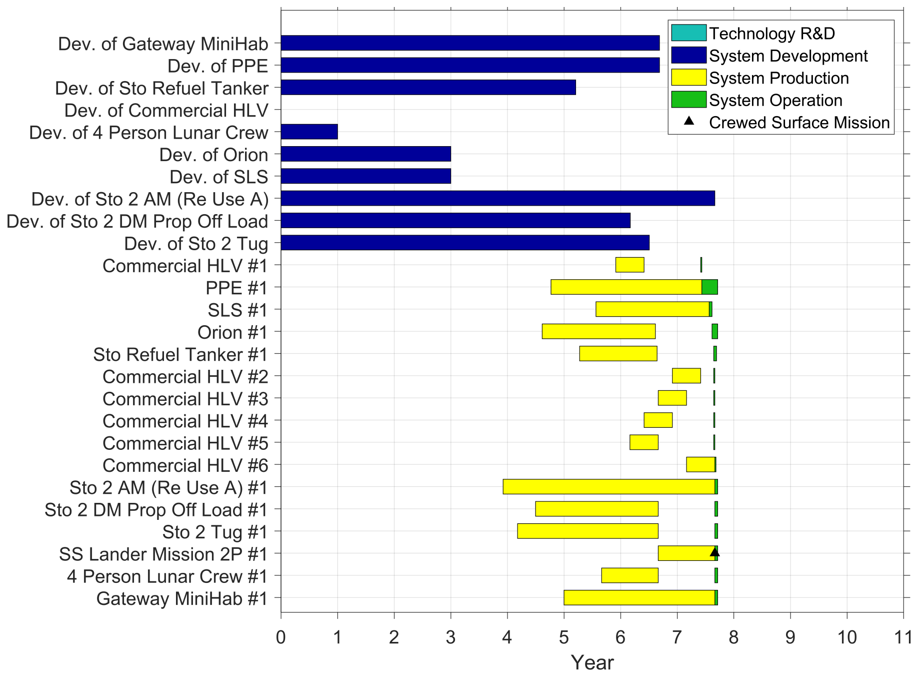


Fig. 7.7.: Storable pump fed propulsion based architecture single mission scheduling

Figures 7.8-7.11 detail the system to system interactions within the architecture. The connectivity and value of capability being transferred is determined by the architecture to create a feasible and optimal solution.

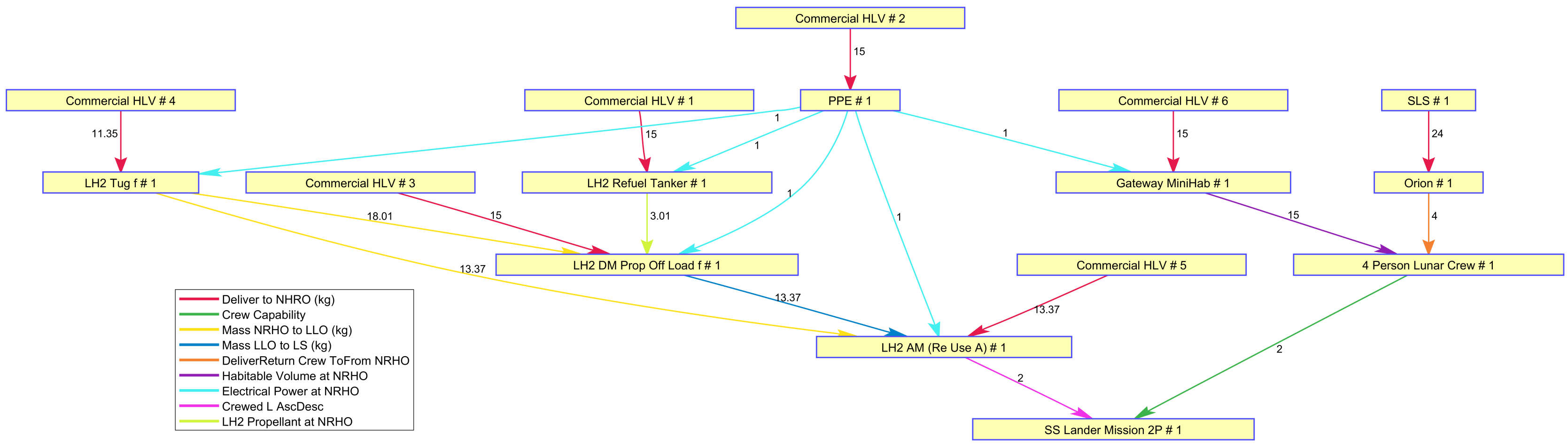


Fig. 7.8.: System interactions of Hydrogen propulsion based architecture single mission with type of capability transfer denoted by color and value of capability transfer denoted with numbers

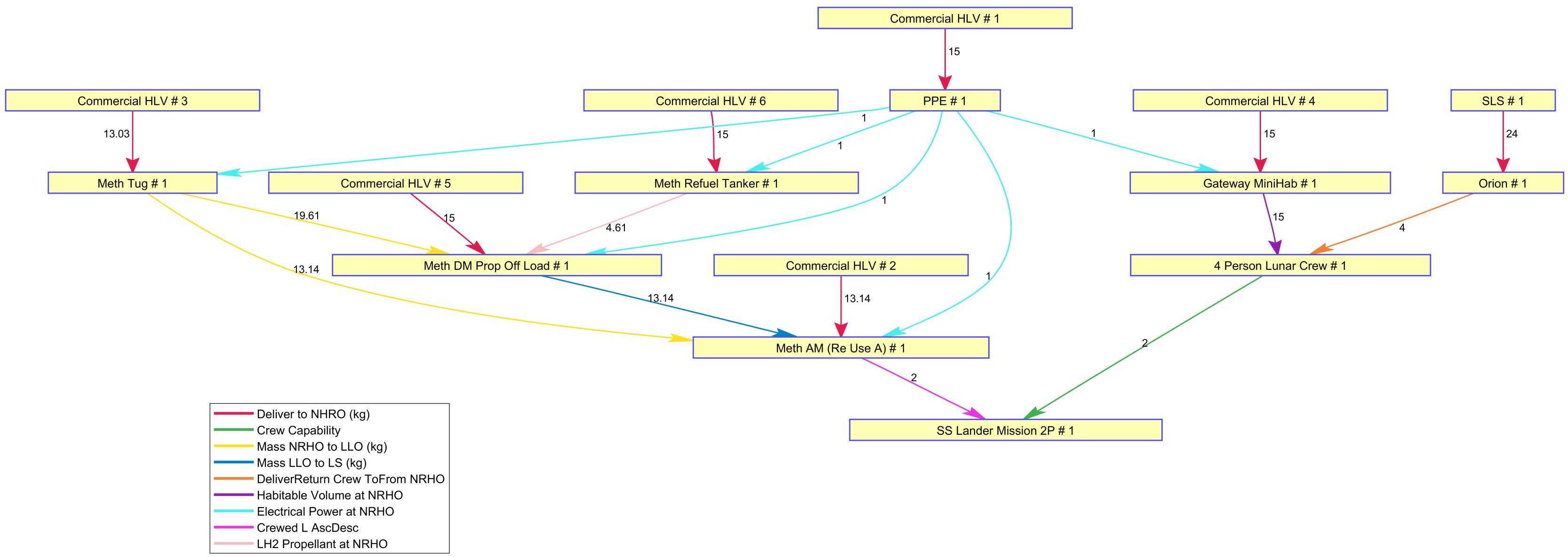


Fig. 7.9.: System interactions of Methane propulsion based single mission scheduling architecture with type of capability transfer denoted by color and value of capability transfer denoted with numbers

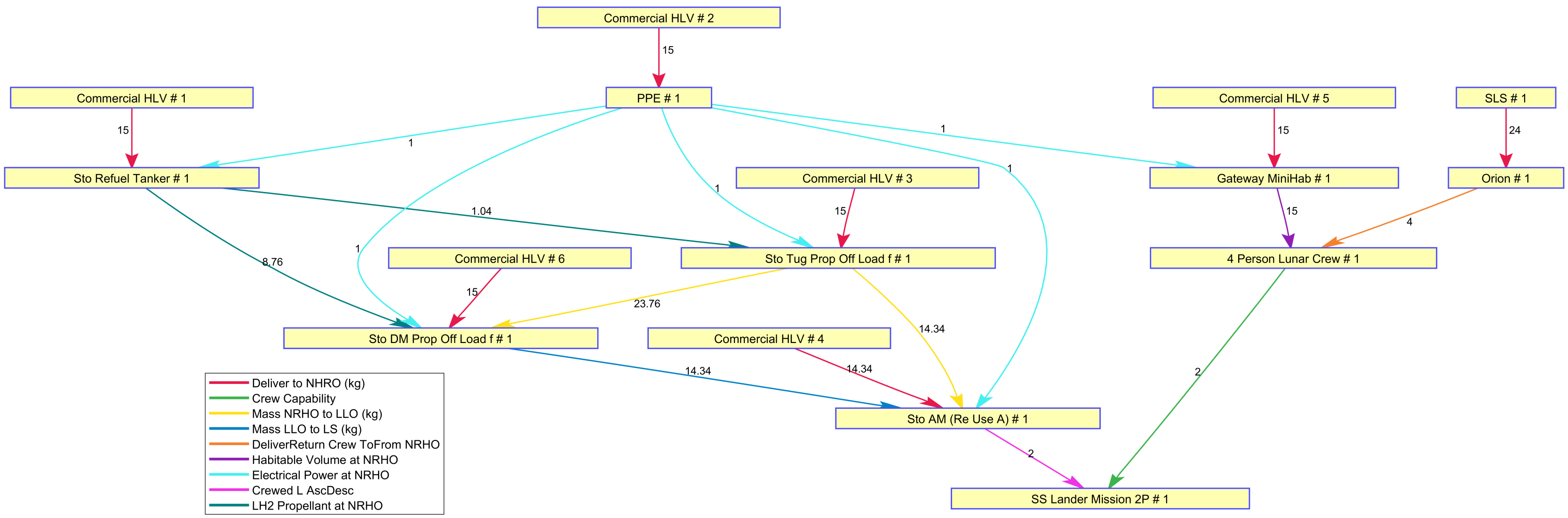


Fig. 7.10.: System interactions of Storable pressure fed propulsion based single mission architecture with type of capability transfer denoted by color and value of capability transfer denoted with numbers

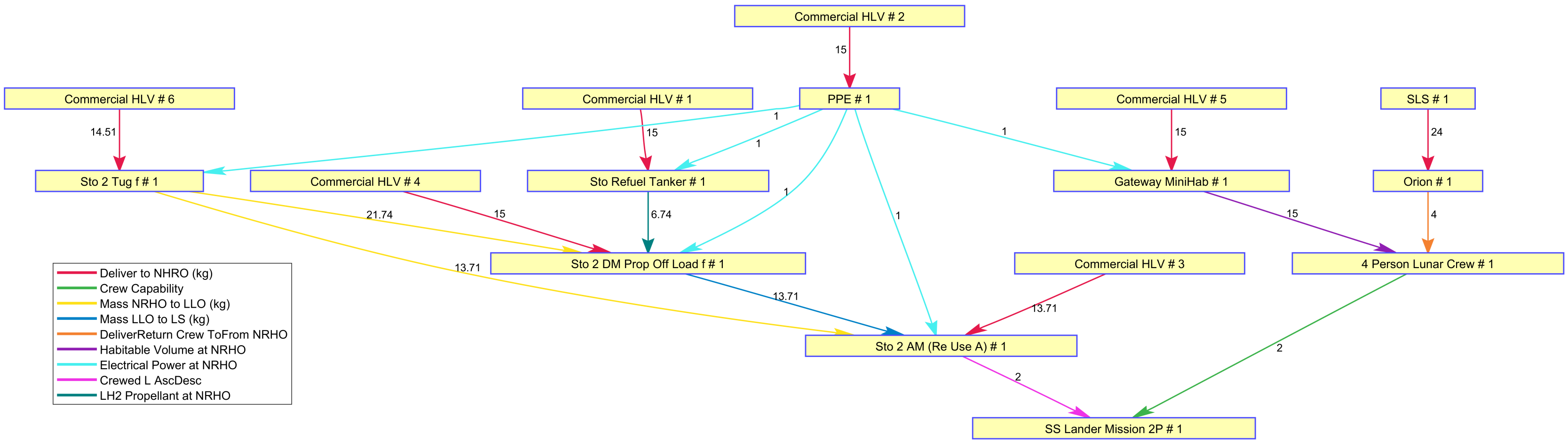


Fig. 7.11.: System interactions of Storable pump fed propulsion based single mission architecture with type of capability transfer denoted by color and value of capability transfer denoted with numbers

7.3.2 Multi Mission Short Stay Demonstration

The optimization was again applied for varying numbers of lunar lander missions to investigate trends in total cost. The year of first mission remains the same and scheduling of subsequent missions depends on SLS and Orion production as well as the ability to launch multiple refueling missions for certain propellant types. The trends in total cost vs number of missions is shown in Figure 7.12. For less than 2 total missions, the pressure fed storable architecture continues to dominate despite requiring a significant number of refueling tankers and commercial HLV. After 2 missions, the cost of refueling elements and their launch vehicles overcomes the cost of DDT&E of the pump fed storable architectures and they become more advantageous in terms of cost. The hydrogen based architecture, while efficient in terms of specific impulse, is initially costlier in terms of system development and production and negatively impacts the total cost for a low number of missions. Due to the power and mass penalties resulting from Cryogenic Fluid Management and insulation, resulting in increased system complexity, the benefits of the higher specific impulse are further reduced. At an architecture level, this manifests in a large up front DDT&E cost and propellant off-loading requiring refueling tankers. At some point between 5 and 6 missions, the the hydrogen based architecture becomes the cost competitive over the pressure fed storable architecture but not as competitive as either the methane or pump-fed storable architectures.

A stakeholder trade-off exists in that while the pressure-fed storable architecture does offer some initial cost savings and an early first mission, it is eventually less sustainable economically than the more advanced architectures. Furthermore, it lacks the ability to eventually leverage In-Situ Resource Utilization that a methane or hydrogen based lander would eventually be able to utilize.

Nonlinear trends can be seen in the depiction of cost versus landings in Fig. 5 as a result of integer numbers of selected systems. For some sets of missions, a refueling element could split capability across multiple missions, resulting in the behavior seen.

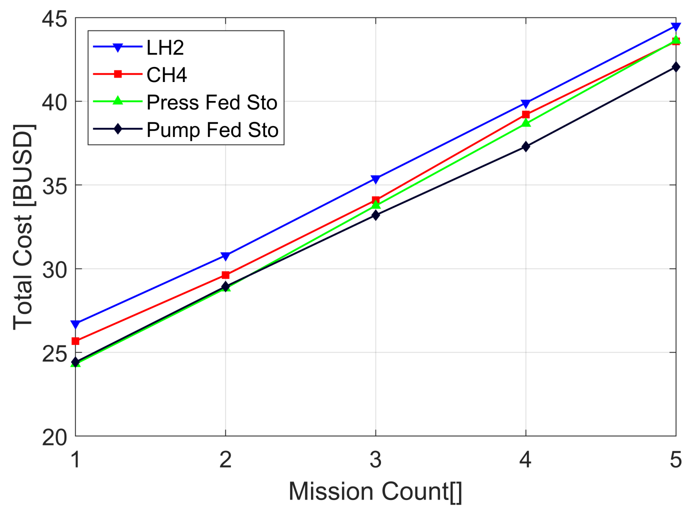


Fig. 7.12.: Total cost of architecture for various number of missions. The pressure-fed storable architecture has the lowest initial cost but the pump-fed storable architecture becomes competitive in total cost after a few missions.

