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A Systematic Process for Assessing Human Spacecraft Designs in Terms of Relative Safety and Operational Characteristics

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A SYSTEMATIC PROCESS FOR ASSESSING HUMAN SPACECRAFT CONCEPTUAL DESIGNS IN TERMS OF RELATIVE SAFETY AND OPERATIONAL CHARACTERISTICS

by

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ABSTRACT

Higdon, Kevin Paul (Ph.D., Aerospace Engineering Sciences)

A Systematic Process for Assessing Human Spacecraft Designs in Terms of Relative Safety and Operational Characteristics

Thesis directed by Associate Professor, David M. Klaus, Ph.D.

The research efforts in this dissertation are focused on reducing uncertainty in the conceptual design phase through a process of establishing a minimum functionality baseline before trading *Safety* and *Operability* in proposed spacecraft configurations. The challenge in human spacecraft development is how to combine the parts into a working design that complies with many requirements for top level mission objectives, safety, and mission success. The design methodologies presented here provides designers and decision makers with additional methods that provide an overall view of candidate design concepts.

This work establishes a definition for a minimum functional design and is the first to group the fundamental mass parameters of a human spacecraft in the categories of *Physics*, *Physiology*, *Safety*, and *Operability*. The minimum functional baseline configuration described in this work is different from previous approaches because it eliminates the bias toward a minimum set of requirements. The amount of *Safety* in the spacecraft is the mass dedicated to safety through similar or dissimilar redundancy, safety components, margins, and dispersions. The amount of *Operability* in the spacecraft is the mass used to perform mission objectives and make functions easier or efficient. Because human spacecraft are highly coupled systems, the introduction of mass in one subsystem has downstream effects on other subsystems that are not easily recognized by designers and the use of rapidly reconfigurable prototypes allows designers

and multidisciplinary teams to utilize Boundary Objects as a means of communication for maturing designs. The mass addition process coupled with the minimum functionality approach creates a tradespace of spacecraft configurations and provides designers with an overall view of how various levels of *Safety* or *Operability* will affect the overall spacecraft mass. The decisions made in the conceptual design phase are critical to the success of the program and uncertainty can lead to unnecessary redesign in later phases. The previous methods can be combined into a conceptual design process that couples easily with typical industry approaches to human spacecraft development. The use of minimum functionality as a precursor to more conventional approaches allows the spacecraft configuration to take shape before detailed CAD and higher fidelity analyses.

DEDICATION

This work is dedicated to my wife, Amy and our son Roman. I appreciate their patience and sacrifice for allowing me to pursue my dream.

He who dwells in the shelter of the Most High will rest in the shadow of the Almighty. I will say of the Lord, "He is my refuge and my fortress, my God in whom I trust."

Psalm 91:1-2

"Because he loves me", says the Lord, "I will rescue him; I will protect him, for he acknowledges my name. He will call upon me, and I will answer him; I will be with him in trouble, I will deliver him and honor him. With long life will I satisfy him and show him my salvation".

Psalm 91:14-16

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DEFINITION OF TERMS

ASCE	American Society of Civil Engineers
CAD	Computer Aided Design
CCDev	Commercial Crew Development
CDR	Critical Design Review
CERR	Critical Events Readiness Review
CEV	Crew Exploration Vehicle
CFD	Computational Fluid Dynamics
CLAMP	Conceptual Lunar Ascent Module Program
CM	Command Module
CO	Collaborative Optimization
CSSO	Concurrent Subspace Optimization
DFMR	Design For Minimum Risk
DoD	Department of Defense
DOE	Design of Experiments
DOF	Degree of Freedom
ECLSS	Environmental Control and Life Support System
EDS	Earth Departure Stage
EOR-LOR	Earth Orbit Rendezvous – Lunar Orbit Rendezvous
ESAS	Exploration Systems Architecture Study
ESD	Event Sequence Diagrams
ETA	Event Tree Analysis

EVA	Extravehicular Activity
FEA	Finite Element Analysis
FMEA	Failure Modes and Effects Analysis
FMECA	Failure Modes and Effects Criticality Analysis
FoS	Factor of Safety
FRR	Flight Readiness Review
FTA	Fault Tree Analysis
GAO	Government Accountability Office
GINA	Generalized Information Analysis
INCOSE	International Council on Systems Engineering
ISS	International Space Station
JSC	NASA Johnson Space Center
LAT	Lunar Architecture Team
LDAC	Lunar Design Analysis Cycle
LEM	Apollo Lunar Excursion Module
LEO	Low Earth Orbit
LL	Lunar Lander
LLO	Low Lunar Orbit
LLPS	Lunar Lander Preparatory Study
LM	Lunar Module
LOC	Loss of Crew
LOI	Lunar Orbit Insertion
LOM	Loss of Mission

LRU	Line Replaceable Unit
LSAM	Lunar Surface Access Module
MATE	Multi Attribute Tradespace Exploration
MCR	Mission Concept Review
MBDO	Metamodel Based Design Optimization
MDO	Multidisciplinary Design Optimization
MIT	Massachusetts Institute of Technology
MLD	Master Logic Diagrams
MMOD	Micrometeoroid and Orbital Debris
MOO	Multi Objective Optimization
MPCV	Multi-Purpose Crew Vehicle
NASA	National Aeronautics and Space Administration
ORR	Operational Readiness Review
PDR	Preliminary Design Review
PFAR	Post Flight Assessment Review
PLAR	Post Launch Assessment Review
PRA	Probabilistic Risk Assessment
PRR	Production Readiness Review
RBD	Risk Based Design
RBDO	Reliability Based Design Optimization
RBF	Radial Basis Functions
RPN	Risk Priority Number
RS	Response Surface

RSM	Response Surface Methodology
SE	Systems Engineering
SoS	Systems of Systems
SAR	System Acceptance Review
SDR	System Definition Review
SIR	System Integration Review
SLS	Space Launch System
SRR	Systems Requirements Review
SSPAC	Space Systems, Policy, and Architecture Consortium
SVR	Support Vector Regression
SVM	Support Vector Machine
TEI	Trans-Earth Injection
TRR	Test Readiness Review
TLI	Trans-Lunar Injection
VSE	Vision for Space Exploration
ZBV	Zero Based Vehicle

CHAPTER 1

INTRODUCTION

Historically, the development of human spacecraft has followed an approach that develops low-volume and high-cost designs for a given or preferred spaceflight mission. Human spacecraft are especially unique because the amount of development and cost is often driven by the need to simultaneously meet a host of requirements to meet performance and mission objectives while at the same time balancing objectives such as risk and safety. Because of the large number of requirements typically associated with human spacecraft, a critical need in the early stages of conceptual development are standardized and repeatable processes for trading performance, safety, and operational objectives in a conceptual design process that can be easily implemented by spacecraft designers.

The research presented in this dissertation builds upon previous conceptual design approaches to introduce a new design methodology focused on the early conceptual design phase of human spacecraft. Because the conceptual design phase is characterized by a large amount of uncertainty (Jilla and Miller, 2004), the approach described here provides valuable information to designers in the early conceptual design phase such that informed decisions can be made as to the expected risk and performance of the spacecraft in later stages of development. The context of this approach is focused on the spacecraft designer, of which this author has firsthand experience and has learned many valuable lessons associated with the design, development, and integration of human spacecraft.

The challenge in human spacecraft development is creating a system that meets a large number of requirements in order to carry out the spaceflight mission. The spacecraft must transport the crew to a particular destination and return, keep the crew alive, keep the crew safe, and perform the mission operation objectives. Because many unknowns exist during the concept exploration phase, the design approach is typically unstructured with design teams pursuing single design point options. A standardized approach that allows designers to investigate a wider range of options in conceptual design is needed such that potential problems can be avoided in later phases. As observed in the development of the Orion spacecraft, unrealistic assumptions made during the early phases created problems in later stages of the design and forced a complete redefinition of the spacecraft in order to achieve mass and performance objectives (Hu *et al.*, 2008).

A human spacecraft must perform many independent tasks during a spaceflight mission. The mass associated with performing the tasks can be grouped into four fundamental parameters. These parameters are associated with transporting the crew, keeping the crew alive, keeping the crew safe, and performing the mission objectives. The combination of these four fundamental mass parameters is the foundation of the design methodology and is expressed as:

$$\sum \text{Spacecraft Mass} = \underbrace{f(\mathit{Physics}) + f(\mathit{Physiology})}_{\text{'non-negotiable'}} + \underbrace{f(\mathit{Safety}) + f(\mathit{Operability})}_{\text{'tradespace'}} \qquad (1)$$

The design methodology presented in this dissertation begins with a "Minimum Functionality" spacecraft configuration as a baseline for trading *Safety* and *Operability*. The minimum functionality design configuration is defined by the required *Physics* to fly the mission and the *Physiological* needs of the crew as a lower boundary condition for design exploration, iteration, and maturation. The minimum functionality configuration would never be utilized as a realistic 'flyable' design because it does not contain mitigations for failures and contingencies. However, in an academic sense, it establishes the absolute lowest spacecraft configuration as a boundary point for trading of mass, risk and operational objectives.

Using the minimum functionality spacecraft configuration as the baseline point of departure, the spacecraft configuration tradespace consists of various levels of *Safety* and *Operability* (or mission objectives) in the spacecraft to evaluate risk and trade mission objectives. The addition of *Safety* is achieved by adding failure tolerance through similar or dissimilar components, safety components, and including factors of safety in components that are designed for minimum risk. The addition of *Operability* is achieved by adding components dedicated for conducting mission operations functions beyond the transportation of crew and mitigating contingencies. Throughout the remainder of this dissertation, the italicized variables of *Physics*, *Physiology*, *Safety* and *Operability* refer to the mass associated with the parameter.

In order to trade and evaluate *Safety* and *Operability*, two figures of merit: the "Safety Index" and "Operability Index" are introduced as metrics for comparing the amount of risk and functionality in conceptual spacecraft design configurations. The Safety Index is not intended to replace current reliability or Probabilistic Risk Assessment (PRA) approaches for spacecraft design, but is an early tool for designers to understand the impact of mass additions on the *Safety* of the spacecraft and reduce the need for detailed PRA in the early stages of design; when the design is changing rapidly and least amount of information is known. The use of an Operability Index provides designers with a metric to evaluate the amount of additional functionality associated with a particular spacecraft configuration. The methods presented in this work will demonstrate a design methodology that can be coupled to existing design processes and procedures for reducing risk and increasing safety in conceptual human spacecraft designs.

1.1 CHALLENGES IN CONCEPTUAL HUMAN SPACECRAFT

Overcoming uncertainty is the greatest challenge in the development of human spacecraft. Spacecraft designers commonly use proven flight hardware and heuristics combined with knowledge and experience to explore the feasibility of potential ideas and concepts. This approach is necessary because human intuition, creativity, and imagination are required in the beginning of the design process to foster ideas that can be developed into feasible solutions. The conceptual design phase is also one of the least standardized but perhaps one of the most important design phases (Jilla and Miller, 2004; Pacheco *et al*., 2003). In the context of a human spacecraft design, the need to reduce uncertainty in the early stages is especially important because the vehicle will be carrying humans and will operate in a combination of environmental conditions that are not easily replicated on the ground.

A secondary challenge in the development of human spacecraft is that new aerospace vehicles are often designed to "push the envelope". The need for new technology development is commonly required for the spacecraft to perform the intended mission, but also creates the need for additional resources or development time to reduce risk and uncertainty with unknown and unproven technology. Thus, when a spacecraft designer or design team begins to explore an idea or concept, they are confronted with the stark reality of "How do I proceed from the mission objectives to a realistic spacecraft design?" and "Do we have the existing technology to make it work?" As a spacecraft designer who has struggled with these questions, the first answers are very difficult to answer because of the uncertainty in design assumptions. The methods used to conceive concepts are highly dependent upon the creative skills and experience of designers or design teams (Bryant, *et al*., 2005).

Conceptual design concepts must be flexible and adaptable for unforeseen and unknown problems. Spacecraft designers recognize the risk of "locking in" a particular design configuration too early and the consequences that can follow in the form of redesign, rework, or completely new designs. Designs must be safe, robust, and reliable; but without the added penalty of over-designing the mass through additional complexity. A common pitfall is to optimize concepts during this early phase to specified mass growth margin tables without considering flexibility or adaptability as the design matures. Because spacecraft subsystems are highly coupled, additional complexity can be introduced in the form of additional requirements creep and misunderstood performance goals (Miller *et al*., 2008).

Perhaps the greatest challenge to spacecraft designers is that in addition to mass, performance, and integration uncertainty, a human spacecraft design must meet a minimum threshold for risk and reliability in order to fly crew. The most common method of quantifying a "Safe" vehicle is through the use of a PRA (Stamatelatos, 2002). The safety of a design concept is usually attributed to parameters such as the probability of Loss of Crew (LOC) or Loss of Mission (LOM). Tumer *et al*., (2005) best described the problem with conceptual design and risk based design approaches in aerospace applications as:

> *"Due to the risky nature of space missions, NASA centers have adopted a variety of techniques – developing tools, procedures, and guidelines to mitigate risk. Most of these techniques, however, require significant amounts of detailed and possibly quantitative information, making them inapplicable to early stages of design."* (Tumer *et al*., 2005)

When little information is known about the reliability of the components that make up the spacecraft subsystems, assumptions about the targeted reliability of unknown or untested components are used as a placeholder until information is gathered through testing or unmanned flight tests. Although this approach provides an initial conservative target for risk, the exact calculation of the reliability of a spacecraft is usually done much later after a conceptual design has been chosen and the design is moving toward a Preliminary Design Review (PDR). In addition, the early targets for reliability might not be achievable as more information is obtained, thus creating the need for redesign or rework within a preferred single design approach.

Many different and competing spacecraft concepts and configurations must be explored in a timely manner during the conceptual design phase to determine the most feasible solutions. In many cases, a single point design is not the most ideal candidate at the end of the conceptual design phase (McManus *et al*., 2004). Optimizing too early will lead to multiple redesigns when additional information is discovered through testing or detailed analysis. A high fidelity detailed analysis of each concept is not the best use of time in this stage (Messac and Mullur, 2008). However, a certain level of confidence in the conceptual design must be achieved in order to mature the design in the preliminary design phase. Mass growth after conceptual design can either make or break a spacecraft and often incorrect assumptions about the technology, integration, and configuration of the spacecraft contribute to mass growth and reduced performance (Thompson *et al*., 2010).

Risk management procedures are used to mature the design in order to mitigate issues of technical, programmatic, or safety risk. The limitation of this approach is that it requires additional design effort beyond the conceptual design phase to accept risk or force additional reliability testing of components if targeted goals are not achievable. The uncertainty with spacecraft integration can lead to concepts that are “over-designed” while adding a layer of unneeded complexity to the entire development process. The goal of any conceptual design activity should be to provide a quality design that will meet the mission objectives and safety requirements.

1.1.1 Background and Purpose of Study

At the start of this research, the Constellation program was in full swing and NASA was busily designing the early concepts for the yet to be named Altair Lunar Lander. A question given to the Bioastronautics group at the University of Colorado was to determine the minimum functional mass for a lunar ascent vehicle based on NASA’s Exploration Systems Architecture Study (ESAS) concepts. In the fall of 2006, a team of graduate students embarked on the development of a full scale Lunar Lander Habitat mockup (Higdon and Klaus, 2008) and human spacecraft conceptual design process to perhaps assist NASA or other industry partners with design processes and information that would be useful in the development of conceptual human spacecraft. The initial prototyping activity was the beginning of this thesis and helped to shape the following research activities. Over the past 6 years, this research has expanded into a human spacecraft design project class and integrated into the Bioastronautics curriculum in the Department of Aerospace Engineering Sciences at the University of Colorado.

The research described in this work is focused on the conceptual design phase of human spacecraft to understand how designers should approach the development of early conceptual human spacecraft and the integration of the subsystems and components in order to maximize *Safety* and *Operability* without over designing the mass through unneeded complexity. A design methodology is presented that enables spacecraft designers to investigate a tradespace in order to minimize total spacecraft mass, increase *Safety*, and maximize mission *Operability*. Four key areas of research were investigated to create a design methodology focused on the conceptual design of human spacecraft. These four areas are:

1) Define a Minimum Functionality design methodology based on the four fundamental parameters of *Physics*, *Physiology*, *Safety*, and *Operability* in human spacecraft designs. (Chapter 3)
2) Examine rapidly reconfigurable physical prototyping methods for defining human factors early in the conceptual design and exploring subsystem integration uncertainties. (Chapter 4)
3) Develop mass addition guidelines and tradespace exploration methods to identify regions in the objective tradespace for future investigation and concept development. (Chapter 5)
4) Develop two figures of merit named the "Safety Index" and "Operability Index" for comparing spacecraft configurations without knowledge of subsystem and component reliabilities. (Chapter 6)

The combination of these four research activities form a design methodology based on a minimum functionality approach in order to explore the highly coupled aspects of a spacecraft design, reduce uncertainty in the configuration, and provide much needed information to decision makers.

1.1.2 Theoretical Framework

The design methodology of this research is in agreement with NASA guidelines as described in Miller *et al*. (2008), but the difference in this research compared to the NASA practices is “how” designers should approach the problem of conceptual human spacecraft development. Based on practical experience with the development of conceptual designs for human spacecraft subsystems, much of the problem in the spacecraft design process is guiding spacecraft designers in the early stages and providing an understanding of how design changes affect the overall performance, *Safety*, or *Operability*.

It is commonly known that the conceptual design phase is the most important phase for determining the overall cost of a program (NASA, 1995; Adelstein *et al*., 2006; Miller *et al*., 2008). According to NASA, conceptual designs are offered to demonstrated feasibility and support programmatic estimates (NASA, 1995). However, the processes used in conceptual design are often unstructured and not well understood (Jilla and Miller, 2004; Pacheco *et al*., 2003). In addition, the conceptual design phase is difficult to translate into a methodology that is useful to both experienced and inexperienced designers (Bryant, *et al*., 2005). Because of the lack of standardized process and difficulty communicating ideas, the conceptual design phase contains many uncertainties (Hastings and McManus, 2004; Chudoba and Huang, 2006; German and Daskilewicz, 2009).

Predicting mass growth following the conceptual design phase has been a common problem for aerospace systems. Many programs were not as successful due to mass growth, technical difficulties, and incorrect assumptions during the conceptual design phase (Thompson *et al*., 2010). As observed with the recent cancellation of the Constellation program, mass growth in Orion and Ares I was attributed with overcoming technical challenges that had not been anticipated in conceptual design (Chaplain, 2008). During the Apollo program, the Lunar Module experienced a 50% growth in mass from the initial conceptual design while the Apollo Command and Service Module experienced a 42% increase in mass during its development (Kelly, 2001; Thompson *et al.*, 2010).

Multidisciplinary Design Optimization is an area of research that develops systematic approaches for the design of complex systems governed by interacting physical phenomena (Alexandrov, 2005). When first introduced in the 1990's, the use of MDO methods for complex engineering problems had great promise for solving and optimizing problems. However, the true use of MDO has been limited to mostly researchers and has not been widely used in realistic engineering problems (Alexandrov, 2005). The central challenge in MDO is balancing the use of coupled high-fidelity models with the amount of computational time required to generate an optimized solution (Messac and Mullur, 2008).

Tradespace exploration is a method used to explore the various objectives in order to understand the relationship to the design space variables. According to Ross (2006), tradespace exploration in the conceptual design phase may empower designers to overcome challenges associated with tendencies to reduce the design space and overlook potential design space solutions. Ross (2006) conducted interviews with industry and learned that broad tradespace exploration is rare and often done in an ad hoc manner.

The findings by Tumer *et al.*, (2005) describe issues with NASA approaches for incorporating risk based design decisions early in the conceptual design process. An earlier NASA report by Knight and Stone (2002) also highlighted many of the needs within NASA for risk based design. Knight and Stone (2002) suggested that risk based design methods make the design process more robust given that a systems-level understanding is incorporated and detailed knowledge about the subsystems is utilized.

The typical approach used by NASA for understanding risk in the early stage of conceptual design is through the use of failure and risk analysis methods. A common method used within NASA to assess risk is PRA. Probabilistic Risk Assessment was identified as a need in 1996 in order to support decisions for Space Shuttle upgrades (Stamatelatos, 2002). The use of PRA identifies what can go wrong, how frequently it will happen, and what are the consequences (Tumer *et al*., 2005). The issue with PRA is that it requires a significant amount of information before a detailed analysis of the risk can be completed.

Minimum functionality design is a design approach where a baseline configuration is defined before trading other factors in human spacecraft designs. Minimum functionality recently gained attention due to its use by the Altair Lunar Lander project. The design approach for minimum functionality in the development of the Altair Lunar Lander started with a single point baseline design point of departure for cost and risk trades in order to justify mass add-backs to the subsystems in the form of additional redundancy and safety (Cohen, 2009). In addition, the Orion Crew Exploration Vehicle used a similar approach of creating a baseline design with a set of minimum functions called the Zero Based Vehicle (ZBV) (Hu *et al*., 2008). According to NASA's Design, Development, Test, and Evaluation Considerations for Safe and Reliable Human Rated Spacecraft Systems, a minimum functional design is the simplest, most robust, and highest performance design option as the starting point for assessing fault tolerance (Miller *et al*., 2008). Much of the confusion between different minimum functionality approaches is how the minimum or starting point configuration is defined before trading other aspects of safety, reliability, performance, and cost.

A human lunar spacecraft is very much different from other spacecraft designs such as capsules and lifting bodies. Although there are many similarities to conventional Low Earth Orbit (LEO) operational spacecraft, a lunar spacecraft must operate in a different set of environments on the surface of the moon including thermal cycling, radiation, dust, lighting, and micrometeoroids (Cohen, 2009). Significant lessons were learned during the Apollo missions that will assist designers in the development of future lunar spacecraft.

The Apollo Lunar Module (LM) was conceived, designed, and manufactured by the Grumman Corporation (Kelly, 2001). The most remarkable aspect of this achievement is that the LM evolved from a conceptual idea to operational hardware on the lunar surface in a period of less than 9 years. The conceptual design launch weight of the LM was initially proposed at 22,000 lbs and grew to 33,000 lbs by the time of Apollo 11 (Kelly, 2001). The major factors that drove LM mass during the preliminary design phase were reliability requirements, mission operational requirements, and configuration definition.

The proposed design objectives of the Altair Lunar Lander were very different from the Apollo LM. The issues with minimizing the mass of the Lunar Lander were well known at the time of the Exploration Systems Architecture Study report and led to the minimum functionality design approach. The technological differences between the Apollo and Constellation programs were significant in regards to the amount of operational capability that was to be included in the Altair design. Because the objectives of the Constellation program were more challenging than Apollo, the large number of development projects was one of the biggest hurdles in the Constellation program (Chaplain, 2009).

1.2 OBJECTIVES

The objectives of this research are to explore the conceptual human spacecraft design process in order to develop methodologies for evaluating and comparing spacecraft configurations during the early stages of development. The minimum functionality design methodology is the primary focus of the research. Within the overall design process, four key areas were investigated.

1. *Chapter 3*: Define a Minimum Functionality design methodology based on the four fundamental parameters of *Physics*, *Physiology*, *Safety*, and *Operability* in human spacecraft designs including:
    a. Defining the minimum functional baseline functions and mass of a given spacecraft configuration,
    b. Defining the functions and components associated with *Safety*, and
    c. Defining the functions and components associated with *Operability*.

2. *Chapter 4*: Utilize rapidly reconfigurable physical prototyping in parallel with minimum functionality design to explore subsystem integration, human factors, and design communication. Specific methods of this objective include:
    a. Functionally decomposing spacecraft subsystems,
    b. Using physical and physiological relationships to define a minimum functionality baseline configuration,
    c. Incorporating a rapidly reconfigurable prototype as a means of exploring configurations in a full scale, hands on manner as a pre-validation step toward developing Computer Aided Design (CAD) models, and

d. Providing a point of departure for mass additions to address levels of uncertainty and degrees of risk and reliability.

3. *Chapter 5*: Develop mass addition guidelines and tradespace exploration methods to identify regions in the objective tradespace for investigation and concept development. Specific methods of this objective include:
   a. Quantifying the minimum functionality baseline configuration in different types of lunar ascent spacecraft,
   b. Determining spacecraft mass at various levels of *Safety*,
   c. Determining spacecraft mass at various levels of *Operability*, and
   d. Defining preferred regions in the spacecraft configuration tradespace based on the objectives of minimizing total spacecraft mass, increasing *Safety*, and increasing *Operability*.

4. *Chapter 6*: Develop two figures of merit named the "Safety Index" and "Operability Index" for comparing spacecraft configurations without knowledge of subsystem and component reliabilities. Specific methods of this objective include:
   a. Comparing the Safety Index score between spacecraft configurations as an indicator of increased redundancy and safety components.
   b. Comparing the Operability Index score between spacecraft configurations as a measure of meeting top level mission objectives.

1.2.1 Importance of Research

This research described in this dissertation is a minimum functionality design methodology that could be utilized by NASA or industry as an effective means for reducing uncertainty and understanding risk before establishing design requirements. Uncertainties in the conceptual design phase cannot be completely eliminated and the focus of this phase is to explore as many configurations as possible to reduce the amount of uncertainty in the design assumptions. The greatest risk in any human spacecraft development process is the use of incorrect design assumptions that create unnecessary requirements and increase complexity. The scope of this research is intended to bring designers back to the fundamentals of spacecraft design and propose a simple and easy to implement process for reducing uncertainty and risk in conceptual human spacecraft.

This research is being presented during a time of change at NASA. The previous government "oversight" approach is being changed to a new paradigm of "insight" human spacecraft design and development where NASA partners with industry in the development of human spacecraft. The recent awarding of Commercial Crew Development (CCDev) contracts represents a fundamental shift in the way NASA plans to conduct business in the future. Because of political and economic factors, companies who develop commercial spacecraft must utilize efficient methods that reduce the amount of development time.

1.2.2 Scope of Research

Maturing a conceptual design to a working design requires a large amount of information. In the early stages, designers use limited information in order to develop a concept that will be iterated as the design matures. A common theme in this early phase is that unknowns in the design will be further investigated upon the results of detailed analysis. Although this approach gives a designer the flexibility in the initial phases of the design, this seemingly logical method also "locks" in a design too early based on previous assumptions and limited information about new technology. A danger of this approach is that previous assumptions are carried over into further phases of the design without proper communication as to the intent of the design logic.

When a designer makes assumptions in order to begin the design process, there is a certain amount of risk that is carried in the design until more information becomes available through higher fidelity analysis, testing, or demonstration. An experienced designer will anticipate changes and potential shortfalls, but this practice is often unstructured and varies among different designers and design teams. A design that is too heavy on risk reduction would likely be too heavy for the launch vehicle or propulsion subsystem; and a spacecraft that is at a bare-bones minimum mass would likely be too unsafe to fly. Instead of focusing on single point concepts, designers should consider the limitations, risks, and uncertainties of many potential design choices. This research describes the following studies to advance design approaches and methods in the early conceptual design of human spacecraft.

- Definition of a Minimum Functionality design methodology for quantifying *Safety* and *Operability* mass in human spacecraft configurations (Chapter 3).
- Investigation of rapidly reconfigurable prototypes for reducing risk and uncertainty in subsystem integration and human factors configurations (Chapter 4).
- Development of guidelines for mass addition and tradespace exploration to evaluate safety and operational functionality in conceptual human spacecraft (Chapter 5).
- Definition of two figures of merit for scoring *Safety* and *Operability* in human spacecraft configurations (Chapter 6).

This author has firsthand experience with conceptual design in the framework of the traditional government "oversight" development processes and with the government "insight" assisted development processes in CCDev. Both approaches to design and development are very similar in a technical sense; but the main difference between the two is related to economic factors. It is hoped that the research and processes developed in this work can be used to guide the designers of future human spacecraft.

CHAPTER 2

LITERATURE REVIEW

Although much has been learned during the past 50 years in human spaceflight, the risks associated with flying humans in space still remain. A human spacecraft is unique compared to other aerospace systems because of the complexity of integrating the human element in the spacecraft while simultaneously optimizing subsystem components. Human spacecraft must meet minimum safety requirements for risk and reliability in order to mitigate potential contingencies and bring the crew home safe.

The following literature review summarizes the background and design processes associated with the development of human spacecraft. A review of Systems Engineering in a NASA human spaceflight context followed by Conceptual Design, Multidisciplinary Design Optimization, Minimum Functionality, and Risk Based Design are included in this chapter. Because the application of this research is focused primarily on the design of conceptual Lunar Ascent spacecraft, a review of the Apollo and Constellation Lunar Lander designs is presented.

2.1 SYSTEMS ENGINEERING

Since the 1960's, NASA has utilized Systems Engineering (SE) principles and methods in the design, development, and operation of human spacecraft. The role of Systems Engineering practices in aerospace systems has evolved and is now firmly established in the aerospace industry. However, the early conceptual design phase within a Systems Engineering framework is an unorganized process that relies on human creativity to foster ideas and concepts for exploration and maturity. Thus, the challenge of designing for uncertainty, risk, human factors, and extreme environments dictates the need for detailed systematic conceptual design methods that are closely coupled to Systems Engineering practices. A large majority of design and product development literature recognizes the importance of decision making during the conceptual design phase and several approaches have been developed to assist design engineers with tools, methods, and processes to reduce uncertainty in conceptual design.

2.1.1 Systems Engineering Background

Systems Engineering extends product development beyond traditional engineering analysis and some might argue that Systems Engineering is a "management" philosophy. This viewpoint is likely due to the emergence of SE as a distinct discipline associated with the management of technological projects during and after World War II (Emes *et al*., 2005). As noted by Mumford and Bishop (2004) on the role of SE in Extra Vehicular Activity (EVA) design, they describe SE as a "catch-all" for all the functions that do not fit nicely in the traditional engineering disciplines and that SE is usually responsible for all of the "-illities" such as manufacturability, maintainability, and operability.

A thorough review of the many definitions and scope of SE was given by Emes *et al*., (2005). These authors point out that the International Council on Systems Engineering (INCOSE) definition of Systems Engineering "includes no description of what is meant by a system and has no reference to engineering; it also makes no assumption that SE is relevant only to machines or technical systems. In addition, Emes, *et al.*, (2005) describe the need to actively "brand" Systems Engineering as its own engineering discipline. However, they also recognize the difficulties with this viewpoint because SE overlaps fields of Operations Research, Systems

Analysis, Project Management, System Dynamics, and Soft Systems Methodology. Because of the lack of a common definition of SE, many organizations have created their own interpretation of SE and the wording of the various definitions is not consistent (Emes *et al*., 2005). For example, NASA describes Systems Engineering as:

> *"A methodical, disciplined approach for the design, realization, technical management, operations, and retirement of a system. – Systems engineering is the art and science of developing an operable system capable of meeting requirements within often opposed constraints" (NASA, 2007a).*

INCOSE defines Systems Engineering as:

> *"An interdisciplinary approach and means to enable the realization of successful systems. It focuses on defining customer needs and required functionality early in the development cycle, documenting requirements, then proceeding with design synthesis and system validation while considering the complete problem"(INCOSE, 2010).*

As observed by the difference between the NASA and INCOSE descriptions, a single definition of Systems Engineering is not universal. The authors of the NASA Systems Engineering handbook agree that there are differences of opinion and interpretations of Systems Engineering. The commonality between the many interpretations and definitions is that SE is an approach used by an organization to meet customer needs and requirements.

Adding to the difficulty of defining exactly what is SE, the adoption of SE has also been restricted by its limited appeal to universities "because Operations Research and Systems Engineering borrowed their methods from other disciplines and were common sense – that is, procedural – their claims to academic legitimacy were tenuous" (Johnson, 1997). Although some might argue for or against this viewpoint at the university level, the main question is how do students learn SE fundamentals? The most effective method for students to learn SE fundamentals is through experience with design projects, courses, or intern job experience. Regardless of the different interpretations, students who plan to work or research in the aerospace industry will need to be familiar with Systems Engineering principles.

NASA approaches SE as a robust approach to design, create, and operate systems (NASA, 1995). The objectives are to verify that the system is designed, built, and operated so that it accomplishes its purpose in the most cost-effective manner, considering performance, cost, schedule, and risk. These four measures are the main attributes to consider in the development and operation of a system. Shown in Figure 1 is an example of how Performance, Risk and Cost are coupled in Systems Engineering.

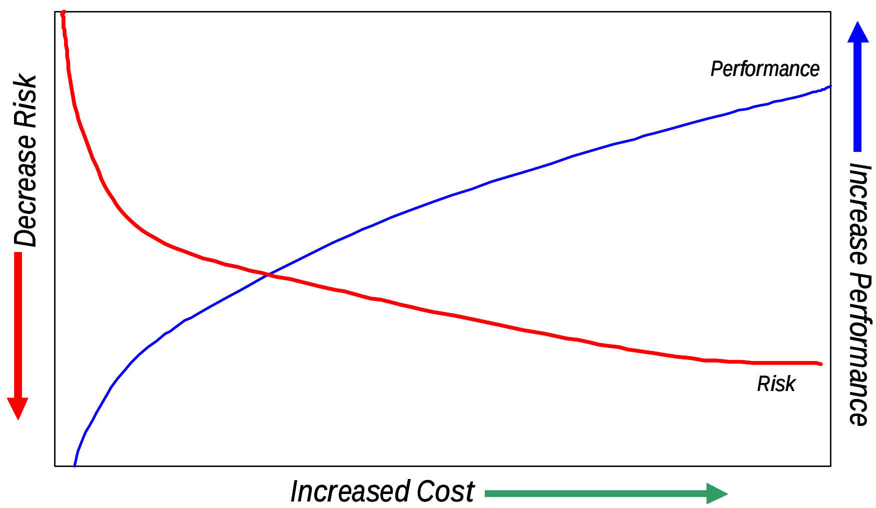


Figure 1: Systems Engineer's dilemma.

Based on these relationships, it is obvious that to achieve increased performance requires higher costs to reduce risk. Schedule can be thought of as a 'cost' and is closely coupled to overall cost. Strictly focusing on the engineering design (performance) in a traditional engineering discipline approach will neglect the other measures of cost, schedule, and risk. Thus, a decision maker must trade options when one of the measures presents a challenge to the entire system. The relationship between the four measures is described as the "Systems Engineer's Dilemma" (NASA, 1995). These relationships are summarized as:

- To reduce cost at constant risk, performance must be reduced;
- To reduce risk at constant cost, performance must be reduced;
- To reduce cost at constant performance, higher risks must be accepted, and;
- To reduce risk at constant performance, higher costs must be accepted.

The management of the multidisciplinary and complex engineering systems as they are matured and developed in a Systems Engineering context is crucial to the reduction of cost while balancing performance and risk for the overall system. If performance measures are not met within a specified schedule and cost, the mission might not be feasible or could add significant risk to the program.

2.1.2 Systems Engineering Development Phases

The entire Systems Engineering development process is divided into a series of life cycle phases that are intended to mature design concepts and reduce uncertainty through a series of technical reviews. The NASA Procedural Requirements NPR 7120.5D defines the major phases of the project life cycle as:

- Pre-Phase A – Advanced Studies
- Phase A – Preliminary Analysis
- Phase B – Definition
- Phase C – Design
- Phase D – Development

- Phase E – Operations
- Phase F – Decommissioning

Major reviews are conducted at the end of each phase to verify the design or plans are acceptable for moving into the next phase. The most common reviews include the Mission Concept Review (MCR), System Requirements Review (SRR), System Definition Review (SDR), Preliminary Design Review (PDR), Critical Design Review (CDR), Test Readiness Review (TRR) and Flight Readiness Review (FRR). These reviews form "Control Gates" for decision makers to assess the measures of the system moving forward. Shown in Table 1 are the major "Control Gates" in each Life Cycle phase (NASA, 2007a).

Table 1: Control gates for Life Cycle phases.

Pre-Phase A	**Phase A**	**Phase B**	**Phase C**	**Phase D**	**Phase E**	**Phase F**
Mission Concept Review (MCR)	System Requirements Review (SRR)	Preliminary Design Review (PDR)	System – Level Critical Design Review (CDR)	Test Readiness Review (TRR)	Post Launch Assessment Review (PLAR)	Decommissioning Reviews
	System Definition Review (SDR)		Production Readiness Review (PRR)	System Acceptance Review (SAR)	Critical Events Readiness Review (CERR)	
			System Integration Review (SIR)	Operational Readiness Review (ORR)	Post Flight Assessment Review (PFAR)	
				Flight Readiness Review (FRR)		

The formulation of the design ends at PDR when the design enters an implementation phase. A baseline concept of system requirements, verification requirements, concepts of operations, design specifications, and project plans are presented as part of PDR at the end of Phase B. According to the NASA SE Handbook (1995), if costs are underfunded in Phases A and B, then overruns are likely in the rest of the program. Thus, it is critical to examine all aspects of a spacecraft design to avoid having to "re-design" the spacecraft later in Phase C.

During Pre-Phase A, the purpose of the initial activity is to "uncover, invent, create, concoct, and devise" a broad spectrum of ideas and alternatives for missions from which new projects can be selected (NASA, 2007a). This phase is where concepts are first identified and explored. Technology needs and readiness levels are assessed in order to provide information for a Mission Concept Review (NASA, 2007b).

During Phase A, the feasibility of a new conceptual system is evaluated. The top level requirements are developed and demonstrations of credible, feasible designs are presented. In addition, the necessary systems engineering tools and models are acquired. The activity in Phase A reexamines the information gained from Pre-Phase A to provide justification for placing a new system in NASA's budget. A central goal of this phase is to analyze mission requirements and establish mission architecture (NASA, 2007b).

The purpose of Phase B is to establish the initial system baseline that includes a decomposition of the system and subsystem design specifications for both flight and ground elements (NASA, 2007a). The establishment of a baseline system is used to project schedule, cost, and business management plans. Configuration management procedures are implemented beginning with the new baseline. To decompose the spacecraft into subsystems, a functional decomposition is used to identify functions that map to specific hardware, software, and personnel. A significant amount of time in this phase is dedicated to tradespace exploration of designs and architectures. These studies are iterative processes that analyze information at the system and subsystem levels. In addition, risk drivers are identified and proposed mitigation plans are developed for each risk (NASA, 2007b).

Before a baseline design is chosen, operational concepts should be validated by a level of technical detail that is beyond the level of detail of Phase A. Subsystem and System level PDRs are held close to the end of Phase B. These reviews examine the processes and analyses used to develop the baseline design requirements and "design-to" specifications. Any issues identified at PDR should be addressed with specific plans before the spacecraft design progresses into Phase C, Final Design and Fabrication.

2.1.3 System and Subsystem Decomposition

Decomposing a spaceflight mission into various spacecraft and launch vehicles makes the design of the overall mission architecture manageable. Using the Constellation lunar mission as shown in Figure 2, the following systems were planned for the lunar mission architecture:

- Ares V launch vehicle,
    - Includes Earth Departure Stage and Lunar Lander (LL)
- Earth Departure Stage,
    - Coupled to Lunar Lander and Orion Crew Exploration Vehicle (CEV)
- Lunar Lander,
    - Includes Ascent and Descent Stages
- Orion CEV, and
    - Includes Command and Service Module
- Ares I launch vehicle.
    - Includes Orion CEV

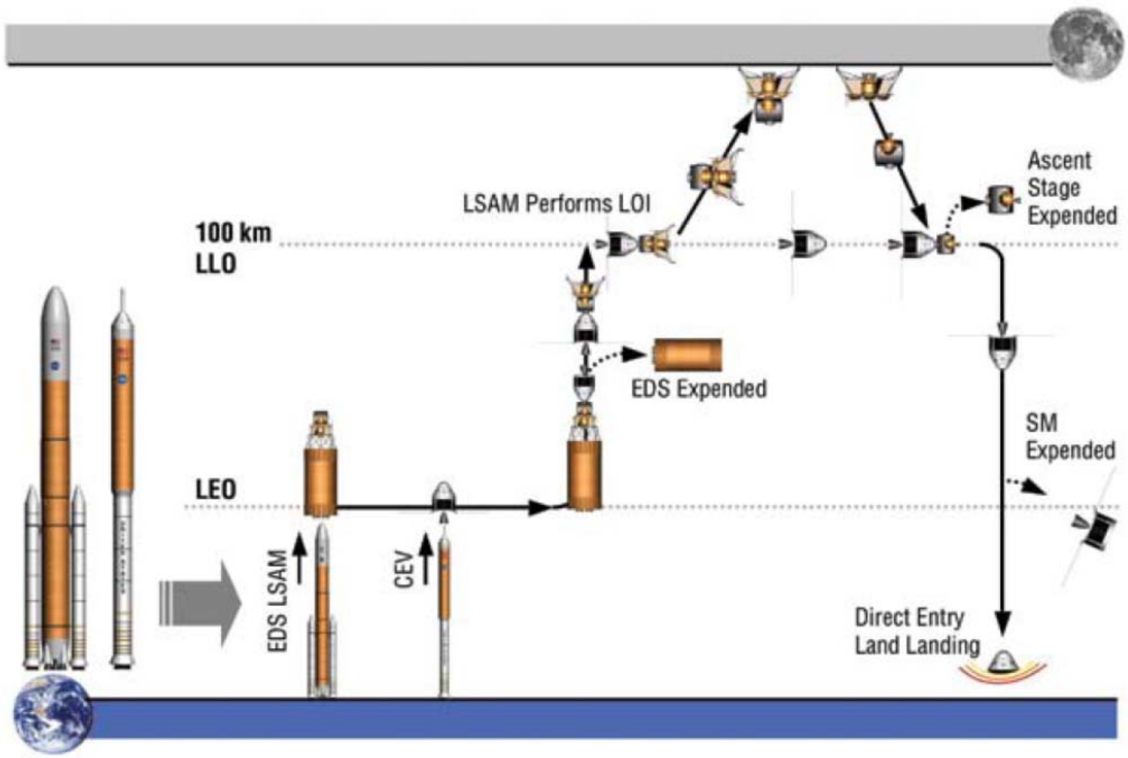


Figure 2: NASA Constellation lunar sortie mission (NASA, 2005).