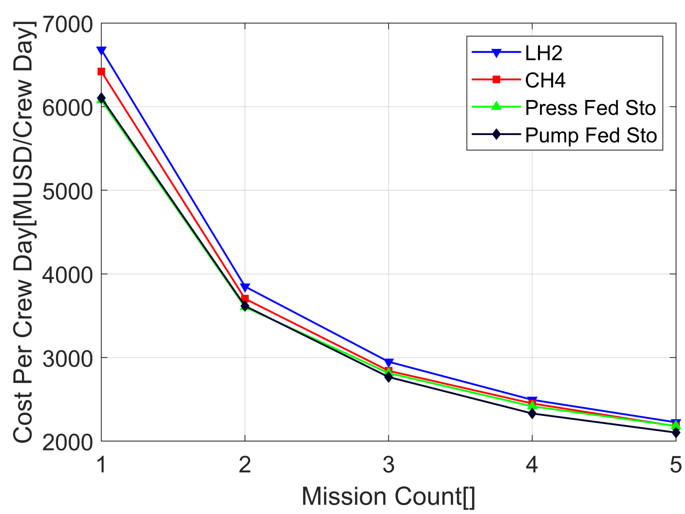


Fig. 7.13.: Cost per crew day of architecture for various number of missions. The pressure-fed storable architecture has the lowest initial cost but the pump-fed storable architecture becomes competitive in total cost after a few missions.

Table 7.8.: Stakeholder metrics and system allocation of a 5 surface mission study

	Architecture	Pump Sto.	Press. Sto.	Methane	LH2
	Total Cost[MUSD]	30,579	30,655	30,995	32,755
	First Mission	2025.0	2024.74	2025.34	2026.05
	# of Surface Missions	5	5	5	5
System Allocation	LH2 TE	0	0	0	5
	LH2 TE Prop Off Load	0	0	0	0
	Meth TE	0	0	5	0
	Meth TE Prop Off Load	0	0	0	0
	Pressure Sto TE	0	0	0	0
	Pressure Sto TE Prop Off Load	0	5	0	0
	Pump Sto TE	5	0	0	0
	Pump Sto TE Prop Off Load	0	0	0	0
	LH2 DM	0	0	0	0
	LH2 DM Prop Off Load	0	0	0	5
	Meth DM	0	0	0	0
	Meth DM Prop Off Load	0	0	5	0
	Pressure Sto DM	0	0	0	0
	Pressure Sto DM Prop Off Load	0	5	0	0
	Pump Sto DM	0	0	0	0
	Pump Sto DM Prop Off Load	5	0	0	0
	LH2 AM	0	0	0	1
	Meth AM	0	0	1	0
	Pressure Sto AM	0	1	0	0
	Pump Sto AM	1	0	0	0
	SLS	5	5	5	5
	Orion	5	5	5	5
	4 Person Crew	5	5	5	5
	SS Crewed Lander Mission	5	5	5	5
	Commercial HLV	18	20	17	16
	LH2 AM (Re Use)	0	0	0	4
	Meth AM (Re Use)	0	0	4	0
	Sto AM (Re Use)	0	4	0	0
	Sto 2 AM (Re Use)	4	0	0	0
	LH2 Refuel Tanker	0	0	0	4
	Meth Refuel Tanker	0	0	5	0
	Sto Refuel Tanker	6	8	0	0
	PPE	1	1	1	1
	Gateway MiniHab	1	1	1	1

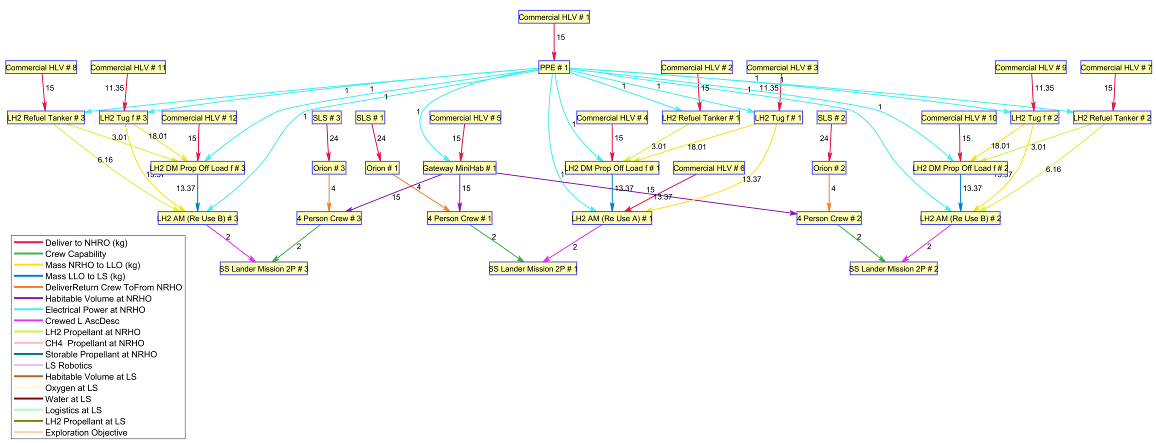


Fig. 7.14.: Example of resulting life cycle scheduling of LH2 Architecture 3 mission

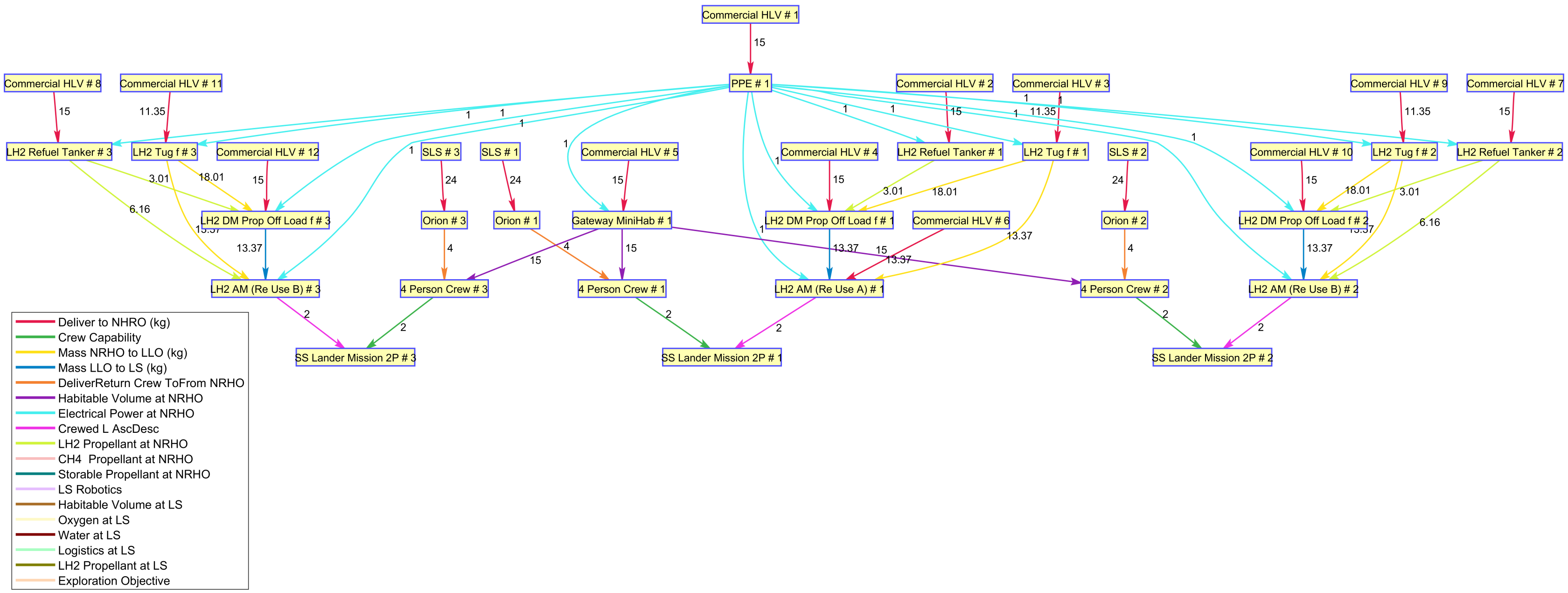


Fig. 7.15.: Example system to system capability transfer of a LH2 architecture with 3 crewed missions. The type of capability transfer is denoted by color and the value of capability transfer is denoted with numbers

7.4 Long Duration Surface Stay

Here we apply the PPO method to the CSL with the intention of assessing long duration missions. Key to these long duration missions is the use of a pre-deployed surface habitat, power systems, and In-Situ Resource Utilization to produce propellant. Given the investment required to sustain the development of these systems, the stakeholder may or may not wish to advance these systems.

The CSL has been modified to include certain long duration stay elements including a surface habitat, power generation, ISRU equipment, logistics containers, lander elements that allow surface refueling and long duration missions. The surface habitat is based on literature representing a minimal habitat and can support 2 person crews for 14 days. The ISRU setup was assumed to provide hydrogen and oxygen propellant based on water harvesting of the lunar surface [72]. While lunar methane ISRU is possible, it is more difficult on the moon and was excluded from this study. The power system was based on the nuclear Kilopower reactor [73]. Logistics containers supply consumables for 2 person crews based on consumption rates for water, oxygen, food, clothes and hygiene products as well as a packing efficiency factor representing the inert mass of the containers [74]. The lander elements for this study remain unchanged with the exception of the addition of an unfueled LH2 Ascent Module. Given that the unfueled LH2 AM is substantially lighter, the rest of the landing architecture will be able to decrease in size and thus cost.

Here we examine a Pareto trade-off between crew days spent on the surface and total mission cost for various numbers of long and short duration missions. The optimization is applied with the modified CSL to minimize total architecture cost with certain requirements for crew duration spent on the surface. This effectively provides a mathematical implementation to compare stakeholder desire for crew time spent on the surface versus total architecture cost. As required crew duration increases, the optimization must select either additional short stay missions or select systems that enable long duration missions.

Shown below in Figure 7.16 is the resulting Pareto frontier of total architecture cost and total days spent on the surface. The total cost and surface duration as well as system allocation are shown in Table 7.9. Points one-three on the Pareto frontier are short stay mission using a pressure-fed storable AM and TE as well as a pump-fed storable DM. It should be noted that this differs from the single propellant study in that this is a mix of two different engine types. The fourth point on the Pareto frontier is a long stay mission using the same lander elements but with a deep space habitat and power system to support long duration stays on the surface. The fifth Pareto point is a long stay mission with storable TE and DM but with a LH2 AM that is refueled with ISRU on the surface. There exists certain points along this architecture where major con-ops decisions are optimal. This analysis demonstrates the ability to conduct trade-off analysis and investigate return on investment from specific technologies. For stakeholders with tight total budgets, the short stay mission type for less than 3 missions is optimal. If the stakeholder has the budget to invest in technologies and is willing to delay the time of the first mission, then a long stay architecture is more optimal in terms of cost per crew day.

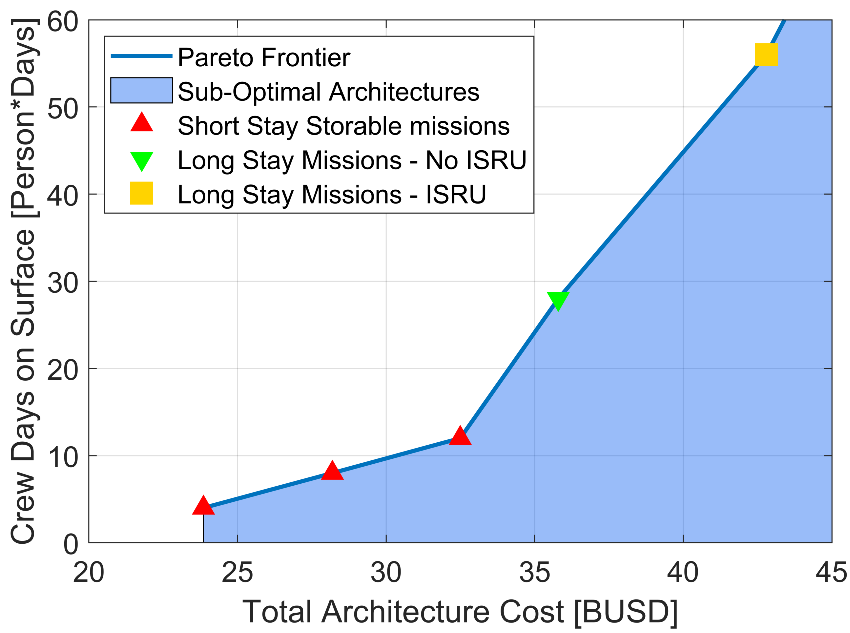


Fig. 7.16.: Pareto frontier of total mission architecture cost and crew duration on the Lunar surface.

Table 7.9.: Table of stakeholder values and system allocation from long stay Pareto frontier

Pareto Point	**1**	**2**	**3**	**4**	**5**
Total Cost	23.85	28.19	32.49	35.79	42.79
Schedule (year)	7.58	7.58	7.66	8.76	9.85
Crew Duration (days)	4	8	12	28	56
Cost Per Crew Day (BUSD/day)	5.96	3.52	2.71	1.28	0.76
LH2 Tug	0	0	0	0	0
LH2 Tug Prop Off Load	0	0	0	0	0
Meth Tug	0	0	0	0	0
Meth Tug Prop Off Load	0	0	0	0	0
Sto Tug	0	0	0	3	3
Sto Tug Prop Off Load	1	2	3	0	0
Sto 2 Tug	0	0	0	0	0
Sto 2 Tug Prop Off Load	0	0	0	0	0
LH2 DM	0	0	0	0	0
LH2 DM Prop Off Load	0	0	0	0	0
Meth DM	0	0	0	0	0
Meth DM Prop Off Load	0	0	0	0	0
Sto DM	0	0	0	0	0
Sto DM Prop Off Load	1	0	0	0	0
Sto 2 DM	0	0	0	0	0
Sto 2 DM Prop Off Load	0	2	3	3	3
LH2 AM (Re Use A)	0	0	0	0	0
Meth AM (Re Use A)	0	0	0	0	0
Sto AM (Re Use A)	1	2	0	1	0
Sto 2 AM (Re Use A)	0	0	3	0	0
SLS	1	2	3	1	2
Orion	1	2	3	1	2
4 Person Lunar Crew	1	2	3	1	2
SS Lander Mission 2P	1	2	3	0	0
LS Lander Mission 2P 15D	0	0	0	1	2
Commercial HLV	6	10	13	13	13
LH2 Refuel Tanker	0	0	0	0	0
Meth Refuel Tanker	0	0	0	0	0
Sto Refuel Tanker	1	3	4	2	2
PPE	1	1	1	1	1
Gateway MiniHab	1	1	1	1	1
Surface habitat	0	0	0	1	1
Logistics Container	0	0	0	0	0
LH2 Empty AM (Re Use A)	0	0	0	0	2
ISRU System	0	0	0	0	1
Reactor Power System	0	0	0	1	1

7.5 Discussion

While this study demonstrates the synthesizing ability of Programmatic Portfolio Optimization, several improvements in data and models would enhance the value to actual decision-makers in this domain.

First, the modified version of the AMCM method, while simple and effective for initial architecture studies, has two major drawbacks. First, the scaling can be very subjective in terms of assessing the systems technical complexity and program complexity. A small change in terms of the complexity ranking can propagate into a large cost or schedule difference. This issue can be reduced with more accurate understanding of system complexity, more granularity of scaling or by accounting for uncertainty as a stakeholder constraint or objective. Second, AMCM is based on traditional NASA procurement methods and does not encapsulate some of the benefits of the commercial procurement strategies utilized of today.

Lastly, this analysis assumed development for the lunar lander systems started in 2018. Many systems have not yet started development and certain commercial lander elements have already begun development. The goal of this study was to demonstrate a methodology, and by selecting a common year for start of development of the lander systems allowed a better comparison. Thus, given the differences in start dates, some propellant choices may be more or less optimal than demonstrated in this dissertation. However, this method could be readily adapted for the more detailed status quo with modifications to the CSL.