Upon review of the Constellation lunar architecture, it is obvious that all the systems are highly coupled and must work together in order to accomplish the overall goals of the lunar mission. The decomposition of a system leads to segments which could be defined as a part of the larger system. For example, if the overall lunar mission architecture is considered a system, then the top-level vehicle architectures are segments, such as Ares V.

In the decomposition of the vehicle architecture, it is obvious that a change in the Lunar Lander weight or configuration would affect the entire lunar architecture. Increases in mass for the Lunar Lander affect not only the Ares V, but also the Earth Departure Stage. Docking requirements between the Lunar Lander and CEV could increase the weight of the CEV, thus affecting the launch weight of the Ares I launch vehicle. Spacecraft designers are aware that changes in one small part of a subsystem can have a large effect on the overall mission design and architecture. In the design of the Apollo Lunar Ascent Module, an increase of 1 lb. increased the mass of the Saturn V launch vehicle by 833 lbs. (Thompson *et al*., 2010).

2.1.4 Systems Engineering Summary

Systems Engineering is a methodology utilized by NASA in the design, development, and operation of human spacecraft. The exact definition and scope of Systems Engineering varies among different organizations, but the central goal is to develop products that meet a customer's needs. In the example of human spacecraft, the various systems and subsystems are highly coupled in terms of performance, cost, risk, and schedule. Thus, a Systems Engineering approach takes a broader view of the entire design in order to balance tradeoffs of the four key parameters. Systems Engineering is more than a traditional management approach, it serves as a leadership role to guide the traditional engineering disciplines in order to integrate the entire system or architecture into a workable solution that meets customer's needs.

2.2 CONCEPTUAL DESIGN

The conceptual design phase is where ideas are first conceived, developed, and explored as potential solutions to meet a customer's need. It is commonly known that the conceptual design phase is the most important phase for determining the overall cost of a program (NASA, 2007a; Adelstein *et al*., 2006; Miller *et al*., 2008). According to NASA, conceptual designs are offered to demonstrated feasibility and support programmatic estimates (NASA, 2007a). The overall purpose is to further examine the feasibility and desirability of suggested new major systems before obtaining funding. In a NASA SE context, Pre-Phase A develops ideas and concepts for future missions and programs and Phase A and B activities are where the proposed conceptual design solutions are developed and explored before PDR. According to Rowell *et al*., (1999), "*the difficulties of early conceptual design are characterized by a low level of system definitions and at this stage the conceptual design parameters are often not well modeled or understood.*" Thus, the need to understand the overall impact of uncertainty, design space, and tradespace exploration is what makes the conceptual design phase very important in terms of designing for additional flexibility as the design matures.

2.2.1 Conceptual Design Uncertainties

The conceptual design phase is one of the least structured in terms of standardized processes and procedures. Jilla and Miller (2004) describe the conceptual design phase as unstructured and with designers "*often pursuing a single concept or modifying an existing idea rather than generating new alternatives.*" McManus *et al*. (2004) also reinforce this viewpoint by describing that traditional approaches used in the U.S. aerospace industry rely on methods that settle on a preferred design early in the process without tools that consider many potential designs. In addition, the approaches used in the conceptual design phase are limited to the aspect of how needs might change during development and operation (McManus *et al*., 2004). The lack of flexibility in later stages of design is one of the central issues with typical conceptual design approaches. Pacheco *et al*. (2003) describe the conceptual design space as not well understood. In their approach of using Bayesian surrogates for uncertainty in design parameters, they address the issue of incomplete knowledge and relationship between the design parameters and the overall system (Pacheco *et al*., 2003). Bryant, *et al.* (2005) describe in their research of conceptual design that the concept generation phase of the design process is difficult to translate in a methodology that is useful to both experienced and inexperienced designers. In their words,

> "*Design success is often heavily weighted on personal experience and innate ability and methods such as brainstorming, intrinsic, and extrinsic searches and morphological analysis are designed to stimulated a designer's creativity, but ultimately still rely heavily on individual bias and experience*" *(Bryant et al., 2005).*

Because the conceptual design phase is where ideas are first explored, standardized processes and procedures are needed to fully explore the design tradespace. A common standardized approach to reducing uncertainty in conceptual design is not universal and is very difficult to quantify because of the multidisciplinary behavior. In a study by Hastings and McManus (2004), the authors categorize the types of uncertainties in the conceptual design phase and the effects on the overall system. These uncertainties include:

- lack of knowledge about the system,
- lack of definition in the system,
- statistically characterized variables or phenomena,
- known unknowns, and
- unknown unknowns.

Hastings and McManus (2004) also point out that Uncertainty causes Risk handled by Mitigation resulting in Outcomes. One can easily follow the logic that carrying uncertainty throughout the conceptual design phase will ultimately lead to risk in later development phases. As noted by Chudoba and Huang (2006), the uncertainty in design knowledge in the early conceptual phase can be related to the lag of design knowledge available to the design knowledge required. Their recommendation is to create a design knowledge based database system to prevent the loss of knowledge in aerospace designs (Chudoba and Huang, 2006). German and Daskilewicz (2009) associate the problem of aerospace conceptual design as a "wicked" problem described in the field of urban planning as "*complex, highly uncertain, and solution dependent dilemmas that do not necessarily have one best answer.*" According to Scheidl and Winkler (2010), the conceptual design phase "*should be supported by good models for the design map, which in turn expresses the relationship between the design parameters and the functional requirements.*" Their work presented a theory on how conceptual models are related to detailed design models and how "good" the conceptual models can be characterized. This research concluded that "*since product development typically envisages a wider range of final product realizations, the knowledge of the validity range of these models is important for a successful systematic design*" (Scheidl and Winkler, 2010). A successful conceptual design must explore the design space and objective tradespace efficiently in order to anticipate future changes in the design. The challenges in conceptual design are compounded in complex engineering systems where multidisciplinary approaches are required for developing design solutions.

Much has been written about processes and methodologies intended to improve the conceptual design process for various applications (Jilla and Miller, 2004; McManus *et al.*, 2004; Pacheco *et al.*, 2003; Bryant *et al.*, 2005; Hastings and McManus, 2004; Chudoba and Huang, 2006; German and Daskilewicz, 2009; Scheidl and Winkler, 2010). Among the vast amount of literature dealing with various approaches in conceptual design, three common themes appear:

- The need to fully explore design concepts and ideas,
- The need to increase knowledge to reduce uncertainty, and
- Development of reduced order modeling.

The large amount of uncertainty and lack of overall system definition contributes to the difficulty of standardizing methodologies and processes for a wide variety of conceptual design problems. However, the common themes among the various approaches in the literature all seem to recognize the need to reduce uncertainty through exploring a wide variety of design concepts and perform the work in a manner that does not require high fidelity modeling development and analysis.

2.2.2 Top Level Mission Objectives

Before starting the conceptual design activity, the top level mission objectives or customer's needs must be clearly defined. This is the foundation for what is expected in the design in terms of performance, risk, schedule, and cost. Establishing firm mission objectives and goals at the beginning of conceptual design activity is crucial for the development and reducing technical and problematic issue later in the design. As observed in the recently cancelled Constellation program, significant redesigns beyond the conceptual design phase proved to be problematic for maintaining cost and schedule goals. According to a Government Accountability Office (GAO) report in 2009, "*NASA is now focused on providing the capability to service the International Space Station and has deferred the capabilities needed for flights to the moon*" (Chaplain, 2009). The change in the top level mission goals to focus on the ISS only and defer the lunar mission in 2009 represented a the fundamental problems in development of human spacecraft. Due to numerous technical issues associated with Orion and Ares I at the time and the critical need for servicing the ISS; this led to a shift in the top level objectives of the Constellation program. Although the original ESAS plan in 2005 outlined the steps toward the lunar mission as being a block upgrades from ISS to lunar mission architecture, the original design of the Orion capsule was primarily focused on a dual role for ISS and lunar missions (NASA, 2005). The GAO report in 2009 was a scathing report on the failures at NASA and the problems associated with maturing a conceptual design. According to the GAO report:

> *"Over the past decade, there have been a number of instances where NASA pursued costly efforts to build a second generation of reuseable human spaceflight vehicles without attaining critical knowledge about requirements and resources. These programs experienced significant problems – including cost and schedule delays. They include the National Aerospace Plane, the X33, and X-34, and the Space Launch Initiative, which were eventually cancelled. - By emphasizing heritage technology, the Constellation program was designed to avoid problems associated with the prior shuttle replacement efforts, which were largely rooted in the desire to introduce vehicles that significantly advanced technologies" (Chaplain, 2009).*

In NASA's defense, the funding required to mature the early concepts from 2005 to 2009 was not enough to overcome the technical challenges. The GAO report in 2009 also highlights this issue by describing that the Constellation program was "poorly funded" and this affected the ability to deal with all of the technical challenges (Chaplain, 2009).

In addition to technical aspects of designing a human spacecraft, the enormous cost and risk largely decide the top level mission objectives. As observed in the Constellation program, when cost and risk were subjectively judged too high in a political atmosphere, NASA changed the top level mission objectives in order to keep the program alive. This change in mission objectives meant that redesign efforts were needed to remove the lunar requirements from Orion, thus creating multiple redesign activities in order to meet the new mission objectives.

The new era of spaceflight is much different from the Apollo program where the mission objectives were much simpler (Man, Moon, and Decade). In addition, the economic landscape for funding major human spaceflight programs is very different from the 1960's. Unfortunately, cost is a huge factor in the fate of many of NASA's programs (Chaplain, 2009). The challenge for spacecraft designers in today's human spacecraft development world is to mature concepts quickly and efficiently within the given top level mission objectives before cost and schedule drive significant redesign.

2.2.3 Functional Decomposition

Given a set of clearly defined top level mission objectives, the next logical step in the spacecraft conceptual design process is to determine the activities that need to take place for the mission to be successful. This is potentially one of the most crucial aspects of design where designers must understand what the system needs to do in order to explore potential solutions.

A functional decomposition process identifies and states the "functions" or the activities that need to take place in the overall system architecture during a given mission. The functional decomposition activity is not a single activity to be completed in the early stage of design. As the design concepts are matured, additional functionality can be added to the design to increase performance, reduce risk, and mitigate contingencies. Similar to the process used in system and subsystem decomposition, a functional decomposition maps individual functions to the lowest levels in the spacecraft subsystems. A key aspect of functional decomposition is that the activities of the spacecraft are defined in a process that is separate from how the functions will be performed. The solutions to the individual functions in the form of "how" is a large portion of the uncertainty in the spacecraft design. It is easy to determine "what" needs to be done, but the difficult part in spacecraft design is determining "how" the functions will be performed. Spacecraft designers consider heuristics, technologies, and materials that will eventually match to the individual functions. During Phases A and B, proof of concepts are studied and traded in order to establish a functional baseline concept. This activity is a form of technology and risk mitigation that evaluates uncertainties before Preliminary Design Review (NASA, 2007a).

The methods used by NASA for functional decomposition are detailed in the document Design, Development, Testing, and Evaluation (DDT&E): Human Factors Engineering (Adelstein *et al*., 2006). According to NASA, the purpose of identifying functions is that it helps to define requirements and resources that are coupled to the functions such as hardware, software, and human elements. The key steps in functional analysis are:

- Determine the objectives, performance requirements, and constraints of the design;
- Define the activities that must be accomplished to meet the objectives and required performance;
- Define the relationships between functions and subsystems; and
- Define tradeoff priorities and constraints (Adelstein *et al*., 2006).

The development of functional models is one of the first steps in systematic methods for product development (Cowan *et al*., 2006; Pahl and Beitz, 1996; Ulrich and Eppinger, 2000, Roy *et al*., 2008; Kitamura and Mizoguchi, 2003). According to the Department of Defense (DoD) Systems Engineering Fundamentals (2001), functional decomposition is described as a "*top-down process of translating system level requirements into detailed functional and performance design criteria*" (DoD, 2001). This interpretation is slightly different from NASA's definition of functional decomposition, but the same approach is used to define the functions and expected performance at lower levels of the subsystems.

One of the drawbacks associated with functional decomposition identified by Kitamura and Mizoguchi, (2003) is that the knowledge about functionality is usually scattered among designers in the conceptual design phase because the individual designers are concerned about how their specific function will perform the prescribed sub-(micro) function. Knowledge about functionality should be shared among different groups and this viewpoint is reinforced by Ingham *et al*., (2006) who state that subsystem level functional decomposition fails to "*scale in the tangled web of interactions typically encountered in complex spacecraft designs.*" In research conducted by Ingham *et al*. (2006), they propose a "State Analysis" for the development of system and subsystem explicit models. A recent study by Camelo and Mulet (2010) offers a method that examines the relationships between functions in order to address the cross coupling aspect of functional decomposition. However, they recognize that not all of the relationships between all of the functions can be fully explored to prevent "*combinatorial explosion*" (Camelo and Mulet, 2010). Regardless of the challenges, functional decomposition remains a key process in the conceptual design phase to understand the fundamental needs of the spacecraft design in order to properly trade potential solutions.

2.2.4 Boundary Objects

Obtaining knowledge in the conceptual design phase is a key activity in the investigation of solutions that match to a functional decomposition. According to Carlile (2002), knowledge is a "*barrier to and a source of innovation.*" To describe how knowledge can flow across boundaries, Star (1989) coined the term "Boundary Objects". Boundary Objects are representations of knowledge used to improve communication and understanding between different groups (Star 1989; Carlile 2002; Carlile 2004; Dare *et al*., 2004;). Star (1989) categorized four types of boundary objects that are shared and sharable across different problem solving contexts. These four types are:

- Repositories – Cost databases, CAD databases, parts libraries;
- Standardized forms and methods – report findings, problems solving;

- Objects or models – sketches, drawings, prototypes, mockups, simulations; and
- Maps of boundaries – Gantt charts, process maps, workflow matrices.

According to Carlile (2002), the use of objects or models depicts or demonstrates current or possible "form, fit, and function" of the differences and dependencies identified at the boundary. A boundary object is considered "good" if it can represent knowledge in a manner that is easily used in a problem solving setting. Carlile (2002) identified three characteristics of a boundary object that made it useful in problem solving.

- An effective boundary object establishes a shared syntax or language for individuals to represent their knowledge.
- An effective boundary object provides a concrete means for individuals to specify and learn about their differences and dependencies across a given boundary.
- An effective boundary object facilitates a process where individuals can jointly transform their knowledge.

These three characteristics were identified from a study of 65 observations of Boundary Objects (Carlile, 2002). It is easy to understand how Boundary Objects can be used to share knowledge among groups, but how can the concept be applied effectively in conceptual design? In a study conducted by Brereton and McGarry (2000) on engineering design thinking and communication, they proposed a design problem to groups of engineering undergraduate students to observe what methods the students used to solve the problem. Of their findings, they concluded that design thinking is heavily dependent upon experiences with physical objects and materials. In the design process, hardware or prototypes becomes a "compelling medium" for thinking. Students used simple objects to clarify their thoughts and ideas, thus making hardware the "object" that was used in communication. Listed below are seven reasons described by Brereton and McGarry (2000) for the use of hardware in engineering design.

- Hardware is tangible; it can be seen and touched.
- It gives physical presence to conceptual models.
- Its behavior reveals errors in conceptual models.
- It behaves in unpredicted ways which provokes the user to explore it.
- It behaves in different ways in different environments and different contexts of use.
- Interaction with hardware and integration of hardware components reveals properties and limits of the hardware and hardware components.
- It is integral to communications affecting the course of inquiry, idea generation, discovery, and the dynamics of group interactions.

Carlile (2002) describes that the role a Boundary Object assumes in new product development is that it helps to establish a "boundary infrastructure" or "boundary process". The main idea is that knowledge is communicated among individuals through a visual means. It is worth noting that Carlile (2002) also points out that CAD modeling can be a useful Boundary Object in one setting but is useless in another setting. In order for a Boundary Object to communicate knowledge, it must have the flexibility to adapt to changes in the design process.

2.2.5 Physical Prototyping

Physical prototypes, mockups or objects can be used to communicate information during conceptual design. According to a research study conducted by Brereton and McGarry (2000), the use of physical objects (hardware) and prototyping materials in support of design thinking and communication has the following roles:

- Hardware is a Starting Point.
  - Hardware is easily noticed, remembered, seen and touched.
  - It offers a basis for comparison.
- Hardware is a Chameleon.
  - Hardware is always in a context of use.
  - What the hardware reveals depends upon the context of use.
- Hardware is a Thinking Prop.
  - Hardware objects have all sorts of properties that afford different actions.
- Hardware is an Episodic Memory Trigger.
  - Episodes of experiences with physical objects serve as memory devices.
- Hardware is an Embodiment of Abstract Concepts.
  - Observing and testing hardware reveals fundamental concepts, physical embodiments of abstract concepts, and unanticipated design issues in hardware behavior.
- Hardware is an Adversary.
  - Challenging theoretical model predictions against hardware behavior reveals discrepancies and provides clues to modeling errors. This reveals theoretical assumptions and causal relations.

- Hardware is a Prompt.
  - Device behavior prompts student questions and suggests experiments. Through repetitive interaction with hardware, students bring order, distilling out key operational parameters, and their relationships.
- Hardware is a Medium for Integration.
  - Integrating components in their functional context reveals practical limits of use, characteristics of operation, methods of connection, causal relations, and physical quantities.
- Hardware is a Communication Medium.
  - Hardware is integral to learning communications, affecting the course of inquiry, idea generation, discovery and the dynamics of group interaction. Hardware is used to command attention, to demonstrate, and to persuade.

The findings of Brereton and McGarry (2000) emphasize the need for hardware as a tool for visualizing abstract concepts and exploring the design space. In a research study of adaptive design using system representations, Dare *et al*. (2004) defined Boundary Objects as "*representations of knowledge that can improve communication and understanding between groups or organizations with different lexicons and cognitive foundations.*" Dare *et al*. (2004) conducted eight case studies of Air Force programs and found the use of Boundary Objects to be effective in helping stakeholders' bridge knowledge boundaries and establish understanding. The use of prototypes as a Boundary Object facilitates communication between multidisciplinary groups and helps to resolve discrepancies before a design configuration is selected. Having a physical (full-scale) system that designers can see and touch reduces the possibility of misinterpretation where CAD drawings or other computer representations may be viewed incompletely or in a different ways by different members of the design team. Before the use of computational design methods, hardware and mockups were the primary means for understanding design concepts. During the design phase of the Apollo Lunar Module, many uncertainties were discovered in preliminary design with the use of mockups (Kelly, 2001). In today's fast pace computational design world, mockups are typically constructed after the conceptual design is nearing completion. However, the earlier new information is available, the higher its value (Krishnan *et al*., 1997).

A mockup of proposed lunar habitats was constructed by NASA Johnson Space Center (JSC) to demonstrate integration of several subsystems for near-term exploration habitats (Kennedy, 2006). Although mockups are not specifically required by NASA at PDR, the use of mockups is encouraged to allow testing and demonstration of technologies that would be useful for the mission. Kennedy (2006) noted in the development of the lunar habitat mockup, the value of the testing will be realized while developing the test. In his words, "*the sheer act of determining the interfaces and integration to bring the subsystems together will help to identify the gaps that must be overcome when designing the actual subsystems*" (Kennedy, 2006).

Research conducted by Mulenburg and Gundo (2004) described a method titled "Design-by-Prototype" to simplify the conceptual design process. This method is used by NASA Ames Research Center to create unique one-of-a-kind research hardware for small high risk projects by eliminating much of the formal engineering design process. Mulenburg and Gundo (2004) quote Frame (2002) as stating, "*…it is impossible to pre-specify requirements precisely*" and the use of prototypes early in the design process can help to validate design requirements through demonstration, experiment, or testing (Mulenberg and Gundo, 2004). Although prototyping is not a new concept, its use in different contexts is prevalent in the literature. The main reason for using mockups, hardware, or prototypes is that uncertainties in the conceptual design can be investigated in a manner that increases communication between multidisciplinary groups.

2.2.6 Computer Aided Design and Virtual Prototyping

Computer Aided Design is a well established engineering tool used in the design of structures and assemblies. Commercial software packages such as SolidWorks, Pro-Engineer, CATIA, and I-DEAS are powerful analysis programs for modeling physical (mass and geometry) characteristics. The disadvantage is that they are becoming more complex and require specialized training. Because of increasing computational abilities available to many companies, computational analysis packages are becoming the standard for mechanical design (Wang and Shan, 2007).

CAD models serve the purpose of representing mass and geometry of a design in a digital format. This information can be utilized by high fidelity analysis programs such as Finite Element Analysis (FEA) and Computational Fluid Dynamics (CFD) where the design is evaluated in simulated environmental conditions to predict responses. A vast amount of literature has been published on the topic of FEA and CFD. However, in the context of this research, the study of CFD and FEA is not the goal. Instead, this research will evaluate early design methods that validate the physical configuration of a human spacecraft using a human-in-the-loop for pre-validation of CAD models. The use of high fidelity analysis after conceptual design is necessary for obtaining higher degrees of accuracy, but limits the number of design configurations that can be evaluated. Wang and Shan (2007) reported that it takes Ford Motor Company about 36 to 160 hours to run one crash simulation with their FEA models. The huge computational expense has led many researchers to investigate the use of approximation models, described as metamodels, as an alternative to high fidelity and time intensive computational codes. Because of the large amount of time to develop CAD models and subsequent FEA models, the use of high fidelity tools is better suited for Phase B after conceptual designs have reached a level of maturity.

According to Cecil and Kanchanapiboon (2007), the use of CAD has been incorrectly used in the literature to represent virtual prototyping. Their definition of a virtual prototype is based on the following characteristics.

- Virtual prototypes must possess accurate geometry, topology, and appearance, reflecting characteristics of the target part, object, system or environment.
- Virtual prototypes should be capable of simulating engineering or science based characteristics including behavior with real time responses.
- Virtual prototypes are digital or computer based representations.
- Virtual prototypes must possess the ability to interface virtual reality technology and graphics including supporting semi-immersive or immersive applications (Cecil and Kanchanapiboon , 2007).

Upon review of the four characteristics, it is easy to determine that Virtual Prototyping is very much different from Computer Aided Design. Because CAD models are also digital representations of the parts and assemblies, this is most likely the cause of the misinterpretation in the literature. CAD models are limited in their ability for a user to immerse in the environment. Especially during the conceptual design phase, as CAD models are matured to explore configurations, they do not always possess accurate geometrical specifications due to incomplete knowledge about the subsystem interfaces.

As computing power increases and technology becomes available, the future of Computer Aided Design is likely to utilize Virtual Prototyping to a greater extent. According to Choi and Chan (2004), virtual reality can be used to determine bottlenecks in manufacturing or production without the use of expensive physical prototypes. The overall goal of virtual prototyping is to simulate in a digital world the many physical aspects of a design. CAD and Virtual Prototyping are tools that can be utilized in the conceptual design phase for maturing designs and communicating design knowledge using a Boundary Object.

2.2.7 Spacecraft Mass Growth

Predicting mass during the conceptual design phase is a common challenge for aerospace systems. The Space Shuttle and the Apollo Lunar Module are two examples of flight programs that experienced significant mass growth issues but were successful. Other programs such as X-30 and X-33 were not as successful due to mass growth and technical difficulties due to incorrect assumptions during the conceptual design phase (Thompson *et al.*, 2010). As observed in the Constellation program, mass growth in Orion and Ares I was attributed with overcoming technical challenges that had not been anticipated in conceptual design (GAO, 2008). During the Apollo program, the Lunar Module experienced a 50% growth in mass from the initial conceptual design while the Apollo Command and Service Module experienced a 42% increase in mass during its development (Kelly, 2001; Thompson *et al.*, 2010). The increase in payload mass proved to be difficult for the Saturn V as its first stage dry mass increased approximately 33% (Thompson *et al.*, 2010).

A recent review of mass growth in spacecraft conducted by Thompson *et al*. (2010) evaluated the reasons for mass growth after the conceptual design phase. This work summarizes the history and issues with mass growth in human spacecraft and the challenges for future designers. According to Thompson *et al*. (2010), manned spacecraft typically grow from 15% to 50% during development. Reasons for the growth include "*optimistic and inaccurate initial estimates, inadequate guidelines on mission specific modifications to the standards and significant subsequent changes to requirements and designs*" (Thompson *et al*., 2010).

To overcome the uncertainty in mass growth in early conceptual design, designers typically use industry and government guidelines to plan margin allocations. This approach utilizes a percentage of the mass based on the maturity of the design in the development cycle (Thompson *et al*., 2010). Although this serves to create allowable mass "boundaries" for identifying future mass impacts and potential threats, this method is highly dependent upon the first assumptions made during conceptual design. Thompson *et al*. (2010) recommend that industry guidelines should be used as initial guidance and that "*a structured and documented method should be employed to identify the mass and performance risks introduced through uncertainty due to lack of knowledge.*" Methods that consider the sensitivity of conceptual design mass growth should be developed and utilized earlier in conceptual design to reduce the likelihood of impacts on cost and schedule. A study conducted by Thunnissen (2004) also concluded that spacecraft mass increases approximately 50% from the conceptual design phase. This is very significant when compared to the accepted margin of 30% based on typical approaches (Thunnissen, 2004). Because aerospace systems are complex and multidisciplinary, not all of the uncertainty in the conceptual design can be quantified or predicted without testing and high fidelity analysis. The lack of knowledge during the early phase of design is a key contributor to uncertainty and the greater the uncertainty, the more likely that redesign efforts will be required as the design concept is matured.

2.2.8 Conceptual Design Summary

The conceptual design phase is a very significant phase in the development of human spacecraft. It is this phase where design decisions are made with limited information and assumed risk that decides the success of the spacecraft development program. Because of the lack of knowledge and the need to explore many design solutions in a time-efficient manner, many recognize this phase as one that is unstructured in terms of design processes to assist the designer.

A central need in any conceptual design process is clearly defined top level objectives that remain constant throughout the development process. Changing the top level objectives during the development program creates unnecessary redesign that usually requires a step back to the conceptual design process in order to redevelop the existing design. Once the top level objectives are clearly stated and defined, a functional decomposition of the activities in the spacecraft is created to assist with trade studies of technology choices to perform the functions. The functional decomposition serves as a direct link between the various subsystems to the top level objectives to specify what the spacecraft systems must do to perform a successful mission.

Increasing knowledge and communication during the conceptual design process has been identified as a key contributor to the success of the life cycle development. Design methods that utilize Boundary Objects to facilitate communication and knowledge among design teams are useful in terms of reducing uncertainty among teams with varying backgrounds. The use of physical prototypes and hardware is a powerful communication tool in the context of Boundary Objects. Hardware serves many roles in the early stages of design, but the most powerful aspect is that it can be seen and touched to reveal potential errors in conceptual design models.

The use of Computer Aided Design is widespread in the aerospace industry. The use of digital representations of mass and geometry allows designers to visualize concepts before costly efforts of physical prototyping. However, the development of CAD is very time consuming and does not always reveal issues with component interfaces. The use of Virtual Prototyping is now being explored as an extension of CAD modeling to allow a designer to "immerse" in the design domain to uncover integration issues that would have traditionally been discovered in physical prototyping.

Spacecraft mass growth is a key issue in any human spacecraft design. Historically this growth has been caused by top level requirements creep, uncertainty in the conceptual design, and pushing the limits of technology development. Various spaceflight programs managed mass growth differently during the life cycle development and few were not as successful. The addition of mass beyond conceptual design is one that is anticipated in future development phases, but should be considered in the early conceptual design to allow flexibility as the design matures.

2.3 MULTIDISCIPLINARY DESIGN OPTIMIZATION

Multidisciplinary Design Optimization (MDO) is described by Sobieszczanski-Sobieski and Haftka (1997) as "*a methodology for the design of systems in which strong interaction between disciplines motivates designers to simultaneously manipulate variables in several* *disciplines*". MDO is an area of research that develops systematic approaches for the design of complex systems governed by interacting physical phenomena (Alexandrov, 2005). In today's computational world, many researchers are using MDO methods as a means for developing and optimizing complex systems. The purpose of MDO is to bring all of the various subsystem disciplines together to simultaneously optimize a "multidisciplinary" system level problem. To optimize a solution means to direct the analysis process toward solutions that will balance the needs of the subsystems in a quicker timeframe. According to Sobieszczanski-Sobieski and Haftka (1997), the difficulty in adopting MDO among various groups lies in the ability of teams and codes to exchange data. Sobieszczanski-Sobieski and Haftka (1997) defined three categories of MDO in the research literature at the time.

1. The combination of two or three disciplines that can spawn a new discipline that focuses on the interaction between the disciplines such as aeroelasticity or thermoelasticity.
2. The use of simple analysis tools in the Conceptual Design phase in a single, usually modular computer programs to reduce computational burden.
3. The organizational and computational challenges for coupling various disciplinary high fidelity codes among dispersed groups and teams.

When first introduced in the 1990's, the use of MDO methods for complex engineering problems had great promise for solving and optimizing problems. However, the true use of MDO has been limited to mostly researchers and has not been widely used in realistic engineering problems (Alexandrov, 2005). One of the reasons that MDO has not been fully adopted is that MDO lies on the border between applied mathematics and engineering

(Alexandrov, 2005). Sobieszczanski-Sobieski and Haftka (1997) pointed out that the use of MDO actually costs more than the collection of single discipline optimizations.

A MDO analysis is a collection of mathematical modules that represent the various parts of the overall system such as a physical phenomena or part (Sobieszczanski-Sobieski and Haftka, 1997). Because a MDO problem can have a large number of design variables due to the many disciplines, the dimensionality of the problem is very difficult to reduce (Koch *et al*., 1999). In many engineering and complex problems, the relationships between the various disciplines are not always clear to the designers in specific areas.

The central challenge in MDO is balancing the use of coupled high-fidelity models with the amount of computational time required to generate an optimized solution. In today's engineering environment, high-fidelity FEA and CFD simulations can take days or weeks to complete a single analysis. For MDO problems, several analysis runs are needed for convergence of an optimized solution. Although computational power has increased over the past few decades, the complexity of high fidelity codes has increased as well. According to Messac and Mullur (2008), the high cost of computer times are unreasonable for obtaining optimized solutions from complex models.

MDO optimization methods are based upon two types of decomposition: hierarchical and non-hierarchical. The difference in the two approaches is how the subsystems or levels are optimized. In non-hierarchical, the overall design problems is optimized at the system level where the hierarchical level, the individual subsystems are optimized before the system level optimization. Methods such as Concurrent Subspace Optimization (CSSO), Collaborative Optimization (CO) use the hierarchical decomposition.

Many researchers have recognized the difficulty of coupling many different types of high-fidelity discipline specific codes and have created approximation techniques intended to reduce the computational time. These techniques are often referred to as "surrogate" or metamodels (model of a model). Because the surrogate or approximation models are based on an objective function created from a finite set of data points originally derived from high-fidelity models, there is an amount of uncertainty introduced in the design solutions. According to Martin and Simpson (2006) in deterministic approaches to MDO, the uncertainties are ignored and not quantified. Therefore, the particular surrogate modeling approach used should consider the random uncertainty in the design space variables.

A vast amount of literature has been published pertaining to Multidisciplinary Design Optimization and development of newer methods that extend the concepts of MDO. Many of the newer methods focus on areas such as Reliability Based Design Optimization (RBDO), Robust Design, Uncertainty Quantification, and Visual Design Steering. For example, Agarwal and Renauld (2004) utilized Response Surfaces (RS) in RBDO to reduce the computational burden of calculating reliability in MDO. Allen *et al*. (2006) developed Robust Design processes in a multidisciplinary context for materials design. Du and Chen (2005) utilized a method called Collaborative Reliability Analysis to improve reliability based design in multidisciplinary designs under uncertainty. Winer and Bloebaum (2002) utilized Visual Design Steering to capture and enable designer insights; which allows a designer to make decisions before, during, or after optimization to effectively steer the design process. All of these techniques build on the principles of MDO in order to develop more computationally and accurate analysis methods.

MDO problems are usually large and complex; requiring the use of higher fidelity analysis codes. Because of the large computational demands, MDO codes are largely optimization programs that attempt to reduce the computational burden and guide the analysis toward an optimized solution. However, because many high fidelity codes are discipline specific such as FEA for structures and CFD for aerodynamics and fluids, a significant amount of human effort are needed to couple the input and output results among the different discipline specific teams. This problem was highlighted by Sobieszczanski-Sobieski and Haftka (1997) and although many methodologies have been developed to overcome some of the challenges in MDO, a large amount of research still remains before MDO becomes more "user friendly." According to Alexandrov (2005), "*there are fundamental analytical and computational obstacles that must be overcome before MDO can make a wide-spread impact on the practice of design.*" MDO is a powerful tool for design, but the difficulties in its use among designers with various backgrounds is likely one of the reasons it will remain largely in a research oriented setting.

2.3.1 Metamodeling

Metamodeling is the process of creating a model that is an approximation of a complex analysis code model. Simpson *et al*., (2001) describe metamodels as "*statistical approximations of expensive computer analyses facilitating multidisciplinary, multiobjective optimization and concept exploration.*" The process of creating metamodels evolved from classical Design of Experiments (DOE) theory and is called "metamodeling" (Wang and Shan, 2007). Because of the increased efficiency in computational times, metamodeling is used to approximate complex higher fidelity analysis codes such as FEA or CFD. Metamodeling extend the capability of MDO analysis such that many more design configurations can be analyzed and understood. Wang and Shan (2007) outlined five benefits of Metamodel Based Design Optimization (MBDO).

- It is easier to connect proprietary and often expensive simulation codes.
- Parallel computation becomes simple as it involves running the same simulation for many data points.
- Building metamodels can better filter numerical noise than gradient methods.
- The metamodel renders a view of the entire design space.
- It is easier to detect errors in simulation as the entire design domain is analyzed.

Although the benefits of metamodeling seem very promising, Yang and Shan (2007) point out that the use of metamodeling techniques in the design engineering community seems to lag the research community. Building metamodels is mathematically intensive and this could be a reason why metamodeling has not been as widespread among the practicing design engineering community (Wang and Shan, 2007). During the AIAA/ ISSMO Symposium on Multidisciplinary Analysis and Optimization in 2002, an Approximations Methods Panel was held with the current leading researchers in the field of approximation methods to summarize the current and future research efforts at the time. A key challenge given by the panel to the academic community was to help educate engineers on how to use metamodels (Simpson et al., 2004).

Researchers have recognized the issue with the mathematical complexity involved with metamodeling and have attempted to develop many different methods of metamodeling to assist designers with optimization of MDO problems. For example, Yang *et al*., (2005) studied five different types of metamodels (Stepwise Regression, Moving Least Squares, Kriging, Multiquadratic, and Adaptive and Interactive Modeling System) based on response surface methods to simulate vehicle frontal impacts. This work demonstrated that metamodeling offers designers a quick tool for evaluating design alternatives using Pareto plots and curves. Wang and Shan (2007) reviewed metamodeling techniques and compiled a list of the most commonly used metamodeling techniques as shown in Table 2.

Table 2: Commonly used metamodeling techniques (Wang and Shan, 2007).

Experimental Design / Sampling Methods	**Metamodel Choice**	**Model Fitting**
Classic Methods • Fractorial • Central Composite • Box-Behnken • Alphabetical optimal • Plackett-Burman Space Filling Methods • Simple Grids • Latin Hypercube • Orthogonal Arrays • Hammersley Sequence • Uniform Design • Minimax and Maximin Hybrid Methods Random or Human Selection Importance Sampling Directional Sampling Discriminative Sampling Sequential or adaptive methods	Polynomial (linear, quadratic) Splines (linear, cubic) Multivariate Adaptive Regression Splines Gaussian Process Kriging Radial Basis Function Least interpolating polynomials Artificial Neural Networks (ANN) Knowledge Base or Decision Tree Support Vector Machine (SVM) Hybrid Models	Weighted Least Squares Regression Best Linear Unbiased Predictor (BLUP) Best Linear Predictor Log – likelihood Multipoint Approximation (MPA) Sequential or adaptive metamodeling Back propagation (for ANN) Entropy (inf. –theoretic, for inductive learning on decision tree)

The columns in Table 2 describe the three areas of metamodeling. First, the particular Experimental Design and Sampling method is used to gather design points from a complex high fidelity code such as FEA or CFD. The advantage of sampling is that it minimizes the number of high fidelity complex code runs needed to produce a "sample set" of information that will be used to develop the metamodel. Second, a particular metamodel choice is used as a surrogate for the high fidelity complex code. Lastly, a technique is used to "fit" or validate the metamodel against the sample data. This area is of particular interest because the validation of a metamodel is crucial to quantifying uncertainty with the metamodeling technique. As pointed out by Wang and Shan (2007), metamodel validation shares many of the same challenges associated with verification and validation of traditional computational models.

The topic of metamodeling is very extensive and much literature has been published in various applications. The critical disadvantage to the use of the particular metamodeling approach is the quality of the metamodel (Yang *et al*., 2005). The use of response surface methods in metamodeling techniques such as the one performed by Yang *et al*. (2005) is just one of the many different metamodeling approaches described in the literature. Much literature has been published on metamodeling techniques and accuracy including: Jin, *et al.* (2001); Simpson *et al.* (2001); Martin and Simpson (2003); Clarke *et al*. (2005); one of the most common metamodeling approximation methods, Kriging, has been extensively documented in the literature (Kleijnen, 2007).

The use of metamodeling for optimizing and exploring the design and tradespace is very advantageous when compared to the computational costs associated with coupling several disciplinary high fidelity complex codes. However, the reduction in computational time must be balanced against the loss of accuracy with a lower fidelity approximation code such as a metamodel. If the intent of the design process is to gather information about the entire design and objective tradespace, such as in the conceptual design phase; then a metamodeling approach is the preferred methodology for reducing the number of potential design solutions to a level where the use of a higher fidelity code is needed for design refinement. The difficulties and mathematical rigor of sampling, programming, and fitting a metamodel is not well understood among practicing design engineers and the gap between the research and design community in metamodeling will continue until more powerful and easier to use software packages are developed to assist design engineers with complex engineering analyses.

2.3.2 Tradespace Exploration

Tradespace exploration is a method used to explore the various objectives in order to understand the relationship to the design space variables. According to Ross (2006), tradespace exploration in the conceptual design phase may empower designers to overcome challenges associated with tendencies to reduce the design space and overlook potential design space solutions. In other words, a thorough exploration of the objective tradespace during the early stages of conceptual design is a powerful tool to uncover potential challenges and reduce uncertainty in the design. A designer needs to understand the impacts of configuration changes on the overall spacecraft design. Ross (2006) conducted interviews with industry and learned that broad tradespace exploration is rare and often done in an ad hoc manner.

A full tradespace exploration evaluates many more spacecraft configurations than a small set of preferred options. In the work by Shaw (1998), an approach named the General Information Analysis (GINA) was proposed to explore a large set of space system design options in terms of generic metrics for comparison among different concepts. The PhD dissertation by

Jilla (2002) utilized the GINA process and added Multi-Objective Optimization (MOO) techniques to fully explore the design and objective tradespace for distributed satellite systems.

According to Hastings (2004), a Multi Attribute Tradespace Exploration (MATE) tool was developed at the Massachusetts Institute of Technology (MIT) to analyze system architectures with the goal of maximizing system attributes. This process focuses on the needs of the stakeholders and the driving preferences are captured in attributes using Multi Attribute Utility Theory and forms a preference space for which potential systems will be evaluated (Ross *et al*., 2002). Ross (2003) further developed the MATE process into MATE-CON; which was an extension into Concurrent Design. Ross *et al*. (2004) also developed this approach as a front end for space system development. What the MATE process identified is the need for fully exploring a tradespace given the specific value attributes a customer desires.

In the PhD dissertation by Ross (2006), the results of a survey with industry suggested that other "illities" such as sustainability, flexibility, scalability, agility, and adaptability are poorly addressed. The topic of tradespace exploration in the literature is rather limited in aerospace applications. The likely reason for the lack of other research beyond the work conducted at MIT is most likely due to how industry perceives the need for tradespace exploration in early conceptual design.

In the design of human spacecraft, a variety of objectives must be explored and compared for future design tradeoffs. Unlike the MATE process, the minimum functionality methodology proposed in this research begins with a point (minimum) design configuration that must meet minimum physics and physiology requirements. This baseline configuration is based in part due to customer or top level objectives but is not "locked down" such that further concepts for minimum functionality could be evaluated in a timely manner. The evaluation of a *Safety* and

Operability tradespace provides designers with valuable information on the relationships between Safety and Operability in the spacecraft configuration.

2.3.3 Design Space Exploration

Design space exploration is similar in many respects to tradespace exploration where the input design variables are evaluated to determine relationships among various objectives. According to Acar (2010), metamodels are widely used in design space exploration and there is a need for developing techniques that increase the accuracy of the metamodel predictions. Nixon (2006) developed a systematic design space exploration process to keep the design effort manageable in the conceptual design phase. In the PhD dissertation by Villeneuve (2007), a concept and technology selection methodology for complex architectures was developed to quantify and explore the design space simultaneously. The interesting part of the research conducted by Villeneuve was in how the design search space was developed using integrated graph theoretical concepts and Ant Colony optimization. Because a complex engineering problem can have a set of hundreds of design variables, typical design space exploration methodologies are used to "search" the design space in order to determine potential designs that could be overlooked in a traditional design process.

2.3.4 Multidisciplinary Design Optimization Summary

Multidisciplinary Design Optimization is a design approach that simultaneously evaluates variables in many disciplines in order to develop a system level design. Many engineering problems are complex and multidisciplinary in nature. An MDO approach couples the discipline specific models for development of optimized designs. A challenge in MDO methods is how to balance the use of coupled high-fidelity models with the amount of computational time required. Because many discipline specific codes are high fidelity and complex, many researchers utilize metamodels as surrogate models in place of the higher fidelity models. Especially in the early conceptual design stage, the use of metamodels allows designers to study a larger number of configurations, but at the cost of reduced accuracy. Researchers have studied the uncertainty in many metamodeling approaches and proposed methods that provide greater accuracy.

A multiobjective optimization problem is a specific form of MDO because of the many and often competing objectives that must be balanced. In many cases, the output of a MOO problem is given as a set of objectives that form a Pareto frontier of non-dominated solutions. The choice as to which solution is the preferred solution is left to decision makers who evaluate the various solutions through various methods such as tradespace exploration or design space exploration.