8. DEMONSTRATION OF METHODS - MARS SURFACE EXAMPLE SCENARIO

This study applies the PPO methodology to the design of a human Mars mission and highlights the scheduling(E1) and variable capability(E2) enhancements. Steps from Figure 4.4 were followed in that a functional analysis was conducted that identified key capabilities and systems required to accomplish a human landing. Unique system choices were identified from which the capabilities, requirements, technology dependencies and life-cycle cost/schedule components were assessed. The Programmatic Portfolio Optimization was then modeled using the resulting Candidate System Library and optimized with the commercial Gurobi solver. Optimal portfolios of systems are found for various stakeholder objectives with focus on total cost, earliest landing attempt and Cost Per Crew Day(CPCD) spent on mars.

The scenario posed here compares heavily with NASA's Design Reference Architecture 5(DRA) [32] and examines 6 of the suggested architecture trades. This scenario assumes a timeline that allows for multiple human landings on mars over the course of 15 years. An assumption was made for this case study that certain technologies such as Nuclear Thermal Rockets(NTR) and In-Situ-Resource-Utilization(ISRU) were not developed prior to the start of this mars campaign. The demonstration of results includes a a pareto tradeoff between cost and performance(crew days on Mars surface) as well as a pareto tradeoff between total cost and earliest landed mission.

8.1 Capabilities

A functional analysis resulted in a better understanding of interaction of the capabilities within the architecture. The resulting list of capabilities is defined in Table 8.1 and graphically represented in the capability map shown in Figure 8.1. A key theme of the capabilities for this architecture was the transfer of mass between locations and the availability of resources including propellant and crew consumables. A balance between capturing a high degree of detail and creating a simplified model that allows reasonable convergence times was captured in the list of capabilities.

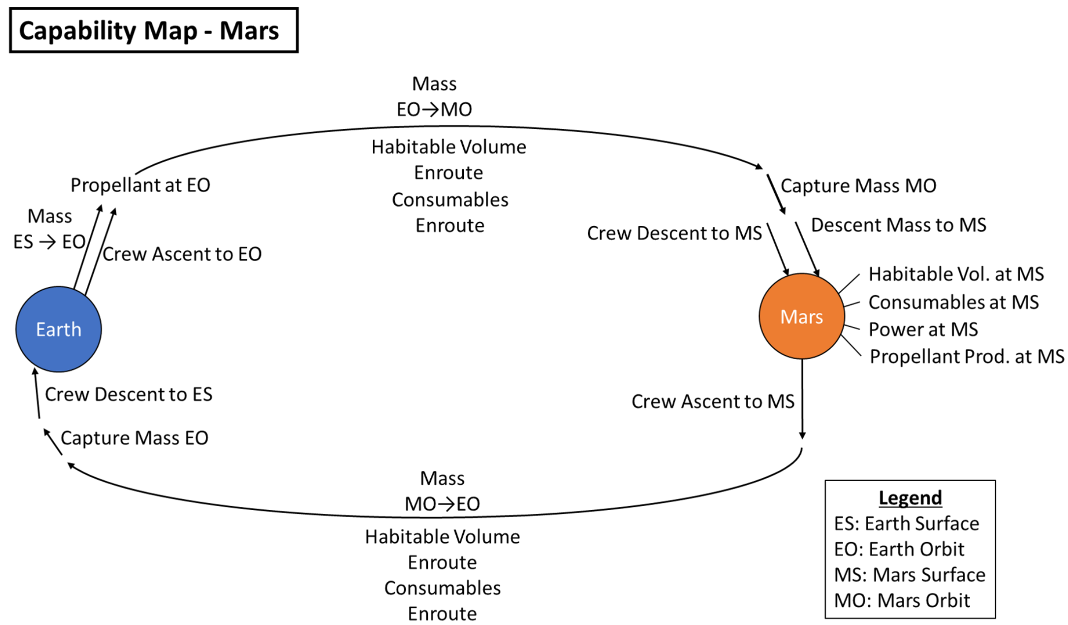


Fig. 8.1.: Capability map of Mars architecture

Table 8.1.: Mars architecture capability descriptions

Capability	**Description**
Exploration	A top level capability equal to the crew duration spent on the surface of Mars (days)
Crew Earth Ascent and Entry	Transfer and habitability of crew between Earth surface and Earth Orbit and vice versa(n)
Mass ES to EO	Mass delivered to Earth Orbit from Earth Surface (t)
Habitable Volume in Transit	Habitable volume in transit from Earth to Mars and back(m^3)
Habitable Volume at MS	Habitable volume on Mars Surface(m^3)
Mass Transfer EO to MO FT	Mass delivered to Mars Orbit from Earth Orbit via a Fast Transit trajectory (t)

continued on next page

Table 8.1.: *Mars architecture capability descriptions continued*

Capability	Description
Mass Transfer MO to EO FT	Mass delivered to Earth Orbit from Mars Orbit via a Fast Transit trajectory (t)
Mass Transfer EO to MO LE	Mass delivered to Mars Orbit from Earth Orbit via a Low Energy trajectory (t)
Mass Transfer MO to EO LE	Mass delivered to Earth Orbit from Mars Orbit via a Low Energy trajectory (t)
Capture Mass at MO FT	Orbit capture at Mars from a Fast Transit trajectory(t)
Capture Mass at MO LE	Orbit capture at Mars from a Low Energy trajectory(t)
Capture Mass at EO FT	Orbit capture at Earth from a Fast Transit trajectory(t)
Mars Crew Descent	Ability to deliver crew members to Mars surface from Mars Surface (n)
Mars Crew Ascent	Ability to deliver crew members to Mars Orbit from Mars Surface (n)
Deliver Mass MO to MS	Ability to deliver mass from Mars Orbit to Mars Surface (kg)
Habitable Volume in Transit	Habitable Crew Volume in Transit Orbit (m^3)
Habitable Volume MS	Habitable Crew Volume on Mars Surface (m^3)
LH2 Propellant at LEO	LH2 Propellant available in Earth Orbit (kg)
LH2 Propellant at MS	LH2 Propellant created on Mars Surface (kg)
Launch Window Capability	Ability to deliver a payload to Mars ()
Consumables at EO	Consumables required for crew duration at Earth Orbit
Consumables at MS	Consumables required for crew duration at Mars Surface

8.2 Candidate System Library

This example explores a full scale Human mission to Mars that is meant to directly compare to NASA's DRA 5 [32] and thus has a Candidate System Library that contains the similar component systems. The CSL was assembled for this study following the steps outlined in Figure 4.4. From the functional analysis, general classes of systems were identified from which several unique systems were identified.

The CSL includes many potential systems but is non-exhaustive. The majority of potential options were centered around the major decisions of DRA 5. The design tree is shown in Figure 8.2 and includes decisions based on: which mission trajectory class, which propulsion type, which Mars capture method, whether to predeploy cargo and assets to the Martian surface prior to crewed missions, and whether to use ISRU for propellant production. Six out of eight of the conjunction class missions are examined in this study. Similar to the DRA 5 analysis, an assumption was made in the system sizer to exclude systems with gross masses surpassing 800 t as this was deemed logistically challenging and would be subject to gravity losses. Thus, the chemical propulsive options with propulsive capture at Mars are excluded. The 6 cases are differentiated in their design decisions in Table 8.3.

Key to the formation of the CSL is to identify and quantify each systems capabilities, requirements, connectivity/compatibility issues, as well as life-cycle cost and schedule components for development, production and operation. Each of these characteristics has been estimated through the use of parametric sizing tools, subject matter experts, available literature, and historical relations. This list of systems with their attributes form the candidate system library from which combinations of systems can then be formed into feasible architectures.

An overview of the candidate systems used in this study is shown in Table 8.2 with their respective capability type, requirement type, and technology dependency. More detailed attribute values for capability, requirements, cost, and schedule components can be found in Tables C.1-C.3 in the appendix. Many systems capabilities and requirements are known and have fixed values. Other systems, like the transit propulsion systems are allowed to scale in terms of their capabilities and requirements and are sized within the optimization using the variable capability constraints. The last column of Table 8.2 details the method for sizing the systems based on their capabilities and requirements with the associated references. It should be noted that every effort was made to use publicly available information based on conference papers, journal articles, textbooks, and technical reports for the system sizing. A future user could readily use a proprietary or government sizing tool to replace these values.

Table 8.2.: Overview of Mars Candidate System Library

General Class	Unique System	Sub Categories	Capabilities	Requirements	Technology Dependency
Crew Transit Systems	• Deep Space Habitat	• n/a	• Habitable Volume in Transit	• Mass launched to Earth Orbit • Mass EO to MO FT • Mass MO to EO FT • Crew consumables at LEO (only re-use)	
	• Crew Exploration Vehicle (Orion CM)	• n/a	• Crew Earth ascent and entry	• Mass launched to Earth Orbit • Mass EO to MO FT • Mass MO to EO FT • Capture at Earth	
	• Logistics Module	• n/a	• Crew consumables at LEO	• Mass launched to Earth Orbit	
Launch Vehicles	• Government super heavy launch vehicle • Commercial heavy launch vehicle	• n/a	• Mass launched to Earth Orbit		
Transit Propulsion	• NTR Transit Propulsion • Chemical Transit Propulsion	• Fast transit trajectory (FT) • Low energy trajectory (LE) • Propulsive capture (PC) • Aero capture (AC)	• Mass EO to MO FT (if FT) • Mass MO to EO FT (if FT) • Mass EO to MO LE (if LE) • Mass MO to EO LE (if LE) • Capture at Mars Orbit FT (If PC and FT) • Capture at Mars Orbit LE (If PC and LE)	• Mass launched to Earth Orbit • Capture at Mars Orbit FT (If AC and FT) • Capture at Mars Orbit LE (If AC and LE) • Mars Launch Window	• NTR Development • Cryogenic Fluid Management • Advanced Aerocapture
Surface Systems	• Mars Surface habitat	n/a	• Habitable Volume Mars Surface (m^3)	• Mass launched to Earth Orbit • Mass EO to MO LE • Mass MO to EO LE • Mass MO to MS	
	• Power systems	n/a	• Power at MS	• Mass launched to Earth Orbit • Mass EO to MO LE • Mass MO to EO LE • Mass MO to MS	
	• In Situ Resource Utilization plant	n/a	• LH2 Propellant at MS	• Mass launched to Earth Orbit • Mass EO to MO LE • Mass MO to EO LE • Mass MO to MS • Power at MS	• ISRU development
	• Surface logistics module	n/a	• Crew Consumables at Mars Surface	• Mass launched to Earth Orbit • Mass EO to MO LE • Mass MO to EO LE • Mass MO to MS	
Mars Ascent Entry	• Cargo Lander • Crew Mars Descent Module • Crew Mars Ascent Module	• Launched with propellant • Launched without propellant		• Mass launched to Earth Orbit • Mass EO to MO LE • Mass MO to EO LE • Mass MO to MS	• Advanced Aerocapture development

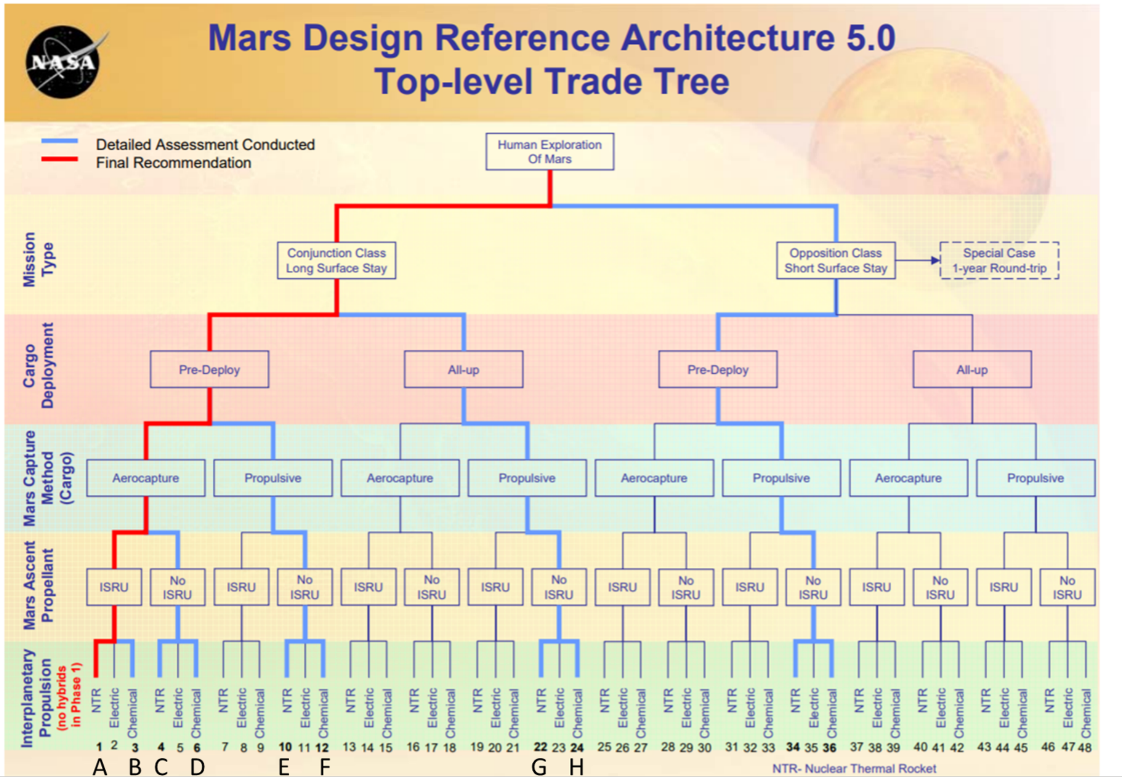


Fig. 8.2.: Design Reference Architecture 5.0 Trade Tree [32]

Table 8.3.: Design Reference Architecture 5.0 Trade Tree Options

ID	Propulsion	Deployment	Capture	ISRU
A	NTR	Pre	Aero	Yes
B	Chem	Pre	Aero	Yes
C	NTR	Pre	Aero	No ISRU
D	Chem	Pre	Aero	No ISRU
E	NTR	Pre	Propulsive	No ISRU
G	NTR	All Up	Propulsive	No ISRU

The relationships between the propulsion systems capabilities and requirements was estimated with a custom sizing tool that incorporated the rocket equation, several assumptions for the delta-V cost of various maneuvers, and properties of different propulsion types.

Both Nuclear Thermal Rocket propulsion and traditional liquid chemical rocket propulsion were assessed in this study. Delta-V differences in the sizing included whether the system was required to: a) conduct propulsive capture at mars, b) propulsive capture at earth, c) whether it was responsible form moving mass from Earth Orbit to Mars Orbit as well as Mars Orbit to Earth Orbit, and d) whether it used a Fast Transit(FT) trajectory or a Low Energy(LE) trajectory. Further details on assumptions and values are found in the appendix. From this sizing, relationships between capability(payload delivered to Mars) and the variable requirements(mass delivered to EO, propellant required at EO, and required capture mass) were produced. These relationships are detailed in Figures XXX-XXX in Appendix XXX

Estimation of other system capabilities, requirements, and cost/schedule components was a mixture of literature review, propulsive sizing and expert opinion. References for the values are detailed in the last column of Table 8.2. Cost and schedule components were again estimated using the AMCM approach detailed in section 4.5.1. Further details of the sizing for the respective systems can be found in Appendix Section C. A critical issue discovered was the difficulty posed by export control in regards to cost/schedule estimates. While there are more accurate tools to estimate cost and schedule than those demonstrated here, the AMCM method provides some a reasonable accurate estimate while being agnostic of proprietary or sensitive data.

Each system was examined as to whether it requires any technologies that were below a NASA Technology Readiness Level(TRL) 6 [68]. For each technology the cost and schedule required to raise the technology to TRL 6 was either estimated or determined through literature. Several methods exist to estimate cost and schedule components for these technologies, but are often proprietary and not available to the public. For instance the Technology Cost and Schedule Estimating Tool(TCASE) developed by NASA for this explicit purpose is tightly controlled and not available to the public [57]. Additionally, any paper using these values would not pass export control due to the sensitive nature of cost values. To illustrate the ability to include technology scheduling within the PPO method, estimated values for cost and schedule from literature are used instead.

8.3 Single Mission Comparison

Given the constraints as previously described and the Candidate System Library, the optimization problem can be formulated via the YALMIP package [61] and solved using the Gurobi Solver [62]. To compare the different DRA architecture cases, an optimization problem for each mission case is formulated by constraining the use of specific systems. For instance, for case A the use of NTR propulsion, ISRU, Aerocapture, and pre-deploy of assets are set as required systems within the constraints. For each case a single surface mission is required.

The high level results of the optimization are shown in Table 8.4. The chem based architectures offer the earliest possible mission opportunity due to their comparatively less complex propulsive technologies. The lowest cost option is case G, the NTR propulsive capture, non ISRU, and all-up architecture. This partly stems from requiring less development of required technologies like aerocapture and ISRU but will require substantially more propellant refueling than the other NTR missions. All-up architectures benefit from requiring the development of only a single type of transit propulsion(Fast Transit) where a pre-deploy is required to develop a cargo(Low Energy) and a crew variant(Fast Transit). This may not accurately portray reality as the differences between a cargo propulsion system and the crewed propulsion may be minor and focused only on the magnitude of velocity change required(Fast Transit vs Low Energy). Future improvement to the method could investigate this further.

The Pareto frontier of cost versus first landing attempt is shown in Figure 8.3. Given that schedule is partially dictated by launch windows there is only a small variability in the schedules with 3 main values. For instance, an architecture may be ready but could have to wait a year for the next available launch window. Schedule is heavily differentiated by the choice of propulsion(NTR vs Chem) as the development time of the NTR systems are much lengthier than the chemical propulsion systems. Lastly, the all-up vs pre-deploy trade allows some architecture like case G to land crew over 2 years(one launch opportunity) earlier than the similar pre-deploy architecture case E. The zone of sub-optimal portfolios contains many feasible architectures that are sub-optimal but are not plotted in the figure.

Table 8.4.: Comparison of stakeholder values resulting from optimization of DRA mission cases. Results include total cost and year of first mission.

ID	Propulsion	Deployment	Capture	ISRU	Total Cost	Year of First Mission
A	NTR	Pre	Aero	Yes	104.27	15.53
B	Chem	Pre	Aero	Yes	97.97	11.26
C	NTR	Pre	Aero	No ISRU	103.36	15.53
D	Chem	Pre	Aero	No ISRU	114.95	13.39
E	NTR	Pre	Propulsive	No ISRU	95.64	15.53
G	NTR	All Up	Propulsive	No ISRU	85.25	13.39

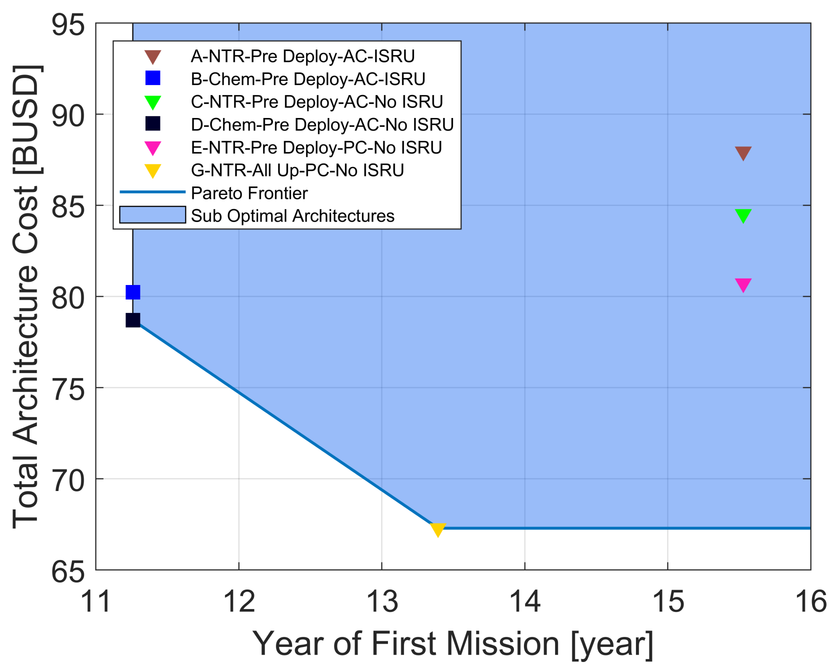


Fig. 8.3.: Pareto frontier of DRA mission cases - cost vs first landing. Variability in schedule of cases is dependent on technology development and launch window opportunities.

A graph of the system-to-system interactions of Case A is shown in Figure 8.4. A hierarchical flow of capability is seen in that launch vehicles lift a payload to earth orbit, the propulsion systems send that payload to Mars orbit, an aerocapture system captures the payload around Mars and lander elements deliver the payload to the surface. On the return

Table 8.5.: System selection results from optimization for DRA mission cases

ID	A	B	C	D	E	G
4 Person Crew Surface	1	1	1	1	1	1
Crewed Mars Surface Mission	1	1	1	1	1	1
Deep Space Habitat	1	1	1	1	1	1
Deep Space Habitat (Re Use)	0	0	0	0	0	0
Orion	1	1	1	1	1	1
Government Super Heavy Lift 1A	3	2	3	2	3	3
Commercial Heavy Lift	7	15	7	15	7	9
NTR TransProp 2W PC FT Var	0	0	0	0	1	0
NTR TransProp 2W PC LE Var	0	0	0	0	0	0
NTR TransProp 2W AC FT Var	1	0	1	0	0	0
NTR TransProp 2W AC LE Var	0	0	0	0	0	0
NTR TransProp 1W PC FT Var	0	0	0	0	0	2
NTR TransProp 1W PC LE Var	0	0	0	0	2	0
NTR TransProp 1W AC FT Var	0	0	0	0	0	0
NTR TransProp 1W AC LE Var	2	0	2	0	0	0
LH2 TransProp 2W PC FT Var	0	0	0	0	0	0
LH2 TransProp 2W PC LE Var	0	0	0	0	0	0
LH2 TransProp 2W AC FT Var	0	1	0	1	0	0
LH2 TransProp 2W AC LE Var	0	0	0	0	0	0
LH2 TransProp 1W PC FT Var	0	0	0	0	0	0
LH2 TransProp 1W PC LE Var	0	0	0	0	0	0
LH2 TransProp 1W AC FT Var	0	0	0	0	0	0
LH2 TransProp 1W AC LE Var	0	1	0	1	0	0
Cargo Descent Vehicle FT	0	0	0	2	2	2
Cargo Descent Vehicle LE	2	2	2	0	0	0
Crew Descent Vehicle LE	1	1	1	1	1	0
Crew Ascent Vehicle Full LE	0	0	0	1	1	0
Crew Ascent Vehicle Full FT	0	0	1	0	0	0
Crew Ascent Vehicle Empty	1	1	0	0	0	0
Crew Descent Vehicle FT	0	0	0	0	0	0
Crew Ascent Descent Vehicle	0	0	0	0	0	1
Surface Habitat LE	1	1	1	1	1	0
Surface Habitat FT	0	0	0	0	0	1
CFM Technology Development	0	0	0	0	0	0
NTR Technology Development	0	0	0	0	0	0
Launch Window	10	10	10	10	12	8
Aero Capture FT	1	1	1	1	0	0
Aero Capture LE	2	1	2	1	0	0
Adv Aerocapture Technology Development	0	0	0	0	0	0
LH2 Tanker	1	10	1	9	1	5
ISRU System	1	1	0	0	0	0
Mars Surface Power	1	1	0	0	0	0
Logistics Container MS FT	0	1	0	1	1	1
Logistics Container MS LE	1	0	1	0	0	0

to earth: crew are transported in the deep space habitat which is delivered by the propulsion system. The crew land in the Orion vehicle that they originally landed in. A gantt chart of the lifecycle phase scheduling of case A is shown in Figure 8.5. The relationship between technology development and system development is evident in the start and end times of the respective phases. Since Case A is a predeploy architecture, the surface habitat and ascent vehicle are sent on a Low Energy transit the preceding opportunity. Graphs and gantt charts of other missions cases can be found in the appendix.

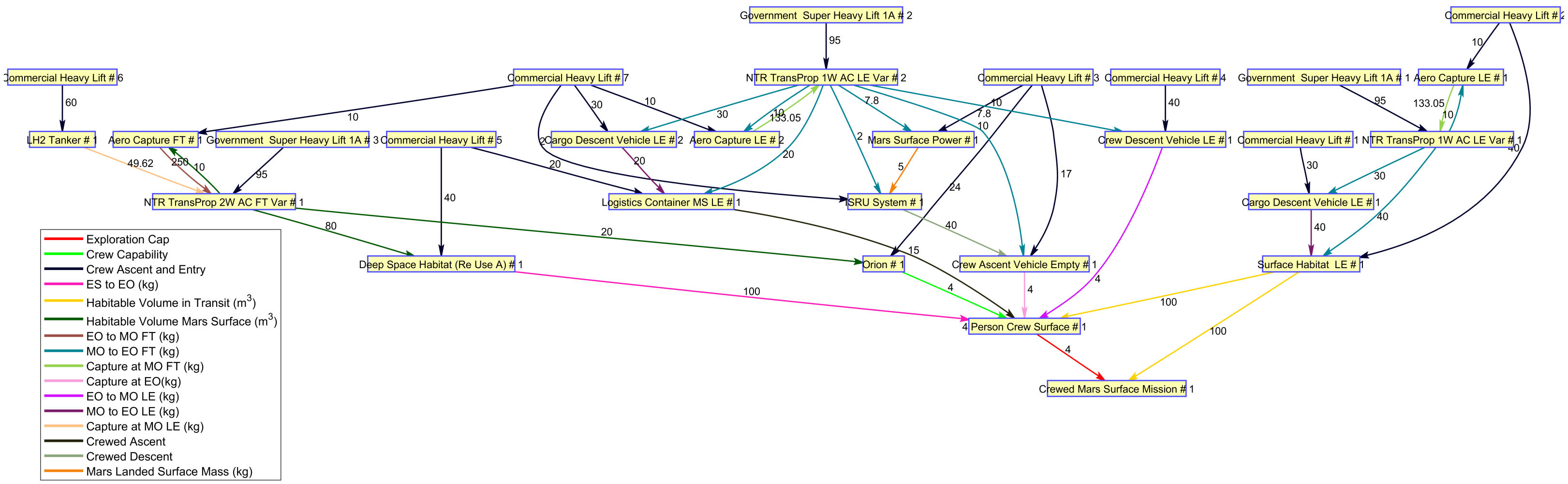


Fig. 8.4.: System-to-system interaction of single mission case A architecture. Capability transfer type is denoted with color and capability transfer value is represented by the numbers over the arrows. For instance, Government Super Heavy Lift 1A #2 provides 95 units of the ES to EO capability to the NTR TransProp 1W AC LE Var #2. AC = AeroCapture, LE = Low Energy,FT = Fast Transit, Var=Variable capability system.

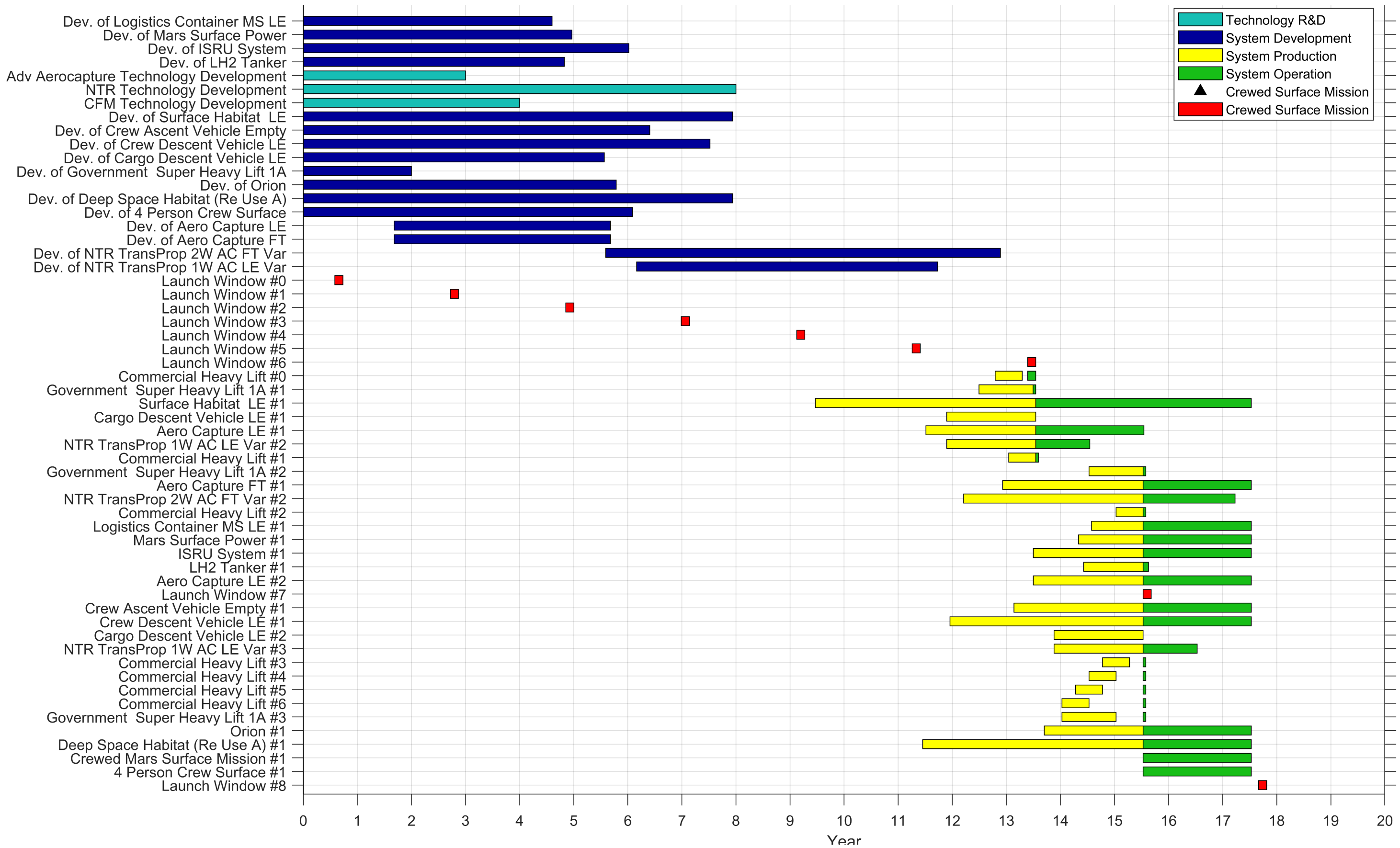


Fig. 8.5.: Life-cycle phase scheduling of single mission case A architecture. The first mission is dependent on the development and production of all required systems as well as the timing of launch windows. Pre-deploy of lunar surface systems(Surface habitat, power, ISRU and ascent vehicle) is delivered on the preceding launch window before crew arrival.

8.4 Multi Mission Study

The optimization was again tested for each DRA case with the requirement for progressively increasing number of missions. The Cost Per Crew Day for each mission can be calculated for each number of missions and each DRA case using the total architecture cost, crew size and expected surface duration(500 days). A plot of the trends of the is shown in Figure8.6 and 8.7. As can be seen, the lowest cost option for a single mission is not the lowest cost option for multiple missions and illustrates the trade-off between development costs and return on investment. For example, while the NTR missions(Excluding case G) are initially more expensive than their chemical counterparts for a single mission, the total cost of several NTR missions is lower than their corresponding chemical architectures for the same number of missions. It's clear that if the stakeholder made the system selection based on only one mission, they would fail to realize the cost savings of some of the high tech systems like NTR and aerocapture.