Tradespace exploration is a method used to explore the relationship between objectives and design space variables. According to Ross (2006), broad exploration of the tradespace is often overlooked or done on an ad hoc basis. A methodology for tradespace exploration called MATE was developed by MIT to evaluate tradespaces based on customer supplied value attributes. Design space exploration is similar in many respects to tradespace exploration but the emphasis is on the study of design space regions and search strategies to develop and optimize design space variables. Several PhD dissertations have been conducted on the topic of design space exploration and the need for fully exploring both the objective and design tradespace of a Multidisciplinary Design Optimization problem during conceptual design.

2.4 RISK BASED DESIGN

According to Tumer *et al*. (2005), "*Risk Based Design (RBD) can be defined as a design process that formally identifies the risk elements during the mission design phase and continuously optimizes investments and decisions to mitigate those risks.*" The work by Tumer *et al*. (2005) provided the groundwork for addressing risk in NASA programs and highlighted a few promising research directions for RBD based on their review of risk practices in NASA.

- Risk Based Design methods at NASA are based on reliability analyses applied to design problems. This approach is difficult in the early design stages because information is vague and probabilities are difficult to assign.
- There is a need for more advanced risk informed methods. Newer methods should treat risk as a tradable resource to characterize, balance, and minimize risk in the uncertain stage of conceptual design.
- Designers must understand the design process, risk analysis practices, and the risk management efforts at NASA. Risks and failure modes should be identified related to design decisions. Decisions should also be based on the risk and failure information.

The findings by Tumer *et al*. (2005) describe the issues with the current NASA approaches for incorporating risk based design decisions early in the conceptual design process. The goal of their work was to lay the foundation for more collaboration between NASA researchers and the academic research community. An earlier NASA report by Knight and Stone (2002) also highlighted many of the needs for risk based design. Knight and Stone (2002) suggested that risk based design methods make the design process more robust given that a systems level understanding is incorporated and detailed knowledge about the subsystems is utilized. They also point out that just the use of rapid modeling and analysis tools will not always result in successful designs (Knight and Stone, 2002).

The use of risk based design methods was described in NASA's Design, Development, Test, and Evaluation (DDT&E) Considerations for Safe and Reliable Human Rated Spacecraft Systems (2008). One of the guiding principles for developing a human rated spacecraft system was to use risk based design loops in the conceptual design phase to iterate the operations concept, the design and the requirements for meeting minimum objectives at minimum complexity (Miller *et al*., 2008).

The typical approach used by NASA for understanding risk in the early stage of conceptual design is through failure and risk analysis methods using traditional reliability tools. Tumer *et al*. (2005) identified the most common methods used for reliability analysis including:

- Failure Modes Effect Analysis (FMEA),
- Fault Tree Analysis (FTA),
- Event Tree Analysis (ETA),
- Failure Modes and Effects Criticality Analysis (FMECA),
- Event Sequence Diagrams (ESD),
- Reliability Block Diagrams (RBD), and
- Master Logic Diagram (MLD).

In addition to the reliability methods listed, a method used within NASA to assess risk is Probabilistic Risk Analysis. The use of PRA identifies what can go wrong, how frequently it will happen, and the likely consequences (Tumer *et al*., 2005). The issue with PRA is that it requires a significant amount of information before a detailed analysis can be completed. In the early stages of conceptual design, many factors such as the particular type of technologies in subsystems, the probability of failure of components, and the exact layout and integration of the subsystem components many not be fully defined. If PRA could be used in the early stage of conceptual design, it would be a powerful decision tool; but the lack of information and the large amount of uncertainty in the design details prevents its use in the early conceptual design phase.

Because of the many different methods available to analyze reliability and risk, designers are unsure about how to integrate the methods into their current design processes (Tumer *et al*., 2005). Many designers do not have full confidence in early PRA results because of the large amount of assumptions used to generate the results. In addition, the PRA results are difficult to understand and designers do not always have the resources for complicated risk analyses (Tumer *et al*., 2005). To overcome these issues, it has been suggested that risk analysts work closely with design engineers to assist with design level decisions in order to effectively steer the design in the early stages.

The work by Tumer *et al*. (2005) was intended to build collaboration with the academic community to develop additional methods that would assist design engineers in the early stages of the design. A central need identified by Tumer et al. (2005) is "*a comprehensive risk-informed design tool that can be practically used in all phases of conceptual design for low volume high cost space missions*." In addition, Tumer *et al*. (2005) emphasize the use of multiobjective optimization problems for generating Pareto frontiers in aerospace systems. In their words, *"NASA therefore, still lacks a formalized and universal design tool that can quantitatively formulate an aerospace system design problem as a multiobjective optimization process, and tradeoff risk in a multiobjective sense"* (Tumer et al., 2005).

A recent work by Mehr and Tumer (2006) developed a new risk based design decision making method referred to as Risk and Uncertainty Based Concurrent Integrated Design Methodology (RUBIC). This method utilized concepts from portfolio optimization theory and continuous resource management to provide a mathematical rigor for risk based decision making. In their approach, they assumed that risk of an element is not independent of other elements in the system and that risk can be traded between elements and subsystems based on weighting the criticality of different elements and subsystems (Mehr and Tumer, 2006).

2.4.1 Reliability Analysis

NASA defines Reliability as: "*the probability that an item will perform its intended function for a specified interval under stated conditions. The function of an item may be composed of a combination of individual sub-functions to which the top level reliability value can be apportioned"* (NASA, 1998). Reliability analyses are the tools and activities that are used to calculate the reliability in a system.

A common tool for understanding potential areas of failure is FMEA; which has been extensively used for systems safety and for the reliability analysis in many industries (Sharma *et al*., 2007). FMEA is a top down approach that focuses on the loss of functionality in a given design. In FMEA, the failure modes of the system are identified that could affect the system and its performance. Failure modes are typically scored according to criticality, likelihood of detection, and severity of the failure. The three scores are multiplied to obtain a Risk Priority Number (RPN). For each high RPN, risk management techniques are used to mitigate the identified risk as the design is matured.

The difficulty with FMEA approaches is that for complex systems, a large amount of data is required and the data that is available is subject to uncertainty (Sharma *et al.*, 2007). To overcome the issue with uncertainty, Sharma *et al*. (2007) developed a knowledge based approximate reasoning methodology that can be coupled with traditional FMEA.

Another failure mode approach that focuses on specific components in functions is known as a FMECA. A FMECA is a bottom up reliability analysis that evaluates all of the major components in order to identify single point failures and hazards overlooked in other system analysis techniques (FAA, 2005). The disadvantage of FMECA is that it does not account for multiple and coupled faults and failures; thus it could give an optimistic estimate of system reliability if a quantitative approach is used. The FMECA should be used with other analyses such as FTA to develop reliability estimates (FAA, 2005).

A Fault Tree Analysis is a top down approach that illustrates the sequence of events that lead to an unfavorable event and provides a quantitative estimate of system reliability (FAA, 2005). FTA uses logical symbols to graphically show the progression of a failure on other components in the system. There may be different outcomes to a single event depending upon how the failures are controlled or mitigated through fault tolerance. A FTA is conducted when a design concept is mature such that failure rates and probability information is available. A detailed explanation of NASA's approach for FTA can be found in the *Fault Tree Handbook with Aerospace Applications* (NASA, 2002).

Much like the FTA, an ETA is a bottom up approach that graphically explores system responses to an initiating event and assesses the probability of the outcome (FAA, 2005). The use of ETA allows designers to assess multiple coexisting faults and failures and identifies vulnerabilities. These vulnerabilities might contain failure propagation throughout the system. An ETA is specific to a single initiating event, so multiple analyses could be required for a system. As with FTA, an ETA requires a large amount of data associated with the design and interfaces. An ETA is likely to be conducted when the design is mature beyond the conceptual design phase.

Another common technique that is used in tradeoffs related to system safety is the use of Reliability Block Diagrams. A RBD illustrates the logical connection between subsystem components and can be used to mathematically model the system probability; thus it is useful for evaluating various potential configurations (FAA, 2005). In the example of early conceptual design, designers would assign reliability targets for specific components in the subsystems if data was not available. The disadvantage to this approach is that a considerable amount of modeling needs to be developed in order to understand reliability sensitivities in the system. Not all systems can be easily modeled using RBD; including non-hardware failure mitigation measures (FAA, 2005).

The various reliability analysis methods approach the quantification of system reliability in a different manner. Most of the methods require complementary analysis such as FMECA should be used with FTA and ETA. RBD should also be used with ETA to uncover all of the potential failures and hazards that could be overlooked with a single approach.

2.4.2 Probabilistic Risk Assessment

Probabilistic Risk Assessment was identified as a need in 1996 in order to support decisions for Space Shuttle upgrades (Stamatelatos *et al*., 2002). The reason for adopting PRA was to quantify the amount of safety improvement based on the proposed upgrades. According to NASA NPR 8705.5, PRA is defined as:

> "*A comprehensive, structured, and logical analysis methodology aimed at identifying and assessing risks in complex technological systems. PRA is generally used for low probability, high consequence events for which limited statistical data exist. – Its application is targeted at risk environments common within NASA that may involve the compromise of safety, inclusive of the potential loss of life, personal injury, and loss or degradation of high value property"* (NASA, 2004).

The emphasis on safety, performance, and mission success is the primary focus of PRA. Although it is not a risk management tool, it provides information to decision makers about potential issues in the design that could compromise safety. Risks must either be mitigated through continuous design or testing; or the risk is accepted as part of the overall risk strategy. PRA informs the decision makers of the sources of risk in order to provide information that will determine costs associated with reducing uncertainty (NASA, 2004).

The PRA process is different from traditional reliability analysis because it is focused on answering three basic questions. (1) What can go wrong? (2) How likely is it? (3) What are the consequences? (NASA, 2004) NASA uses three different levels of PRA depending upon the scope and maturity of the design. These levels are listed as the following:

- <u>Full Scope PRA</u> contains all major PRA components and addresses all applicable end states that lead to loss of crew, accidental exposure, potential illness or death of public or ground based personnel, loss of ground facilities, loss of space based facilities, mission abort, loss of mission, and mission reconfiguration.

- Limited Scope PRA defined on a case by case basis so that the results can provide specific answers to identified mission critical questions and safety concerns rather than an assessment of all risks.
- Simplified PRA identifies and quantifies major risk contributors and applies to systems of less available data. Contains a reduced set of scenarios or simplified scenarios designed to capture only essential top level mission risk contributors (NASA, 2004).

The various levels of a PRA analysis cover a wide range of risk assessments. In the conceptual design phase where data is limited and sometimes unknown, the Simplified PRA is the most logical choice for identifying risks that are related to top level mission objectives. As mentioned earlier, a PRA does not mitigate risk but serves to identify where the issues are located in the design. Whether it is lack of information or incorrect design assumptions, the PRA is used to uncover design faults such that resources can be allocated as the design matures. A comprehensive guide to the use of PRA can be found in the Probabilistic Risk Assessment Procedures Guide for NASA Managers and Practitioners by Stamatelatos *et al*. (2002).

According to Tumer *et al*. (2005) one of the main drawbacks to the use of PRA for early conceptual design is the limited amount of data. The use of knowledge bases of quantitative data to derive probabilities has been suggested as a possible solution for reducing uncertainty in PRA. This approach seems rather logical, but lessons learned on historical development programs may be unable to predict the future behavior of the equipment (Sharma *et al*., 2007). The use of PRA approaches are better suited for mature designs and the use of PRA in the conceptual design phase could be very uncertain and cumbersome given the final design configuration is in a state of flux.

2.4.3 Risk Based Design Summary

Risk based design is a process for identifying and mitigating risks in the early design phase. A key issue identified by Tumer *et al*. (2005) is the need for newer approaches that couple traditional reliability analyses with a more risk informed approach in the conceptual design phase. Many of the typical reliability analysis and risk assessment tools used by NASA are applicable to designs that have reached a level of maturity such that probabilities of failure in components, subsystem configurations, and operation concepts are fully defined. The reason why traditional approaches do not work well for the conceptual design phase is due to the large amount of uncertainty and design flexibility. The detailed nature of PRA does not lend itself to very simple assessments of risk and often designers do not accept the results based on the assumptions used to generate the analysis. Other reliability methods also suffer from the same problem as PRA in conceptual design. The lack of information and design definition in the early stages prevents an accurate assessment of the overall risk posture of a spacecraft design.

The work presented in this research is intended to provide a method to allow designers to quantify and trade the amount of risk through a "Safety Index" that will allow comparison of various spacecraft configurations in the early stages of conceptual design. The use of this index is not intended to replace PRA methods, but provide designers with an easy to understand and implement process that provides a "measure of goodness" related to future PRA analysis of matured spacecraft designs.

2.5 MINIMUM FUNCTIONALITY DESIGN

Minimum functionality design is a design approach where a baseline configuration is defined before trading other factors in human spacecraft designs. Minimum functionality recently gained attention in the Altair Lunar Lander project. According to the NASA Broad Agency Announcement for the Constellation Lunar Lander Development Study in 2008:

> "*This design is intended to provide as close to a minimum configuration reasonable to evaluate in order to conclusively buy down risk through a deliberate "add-back" process - Minimum functionality means that the LDAC-1 Lander design does not incorporate any additional capabilities beyond those required to perform the reference mission*" (NASA, 2008).

The design approach for minimum functionality in the development of the Altair Lunar Lander started with a single point baseline design point of departure for cost and risk trades in order to justify mass "add-backs" to the subsystems in the form of additional redundancy and safety (Cohen, 2009). The phrase "minimum functionality" as a combination of the words "minimum" and "functionality" carries various meanings among different organizations. For example, NASA defines minimum functionality as a minimum configuration spacecraft that will perform a successful baseline mission without considering failures or contingencies (NASA, 2008). This methodology was first introduced in the Lunar Development Analysis Cycle One (LDAC-1) for the Altair Lunar Lander (Cohen, 2009). In addition, the Orion Crew Exploration Vehicle used a similar approach of creating a baseline design with a set of "minimum functions" called the "Zero Based Vehicle" (Hu *et al*., 2008). According to NASA's Design, Development, Test, and Evaluation Considerations for Safe and Reliable Human Rated Spacecraft Systems, a minimum functional design is the simplest, most robust, and highest performance design option as the starting point for assessing fault tolerance (Miller *et al*., 2008). In each of these approaches, the common theme is that a "minimum" spacecraft configuration is defined to meet the desired requirements for a baseline "minimum" mission. The minimum functional and minimum mass configuration serves as a starting point boundary configuration for adding mass to reduce risk through increasing fault tolerance, safety, and mission specific operability.

Much of the confusion between different minimum functionality approaches is how the minimum or starting point configuration is defined before trading other aspects of safety, reliability, performance, and cost. In NASA's design methodology, a minimum baseline vehicle included minimum acceptable Factors of Safety (FoS). In the Orion ZBV, the intent was to reduce the mass of an existing spacecraft design to a minimum mission configuration (Hu *et al*., 2008). Both of these approaches prescribe minimum functionality according to "minimum mission requirements" which included aspects of mission operability. The research presented here builds upon the previous concepts of minimum functionality design to define a standardized approach that can be used across various conceptual spacecraft design activities.

2.5.1 NASA's Minimum Functionality Design Approach

Because the Lunar Ascent Module contributed the most to the overall mass of the lunar mission architecture, a reduction in mass of this spacecraft reduces the required Earth launch mass significantly. Using the Apollo Lunar Excursion Module as an example, a 1 lb. increase in the Lunar Ascent Module mass increased the Saturn V gross launch vehicle mass by 883 lb. (Thompson *et al*., 2010). Thus, a need to design the Lunar Ascent Module to its minimum optimized mass is one of the reasons NASA used a "non-traditional" spacecraft design philosophy. This design approach first defined a minimum functionality baseline spacecraft that was assumed to be "non-flyable" because of single strings in the minimum number of subsystems required to complete a minimum implementation mission (Dorris, 2008).

NASA utilized a set of development cycles to design the Altair Lunar Lander in a true risk informed approach. In the LDAC-1 cycle, a "minimum functional" vehicle was developed to meet a minimum set of baseline mission requirements. According to Cohen (2009), the designers and systems engineers determined what would be the minimum configuration based on single-string redundancy for the subsystems without accounting for failures through fault tolerance or other contingencies (Cohen, 2009). One of the issues identified during this early analysis cycle was a lack of expertise in spacecraft developers. The lack of recent human spaceflight projects did not provide the required experience base for understanding all of the critical issues in spacecraft development (Dorris, 2008).

In the LDAC-2 design cycle, the previous baseline configuration was matured in order to reduce risk through cost/risk trades to determine the most significant contributors to LOC. In the LDAC-3 design cycle, the largest contributors to LOM were identified and cost/risk trades were performed to optimize the design (Dorris, 2008). Mass additions through increased subsystem redundancy and fault tolerance were justified by stakeholders in the various subsystems (Cohen, 2009)

The non-traditional approach used by NASA was a different methodology than the typical government oversight programs of industry charged design and development process. The minimum functionality approach was also based on a belief that a traditional Systems Engineering approach was too costly (Cohen, 2009). The intent of the initial government study was to mature the design to a point in the lifecycle such as Systems Requirements Review or System Definition Review before handing the design over to industry for further development.

According to Cohen (2009), the minimum functional approach created concerns that the minimum functionality design was too small and would affect crew productivity. The focus on minimizing everything in the spacecraft included minimizing habitable volume as well. Other concerns such as an EVA porch and line of sight for landing were noted as challenges that would need to be addressed (Cohen, 2009). The issues raised by Cohen (2009) were known by NASA and based on the recommendations given by Northrop Grumman on the design of the Altair Lunar Lander; one can easily understand that a redesign effort is likely after the design is handed over to industry. Thus, the initial work conducted by NASA served two purposes, first, it trained a workforce at NASA in spacecraft development and second, it presented a preferred design configuration by NASA to reduce unnecessary trade studying in later phases of development.

NASA asked industry to provide feedback to the LDAC-1 minimum functional configuration. The work by Cohen (2009) presented several findings for design issues related to crew productivity, pilot view angle, separate habitat and airlock, required habitable volume, required airlock volume, pressurized payload implications for lunar surface science, and symmetry about the thrust axis. Although NASA proposed a minimum functional design for the LDAC cycles, there were issues raised about human factors being an important part of the top level objectives (Cohen, 2009). A full understanding of the role of human factors and the required habitable volume needed for sustaining humans for a long period of time such as an extended lunar sortie is one of the issues that should be addressed in any minimum functional design approach.

The work performed by NASA in the development of the Altair Lunar Lander was utilized a risk based design approach and designed the vehicle using a bottom-up philosophy in order to understand the impacts of subsystem redundancy on performance, safety, risk, and mission operability. Although NASA succeeded in developing a baseline design concept, issues with design maturity seem to be evident as expressed by Cohen (2009). One may imagine that a future Altair Lunar Lander might be different from the initial concepts developed by NASA. One of the lessons learned from the activities performed by NASA is that the minimum functionality process needs to be flexible enough such that design issues can be investigated earlier. The minimum functionality design methodology presented in this research builds upon the previous lessons to demonstrate how human factors can be incorporated in the early stage of the conceptual design process.

2.5.2 Orion Zero Based Minimum Functionality Approach

Because of mass concerns with the Orion Crew Exploration Vehicle at the end of the first Design Analysis Cycle (DAC-1), a minimum functionality approach was utilized to "scrub" the mass of the entire spacecraft before System Definition Review (Hu *et al.*, 2008; Jordan, 2009). The Orion approach was in many respects similar to NASA's minimum functionality design, but the definition of the minimum configuration spacecraft was slightly different than an all single string subsystems as defined by the LDAC-1 design cycle. In the minimum functionality Orion configuration, components were removed from the existing spacecraft design to reach a configuration without "non-essential" components. Similar to the NASA approach, the minimum configuration was not a "flyable" configuration, but served as a point of departure for adding back subsystem components. In the Orion ZBV, not all the subsystems were single string, the redundancy was reduced to one fault tolerance for safety critical subsystems and zero fault tolerance for mission success (Jordan, 2009).

The weight reduction effort utilized the Orion Vehicle Engineering and Integration Working Group to lead the process for adding subsystem components back to the spacecraft (Hu *et al*., 2008; Jordan, 2009). Components and functions to be added to the ZBV were prioritized according to safety, robustness, and mission objectives (Jordan, 2009). All safety critical items were considered mandatory add backs. A series of operational assessments was conducted and subsystem components were ranked according to impact on operations.

Although the ZBV activity was similar to NASA's minimum functionality design, the two approaches shared the common methodology of justifying add backs to the configuration through safety, performance, risk, and mission objectives. The minimum functional configuration was different between the two approaches, but the concept of establishing a baseline for subsequent trading of design attributes was the common theme. A total of 2500 kg was removed from the Orion spacecraft as a result of the ZBV activity and maintained needed functionality for lunar sortie missions (Hu *et al*., 2008).

2.5.3 NASA Processes for Developing Safe and Reliable Human Spacecraft

The focus of minimum functionality design is to first establish a baseline configuration before the trading of performance, safety, risk, and mission objectives. A minimum functional spacecraft configuration would never be considered as a realistic flyable spacecraft design until the risk and reliability is within acceptable levels for human rated requirements. During the Altair LDAC-2 and 3 cycles, the addition of subsystem components increased safety and reduced risk in order to meet reliability thresholds.