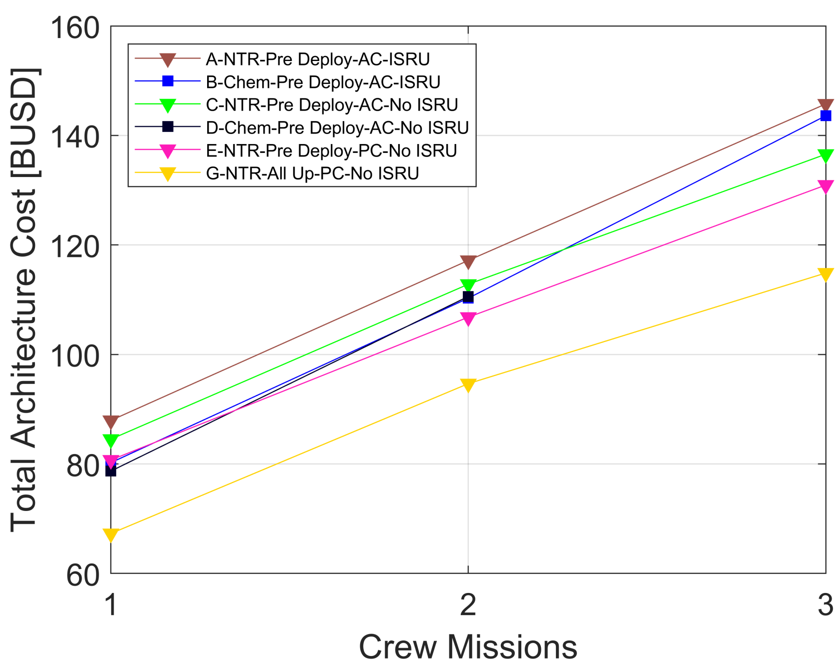


Fig. 8.6.: Cost vs Mission count

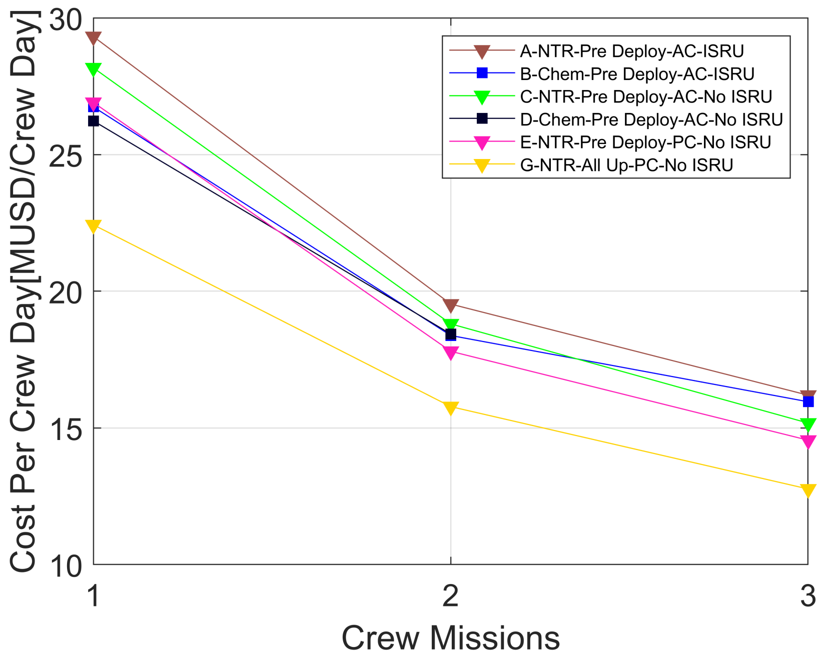


Fig. 8.7.: Cost per crew day vs Mission count

9. DEMONSTRATION OF METHODS - MOON TO MARS SURFACE EXAMPLE SCENARIO - PROGRESS AND REMAINING CHALLENGES

In this study, an example of the stepping stones approach applied to a Moon-to-Mars architecture is demonstrated. Highlighted in this example are the E1, E2, and E3 enhancements with particular emphasis on the time dependent objective E3 enhancement. An assumption is made that for the first 15 years the stakeholders want to maximize person-days spent on the Moon and for the second 15 years to maximize person-days on Mars.

The Candidate System Library for this study is simple the combination of the Lunar CSL and the Martian CSL. The Programmatic Portfolio Optimization was then modeled using the resulting CSL and optimized with the commercial Gurobi solver. Optimal portfolios of systems were sought for various stakeholder objectives with focus on minimizing total cost and maximizing surface duration on the Moon and Mars.

9.1 Candidate System Library

The Candidate System Library for this study is made up of systems for both lunar and Mars missions and utilizes the CSL of both the Moon and Mars studies shown in Tables 7.2 and 8.2. Of particular interest are the technologies that overlap between the two libraries as selecting common technologies is expected to reduce overall cost. The list of common technologies includes: Cryogenic Fluid Management, precision landing, nuclear power, and In-Situ Resource Utilization.

The PPO problem is formulated as described with the E1, E2 , and E3 enhancements implemented as constraints. An objective function is formed that maximizes the total number of days spent on the Moon, maximizes the total number of days spent on Mars, and minimizes total cost of the entire multi destination architecture. The Multi domain constraints are applied such that for the first 15 years only lunar missions are allowed and following that Mars missions are allowed for 15 years.

9.2 Difficulties and Further Work

Several issues were discovered in the stepping-stone study that make optimization cases of this size impractical and in some cases impossible. It was found that large problems of this size with all three enhancements take days to weeks to solve. The long run time of the stepping-stone problem precludes both the ability for extensive exploration of the design space where many sequenced iterations of the case may be solved with varying system designs, objectives, and constraints. It may preclude the adaption to a concurrent design team where decisions are iterated within hours.

This work was meant to demonstrate the Programmatic Portfolio Optimization method on an applicable Moon to Mars scenario. Given the size of the Lunar architecture design space as well as the Mars architecture design space, the resulting combined problem is relatively massive. Because X_{cij} scales in two dimensions with the number of systems, the resulting problem scales quadratically with the number of systems. In practice, given the CSL of both missions, the resulting problem takes days before reaching any reasonable convergence value and weeks to reach a Mixed Integer Programming bound gap of less than 1%.

This is impractical for a number of reasons. In the application of this method, it was useful to vary either the objective function, systems within the candidate system library, or architecture requirements to examine trade-offs within the design space. Week long run times make this sort of analysis impossible. Additionally, this method is best applied as an iterative process where initial inputs such as the CSL are examined and improved, which is hindered with long run times.

Several modifications were made to improve computational efficiency with the goal of applying the multi-domain enhancement in a practical solve time. Many parameters of the Gurobi solver were modified in an attempt to improve solving time and follow guidelines laid out by Klotz et al. [75]. The behavior of the solver can be characterized as a "lack of progress in the best bound" and as such modification to the solver parameters governing branching and cut generation were explored. A medium sized case with a 10 minute solve time was used as a test of parameter improvement. With no modifications, this case served as a control for comparison. Each parameter was modified, tested, and compared with the control for solver improvement as well verifying the optimal solution was found. Other than convergence thresholds, three parameters were found to have the maximum effectiveness: branch variable selection strategy, global cut generation control, and the high level solution strategy parameter. "Pseudo Shadow Price Branching" was found to be the most effective as the branch variable selection strategy. A "moderate" level of global cut control was found to be most effective to limit excessive cutting. Additionally, Gurobi has a parameter that allows the user to control the high-level solution strategy called "MIPFocus." Each value was tested for improvement with the best performance using a value of either 1("finding feasible solutions quickly") or 3("Improving a slow moving bound") [62]. Some differences in application scaling of the MIPFocus variable in that a value of 1 was more more effective with relatively small-medium sized problems where was a value of 3 was more effective with relatively larger problems. The combination of parameters led to an improvement of 20-30% as applied to the test case over the default settings. However, this improvement may not be the same for smaller or larger problems.

Every constraint and parameter of the PPO method was systematically examined with the goal of tightening the problem formulation as recommended by Klotz et al. [75]. This problem has several Big-M formulations in both the operational constraints as well as the schedule constraints. Big-M constraints are used to implement integer logic and can be cause significant issues for mixed integer solvers [76]. Several Big-M constraint formulations were improved using the Implies function within YALMIP [61] which can allow an even tighter bound on the Big-M formulation. In terms of values within the Big-M constraints, considerable care was given to selecting the absolute lowest value of M in order to tighten the problem formulation. For schedule constraints this involved setting an appropriate end time for the architecture and for operational constraints this involved determining the max value of every type of capability present within the CSL. Together, this led to the largest difference with an overall improvement of over 80% in solve time.

All of the results demonstrated in this paper result from applying the version 9.0.2 of the Gurobi commercial solver which is very efficient for small to medium PPO problems. Other solvers including CPLEX, the generic Matlab integer linear programming solver, and many others were investigated through either literature review or direct testing [77, 78]. Gurobi was found to offer the best performance. For larger PPO problems like the Moon to Mars scenario, the solution time is on the order of days to weeks. While Gurobi and computer processors will continue to improve, a custom solver utilizing column generation and an improved constraint pricing strategy may offer greater improvement. This would be a considerable and difficult task to implement. Both CPLEX and Gurobi have a history of substantial annualized improvements to the efficiency of their solvers and may eventually be able to solve this problem type ans scale in a reasonable amount of time.

10. CONCLUSIONS AND FUTURE WORK

10.1 Summary

The work presented in this dissertation demonstrates an approach to finding portfolios of systems that optimize stakeholder objectives like cost, schedule, robustness, and performance as well as forming feasible architectures by accounting for system-to-system interactions and life-cycle scheduling. The large number and significant variety of systems available for space exploration missions produce countless potential architecture combinations. Compounding this are the scheduling intricacies of system life-cycle phases, time dependent operational dependencies, as well as the uncertainty associated with each system and technology in terms of cost, schedule, and performance. Traditional architecting emphasizes the individual design of component systems over the wide-ranging and robust assessment of architecture options early in mission design. A top down method that can assess the capabilities, requirements, and risks associated with the diversity of available space systems and form optimal portfolios of interdependent systems is necessary.

Several stakeholder analysis needs were discovered that were not fulfilled by the status quo in literature. The bottom up approach that is prevalent in most architecture studies limits SoS managers from effectively comparing possible combinations of systems and technologies in terms of high level stakeholder objectives like cost, schedule, robustness, and performance. Robust Portfolio Optimization could be a method to make those comparisons. However, the current version of RPO lacks the ability to assess the schedule impacts of certain systems on the resulting architecture as well as the ability to optimally size systems within the optimization. Current space technology assessment methods focus on how directly related systems are impacted or examine a correlation with a stakeholder objective. The inclusion of technology dependencies within the Programmatic Portfolio Optimization formulation allows the comparison of technologies when they are integrated within the architecture.

The primary research thrusts produced several enhancements were made to the Robust Portfolio Optimization formulation to create the Programmatic Portfolio Optimization formulation with the goal of solving the identified literature gaps. The three main contributions from these enhancements addressed literature gaps.

1. Life-cycle Phase Scheduling - Scheduling was introduced by accounting for system life-cycle phases including development, production, and operation. New constraints relate the various phases to each other and account for the duration of each phase. The scheduling problem is integrated with the operational realm through constraint logic. Finally, technology dependency dependencies were added to account for low Technology Readiness Level of certain required hardware.

2. Variable Capability sizing - Previous versions of Robust Portfolio Optimization required fixed values for the capabilities of a system prior to optimization. In certain scope areas such as space systems, there often exists a chain of system dependency where a small change in a single system has a cascading effect throughout the architecture. Allowing systems to be sized within the optimization allows for a more optimal architecture and more accurate assessment of system options.

3. Multi Domain Optimization - The ability to focus on different periods of time allows for optimization of evolving stakeholder interest. This is particularly useful for a Moon to Mars type architecture where mission objectives change with time. This enhancement allows the assessment of technology value over different time domains.

Several examples illustrate the application of the methods to different scenarios. The examples in Chapter 5 are simplistic scenarios that focused on demonstrating the three enhancements at the most basic level. The Candidate System Library for these was composed of fictitious systems with minimal complexity to help demonstrate the principles.

The second example explored an architecture with a focused on using a lunar orbital station for crew missions and demonstrated the scheduling enhancement along with some of the original features of Robust Portfolio Optimization. This example highlighted the scheduling enhancement by comparing different propulsion options for reaching the lunar station and the resulting impact on mission schedule, stakeholder value, and total architecture cost. Additionally, two adaptations of the robustness metric were examined in terms of operational and financial robustness.

The third example examined a Lunar surface architecture using the lunar station and crewed lander elements. This example highlighted both the scheduling enhancement as well as the variable capability enhancement. Different options for lander elements were explored and examined in terms of cost and schedule. Additionally, trends in sustainability and return on investment of technologies were explored.

The fourth example examined a Mars surface architecture using the lunar station and compares the recommended cases from NASA's Design Reference Architecture 5. This example highlighted both the scheduling enhancement as well as the variable capability enhancement. Architecture cases were explored in terms of cost and schedule. The final recommendation from the DRA 5 assessment was neither the optimal choice for cost nor schedule for a single mission. Trends in cost were explored for multiple missions and the return on investment from technologies like Nuclear Thermal Rockets and ISRU was examined.

The last example explored a Moon to Mars campaign and attempted to highlight the scheduling adaptation and the multi domain optimization enhancement. Technologies and systems were identified that benefit both the Lunar and Martian architectures and included in the Candidate System Library. A limitation was discovered in that applications of the PPO to problem of this scale results in solution times on the order of days to weeks. A number of remedies were attempted with limited success. A number of future work areas was identified that could improve the efficiency of the model and reduction of solve time.

10.2 Further Work

This work was meant to demonstrate the Programmatic Portfolio Optimization method. Several areas to further advance the method were identified.

- <u>More Accurate Assessment of System Properties</u> - Every effort was made to characterize each system as accurately as possible in terms of capability, requirements, cost, and schedule. However, without access to an experienced design team or proprietary sizing tools, many estimates and assumptions had to be made. Since the data used for many systems is representative, but not authoritative, the results are examined from the perspective of evaluating efficacy of the method. If this method is applied at a decision making level, more accurate tools and methods should be used to characterize the various system options. For instance, NASA has a tool and database called Technology Cost and Schedule Estimation(TCASE) to document the cost and schedule components of potential technologies. This database could easily be used as an input to PPO but is not available to the academic community. Additionally, tools like the Beyond LEO Architecture Sizing Tool(BLAST) were used for some but not all of the sizing in this demonstration. Application of BLAST or similar tools would improve the accuracy of system sizing.

- More Efficient Solver - All of the results demonstrated in this dissertation result from applying the Gurobi commercial solver which is very efficient for small to medium PPO problems. For larger PPO problems like the moon to mars scenario, the solution time is on the order of days to weeks. This may be impractical for some mission planning teams. While Gurobi and computer processors will continue to improve, a custom solver utilizing column generation and a constraint pricing strategy may offer greater improvement.

- Alternate Systems and Concepts - The goal of this study was not to discover the best architecture but to demonstrate a method that could be used to do so. In terms of the Martian architectures, several concepts including electric propulsion, hybrid propulsion, several forms of mars ascent vehicles, and several forms of Entry Descent and Landing were not explored. In terms of the Lunar architecture, the rapid pace of the Artemis Mission limits some of practical design space given that funding and development has already been allocated. However, assessment of other system options, especially surface systems, can still be assessed and may highlight a beneficial strategy.

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APPENDICES

A. APPENDIX A LUNAR ORBIT APPENDIX

This appendix includes relevant details for the Lunar Orbit scenario(Chapter 6).

Table A.1.: Phased life cycle cost and schedule estimates for each potential system.

System	Estimaton Method	Dry Mass [kg]	Difficulty Inputs						Gen.	Resulting Difficulty	DDT&E Cost [MUSD16]	DDT&E Duration [years]	Prod. Cost [MUSD16]	Prod. Duration [years]
			Technical Complexity ▼	Planned years of Operation	Risk Tolerance ▼	Human Rated ▼	Program Complexity ▼	System Type ▼						
4 Pers. Crew(30 Day)	Hist. Estimate	0	High	10	Medium	Yes	High	Lunar Orbit	1	1.72	100	1	50	1
4 Pers. Crew(60 Day	Hist. Estimate										100	1	50	1
4 Pers. Crew(90 Day)	Hist. Estimate										100	1	50	1
4 Pers. Crew(120 Day)	Hist. Estimate										100	1	50	1
Deep Space Habitat 1	AMCM (Rolley)	8000	Medium	10	Medium	Yes	Low	Lunar Orbit	2	0.22	2053	6	411	2
Deep Space Habitat 2	AMCM (Rolley)	15000	High	10	Medium	Yes	Medium	Lunar Orbit	1	1.22	6263	7	1253	3
Deep Space Habitat 3	AMCM (Rolley)	5000	High	10	Medium	Yes	Medium	Lunar Orbit	1	1.22	3033	7	607	3
Orion	AMCM (Rolley)	23000	High	1	Low	Yes	High	Lunar Orbit	1	1.57	9700	8	1940	4
Commercial HLV	Market										0	150	150	1
Commercial SHLV	Market										1000	200	400	1
Commercial MLV	Market										0	100	100	1
Government SHLV B	Gov. Estimate										15000	1200	800	1
Government SHLV A	Gov. Estimate										10000	1000	500	1
Logistics Module 1	AMCM (Rolley)	3500	Low	1	Medium	Yes	Low	Lunar Orbit	1	-0.88	930	5	186	1
Logistics Module 2	AMCM (Rolley)	5300	Low	1	Medium	Yes	Low	Lunar Orbit	1	-0.88	1222	5	244	1
Logistics Module 3	AMCM (Rolley)	10000	Low	1	Medium	Yes	Low	Lunar Orbit	1	-0.88	1859	5	372	1
Logistics Module 4	AMCM (Rolley)	16500	Low	1	Medium	Yes	Low	Lunar Orbit	1	-0.88	2587	5	517	1
PPE	AMCM (Rolley)	10000	Medium	5	Medium	Yes	Medium	Lunar Orbit	1	0.39	3291	6	658	2
Science Airlock	AMCM (Rolley)	200	Low	10	Medium	Yes	Medium	Lunar Orbit	3	0.22	155	6	31	2
Crew Airlock	AMCM (Rolley)	500	Medium	10	Medium	Yes	Medium	Lunar Orbit	3	0.72	357	7	71	3
Robotic Arm	AMCM (Rolley)	200	Low	10	Medium	Yes	Medium	Lunar Orbit	3	0.22	155	6	31	2
Prop Storage	AMCM (Rolley)	200	Low	5	Medium	Yes	Low	Lunar Orbit	2	-0.61	124	5	25	1
Lunar Sci Payload	AMCM (Rolley)	200	High	4	Medium	Yes	Medium	Lunar Orbit	1	0.82	303	7	61	3
SEP Prop. System	AMCM (Rolley)	20000	Medium	1	Medium	No	Low	Lunar Orbit	1	-1.38	2344	4	469	1
NTR Prop. System	AMCM (Rolley)	7600	High	1	Low	Yes	High	Lunar Orbit	1	1.57	4671	8	934	4
Chemical Prop. System	AMCM (Rolley)	4000	Low	1	Medium	Yes	Low	Lunar Orbit	3	-0.88	684	5	137	1
Science Lander	AMCM (Rolley)	1000	High	1	High	No	Medium	Lunar Entry	2	-0.93	337	5	67	1
Mission Control Ctr.	Hist. Estimate										100	2	100	2

Table A.2.: Capabilities section of example Candidate System Library as used in calculation of preliminary results.

Systems	Capabilities											
	Exploration Cap.	Crew Earth Surface to Lunar Orbit	Habitable Volume Lunar Orbit (m^3)	Power (kWe)	Deliver to LEO (kg)	Deliver to LO (kg)	Return Crew To Earth	Consumables (kg)	Lab Access	Landed Ability	MCC Control	Return Samples to Earth
4 Pers. Crew(30 Day)	120	0	0	0	0	0	0	0	0	0	0	0
4 Pers. Crew(60 Day	240	0	0	0	0	0	0	0	0	0	0	0
4 Pers. Crew(90 Day)	360	0	0	0	0	0	0	0	0	0	0	0
4 Pers. Crew(120 Day)	480	0	0	0	0	0	0	0	0	0	0	0
Deep Space Habitat 1	0	0	60	0	0	0	0	1200	0	0	0	0
Deep Space Habitat 2	0	0	80	0	0	0	0	1800	0	0	0	0
Deep Space Habitat 3	0	0	20	0	0	0	0	2400	0	0	0	0
Orion	0	4	0	0	0	0	4	960	0	0	0	100
Commercial HLV	0	0	0	0	30000	0	0	0	0	0	0	0
Commercial SHLV	0	0	0	0	63000	0	0	0	0	0	0	0
Commercial MLV	0	0	0	0	23000	0	0	0	0	0	0	0
Government SHLV B	0	0	0	0	130000	0	0	0	0	0	0	0
Government SHLV A	0	0	0	0	95000	0	0	0	0	0	0	0
Logistics Module 1	0	0	0	0	0	0	0	2000	0	0	0	0
Logistics Module 2	0	0	0	0	0	0	0	3200	0	0	0	0
Logistics Module 3	0	0	0	0	0	0	0	6000	0	0	0	0
Logistics Module 4	0	0	0	0	0	0	0	5200	0	0	0	0
PPE	0	0	0	50	0	0	0	0	0	0	0	0
Science Airlock	0	0	0	0	0	0	0	0	5	0	0	0
Crew Airlock	0	0	0	0	0	0	0	0	0	0	0	0
Robotic Arm	0	0	0	0	0	0	0	0	0	0	0	0
Prop Storage	0	0	0	0	0	0	0	0	0	0	0	0
Lunar Sci Payload	0	0	0	0	0	0	0	0	0	0	0	0
SEP Prop. System	0	0	0	0	0	15000	0	0	0	0	0	0
NTR Prop. System	0	0	0	0	0	25000	0	0	0	0	0	0
Chemical Prop. System	0	0	0	0	0	25000	0	0	0	0	0	0
Science Lander	0	0	0	0	0	0	0	0	0	50	0	0
Mission Control Ctr.	0	0	0	0	0	0	0	0	0	0	20	0

Table A.3.: Requirements section of example Candidate System Library as used in calculation of preliminary results.

Systems	Requirements												
	Exploration Cap.	Crew Earth Surface to Lunar Orbit	Habitable Volume Lunar Orbit (m^3)	Power (kWe)	Deliver to LEO (kg)	Deliver to LO (kg)	Return Crew To Earth	Consumables (kg)	Lab Access	Landed Ability	MCC Control	Return Samples to Earth
4 Pers. Crew(30 Day)	0	4	50	0	0	0	4	960	0	0	1	0
4 Pers. Crew(60 Day	0	4	63	0	0	0	4	1920	0	0	1	0
4 Pers. Crew(90 Day)	0	4	71	0	0	0	4	2880	0	0	1	0
4 Pers. Crew(120 Day)	0	4	76	0	0	0	4	3840	0	0	1	0
Deep Space Habitat 1	0	0	0	20	10000	10000	0	0	0	0	1	0
Deep Space Habitat 2	0	0	0	25	15000	15000	0	0	0	0	1	0
Deep Space Habitat 3	0	0	0	10	5000	5000	0	0	0	0	1	0
Orion	0	0	0	0	23000	23000	0	0	0	0	1	0
Commercial HLV	0	0	0	0	0	0	0	0	0	0	1	0
Commercial SHLV	0	0	0	0	0	0	0	0	0	0	1	0
Commercial MLV	0	0	0	0	0	0	0	0	0	0	1	0
Government SHLV B	0	0	0	0	0	0	0	0	0	0	1	0
Government SHLV A	0	0	0	0	0	0	0	0	0	0	1	0
Logistics Module 1	0	0	0	4	3500	3500	0	0	0	0	1	0
Logistics Module 2	0	0	0	4	5300	5300	0	0	0	0	1	0
Logistics Module 3	0	0	0	4	10000	10000	0	0	0	0	1	0
Logistics Module 4	0	0	0	4	16500	16500	0	0	0	0	1	0
PPE	0	0	0	0	10000	10000	0	0	0	0	1	0
Science Airlock	0	0	0	1	200	200	0	0	0	0	1	0
Crew Airlock	0	0	0	1	500	500	0	0	0	0	1	0
Robotic Arm	0	0	0	1	200	200	0	0	0	0	1	0
Prop Storage	0	0	0	1	200	200	0	0	0	0	1	0
Lunar Sci Payload	0	0	0	1	200	200	0	0	0	0	1	0
SEP Prop. System	0	0	0	0	30000	0	0	0	0	0	1	0
NTR Prop. System	0	0	0	0	50000	0	0	0	0	0	1	0
Chemical Prop. System	0	0	0	0	70000	0	0	0	0	0	1	0
Science Lander	0	0	0	0	2000	2000	0	0	1	0	1	50
Mission Control Ctr.	0	0	0	0	0	0	0	0	0	0	0	0

B. APPENDIX B LUNAR SURFACE APPENDIX

This appendix includes relevant details for the Lunar surface scenario(Chapter 7).

Table B.1.: Advanced Mission Cost Model inputs and Phased life cycle cost and schedule estimates for each potential system and technology.

System	Estimaton Method	Dry Mass (kg)	Difficulty Inputs						Generation	Resulting Difficulty	DDT&E Cost [MUSD16]	DDT&E Duration [years]	Production Cost [MUSD]	Production Duration [years]	Operations Cost	AMCM Coef.	VarC AMCM Inert Mass m	VarC AMCM Inert Mass b
			Technical Complexity ▼	Plan yrs of Oper	Risk Tolerance ▼	Human Rated ▼	Program Complexity ▼	System Type ▼										
LH2 Tug	AMCM (Rolley)	4529.9	0.5	1	Medium	Yes	Medium-High	Lunar Orbit	1	0.3722	2613.01	6.3867	522.6029	2.3691	309.8761	5.9861	0.0466	3066.4528
LH2 Tug Prop Off Load	AMCM (Rolley)	4529.9	0.5	1	Medium	Yes	Medium-High	Lunar Orbit	1	0.3722	2613.01	6.3867	522.6029	2.3691	309.8761	5.9861	0.0466	3066.4528
Meth Tug	AMCM (Rolley)	4014.8	0.5	1	Medium	Yes	Medium-High	Lunar Orbit	1	0.3722	2613.01	6.3867	522.6029	2.3691	286.1497	5.9861	0.0490	2411.4406
Meth Tug Prop Off Load	AMCM (Rolley)	4014.8	0.5	1	Medium	Yes	Medium-High	Lunar Orbit	1	0.3722	2613.01	6.3867	522.6029	2.3691	286.1497	5.9861	0.0490	2411.4406
Sto Tug	AMCM (Rolley)	4372.1	0.4	1	Medium	Yes	Medium	Lunar Orbit	1	0.0222	2231.40	5.9667	446.2795	1.9836	258.5011	5.1118	0.0550	2278.2855
Sto Tug Prop Off Load	AMCM (Rolley)	4372.1	0.4	1	Medium	Yes	Medium	Lunar Orbit	1	0.0222	2231.40	5.9667	446.2795	1.9836	258.5011	5.1118	0.0550	2278.2855
Sto 2 Tug	AMCM (Rolley)	4143.9	0.47	1	Medium	Yes	Medium	Lunar Orbit	1	0.0922	2302.98	6.0507	460.5957	2.0581	257.5178	5.2758	0.0527	2274.5806
Sto 2 Tug Prop Off Load	AMCM (Rolley)	4143.9	0.47	1	Medium	Yes	Medium	Lunar Orbit	1	0.0922	2302.98	6.0507	460.5957	2.0581	257.5178	5.2758	0.0527	2274.5806
LH2 DM	AMCM (Rolley)	5989.9	0.6	1	Medium	Yes	Medium-High	Lunar Surface	1	0.4722	2503.83	6.5067	500.7663	2.4852	357.0528	5.7359	0.1252	4315.9529
LH2 DM Prop Off Load	AMCM (Rolley)	5989.9	0.6	1	Medium	Yes	Medium-High	Lunar Surface	1	0.4722	2503.83	6.5067	500.7663	2.4852	357.0528	5.7359	0.1252	4315.9529
Meth DM	AMCM (Rolley)	4753.2	0.6	1	Medium	Yes	Medium-High	Lunar Surface	1	0.4722	2503.83	6.5067	500.7663	2.4852	306.5129	5.7359	0.1166	3221.3300
Meth DM Prop Off Load	AMCM (Rolley)	4753.2	0.6	1	Medium	Yes	Medium-High	Lunar Surface	1	0.4722	2503.83	6.5067	500.7663	2.4852	306.5129	5.7359	0.1166	3221.3300
Sto DM	AMCM (Rolley)	5185.9	0.5	1	Medium	Yes	Medium	Lunar Surface	1	0.1222	2138.16	6.0867	427.6320	2.0904	277.2406	4.8982	0.1400	3178.7757
Sto DM Prop Off Load	AMCM (Rolley)	5185.9	0.5	1	Medium	Yes	Medium	Lunar Surface	1	0.1222	2138.16	6.0867	427.6320	2.0904	277.2406	4.8982	0.1400	3178.7757
Sto 2 DM	AMCM (Rolley)	4970.4	0.57	1	Medium	Yes	Medium	Lunar Surface	1	0.1922	2206.75	6.1707	441.3501	2.1668	278.2295	5.0554	0.1321	3158.5900
Sto 2 DM Prop Off Load	AMCM (Rolley)	4970.4	0.57	1	Medium	Yes	Medium	Lunar Surface	1	0.1922	2206.75	6.1707	441.3501	2.1668	278.2295	5.0554	0.1321	3158.5900
LH2 AM (Re Use A)	AMCM (Rolley)	7110.1	0.8	10	Low	Yes	Medium-High	Lunar Surface	1	1.7167	3504.53	8.0000	700.9052	4.1507	700.9052	0	0	0
LH2 AM (Re Use B)	AMCM (Rolley)	7110.1	0.8	10	Low	Yes	Medium-High	Lunar Surface	1	1.7167	4380.66	8.0000	876.1315	4.1507	876.1315	0	0	0
Meth AM (Re Use A)	AMCM (Rolley)	5972.6	0.8	10	Low	Yes	Medium-High	Lunar Surface	1	1.7167	3123.63	8.0000	624.7254	4.1507	624.7254	0	0	0
Meth AM (Re Use B)	AMCM (Rolley)	5972.6	0.8	10	Low	Yes	Medium-High	Lunar Surface	1	1.7167	3123.63	8.0000	624.7254	4.1507	624.7254	0	0	0
Sto AM (Re Use A)	AMCM (Rolley)	6015.6	0.7	10	Low	Yes	Medium	Lunar Surface	1	1.3667	2680.10	7.5800	536.0195	3.6409	536.0195	0	0	0
Sto AM (Re Use B)	AMCM (Rolley)	6015.6	0.7	10	Low	Yes	Medium	Lunar Surface	1	1.3667	2680.10	7.5800	536.0195	3.6409	536.0195	0	0	0
Sto 2 AM (Re Use A)	AMCM (Rolley)	6015.6	0.77	10	Low	Yes	Medium	Lunar Surface	1	1.4367	2766.07	7.6640	553.2145	3.7403	553.2145	0	0	0
Sto 2 AM (Re Use B)	AMCM (Rolley)	6015.6	0.77	10	Low	Yes	Medium	Lunar Surface	1	1.4367	2766.07	7.6640	553.2145	3.7403	553.2145	0	0	0
SLS	Literature	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
Orion	Literature	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
4 Person Lunar Crew	Estimate	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
SS Lander Mission 2P	AMCM (Rolley)	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
LS Lander Mission 2P 15D	AMCM (Rolley)	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
Commercial HLV	Market Rate	10000	Low (0)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	2271.14	5.4200	454.2279	1.5302	454.2279	0	0	0
LH2 Refuel Tanker	AMCM (Rolley)	5900	Medium (0.5)	5	Medium	No	Medium-High	Lunar Orbit	1	-0.3611	1325.09	5.5067	265.0174	1.5985	265.0174	0	0	0
Meth Refuel Tanker	AMCM (Rolley)	5250	Medium (0.5)	5	Medium	No	Medium-High	Lunar Orbit	1	-0.3611	1226.84	5.5067	245.3676	1.5985	245.3676	0	0	0
Sto Refuel Tanker	AMCM (Rolley)	4990	Low-Medium (0.25)	5	Medium	No	Medium-High	Lunar Orbit	1	-0.6111	1059.87	5.2067	211.9745	1.3682	211.9745	0	0	0
PPE	AMCM (Rolley)	7500	Medium (0.5)	1	Medium	Yes	High	Lunar Orbit	2	0.6222	1884.89	6.6867	376.9783	2.6643	376.9783	0	0	0
Gateway MiniHab	AMCM (Rolley)	5000	Medium (0.5)	1	Medium	Yes	High	Lunar Orbit	2	0.6222	1442.34	6.6867	288.4670	2.6643	288.4670	0	0	0
Surface habitat	AMCM (Rolley)	12000	Medium-High (0.75)	15	Low	Yes	Medium-High	Lunar Surface	1	2.0000	5625.43	8.3400	1125.0863	4.5870	1125.0863	0	0	0
Logistics Container	AMCM (Rolley)	150	Medium (0.5)	1	Medium	No	High	Lunar Orbit	2	-0.3778	90.80	5.4867	18.1592	1.5826	18.1592	0	0	0
LH2 Empty AM (Re Use A)	AMCM (Rolley)	7110.1	0.8	10	Low	Yes	High	Lunar Surface	1	1.9667	3922.87	8.3000	784.5745	4.5346	784.5745	0	0	0
LH2 Empty AM (Re Use B)	AMCM (Rolley)	7110.1	0.8	10	Low	Yes	High	Lunar Surface	1	1.9667	3922.87	8.3000	784.5745	4.5346	784.5745	0	0	0
ISRU System	AMCM (Rolley)	5000	Medium (0.5)	15	Medium	No	Medium	Lunar Surface	1	0.0556	1313.11	6.0067	262.6216	2.0189	262.6216	0	0	0
Reactor Power System	AMCM (Rolley)	6000	Medium-High (0.75)	1	Medium	No	Medium	Lunar Surface	1	-0.6278	1088.17	5.1867	217.6333	1.3534	217.6333	0	0	0
CFM Technology Development	AMCM (Rolley)	20	Low (0.25)	1	Low	Yes	Low	Lunar Orbit	1	-0.4333	37.58	5.4200	7.5154	1.5302	7.5154	0	0	0
ISRU Technology Development	Literature	0	Medium (0.5)	3	Medium	No	Medium	Lunar Surface	1	-0.7444	1000.00	2.0000	200.0000	0.4962	200.0000	0	0	0
Precision Landing Technology Development	Literature	0	Medium (0.5)	1	Medium	Yes	Medium	Lunar Surface	1	0.1222	1000.00	2.0000	200.0000	0.6869	200.0000	0	0	0