A recent technical report by NASA's Engineering and Safety Center, entitled the Design, Development, Test and Evaluation (DDT&E) Considerations for Safe and Reliable Human Rated Spacecraft Systems, Miller *et al*. (2008) described a methodology for developing safe and reliable human rated systems. In their approach, the safety and reliability requirements are specified through a triad of fault tolerance, bounding failure probability, and adhering to proven practices and standards (Miller *et al*., 2008). Spacecraft systems should be designed with fault tolerance supported by probabilistic safety, reliability, and risk analysis backed up with data and analysis. In a minimum functionality design approach, the determination of fault tolerance and reliability is one of the key activities in order to justify the mass additions.

The NASA guidelines outlined a conceptual design minimum functionality process for spacecraft development in Section 2.3, Conceiving the Right System, Critical Activities Early in the Life Cycle (Miller *et al*., 2008). Similar to the Altair design activity, the NASA guidelines specify a risk based design approach to iterate the operations concept, design, and requirements "*until the system meets mission objectives at minimum complexity and is achievable within constraints*" (Miller *et al*., 2008). The minimum functional approach first begins with a set of top level mission objectives that determine the high level requirements. Second, a minimum set of functions are defined to accomplish the mission objectives; this could be considered the "minimum functionality" needed for the mission. But what the guidelines do not clarify is how to develop a minimum set of functions other than what is needed to accomplish the mission objectives. Third, the simplest conceptual design of the proposed system is developed. In this step, the description of the simplest system is the most robust and highest performance. It is unknown if subsystem redundancy is intended to be a part of this minimum configuration. The description of "most robust" could be interpreted as single highly reliable components or redundant subsystems that provide a high level of reliability. Clarification of "most robust" is needed to assist designers in this critical step of development. The fourth step in the design process adds independent elements in order to increase safety. A decrease in performance is accepted as long as reliability is increased to meet safety. Risk and reliability analyses are conducted to determine potential failure modes and evaluate design options. Next, components are added for mission success and evaluated for safety and risk. Finally, the cost and schedule are identified to mature the conceptual design.

The NASA guidelines for creating a safe and reliable system are very similar to the design and development of the Altair Lunar Lander. Although the approach is not titled "minimum functionality", it is a process that shares many of the key aspects. The only disadvantage to this approach is that it is subject to different levels of interpretation as to what is the "most simple" or "most robust" system. In the example of comparing two different configurations of spacecraft to perform the same mission, the only design attributes that could be realistically compared are mass, risk, reliability, and cost. Judging a spacecraft on these parameters alone will often overlook other necessary "illities" such as complexity, maintainability, and utility.

2.5.4 NASA Human Rated Spacecraft Requirements

Because a minimum functional design would never be considered a flyable design, the addition of redundancy and safety in the design is necessary to achieve levels that satisfy human rating requirements. Human Rating Requirements for Space Systems are defined in the NASA procedural requirements document NPR 8705.2B. Because of the recent Commercial Crew Development activity, NASA needed to specify the requirements for human rating crewed systems. The technical requirements in Section 3 specify safety requirements through fault tolerance or redundancy, control requirements, and abort requirements. Mentioned in paragraph 3.1.2, the requirements are not intended to be "*all inclusive or an absolute prescription for human rating*" (NASA, 2009). The designers of a spacecraft system should evaluate the technical requirements to develop the safest practical system that will meet the mission objectives. The set of human rating requirements is likely the minimum set of requirements that must be satisfied in order to achieve human rating certification. However, many more requirements may be levied on a design depending upon risks in the design or incorporation of newer unproven technologies. Each spacecraft design will likely follow its own process for human rating and achieving NASA certification in regards to safety and reliability.

2.5.5 Minimum Functionality Design Summary

Minimum functionality design is a non-traditional design approach in the conceptual design of human spacecraft. Its purpose is to establish a baseline configuration such that additional mass additions can be justified based on performance, safety, reliability, and mission objectives. The minimum functional configuration would never be considered as a flyable design and serves only as a lower boundary condition for trading additional design parameters. The approach was derived from a risk based approach to justify the needs and requirements for a baseline mission. A similar approach to minimum functionality was used in the Orion program where a design configuration called the Zero Based Vehicle was defined as a lower boundary for subsequent trading of subsystem redundancy and fault tolerance to meet the mission objectives of a lunar mission. Although the differences in the Orion approach compared to the Altair approach were slightly different in implementation, the common theme among the two design activities was the development of a lower baseline configuration in order to justify additional subsystem redundancy and fault tolerance to meet safety and mission requirements.

Deciding the amount of "safety" in a spacecraft is a difficult task and open to different interpretations because the exact safety requirements will depend upon the particular spacecraft design. In response, NASA developed a set of Human Rating Requirements for Spacecraft Systems to define requirements that must be achieved in order to be utilized by NASA. The NASA procedural requirements document NPR 8705.2B serves the purpose of defining the critical safety requirements for obtaining human rated certification. Depending upon the spacecraft design, additional safety requirements may be levied on the design and the requirement specified are not all inclusive. The goal of a minimum functional design process is to first establish a credible design baseline that will be used to evaluate risk reduction and safety improvements. The process of adding components that increase safety and mission objectives should be based on a risk based design approach where the reliability in the overall systems is evaluated in an iterative fashion.

2.6 LUNAR SPACECRAFT DEVELOPMENT

A human lunar spacecraft, commonly known as a Lunar Lander or Lunar Habitat is very much different from other spacecraft designs such as capsules and lifting bodies. Although there are many similarities to conventional LEO spacecraft, a lunar spacecraft must operate in a different set of environments on the surface of the moon including: thermal cycling, radiation, dust, lighting, and micrometeoroids (Cohen, 2009). Significant lessons were learned during the Apollo missions that will assist designers in the development of future lunar spacecraft.

However, there is much to be learned before establishing a human presence on the moon and the goals of the Constellation program were intended to fulfill this purpose.

Development human lunar spacecraft presents many challenges. The design team must consider the number of crew members, where the crew will land, and how long the mission will last. The coordination of systems, people, and resources needed for a mission requires the collaborative efforts of many organizations including government, industry, and academia. The success of the Apollo program was due to the cooperative work of 20,000 companies and 400,000 people across the country (Fries, 1992; Kelly 2001). Because of the large number of people and systems involved in the lunar architecture design, a Systems Engineering approach is necessary to develop and manage the spacecraft architecture from concept to end of life.

The focus of this section provides information about the design and development of the Apollo and Constellation programs. This information will be helpful in understanding the Lunar Ascent Module design concepts presented as part of the research in this dissertation. Although the literature contains vast information about the Apollo Lunar Excursion Module (LEM), a summary of the significant development challenges will be presented. For the Constellation program's Altair Lunar Lander, a summary of the design concept will be presented beginning with the Exploration System Architecture Study. The intent of this section is not intended to provide a comprehensive review of the Lunar Lander literature, but provide information about the difficulties and challenges associated with the development of human lunar spacecraft.

2.6.1 Apollo Lunar Module Program

The goal of the Apollo program was to put a man on the moon and return him safely to Earth. The goals given by President Kennedy in his May 25, 1961 speech before Congress became a reality with the successful mission of Apollo 11 on July 20, 1969. The success of the moon landing was in a large part due to the development of the Apollo Lunar Module. The Apollo Lunar Module was conceived, designed, and manufactured by the Grumman Corporation (Kelly, 2001). The most remarkable aspect of this achievement is that the LEM evolved from a conceptual idea to operational hardware on the lunar surface in a period of less than 9 years. The successful cooperation between NASA and contractor development teams was likely the main reason for the rapid development.

Shown in Figure 3 is an illustration of the LEM. The two parts of the LEM consisted of an Ascent and Descent Stage. The spacecraft was approximately 23 feet tall and 31 feet wide across the extended landing gear. The final vehicle mass was 36,100 lbs at Earth launch with approximately 10,000 lbs of wet Ascent mass upon liftoff from the moon (NASA, 1972). The Ascent Stage served a dual purpose of providing a habitable volume and transportation from the lunar surface to Low Lunar Orbit. The LEM was designed for a crew of 2 and duration of approximately 3 days on the lunar surface. The Descent Stage served the purpose of landing the spacecraft and providing consumables while on the lunar surface.

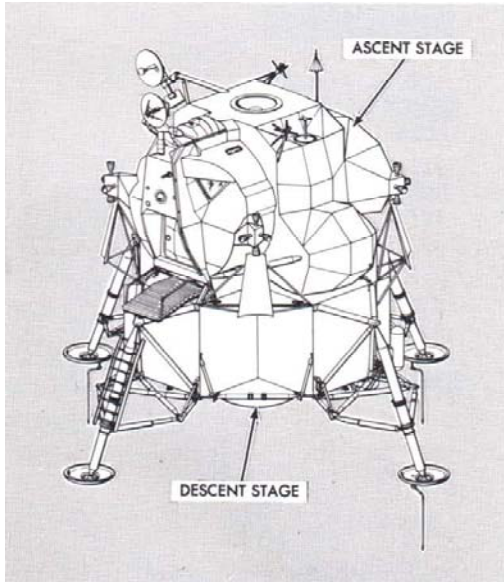


Figure 3: Apollo Lunar Module (NASA, 1972).

The development of the LEM proved to be a challenge for the Grumman Corporation. From the beginning of the LEM program, there were schedule delays, hardware failures, requirements creep, and mass increase after the conceptual design phase. The conceptual design launch weight of the LEM was initially proposed at 22,000 lbs and grew to 33,000 lbs by the time of Apollo 11 (Kelly, 2001). It is interesting to note that the five most important considerations given by Grumman in the conceptual design proposal of the LEM were:

- Propulsion design and development,
- Flight control system design and development,
- Reliability,
- Weight control, and
- LEM configuration.

All of the initial considerations experienced later design challenges that led to an increase in mass of the LEM and were not immediately obvious during the conceptual design phase. Mass became the main driver in the design of the LEM as subsystems were being developed. The mass growth from the conceptual design was largely due to overcoming technical challenges in the development of subsystems and integration. A large amount of mass increase was discovered through the use of mockup prototypes during preliminary design. The major factors that drove LEM mass during the preliminary design phase were reliability requirements, mission operational requirements, and configuration definition. As the design team studied the design reference missions specified by NASA, the understanding of mission operations requirements improved and the mass increased (Kelly, 2001). The final flight design of the LEM launched by the Saturn V fully utilized the allocated payload weight of approximately 36,100 lbs. A comprehensive story of the LEM development from the early concepts to the first lunar landing can be found in *Moon Lander: How we developed the Apollo Lunar Module* (Kelly, 2001).

2.6.2 Constellation Lunar Lander Program

When President Bush outlined the Vision for Space Exploration (VSE) in February 2004, the new exploration spacecraft that would achieve this goal were concepts based on previous studies. The development of the new Constellation program spacecraft would follow a "building block" approach as NASA planned to finish the International Space Station (ISS) and retire the Space Shuttle by 2010. The initial exploration study of the VSE was documented in the NASA ESAS report. In this report, various launch configurations and lunar missions were studied and an approach to the exploration architecture is shown in Figure 4.

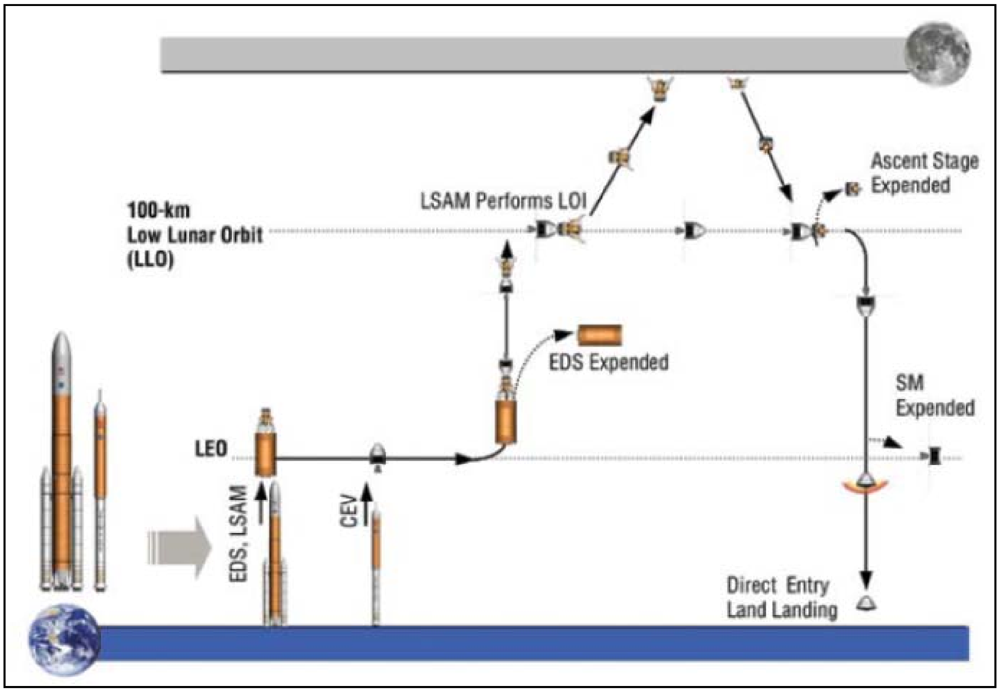


Figure 4: ESAS lunar mission architecture (NASA, 2005).

Unlike Apollo, the Constellation lunar mission architecture utilized two launch systems. The first launch vehicle, (later named Ares I), would consist of a single Space Shuttle derived 5-stack solid rocket booster first stage and a modified J-2X engine for the second stage. This launch vehicle would ferry a crew of 4 in the Orion CEV to LEO. The second launch vehicle, (later named Ares V) would be the next heavy-lift launch vehicle for the United States and would carry the Lunar Lander and Earth Departure Stage to LEO for docking with the Orion CEV. The Ares V concept utilized two 5-stack solid rocket boosters for the first stage and 5 RS-68 liquid engines for continuous burn throughout the first and second stages. The third stage of the Ares V utilized the J-2X engine of the Earth Departure Stage to reach a parking orbit in LEO. The Orion CEV and Ares I were the only systems to reach a PDR level of design maturity; while the Lunar

Lander, Earth Departure Stage, and the Ares V were not fully matured to PDR levels by the time the Constellation program was cancelled.

As outlined in the ESAS report, the Lunar Surface Access Module (LSAM) undocks from the CEV, performs an orbital plane change, and lands on the lunar surface with a crew of four. The Orion CEV would remain unattended in a yet to be determined lunar orbit. After a 7 day sortie, the ascent stage of the LSAM launches the crew into Low Lunar Orbit (LLO) to dock with the CEV. The ascent stage of the LSAM is discarded in LLO and the crew returns to Earth in the CEV. This lunar mission architecture utilized a method known as Earth Orbit Rendezvous – Lunar Orbit Rendezvous (EOR-LOR) and was found by the ESAS study team to provide superior performance and the highest reliability of mission success (NASA, 2005). Shown in Figure 5 is the proposed LSAM Lander concept detailed in the ESAS report.

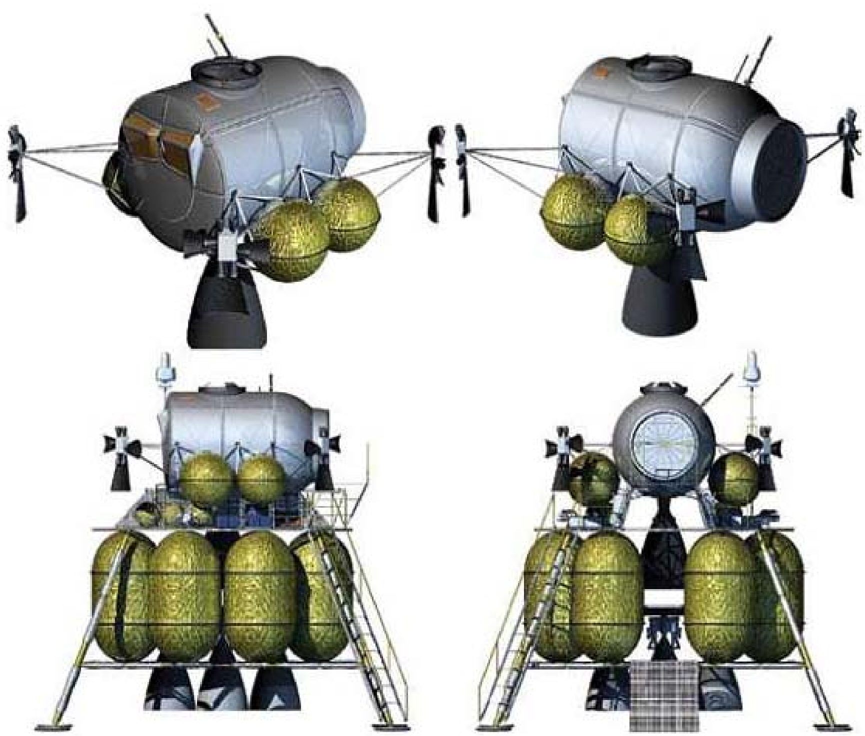


Figure 5: LSAM concept (NASA, 2005).

The proposed design objectives of the Altair Lunar Lander (formerly known as LSAM) were very much different from the Apollo LEM. The Altair Lunar Lander was designed to carry a crew of four, be able to land at any location on the moon, and remain on the surface for up to 2 weeks. The small size and weight of the Apollo LEM was largely driven by the shroud diameter and payload launch capacity of the Saturn V launch vehicle. The Apollo LEM was designed for landings close the Moon's equator and short sortie missions lasting no longer than 3 days. Unlike Apollo, the focus of lunar exploration in the Constellation program was to develop a permanent human presence on the moon. Shown in Figure 6 was the latest configuration of the Altair Lunar Lander concept at the end of the Constellation program.

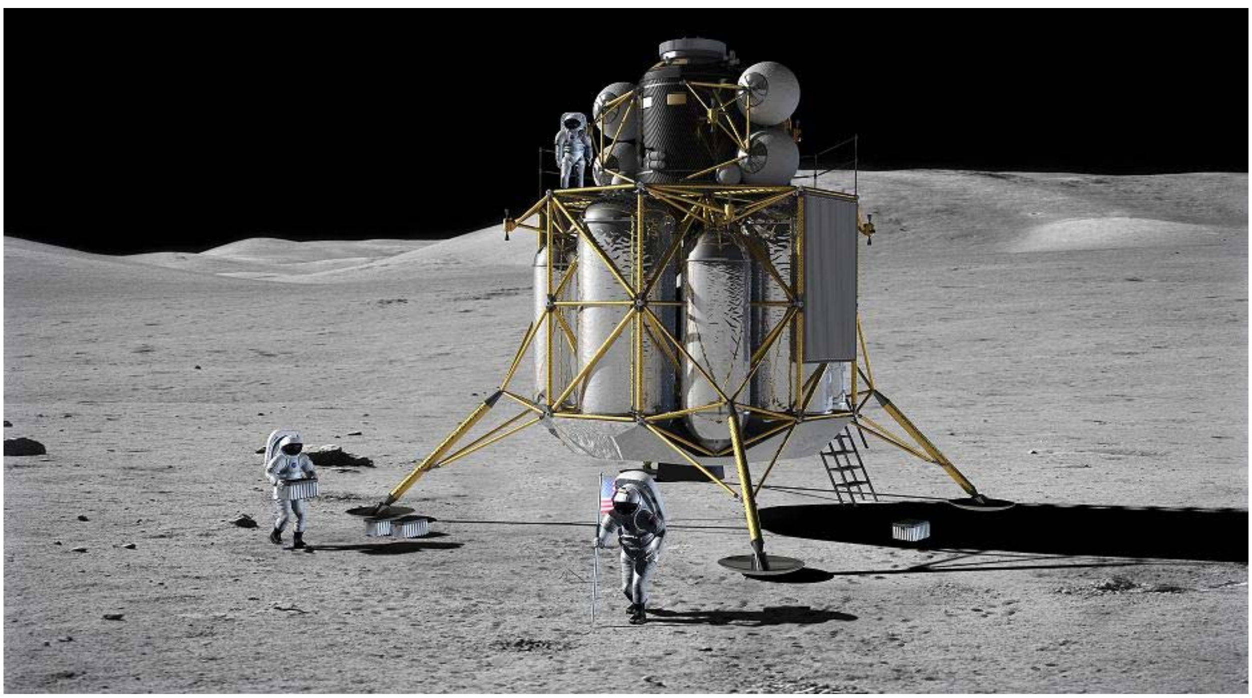


Figure 6: Altair Lunar Lander concept.

In addition to issues related to configuration and design, the cost of developing the Lunar Lander was a limiting factor. According to information presented by the NASA Lunar Lander

Project Office at the 3rd annual Space Exploration Conference, the cost of developing the Lunar Lander will exceed funding capability through Phase B. (Dorris, 2008). The two options that NASA faced were either postponing significant Lunar Lander work or pursuing a different approach. Because of the forecasted budget shortfalls, the Lunar Lander Project Office "streamlined" the systems engineering development process by "studying the design" and validating a "good set of requirements" before SRR.

The technological differences between the Apollo and Constellation programs were significant in regards to the amount of operational capability that was to be included in the Altair design. In Apollo, the objective was to land a man on the moon for short term sortie missions lasting no longer than 3 days. The focus of the Constellation program was to start with sortie missions lasting up to 7 days to develop infrastructure on the lunar surface for a future lunar outpost. The ESAS report concluded that the proposed architecture would have the following advantages when compared to the Apollo program;

- Double the number of crew to the lunar surface;
- Four times the number of lunar surface crew hours for sortie missions;
- Global lunar surface access with anytime return to Earth;
- Enabling a permanent human presence at a lunar outpost;
- Demonstrating systems and technologies for human Mars missions;
- Making use of in-situ lunar resources; and
- Providing significantly higher human safety and mission reliability.

Because the objectives of the Constellation program were more challenging than Apollo, a significant amount of technology development was required. As listed in the ESAS report, the following developmental projects were needed in the following areas: Structures and Materials, Protection, Propulsion, Power, Thermal Controls, Avionics and Software, Environmental Control and Life Support System (ECLSS), Crew Accommodations, Mechanisms, Analysis, Integrations, and Operations.

The large number of development projects was one of the biggest hurdles in the Constellation program (Chaplain, 2009). Because of funding issues and technical challenges, the Constellation program suffered many challenges and was officially cancelled in June 2011. It is unknown if future administrations will change the direction of human exploration back to the moon, but the current development of the Orion Multi Purpose Crew Vehicle (MPCV) as an exploration vehicle is a step in the right direction toward extending human spaceflight beyond LEO.

2.6.3 Lunar Spacecraft Development Summary

Perhaps the greatest issue facing both the Apollo and Constellation programs was the mass and development of the Lunar Lander. The challenges of minimizing the mass of the Lunar Lander were well known at the time of the ESAS report and led to the use of a non-traditional design approach known as minimum functionality design for the Altair Lunar Lander. The Apollo LEM experienced a 50% growth from the initial conceptual design phase due to several issues related to requirements creep, integration, and reliability. The Altair Lunar Lander project recognized the same issues as Apollo and attempted to design the spacecraft within NASA in order to better understand the challenges. However, the technological hurdles and development issues with a limited amount of funding proved to be too much for the Constellation program. Many lessons were learned during the development of the Altair Lunar Lander and it is hoped that this knowledge can be applied to future Lunar Lander designs. Until there is motivation to go back to the moon, the current exploration plan is to use the Orion MPCV for exploration beyond LEO. However, in order to establish a presence on the moon, resources will have to be directed toward the development of a Lander and surface systems.

2.7 LITERATURE REVIEW SYNOPSIS

In the design of a human spacecraft, many different aspects of the design must be considered. The conceptual design phase is driven by human creativity in the exploration of various spacecraft configurations. The task is challenging and many organizations have struggled to bring designs to operation, but the prevalent theme in a large majority of the literature is the need to explore and quantify the uncertainties associated with a conceptual design such that additional flexibility and margin for growth can be accommodated as the design matures. The information presented in this Literature Review explored many of the topics related to conceptual design and human spacecraft.

2.7.1 General Summary of the Literature

Much has been learned over the past 50 years in human spaceflight, but the risks and uncertainties of exploring new environments presents challenges to spacecraft designers. Since the 1960's, NASA has utilized a process known as Systems Engineering for the design, development, and operation of human spacecraft. Systems Engineering emerged from the management of technological projects during World War II (Emes *et. al*., 2005). Because SE is interpreted differently among various organizations, there is not a universal definition (Emes *et al.*, 2005). Systems Engineering is more than a traditional management approach, it serves as a leadership role to guide the traditional engineering disciplines (NASA, 2007a).

The conceptual design phase is one of the most important phases for determining the overall cost of a program (NASA 1995; Adelstein *et al*., 2006; Miller *et al*., 2008). According to NASA, conceptual designs are offered to demonstrated feasibility and support programmatic estimates (NASA, 1995). However, the processes used in conceptual design are often unstructured and not well understood (Jilla and Miller, 2004; Pacheco *et al*., 2003). Traditional methods rely on methods that settle on a preferred design early in the process without considering other many potential designs (McManus *et al*., 2004). In addition, the conceptual design phase is difficult to translate into a methodology that is useful to both experienced and inexperienced designers (Bryant, *et al*., 2005). Because of the lack of standardized process and difficulty communicating ideas, the conceptual design phase contains many uncertainties (Hastings and McManus, 2004; Chudoba and Huang, 2006; German and Daskilewicz, 2009). In order to understand the issues in conceptual design, models that express the relationships between the design parameters and fundamental requirements should be utilized (Scheidl and Winkler, 2010).

Before the start of conceptual design activity, the top level mission objectives or customer needs must be clearly defined. This is the foundation for what is expected in the overall design in terms of performance, risk, schedule, and cost (NASA, 1995). In the example of Constellation, NASA chose to redefine top level objectives based on inadequate levels of funding and technical challenges (Chaplain, 2009).

Given a set of clearly defined top level mission objectives, the next logical step in a spacecraft conceptual design process is to determine the activities that need to take place for the mission to be successful. A functional decomposition process identifies and states the "functions" or the activities that need to take place in the overall system architecture during a given mission. The development of functional models is one of the first steps in systematic methods for product development (Cowan *et al*., 2006; Pahl and Beitz, 1996; Ulrich and Eppinger, 2000, Roy *et al*., 2008; Kitamura and Mizoguchi, 2003). A drawbacks associated with functional decomposition is that the knowledge about functionality is usually scattered among designers in the conceptual design phase (Kitamura and Mizoguchi, 2003; Ingham *et al*., 2006). The large number of relationships among discipline specific functions cannot always be fully explored (Camelo and Mulet, 2010). Regardless of the challenges, functional decomposition remains a key process in conceptual design to understand the fundamental needs of the spacecraft in order to properly trade potential solutions.

Obtaining knowledge in the conceptual design phase is a key activity in the investigation of solutions that match to a functional decomposition. To describe how knowledge can flow across boundaries, Star (1989) coined the term "Boundary Objects". Boundary Objects are representations of knowledge used to improve communication and understanding between different groups (Star 1989; Carlile 2002; Carlile 2004; Dare *et al*., 2004). In a study conducted by Brereton and McGarry (2000) on engineering design thinking and communication, they concluded that design thinking is heavily dependent upon experiences with physical objects and materials and emphasize the need for hardware as a tool for visualizing abstract concepts and exploring the design space. The use of prototypes in the context of Boundary Objects facilitates communication between multidisciplinary groups and helps to resolve discrepancies before a design configuration is selected.

Physical prototyping has been successfully used to develop and understand design integration. During the development of the Apollo Lunar Module, many uncertainties were discovered in preliminary design with the use of mockups (Kelly, 2001). A mockup of proposed lunar habitats was constructed by NASA Johnson Space Center to demonstrate integration of several subsystems for near-term exploration habitats (Kennedy, 2006). Research conducted by Mulenburg and Gundo (2004) described a method titled "Design-by-Prototype" to simplify the conceptual design process. Although prototyping is not a new concept, its use in different contexts is prevalent in the literature. The main reason for using mockups, hardware, or prototypes is that uncertainties in the conceptual design can be investigated in a method that increases communication between multidisciplinary groups.

Computer Aided Design is a well established engineering tool used in the design of structures and assemblies. Computer Aided Design model information can be utilized by high fidelity analysis programs such as Finite Element Analysis and Computational Fluid Dynamics. Wang and Shan (2007) reported that it takes Ford Motor Company about 36 to 160 hours to run one crash simulation with their FEA models. The huge computational expense has led many researchers to investigate the use of approximation models as an alternative to high fidelity and time intensive computational codes. As computing power increases and technology becomes available, the future of Computer Aided Design is likely to utilize Virtual Prototyping to a greater extent. According to Choi and Chan (2004), for process simulation, virtual reality can be used to determine bottlenecks in manufacturing or production. The overall goal of virtual prototyping is to simulate the many physical aspects of a design in a digital world.

Predicting mass during the conceptual design phase has been a common problem for aerospace systems. Programs such as X-30 and X-33 were not as successful due to mass growth, technical difficulties, and incorrect assumptions during the conceptual design phase (Thompson *et al*., 2010). As observed with the recent cancellation of the Constellation program, mass growth in Orion and Ares I was attributed with overcoming technical challenges that had not been anticipated in conceptual design (Chaplain, 2008). During the Apollo program, the Lunar Module experienced a 50% growth in mass from the initial conceptual design while the Apollo Command and Service Module experienced a 42% increase in mass during its development (Kelly, 2001; Thompson *et al*,, 2010). The increase in payload mass proved to be difficult for the Saturn V as its first stage dry mass increased approximately 33% (Thompson *et al*,, 2010). A study conducted by Thunnissen (2004) concluded that margins for spacecraft mass growth should be approximately 50% from the conceptual design phase.

Multidisciplinary Design Optimization is an area of research that develops systematic approaches for the design of complex systems governed by interacting physical phenomena (Alexandrov, 2005). When first introduced in the 1990's, the use of MDO methods for complex engineering problems had great promise for solving and optimizing problems. However, the true use of MDO has been limited to mostly researchers and has not been widely used in realistic engineering problems (Alexandrov, 2005). The central challenge in MDO is balancing the use of coupled high-fidelity models with the amount of computational time required to generate an optimized solution (Messac and Mullur, 2008).

Many researchers have recognized the difficulty of coupling many different types of high-fidelity discipline specific codes and have created approximation techniques intended to reduce the computational time. Metamodeling is the process of creating a model that is an approximation of a complex analysis code model (or a model of a model) (Jin, *et al.,* 2001); Simpson *et al*., 2001; Simpson *et al*., 2004; Martin and Simpson, 2003; Clarke *et al*., 2005); Yang *et al*., 2005; Wang and Shan, 2007). Because the surrogate or approximation models are based on an objective function created from a finite set of data points, there is an amount of uncertainty in the design solutions (Martin and Simpson, 2006).

Tradespace exploration is a method used to explore the various objectives in order to understand the relationship to the design space variables. According to Ross (2006), tradespace exploration in the conceptual design phase may empower designers to overcome challenges associated with tendencies to reduce the design space and overlook potential design space solutions. Ross (2006) conducted interviews with industry and learned that broad tradespace exploration is rare and often done in an ad hoc manner. In the work by Shaw (1998), an approach named GINA was proposed to explore a large set of space system design options in terms of generic metrics for comparison among different concepts. The PhD dissertation by Jilla (2002) utilized the GINA process and added MOO techniques to fully explore the design and objective tradespace for distributed satellite systems. According to Hastings (2004), a Multi Attribute Tradespace Exploration tool was developed at MIT to analyze system architectures with the goal of maximizing system attributes. Ross (2003) further developed the MATE process into MATE-CON; which was an extension into Concurrent Design.

Design space exploration is similar in many respects to tradespace exploration where the input design variables are evaluated to determine relationships among various objectives. According to Acar (2010), metamodels are widely used in design space exploration and there is a need for developing techniques that increase the accuracy of the metamodel predictions. Nixon (2006) developed a systematic design space exploration process to keep the design effort manageable in the conceptual design phase. In the PhD dissertation by Villeneuve (2007), a concept and technology selection methodology for complex architectures was developed to quantify and explore the design space simultaneously.

The findings by Tumer *et al*. (2005) describe issues with NASA approaches for incorporating risk based design decisions early in the conceptual design process. The goal of their work was to lay the foundation for more collaboration between NASA researchers and the academic research community. An earlier NASA report by Knight and Stone (2002) suggested that risk based design methods make the design process more robust given that a systems-level understanding is incorporated and detailed knowledge about the subsystems is utilized.

The typical approach used by NASA for understanding risk in the early stage of conceptual design is through the used of failure and risk analysis methods using traditional reliability tools. In addition to the reliability methods listed, a method used by NASA to assess risk is PRA. Probabilistic Risk Assessment was identified as a need in 1996 in order to support decisions for Space Shuttle upgrades (Stamatelatos *et al*., 2002). The use of PRA identifies what can go wrong, how frequently it will happen, and what are the consequences (Tumer *et al*., 2005). The issue with PRA is that it requires a significant amount of information before a detailed analysis of the risk can be completed. In the early stages of conceptual design, many factors such as the particular type of technology used in subsystems, the probability of failure of components, and the exact layout and integration of the subsystem components many not be fully defined.

A common tool for understanding potential areas of failure is FMEA; which has been extensively used for systems safety and for the reliability analysis in many industries (Sharma, 2007). FMEA is a top down approach that focuses on the loss of functionality in a given design.

Another failure mode approach that focuses on specific components in functions is known as a FMECA. A FMECA is a bottom up reliability analysis that evaluates all of the major components in order to identify single point failures and may identify hazards overlooked in other system analysis techniques (FAA, 2005). The disadvantage for FMECA is that it does not account for multiple and coupled faults and failures; thus it could give an optimistic estimate of system reliability if a quantitative approach is used. A Fault Tree Analysis is a top-down approach that illustrates the sequence of events that lead to an unfavorable event and provides a quantitative estimate of system reliability (FAA, 2005). FTA uses logical symbols to graphically show the progression of a failure on other components in the system. Much like the FTA, an ETA is a bottom up approach that graphically explores system responses to an initiating event and assesses the probability of the outcome (FAA, 2005). The use of ETA allows designers to assess multiple coexisting faults and failures and identifies vulnerabilities.

Another common technique that is used in tradeoffs related to system safety is the use of Reliability Block Diagrams. A Reliability Block Diagram illustrates the logical connection between subsystem components and can be used to mathematically model the system probability, thus it is useful for evaluating various potential configurations (FAA, 2005). In the example of early conceptual design, designers would assign reliability targets for specific components in the subsystems if data was not available. A key issue identified by Tumer *et al.* (2005) is the need for newer approaches that couple traditional reliability analyses with a more risk informed approach in the conceptual design phase.

Minimum functionality design is an approach where a baseline configuration is defined before trading other factors in human spacecraft designs. Minimum functionality recently gained attention due to its use by the Altair Lunar Lander Project. The design approach for minimum functionality in the development of the Altair Lunar Lander started with a single point baseline design point of departure for cost and risk trades in order to justify mass "add-backs" to the subsystems in the form of additional redundancy and safety (Cohen, 2009). In addition, the Orion Crew Exploration Vehicle used a similar approach of creating a baseline design with a set of "minimum functions" called the "Zero Baseline Vehicle" (Hu *et al*., 2008). According to NASA's Design, Development, Test, and Evaluation Considerations for Safe and Reliable Human Rated Spacecraft Systems, a minimum functional design is the simplest, most robust, and highest performance design option as the starting point for assessing fault tolerance (Miller *et al*., 2008). The NASA guidelines outline a conceptual design minimum functionality process for spacecraft development; outlined in section 2.3, Conceiving the Right System, Critical Activities Early in the Life Cycle (Miller *et al*., 2008). Much of the confusion between different minimum functionality approaches is how the minimum (or starting point) configuration is defined before trading other aspects of safety, reliability, performance, and cost.

Human Rating Requirements for Space Systems are defined in NASA NPR 8705.2B. In this document, the human rating certification process, certification requirements, and technical requirements for human rating are specified. The set of human rating requirements is likely the minimum set of requirements that must be satisfied in order to achieve human rating certification.

A human lunar spacecraft, commonly known as a Lunar Lander or Lunar Habitat is very much different from other spacecraft designs such as capsules and lifting bodies. Although there are many similarities to conventional Low Earth Orbit operational spacecraft, a lunar spacecraft must operate in a different set of environments on the surface of the moon including thermal cycling, radiation, dust, lighting, and micrometeoroids (Cohen, 2009). Significant lessons were learned during the Apollo missions that will assist designers in the development of future lunar spacecraft.

The goal of the Apollo program was to put a man on the moon and return him safely to Earth. The Apollo Lunar Module was conceived, designed, and manufactured by the Grumman Corporation (Kelly, 2001). The most remarkable aspect of this achievement is that the LEM evolved from a conceptual idea to operational hardware on the lunar surface in a period of less than 9 years. The conceptual design launch weight of the LEM was initially proposed at 22,000 lbs and grew to 33,000 lbs by the time of Apollo 11 (Kelly, 2001). The major factors that drove LEM mass during the preliminary design phase were reliability requirements, mission operational requirements, and configuration definition. The proposed design objectives of the Constellation Altair Lunar Lander (formerly known as LSAM) were very different from the Apollo LEM. The Altair Lunar Lander was designed to carry a crew of four, be able to land at any location on the moon, and remain on the surface for up to 2 weeks.

Perhaps the greatest issue facing both the Apollo and Constellation lunar architectures was the development of the Lunar Lander. The issues with minimizing the mass of the Lunar Lander were well known at the time of the ESAS report and led to the use of the non-traditional minimum functionality design approach. The technological differences between the Apollo and Constellation programs were significant in regards to the amount of operational capability that was to be included in the Altair design. Because the objectives of the Constellation program were more challenging than Apollo, a significant amount of technology development was needed for the new spacecraft architecture. The large number of development projects was one of the biggest hurdles in the Constellation program (Chaplain, 2009).

The Constellation program suffered many challenges and was officially cancelled in June 2011. It is unknown if future administrations will change the direction of human exploration, but the current development of the Orion MPCV as an exploration vehicle is a step in the right direction for spaceflight beyond LEO. After the cancellation of Constellation, no further plans for Lunar Lander development have been proposed by NASA.

2.7.2 Relationship to Studies in Chapters 3, 4, 5, and 6

The design methodology described in Chapter 3 build upon the previous approaches of minimum functionality and conceptual human spacecraft design. A new minimum functionality design methodology focused on the development of conceptual human spacecraft is developed and verified against the Apollo LEM design. The information gathered from the Altair Lunar Lander LDAC cycles, the Orion ZBV approach, and the NASA DDT&E guidelines for conceptual human spacecraft design using a risk based approach were examined in the development of a systematic minimum functionality approach for conceptual design. Systems Engineering will continue to be utilized within the aerospace industry for development of human spacecraft, but within the context of SE, additional tools and procedures are needed to fully explore the design space and objective tradespace in an efficient manner.

A Conceptual Lunar Ascent Module Program (CLAMP) was developed as a result of the research described in this chapter. The code was verified using Apollo 15 historical data in order to establish the mathematical methods used to quantify *Safety* and *Operability* as a function of overall spacecraft mass. Many sources of Apollo literature were used in the verification of the program. The information gathered from the development of the Apollo Lunar Module and the Constellation Altair Lunar Lander was essential in the development of the program.

The research conducted in Chapter 4 was based on a combination of minimum functionality and conceptual design physical prototyping in the context of building full scale physical prototypes to evaluate human factors as an early step in the conceptual design of human spacecraft. The use of physical prototypes as Boundary Objects described by Star (1989) demonstrated the need for physically evaluating hardware configurations in the context of human factors engineering. The findings by Brereton and McGarry (2000) were demonstrated in this exercise as potential issues with subsystem layout, human factors, and design thinking and communication were discovered. Prototypes of the ESAS baseline Lunar Lander Ascent and Habitation Module were constructed and studied for interface definition, risk identification, and uncertainties. This activity served as pre-validation of a preferred spacecraft geometry and configuration before detailed CAD modeling and design. Throughout the study, the layout and geometry of the conceptual design was reconfigured at least 9 times and used simple reconfigurable materials instead of traditional expensive wood prototypes. The combination of Systems Engineering, minimum functionality, conceptual design, and human factors in this activity led to a new project based design curriculum.

The research conducted in Chapter 5 expanded upon the minimum functionality design methodology presented in Chapter 3 to fully explore the multidisciplinary tradespace. In typical human spacecraft designs, decision makers are required to evaluate and balance many competing objectives. A vast amount of literature has been published on methods designed to make the conceptual design process more efficient in terms of developing Pareto frontiers and exploring the objective tradespace. Building upon the previous methods presented in the literature for minimum functionality, multidisciplinary design optimization, reliability analysis, and risk based design, a mass addition process was developed to explore the design space and objective tradespace of three Lunar Ascent Module spacecraft configurations. The tradespace of *Safety*, *Operability* and Total Spacecraft mass is proposed as an efficient method for evaluating the impacts of mass addition of safety components, redundancy, and operational characteristics.

The methods presented in Chapter 6 defined two figures of merit, the "Safety Index" and the "Operability Index." Using the guidance expressed by Tumer *et al*. (2005), for risk based approaches in conceptual design; this research explored the benefits of the utility parameters as an early indication of reliability without the need for detailed information about the probabilities of failure in the subsystem components. Building upon established reliability and risk approaches as described by Miller *et al*. (2008), the Safety Index and Operability Index provide designers with an easy to utilize figure of merit that indicates potential issues in design configurations.

CHAPTER 3

A MINIMUM FUNCTIONALITY DESIGN METHODOLOGY FOR DETERMINING SAFETY AND OPERABILITY MASS IN HUMAN SPACECRAFT

3.1 ABSTRACT

A systematic methodology is presented for defining a minimum functionality baseline configuration of a human spacecraft. In order to estimate a lower bound for the spacecraft mass, a set of essential functions is coupled to single string subsystems with zero fault tolerance. This minimum functionality baseline is defined to meet the physical requirements needed to transport the crew to the target destination and ensure their physiological needs are met; but without margin, dispersions, redundancy or factor of safety. This constitutes a set of 'non-negotiable' requirements based on fundamental parameters derived from Physics and Physiology. By definition, this represents a technically feasible solution, but results in the 'highest risk' design. Mass additions beyond the minimum functional configuration are allocated to increase *Safety* through redundancy, fault tolerance or factor of safety, or to increase *Operability* through additional mission functionality or improved human-system interfaces, and are determined by risk analyses and design trade studies. This proposed methodology was used to analyze a range of lunar ascent stage spacecraft configurations and a process was developed to allow systematic estimation of mass for the specified spacecraft subsystems. The modeled results are verified by comparison to actual subsystem mass of the Apollo Lunar Module ascent stage.