Table B.2.: Capabilities section of example Candidate System Library as used in calculation of results.

	Capabilities																
Systems	Deliver to NHRO (kg)	Crew Capability (n)	Mass NRHO to LLO (kg)	Mass LLO to LS (kg)	DeliverReturn Crew ToFrom NRHO (n)	Habitable Volume at NRHO (m^3)	Electrical Power at NRHO (kw)	Crewed L AscDesc (n)	LH2 Propellant at NRHO (kg)	CH4 Propellant at NRHO	Storable Propellant at NRHO	Habitable Volume at LS (m^3)	Electrical Power at LS (kw)	Logistics at LS	LH2 Propellant at LS (kg)	Exploration Objective (days)	Variable Capability?
LH2 Tug	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
LH2 Tug Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Meth Tug	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Meth Tug Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Sto Tug	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Sto Tug Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Sto 2 Tug	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Sto 2 Tug Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
LH2 DM	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
LH2 DM Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Meth DM	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Meth DM Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	Yes
Sto DM	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto DM Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto 2 DM	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto 2 DM Prop Off Load	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
LH2 AM (Re Use A)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
LH2 AM (Re Use B)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Meth AM (Re Use A)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Meth AM (Re Use B)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto AM (Re Use A)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto AM (Re Use B)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto 2 AM (Re Use A)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto 2 AM (Re Use B)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
SLS	25.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Orion	0.00	0.00	0.00	0.00	4	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
4 Person Lunar Crew	0.00	4	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
SS Lander Mission 2P	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	4	No
LS Lander Mission 2P 15D	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	28	No
Commercial HLV	15.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
LH2 Refuel Tanker	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	9.10	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Meth Refuel Tanker	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	9.75	0.00	0.00	0.00	0.00	0.00	0.00	No
Sto Refuel Tanker	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	10.01	0.00	0.00	0.00	0.00	0.00	No
PPE	0.00	0.00	0.00	0.00	0.00	0.00	40.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Gateway MiniHab	0.00	0.00	0.00	0.00	0.00	45.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Surface habitat	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	20.00	0.00	0.00	0.00	0.00	No
Logistics Container	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.10	0.00	0.00	No
LH2 Empty AM (Re Use A)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
LH2 Empty AM (Re Use B)	0.00	0.00	0.00	0.00	0.00	0.00	0.00	2.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
ISRU System	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	10.00	0.00	No
Reactor Power System	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	45.00	0.00	0.00	0.00	No
ISRU Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
Precision Landing Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No
CFM Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	No

Table B.3.: Requirements section of example Candidate System Library as used in calculation of results.

Systems	Deliver to NHRO (kg)	Crew Capability (n)	Mass NRHO to LLO (kg)	Mass LLO to LS (kg)	DeliverReturn Crew ToFrom NRHO (n)	Habitable Volume at NRHO (m^3)	Electrical Power at NRHO (kw)	Crewed L AscDesc (n)	LH2 Propellant at NRHO (kg)	CH4 Propellant at NRHO	Storable Propellant at NRHO	Habitable Volume at LS (m^3)	Electrical Power at LS (kw)	Logistics at LS	LH2 Propellant at LS (kg)	Exploration Objective (days)
LH2 Tug	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 Tug Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth Tug	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth Tug Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto Tug	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto Tug Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto 2 Tug	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto 2 Tug Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 DM	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 DM Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth DM	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth DM Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto DM	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto DM Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto 2 DM	0.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto 2 DM Prop Off Load	15.00	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 AM (Re Use A)	13.37	0.00	13.37	13.37	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 AM (Re Use B)	0.00	0.00	13.37	13.37	0.00	0.00	1.00	0.00	6.16	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth AM (Re Use A)	13.14	0.00	13.14	13.14	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth AM (Re Use B)	0.00	0.00	13.14	13.14	0.00	0.00	1.00	0.00	0.00	7.07	0.00	0.00	0.00	0.00	0.00	0.00
Sto AM (Re Use A)	14.34	0.00	14.34	14.34	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto AM (Re Use B)	0.00	0.00	14.34	14.34	0.00	0.00	1.00	0.00	0.00	0.00	8.23	0.00	0.00	0.00	0.00	0.00
Sto 2 AM (Re Use A)	13.71	0.00	13.71	13.71	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto 2 AM (Re Use B)	0.00	0.00	13.71	13.71	0.00	0.00	1.00	0.00	0.00	0.00	7.6577	0.00	0.00	0.00	0.00	0.00
SLS	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Orion	24	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
4 Person Lunar Crew	0.00	0.00	0.00	0.00	4	15.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
SS Lander Mission 2P	0.00	2.00	0.00	0.00	0.00	0.00	0.00	2	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LS Lander Mission 2P 15D	0.00	2.00	0.00	0.00	0.00	0.00	0.00	2	0.00	0.00	0.00	20.00	0.00	0.00	0.00	0.00
Commercial HLV	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 Refuel Tanker	15	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Meth Refuel Tanker	15	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Sto Refuel Tanker	15	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
PPE	15	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Gateway MiniHab	15	0.00	0.00	0.00	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Surface habitat	12	0.00	12	12	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	5.00	0.00	0.00	0.00
Logistics Container	0.15	0.00	0.15	0.15	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
LH2 Empty AM (Re Use A)	7.21	0.00	7.21	7.21	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	6.16	0.00
LH2 Empty AM (Re Use B)	0.00	0.00	7.21	7.21	0.00	0.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	6.16	0.00
ISRU System	5.00	0.00	5.00	5.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	40.00	0.00	0.00	0.00
Reactor Power System	6	0.00	6.00	6.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
CFM Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
ISRU Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Precision Landing Technology Development	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00

C. APPENDIX C MARS SURFACE APPENDIX

This appendix includes relevant details for the Lunar surface scenario(Chapter 8).

Table C.1.: Phased life cycle cost and schedule estimates for each potential system.

System	Estimaton Method	Dry Mass	Difficulty Inputs						Generation	Resulting Difficulty	DDT&E Cost [MUSD16]	DDT&E Duration [years]	Production Cost [MUSD18]	Production Duration [years]	AMCM Coefficient	AMCM Inert Mass m	AMCM Inert Mass b
			Technical Complexity ▼	Planned years of Operation	Risk Tolerance ▼	Human Rated ▼	Program Complexity ▼	System Type									
4 Person Crew Orbital	Training	0	Medium	1	Medium	Yes	Medium	unar Orbit	2	0.12	0.00	6.09	0.00	2.09	4.17	0.00	115.26
4 Person Crew Surface	Training	0	Medium	1	Medium	Yes	Medium	unar Orbit	2	0.12	0.00	6.09	0.00	2.09	4.17	0.00	115.26
Crewed Mars Orbital Mission (30 Day)	Training	0	Medium	1	Medium	Yes	Medium	unar Orbit	2	0.12	0.00	6.09	0.00	2.09	4.17	0.00	100.69
Crewed Mars Surface Mission	Training	0	Medium	1	Medium	Yes	Medium	unar Orbit	2	0.12	0.00	6.09	0.00	2.09	4.17	0.00	100.69
Deep Space Habitat (Re Use A)	AMCM (Rolley)	28000	Medium-High	10	Low	Yes	Medium-High	Mars Orbit	3	1.67	7751.76	7.94	1550.35	4.08	9.00	0.00	97.92
Deep Space Habitat (Re Use B)	AMCM (Rolley)	10000	Medium	1	Medium	Yes	Low-Medium	Mars Orbit	2	-0.13	2023.66	5.79	404.73	1.83	4.64	0.00	97.92
Orion	Gov Estimate	10000	Medium	1	Medium	Yes	Low-Medium	Mars Orbit	2	-0.13	2023.66	5.79	404.73	1.83	4.64	0.00	97.68
Government Super Heavy Lift 1A	Gov Estimate	10000	Medium	1	Medium	Yes	Low-Medium	Mars Orbit	2	-0.13	2023.66	5.79	404.73	1.83	4.64	0.00	97.68
Commercial Heavy Lift	Market	10000	Medium	1	Medium	Yes	Medium-High	Mars Orbit	1	0.37	3254.31	6.39	650.86	2.37	7.46	0.00	141.03
SuperPropSystem	AMCM (Rolley)	25000	Medium-High	2	Low	Yes	Medium-High	Mars Orbit	1	1.13	8398.47	7.30	1679.69	3.32	10.51	0.81	141.03
NTR TransProp 2W PC FT Var	AMCM (Rolley)	20000	Medium-High	2	Low	Yes	Medium-High	Mars Orbit	1	1.13	7248.36	7.30	1449.67	3.32	10.51	0.81	0.00
NTR TransProp 2W PC LE Var	AMCM (Rolley)	20000	Medium-High	2	Low	Yes	Medium-High	Mars Orbit	1	1.13	7248.36	7.30	1449.67	3.32	10.51	0.81	0.00
NTR TransProp 2W AC FT Var	AMCM (Rolley)	20000	Medium-High	2	Low	Yes	Medium-High	Mars Orbit	1	1.13	7248.36	7.30	1449.67	3.32	10.51	0.54	0.00
NTR TransProp 2W AC LE Var	AMCM (Rolley)	20000	Medium-High	2	Low	Yes	Medium-High	Mars Orbit	1	1.13	7248.36	7.30	1449.67	3.32	10.51	0.54	0.00
NTR TransProp 1W PC FT Var	AMCM (Rolley)	20000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	3778.10	5.57	755.62	1.65	5.48	0.54	0.00
NTR TransProp 1W PC LE Var	AMCM (Rolley)	20000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	3778.10	5.57	755.62	1.65	5.48	0.41	0.00
NTR TransProp 1W AC FT Var	AMCM (Rolley)	20000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	3778.10	5.57	755.62	1.65	5.48	0.41	0.00
NTR TransProp 1W AC LE Var	AMCM (Rolley)	20000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	3778.10	5.57	755.62	1.65	5.48	0.27	0.00
LH2 TransProp 2W PC FT Var	AMCM (Rolley)	20000	Low-Medium	2	Low	Yes	Low-Medium	Mars Orbit	1	0.13	4616.79	6.10	923.36	2.10	6.69	2.85	0.00
LH2 TransProp 2W PC LE Var	AMCM (Rolley)	20000	Low-Medium	2	Low	Yes	Low-Medium	Mars Orbit	1	0.13	4616.79	6.10	923.36	2.10	6.69	2.85	0.00
LH2 TransProp 2W AC FT Var	AMCM (Rolley)	20000	Low-Medium	2	Low	Yes	Low-Medium	Mars Orbit	1	0.13	4616.79	6.10	923.36	2.10	6.69	1.13	0.00
LH2 TransProp 2W AC LE Var	AMCM (Rolley)	20000	Low-Medium	2	Low	Yes	Low-Medium	Mars Orbit	1	0.13	4616.79	6.10	923.36	2.10	6.69	1.13	0.00
LH2 TransProp 1W PC FT Var	AMCM (Rolley)	20000	Low-Medium	2	Medium	No	Low-Medium	Mars Orbit	1	-1.31	2406.43	4.37	481.29	0.81	3.49	1.13	0.00
LH2 TransProp 1W PC LE Var	AMCM (Rolley)	20000	Low-Medium	2	Medium	No	Low-Medium	Mars Orbit	1	-1.31	2406.43	4.37	481.29	0.81	3.49	0.71	0.00
LH2 TransProp 1W AC FT Var	AMCM (Rolley)	20000	Low-Medium	2	Medium	No	Low-Medium	Mars Orbit	1	-1.31	2406.43	4.37	481.29	0.81	3.49	0.71	0.00
LH2 TransProp 1W AC LE Var	AMCM (Rolley)	20000	Low-Medium	2	Medium	No	Low-Medium	Mars Orbit	1	-1.31	2406.43	4.37	481.29	0.81	3.49	0.39	0.00
Cargo Descent Vehicle FT	AMCM (Rolley)	10000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	2391.08	5.57	478.22	1.65	5.48	0.00	119.26
Cargo Descent Vehicle LE	AMCM (Rolley)	10000	Medium-High	2	Medium	No	Medium-High	Mars Orbit	1	-0.31	2391.08	5.57	478.22	1.65	5.48	0.00	119.26
Crew Descent Vehicle LE	AMCM (Rolley)	10000	High	1	Low	Yes	Medium-High	Mars Orbit	1	1.32	4982.82	7.52	996.56	3.57	11.41	0.00	119.26
Crew Ascent Vehicle Full LE	AMCM (Rolley)	20000	High	5	Low	Yes	Medium-High	lars Surfac	1	1.58	8133.32	7.84	1626.66	3.95	11.79	0.00	119.26
Crew Ascent Vehicle Full FT	AMCM (Rolley)	20000	High	1	Low	Yes	Medium-High	lars Surfac	1	1.32	7211.54	7.52	1442.31	3.57	10.46	0.00	119.26
Crew Ascent Vehicle Empty	AMCM (Rolley)	20000	High	5	Medium	No	High	unar Orbit	1	0.39	4159.94	6.41	831.99	2.39	6.03	0.00	119.26
Crew Descent Vehicle FT	AMCM (Rolley)	20000	Medium-High	1	Low	Yes	Medium-High	lars Surfac	1	1.07	6442.48	7.22	1288.50	3.23	9.34	0.00	119.26
Crew Ascent Descent Vehicle	AMCM (Rolley)	30000	High	1	Low	Yes	High	lars Surfac	1	1.57	10549.29	7.82	2109.86	3.93	11.70	0.00	119.26
Surface Habitat LE	AMCM (Rolley)	35000	Medium-High	10	Low	Yes	Medium-High	lars Surfac	1	1.67	12217.94	7.94	2443.59	4.08	12.24	0.00	119.26
Surface Habitat FT	AMCM (Rolley)	35000	Medium-High	10	Low	Yes	Medium-High	lars Surfac	1	1.67	12217.94	7.94	2443.59	4.08	12.24	0.00	119.26
CFM Technology Development	Estimate	10000	Medium	1	Low	Yes	Low	unar Orbit	1	0.07	2276.58	6.02	455.32	2.03	5.22	0.00	119.26
NTR Technology Development	Estimate	10000	Medium	1	Low	Yes	Low	unar Orbit	1	0.07	2276.58	6.02	455.32	2.03	5.22	0.00	119.26
Launch Window	N/A	10000	Medium	1	Low	Yes	Low	unar Orbit	1	0.07	2276.58	6.02	455.32	2.03	5.22	0.00	119.26
Aero Capture FT	AMCM (Rolley)	10000	Medium	1	Low	Yes	Medium	Mars Orbit	1	0.57	3552.63	6.62	710.53	2.60	8.14	0.00	119.26
Aero Capture LE	AMCM (Rolley)	10000	Medium	1	Low	Yes	Low	Mars Orbit	1	0.07	2835.31	6.02	567.06	2.03	6.50	0.00	119.26
Adv Aerocapture Technology Development	AMCM (Rolley)	10000	Medium	1	Low	Yes	Low	lars Surfac	1	0.07	2597.01	6.02	519.40	2.03	5.95	0.00	119.26
LH2 Tanker	AMCM (Rolley)	3000	Medium	0.25	Medium	No	Medium	Earth Orbit	1	-0.93	601.51	4.83	120.30	1.10	3.05	0.00	119.26
CH4 Tanker	AMCM (Rolley)	3000	Medium	0.25	Medium	No	Medium	Earth Orbit	1	-0.93	601.51	4.83	120.30	1.10	3.05	0.00	119.26
ISRU System	AMCM (Rolley)	10000	Medium	1	Low	Yes	Low	unar Orbit	1	0.07	2276.58	6.02	455.32	2.03	5.22	0.00	119.26
Mars Surface Power	AMCM (Rolley)	3000	Medium	2	Medium	No	Medium	unar Orbit	1	-0.81	692.19	4.97	138.44	1.20	3.51	0.00	119.26
Logistics Container MS FT	AMCM (Rolley)	2000	Low	2	Low	No	Low-Medium	unar Orbit	1	-1.12	461.48	4.60	92.30	0.95	3.06	0.00	119.26
Logistics Container MS LE	AMCM (Rolley)	2000	Low	2	Low	No	Low-Medium	unar Orbit	1	-1.12	461.48	4.60	92.30	0.95	3.06	0.00	119.26
Logistics Container LEO	AMCM (Rolley)	2000	Low	2	Low	No	Low-Medium	unar Orbit	1	-1.12	461.48	4.60	92.30	0.95	3.06	0.00	119.26

Table C.2.: Capabilities section of example Candidate System Library as used in calculation of preliminary results.

System	Exploration Cap	Crew Capability (n)	Crew Ascent and Entry	ES to EO (t)	Habitable Volume in Transit (m^3)	Habitable Volume Mars Surface (m^3)	EO to MO FT (t)	MO to EO FT (t)	Capture at MO FT (t)	Capture at EO(t)	EO to MO LE (kg)	MO to EO LE (t)	Capture at MO LE (t)	Crewed Ascent (n)	Crewed Descent (n)	Mars Landed Surface Mass (kg)	CH4 Propellant at LEO (t)	LH2 Propellant at LEO (t)	Consumables at MS	LH2 Propellant at MS	Power at MS (KW)	Launch Window	Consumables at LEO (t)
4 Person Crew Orbital	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 Person Crew Surface	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crewed Mars Orbital Mission	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crewed Mars Surface Mission	200.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Deep Space Habitat (Re Use A)	0.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Deep Space Habitat (Re Use B)	0.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Orion	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Government Super Heavy Lift 1A	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Commercial Heavy Lift	0.0	0.0	0.0	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 2W PC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 2W PC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 2W AC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 2W AC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 1W PC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 1W PC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 1W AC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR TransProp 1W AC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 2W PC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 2W PC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 2W AC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 2W AC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 1W PC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 1W PC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 1W AC FT Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 TransProp 1W AC LE Var	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Cargo Descent Vehicle FT	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Cargo Descent Vehicle LE	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	50.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Descent Vehicle LE	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Ascent Vehicle Full LE	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Ascent Vehicle Full FT	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Ascent Vehicle Empty	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Descent Vehicle FT	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Ascent Descent Vehicle	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Surface Habitat LE	0.0	0.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Surface Habitat FT	0.0	0.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
CFM Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Launch Window	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	400.0	0.0
Aero Capture FT	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	250.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Aero Capture LE	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	250.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Adv Aerocapture Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
LH2 Tanker	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	57.0	0.0	0.0	0.0	0.0	0.0
CH4 Tanker	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	57.0	0.0	0.0	0.0	0.0	0.0	0.0
ISRU System	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0
Mars Surface Power	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	30.0	0.0	0.0
Logistics Container MS FT	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	15.0	0.0	0.0	0.0	0.0
Logistics Container MS LE	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	15.0	0.0	0.0	0.0	0.0
Logistics Container LEO	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	13.0

Table C.3.: Requirements section of example Candidate System Library as used in calculation of preliminary results.

System	Exploration Cap	Crew Capability (n)	Crew Ascent and Entry	ES to EO (t)	Habitable Volume in Transit (m^3)	Habitable Volume Mars Surface (m^3)	EO to MO FT (t)	MO to EO FT (t)	Capture at MO FT (t)	Capture at EO(t)	EO to MO LE (kg)	MO to EO LE (t)	Capture at MO LE (t)	Crewed Ascent (n)	Crewed Descent (n)	Mars Landed Surface Mass (kg)	CH4 Propellant at LEO (t)	LH2 Propellant at LEO (t)	Consumables at MS	LH2 Propellant at MS	Power at MS (KW)	Launch Window	Consumables at LEO (t)
4 Person Crew Orbital	0.0	0.0	4.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 Person Crew Surface	0.0	0.0	4.0	0.0	100.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	4.0	4.0	0.0	0.0	0.0	15.0	0.0	0.0	1.0	0.0
Crewed Mars Orbital Mission (30 Day)	0.0	4.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crewed Mars Surface Mission	0.0	4.0	0.0	0.0	0.0	100.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Deep Space Habitat (Re Use A)	0.0	0.0	0.0	40.0	0.0	0.0	40.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Deep Space Habitat (Re Use B)	0.0	0.0	0.0	0.0	0.0	0.0	40.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	13.0
Orion	0.0	0.0	0.0	24.0	0.0	0.0	10.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Government Super Heavy Lift 1A	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Commercial Heavy Lift	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 2W PC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 2W PC LE Var	0.0	0.0	0.0	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 2W AC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 2W AC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 1W PC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 1W PC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 1W AC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
NTR TransProp 1W AC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 2W PC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 2W PC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 2W AC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 2W AC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 1W PC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 1W PC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 1W AC FT Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 TransProp 1W AC LE Var	0.0	0.0	0.0	95.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Cargo Descent Vehicle FT	0.0	0.0	0.0	30.0	0.0	0.0	30.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Cargo Descent Vehicle LE	0.0	0.0	0.0	30.0	0.0	0.0	0.0	0.0	0.0	0.0	30.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Crew Descent Vehicle LE	0.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Crew Ascent Vehicle Full LE	0.0	0.0	0.0	42.0	0.0	0.0	0.0	0.0	0.0	0.0	42.0	0.0	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Crew Ascent Vehicle Full FT	0.0	0.0	0.0	42.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Crew Ascent Vehicle Empty	0.0	0.0	0.0	17.0	0.0	0.0	0.0	0.0	0.0	0.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	40.0	0.0	1.0	0.0
Crew Descent Vehicle FT	0.0	0.0	0.0	40.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Crew Ascent Descent Vehicle	0.0	0.0	0.0	80.0	0.0	0.0	80.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Surface Habitat LE	0.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Surface Habitat FT	0.0	0.0	0.0	40.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	40.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
CFM Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NTR Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Launch Window	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Aero Capture FT	0.0	0.0	0.0	10.0	0.0	0.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Aero Capture LE	0.0	0.0	0.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	10.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Adv Aerocapture Technology Development	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
LH2 Tanker	0.0	0.0	0.0	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
CH4 Tanker	0.0	0.0	0.0	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
ISRU System	0.0	0.0	0.0	2.0	0.0	0.0	0.0	0.0	0.0	0.0	2.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	5.0	1.0	0.0
Mars Surface Power	0.0	0.0	0.0	7.8	0.0	0.0	0.0	0.0	0.0	0.0	7.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Logistics Container MS FT	0.0	0.0	0.0	20.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Logistics Container MS LE	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0
Logistics Container LEO	0.0	0.0	0.0	18.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

Propulsion Sizing Details Propulsion Type Details NTR

- ISP = 1000
- L =0.3

LH2 Chem

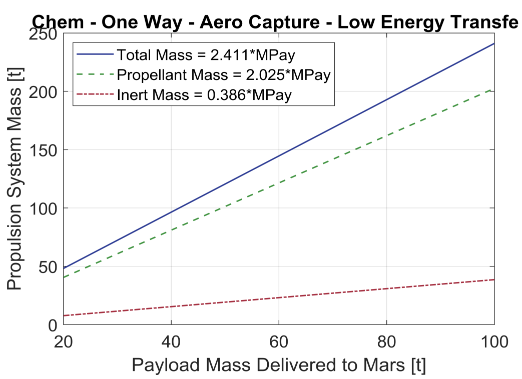
- ISP = 450
- L =0.16

DV Estimates Fast Transit

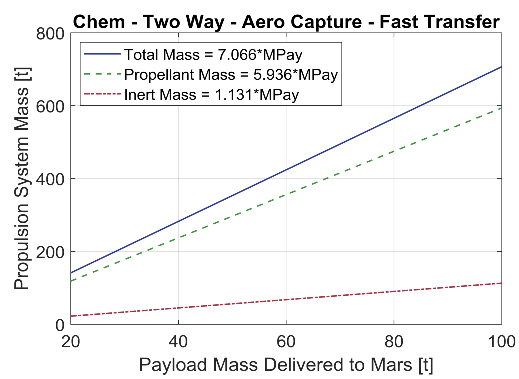
- TMI = 4000 m/s
- TEI = 1500 m/s
- Propulsive Capture = 1500 m/s
- Aerocapture Propulsion DV = 375 m/s

Low Energy Transit

- TMI = 3600 m/s
- Propulsive Capture = 1500 m/s
- Mars Aerocapture Propulsion DV = 375 m/s

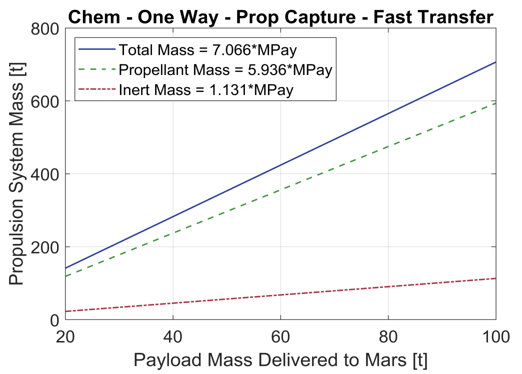


(a) One Way Low Energy

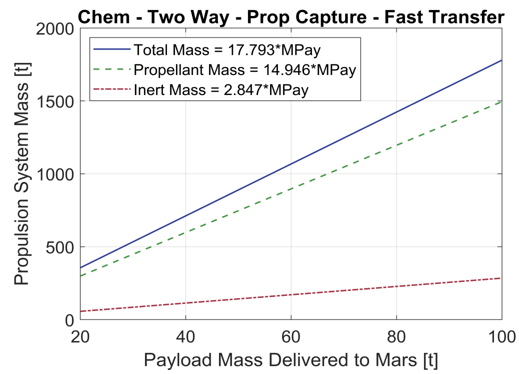


(b) Two Way Fast Transit

Fig. C.1.: Chemical Propulsion - Aerocapture

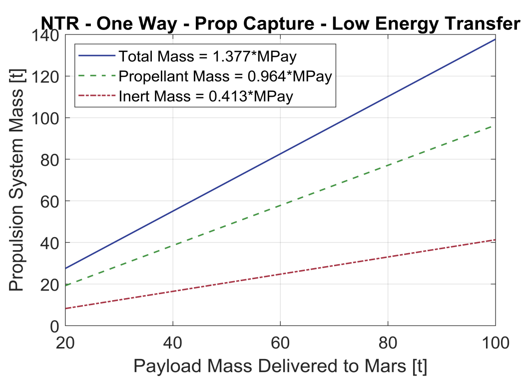


(a) One Way Low Energy

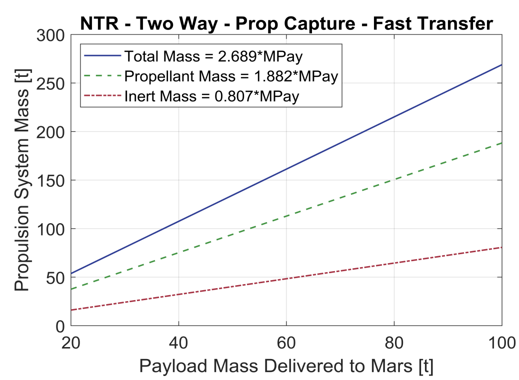


(b) Two Way Fast Transit

Fig. C.2.: Chemical Propulsion - Prop Capture

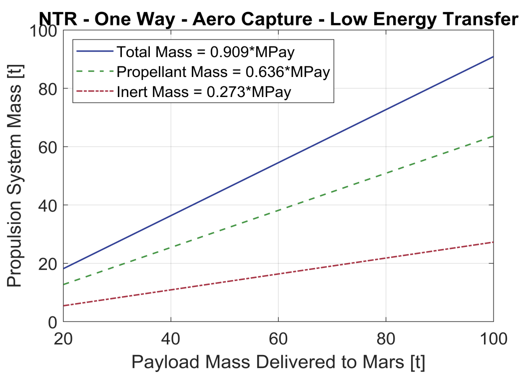


(a) One Way Low Energy

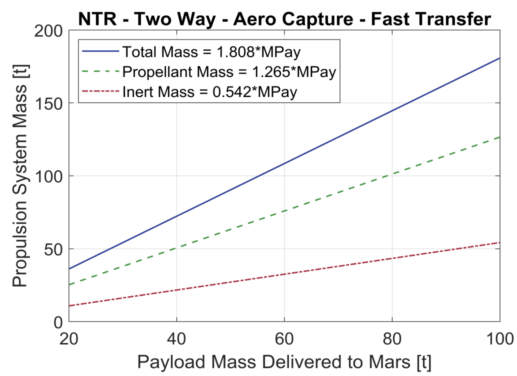


(b) Two Way Fast Transit

Fig. C.3.: NTR Propulsion - Prop Capture



(a) One Way Low Energy



(b) Two Way Fast Transit

Fig. C.4.: NTR Propulsion - Aerocapture