3.2 INTRODUCTION

A "minimum functionality" design process defines a baseline configuration of a spacecraft that can meet the stated mission objectives as a first step before trading parameters such as performance, enhanced objectives, safety, and cost. NASA described a minimum functionality configuration in the Broad Agency Announcement NNJ08ZBT001 for the Constellation Lunar Lander development study. In this context, a spacecraft configuration is defined by the various subsystems and components that make up the overall design. A minimum functional configuration would not be considered a realistically flyable spacecraft by NASA because of the inherent high risk and reduced safety due to the lack of margin, factor of safety and subsystem redundancy. However, it does serve as a conceptual lower boundary for further iteration of the design in order to assess the impact of subsystem mass additions needed to provide fault tolerance and increase safety (Cohen, 2009). According to NASA's Design, Development, Test, and Evaluation Considerations for Safe and Reliable Human Rated Spacecraft Systems, a minimum functional design is the simplest, most robust, and highest performance design option as the starting point for trading safety and reliability (Miller *et al*., 2008). This definition takes into account expected fault tolerance requirements and represents a more realistic vehicle but not a true minimum functionality design. Likewise, the Orion Crew Exploration Vehicle used a similar approach of establishing a baseline design with a set of "minimum functions" called the Zero Based Vehicle (Hu et al., 2008). In each of these approaches, the common theme is that a baseline spacecraft configuration is first defined to meet the desired requirements for a "minimum" mission. However, by this definition, certain risk mitigation and operational ground rules are assumed in the baseline configuration that may not be absolutely essential for mission success.

Instead of describing minimum functionality according to a set of predetermined mission and safety requirements, the minimum functional design methodology presented in this work defines a minimum functionality configuration as one that is just capable of meeting the physical requirements needed to transport the crew to the target destination (e.g., rocket propulsion, structural provisions, minimum habitable volume, no redundancy or factor of safety, etc.) while ensuring that the essential physiological requirements of the crew are also met (e.g., metabolic consumable provisions, waste collection, atmosphere regulation, etc.). These requirements constitute a set of 'non-negotiable' parameters based on the *Physics* and *Physiology* needs of a given mission. Subsequent mass additions and spacecraft growth such as margins and abort capabilities above the 'non negotiable' parameters are allocated to either an increase in *Safety* (reducing risk, increased reliability through redundancy, fault tolerance, robustness, mitigating contingencies, adding margin, increasing factor of safety, etc.) or *Operability* (enhanced mission objectives, increased mission specific capabilities, automation, improved user interface, etc.). The combination of these four fundamental mass parameters forms the foundation of this methodology and can be expressed as:

$$\sum \text{Spacecraft Mass} = \underbrace{f(Physics) + f(Physiology)}_{\textit{'non-negotiable'}} + \underbrace{f(Safety) + f(Operability)}_{\textit{'tradespace'}} \qquad (2)$$

3.2.1 Objectives of Design Methodology

This methodology provides a structured approach for developing and evaluating conceptual spacecraft configurations using a minimum functionality baseline that is not biased toward safety or mission objectives. It also provides designers with a process for quantifying the amount of mass dedicated to *Safety* and *Operability*. Increases in mass due to Factor of Safety for structural and pressure storage components are captured in the *Safety* mass and increases in the mission objectives or automation are captured in the *Operability* mass. The methodology can be adopted for a variety of conceptual designs and provides a means for relative comparison between spacecraft configurations. Many different and often competing spacecraft concepts must be explored in a timely manner during the conceptual design phase to determine feasible solutions. One of the dangers of focusing on a single conceptual design too quickly is that other, more flexible design configurations could be overlooked; leading to designs that are mass constrained in further development. Mass growth after conceptual design is a common issue and incorrect assumptions about the technology, integration, and configuration of the spacecraft can contribute to unforeseen mass growth and reduced performance as a consequence (Thompson *et al*., 2010).

It is clear that human spacecraft must be safe and reliable, but this should be achieved without the added penalty of over-designing the vehicle with unnecessary additional mass and complexity. Because human spacecraft are typically volume-constrained, highly complex systems designed for specific mission objectives, a large amount of uncertainty tends to exist in the early stages of development. Examples of this uncertainty include initial estimates for subsystem mass budgets, spacecraft geometry, and reliability in components. Conceptual designs must be flexible and adaptable enough to accommodate unforeseen problems as the design is matured, because decisions made in this early phase are critical to the cost and success of the spacecraft program (McManus *et al*., 2004). In addition, the conceptual design phase is often described as being unstructured with designers pursuing single designs instead of considering a range of concepts due the vast scope of the overall design challenge and perceived time and effort needed to conduct credible analyses on multiple options (McManus *et al.*, 2004; Jilla and Miller, 2004).

The goal of this work is to demonstrate the application of a proposed systematic methodology for the conceptual design of human spacecraft. The lunar ascent module is a unique type of human spacecraft because it has a large impact on the mass of other systems such as launch vehicles and propulsive units; but is relatively straightforward to define in terms of required functionality within a limited tradespace. The Apollo Lunar Ascent Module was used to verify the design methodology and a modeling approach was developed to define and calculate the mass contributions based on *Physics, Physiology, Safety*, and *Operability* for a conceptual minimum functionality lunar ascent spacecraft. A series of parametric analyses was then conducted using varying levels of *Safety* and *Operability* and the results were combined into a tradespace of spacecraft configurations to provide designers with information about how specific mass impacts are related to the overall spacecraft design.

3.3 BACKGROUND

The design approach used by NASA for the Altair Lunar Lander started with a baseline design point of departure for cost and risk trades in order to justify mass “add-backs” to the subsystems in the form of additional redundancy and safety (Cohen, 2009). Because the Lunar Ascent Module contributed disproportionately to the overall mass of the lunar mission architecture, a reduction in mass of the lunar ascent module would reduce the required Earth launch mass significantly. For an Apollo or Constellation type lunar architecture mission, the lunar ascent module mass is “propelled” five times through the various mission phases of Earth launch, Earth departure, lunar orbit insertion, lunar landing, and lunar ascent. Using the Apollo

Lunar Excursion Module as an example, a 1 lb. increase in the Lunar Ascent Module mass increased the Saturn V gross launch vehicle mass by 883 lb. (Thompson *et al*., 2010). Thus, a need to design the Altair Lunar Ascent Module to its minimum optimized mass was one of the reasons NASA chose a “non-traditional” spacecraft design philosophy. According to the NASA Broad Agency Announcement for the Constellation Lunar Lander Development Study in 2008, the minimum functionality approach was used to buy down risk through an “add-back” process. The non-traditional design approach used by NASA was slightly different from a typical government oversight program of industry charged design and development. The minimum functionality approach was also based on a belief that a traditional Systems Engineering (SE) approach was too costly (Cohen, 2009). The intent of the initial government study was to mature the conceptual design to a point in the lifecycle such as Systems Requirements Review or System Definition Review before handing the design over to industry for further development. The initial work conducted by NASA trained a workforce in spacecraft development and developed a preferred design configuration intended to reduce unnecessary trade studies in later phases of the development cycle.

To avoid later design issues, NASA asked industry to provide feedback to the LDAC-1 minimum functional configuration. The work by Cohen (2009) presented several findings for design issues related to crew productivity, pilot view angle, separate habitat and airlock, required habitable volume, required airlock volume, pressurized payload implications for lunar surface science, and symmetry about the thrust axis. Although NASA proposed the minimum functional design for the LDAC cycles, there were issues raised about human factors being an important part of the top level objectives (Cohen, 2009). A full understanding of the role of human factors and the required habitable volume needed for sustaining humans for a long period of time such as an extended lunar sortie is one of the issues that should be addressed in any minimum functional design approach for human spacecraft.

A minimum functionality design approach was utilized in the Orion Crew Exploration Vehicle at the end of the first Design Analysis Cycle. This approach was utilized to "scrub" the mass of the entire spacecraft before System Definition Review (Hu *et al*., 2008). The minimum functionality configuration was named the Zero Based Vehicle and the process used for defining the ZBV was very similar to NASA's minimum functionality design approach, but differed in how the minimum baseline configuration was defined. The baseline Orion ZBV included a few redundant subsystems based on the ground rules of one fault tolerance for safety and zero fault tolerance for mission success (Jordan, 2009). A risk balancing approach was used to identify vulnerabilities in subsystems and determine mitigation strategies. Using a two round approach for justification of subsystem technologies and desired functionality, a total of 2500 kg was removed from the original DAC-1 configuration (Hu *et al*., 2008).

Quantifying safety in a human spacecraft is not an easy task because of the large number of associated hazards and uncertainties that must be characterized. The NASA Procedural Requirements document NPR 8705.2B defines critical safety requirements for obtaining human rated certification. The technical requirements defined in Section 3 of this document specify safety measures taken through fault tolerance, redundancy, control and abort modes. As mentioned in paragraph 3.1.2, the requirements are not intended to be "all inclusive or an absolute prescription for human rating" and risks in the spacecraft design must be identified and understood. The approach used by NASA for understanding risk in the early stage of conceptual design is through the use of failure analysis, risk analysis, traditional reliability methods, and Probabilistic Risk Assessment. A PRA identifies what can go wrong, how frequently it will happen, and the likely consequences (Stamatelatos, 2002). The issue with PRA is that it requires a significant amount of detailed information before an analysis of the risk can be completed. In the early stages of conceptual design, many factors such as the particular type of technology used in subsystems, the probability of failure of components, and the exact layout and integration of the subsystem components many not be fully defined.

3.3.1 Goals of Minimum Functionality Design

The goal of a minimum functionality design approach is to design a spacecraft that is sufficiently safe and reliable, while balancing the mass requirements related to optimizing performance, safety, risk, and cost. In previous minimum functionality design approaches, the baseline configuration varied according to assumptions made for the "minimum" boundary. However, the design methodology presented in this research takes this idea one step further by defining groupings of mass based on the four fundamental parameters of *Physics*, *Physiology*, *Safety*, and *Operability*. The advantage of this methodology compared to previous methods is that it can be readily modified for different mission applications and allows the relative impact of mass addition to be systematically quantified in the context of total spacecraft mass.

3.4 DEFINING A MINIMUM BASELINE FOR ASSESSING SAFETY AND OPERABILITY

The development of this methodology began in the fall of 2006 when a group of graduate students from the University of Colorado, inspired by a prior class design project analyzing NASA's Vision for Space Exploration, began construction of a full-scale Lunar Lander habitation mockup with goals of better understanding the volumetric constraints of the proposed lunar vehicle architecture and supplying feedback to the stakeholders. Beginning in 2007, the project evolved into a focused Systems Engineering approach in which top level requirements were evaluated early in the program through the use of full-scale, low cost, rapid prototyping techniques. The research conducted between 2007 and 2009 focused on prototyping in the context of multidisciplinary design optimization (Higdon and Klaus, 2008; Klaus and Higdon, 2009). A "rapidly reconfigurable" minimum functionality prototype of the ESAS minimum functionality lunar ascent module was constructed and outfitted with single string subsystems for studying human factors, subsystem integration, layout, and technology choices. The lessons learned in the prototyping activity of the minimum functionality lunar ascent module formed the foundation of this methodology.

The proposed design methodology outlines an approach for quantifying the Minimum Functionality, *Safety*, and *Operability* mass in human spacecraft designs in order to evaluate and trade various spacecraft configurations based on key mass drivers. This approach quantifies *Safety* and *Operability* through an "add back" process that allows designers to understand how increases in redundancy, safety components, or additional mission functionality are related to the overall spacecraft mass. The italicized terms, *Physics*, *Physiology*, *Operability*, and *Safety* are used throughout to refer to the mass associated with each of the parameters. This process is best used in the early stages of conceptual design when the fundamental parameters of spacecraft geometry, number of crew, structural materials, and trajectories are being studied to determine the most feasible design solutions. Shown in Figure 7 is a flow chart of the minimum functional design methodology starting with a set of top level mission objectives.

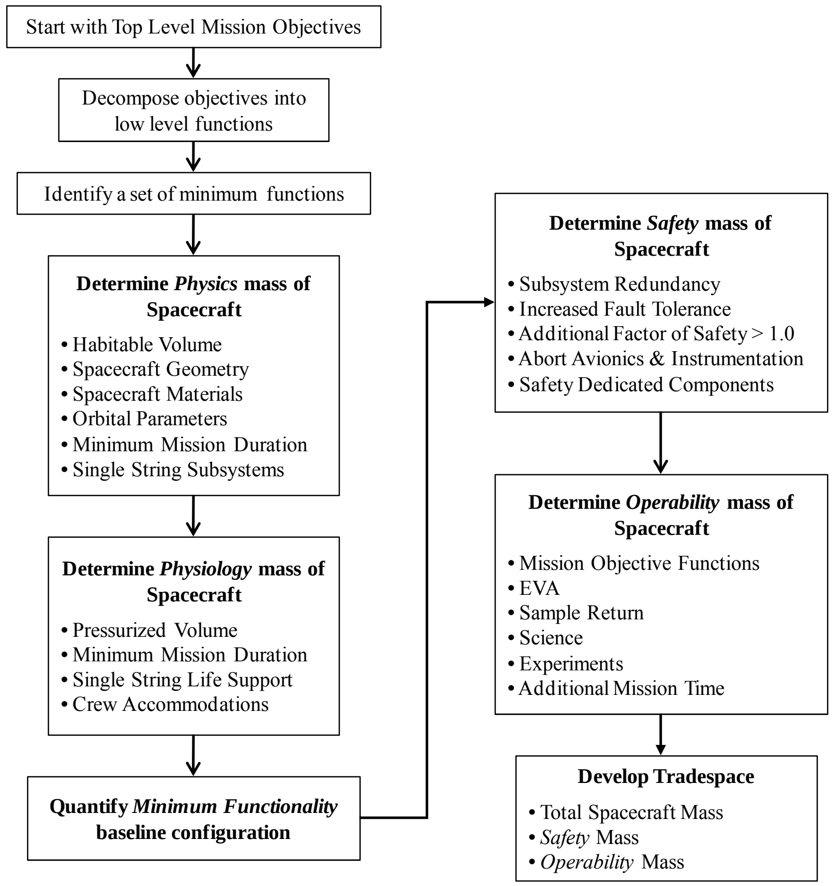


Figure 7: Flowchart of the Minimum Functionality design methodology.

3.4.1 Top Level Mission Objectives

The significant drivers of a human spacecraft design are the top level mission objectives. It can be very confusing to the design teams if the top level mission objectives change in the middle of design activity. In the example of Orion, the top level mission objectives changed from Lunar to ISS, then to an ISS Rescue Vehicle, and now Beyond Earth Orbit. Thus, a need for a solid foundation of the operational environment is a necessary requirement for developing an efficient design. The top level questions that form the basis of any human spacecraft are: What is the destination; Moon, Mars, or Asteroid? How many crewmembers will participate; 2, 3, 4, or more? How much time will crewmembers spend at the destination; 1 day or 6 months? What type of spacecraft will transport the crew during different phases of the mission; Module, Capsule? The answers to these questions determine the type and size of a spacecraft and launch vehicles within the mission architecture. The combination of all the systems in the mission architecture is a very complex problem and becomes a large integration problem as observed with the Apollo Lunar Ascent Module.

3.4.2 Functional Decomposition

The functional decomposition presented in this design methodology is focused on the design of a single individual spacecraft within an overall mission architecture. In the example of a Lunar Ascent Module, its specific functionality is only a small part of the total mission architecture. However, the Lunar Ascent Module is a critical part of the lunar mission and its required functions are necessary for the lunar ascent phase. A functional decomposition matrix was developed to capture all of the necessary activities that need to occur in a spacecraft at the lowest subsystem and Line Replaceable Unit (LRU) component levels. Within the global set of functions for the spacecraft, a subset was identified to define a minimum functionality baseline configuration needed to transport the crew to the desired destination and keep them alive for minimum mission duration. The intent of the functional decomposition activity is to list as many of the required functions as possible given the current knowledge of the mission objectives, while allowing the flexibility to add new functions as the design matures.

Assuming the top level mission objectives are clear, the first step is to create a functional decomposition matrix of the necessary spacecraft subsystems to answer the question "What activities need to occur to make the mission successful?" Using this fundamental question as a guideline leads the designer to the lowest level minimum functionality required in the spacecraft. Examples of low level functions could include: “Remove Carbon Dioxide from Cabin Atmosphere” or “Store Main Engine Fuel”. The use of a verb-noun descriptor to identify the lowest level functions in the spacecraft makes the intent of the function easy to understand and group to a particular subsystem. A list of subsystems and high level functions decomposed into lower level functions is listed in Figure 8.

Human Spacecraft Subsystems

Avionics	**Crew Accommodations**	**Payloads**	**Environmental Control and Life Support System (ECLSS)**	**Power**	**Thermal**	**Structures**	**Propulsion**	**EVA**
Provide Command and Data Handling	Support Human Factors	Return Science Payload	Maintain Atmosphere	Store Power	Collect Heat	Provide Load Bearing Capability	Provide Main Propulsion	Environmental Protection
Provide Navigation	Provide Lighting	Store Tools and Equipment	Provide Food and Water	Distribute Power	Transport Heat	Provide Environmental Protection	Provide Flight Control	Allow External Mobility
Provide Guidance and Control	Provide Restraints and Handholds	Store Photography	Manage Waste	Regulate Power	Remove Heat	Support Subsystems		
Store Spacecraft Data	Maintain Health			Provide Grounding		Provide External Interfaces		
Monitor Vehicle Subsystems	Store Operational Supplies			Provide Overload Protection		Provide Direct External Observation		
Provide Communications				Provide Shielding		Support TPS and MMOD		

Figure 8: Human spacecraft subsystems and high level functions.

The functional decomposition matrix defines all of the necessary activities that need to occur in a spacecraft. Although there are a number of functions that would not be included in a minimum functional design, the intent of the functional decomposition activity is to list as many of the activities as possible given the current knowledge of the mission and allow flexibility to add more functions later as the design matures.

3.4.3 Candidate Technology Selections

After the functional decomposition matrix has been developed and peer-reviewed by the stakeholders, the next step is to identify candidate hardware or technologies to perform the intended functionality. Using the low level function "Remove Carbon Dioxide" as an example, there are several different methods of removing Carbon Dioxide such as Lithium Hydroxide Canisters, Molecular Sieve Beds, and Catalyst Beds. Thus, a few different technologies exist that would not require a development of new technology to meet this function. The tradeoff is deciding the appropriate technology for the spacecraft. Each technology choice has a specific mass, volume, and perhaps power requirement (which leads to additional required mass). Additional questions for selecting a technology include the following:

- How will this technology choice integrate into the current knowledge of the spacecraft design?
- How much maintenance will be required?
- Is the technology scalable or does it require a redesign for this particular application?
- Is this technology proven or will it require significant development activity?
- What are the unknowns associated with this technology?

Many decisions are made concerning the technology choices that contribute to the uncertainty in the overall design. Simply assuming that a particular technology choice has flight history does not always work for the current design. In addition to the mass and volume characteristics, a probability of failure in the technology is considered in the overall subsystem reliability analysis or PRA. The PRA is usually completed at a later time in the design when reliability information about the subsystems is available. In the early stages of conceptual design, assumptions can be used for expected probabilities, but this also adds uncertainty and the PRA must be refined at a later time. The choice of technologies is of prime importance when developing conceptual designs because the decisions made during this process ultimately affect the success or failure of the conceptual design moving forward.

If the total number of combinations of technologies, redundant number of hardware component paths, and spacecraft minimum functions are multiplied to determine the number of spacecraft configurations, the number quickly becomes so great that the total possible number of subsystem hardware and redundant combinations cannot be fully explored in a timely manner. This "curse of dimensionality" is one of the reasons for developing reduced order models or commonly known as a model of a model or "metamodels" in the early stage of conceptual design to examine a reduced order design tradespace.

3.4.4 Minimum Functionality Baseline

The minimum functionality mass is the sum of the 'non-negotiable' mass parameters, *Physics* and *Physiology*;

$$Minimum\ Baseline = f(Physics) + f(Physiology). \tag{3}$$

The first non-negotiable mass parameter, *Physics*, is a function of the following variables and subsystems:

$$Physics = f(Avionics, Power, Thermal, Structures, Propulsion, Duration) \tag{4}$$

The *Physics* mass parameter is defined by incorporating single string subsystems in the Avionics, Power, Thermal, and Propulsion subsystems and the mission duration. The structural mass components are designed at a Factor of Safety of 1.0 for given nominal loads without dispersions. The second parameter, *Physiology*, is expressed as:

$$Physiology = f(Pressure\ Vessel, Life\ Support, Crew\ Accommodations, Duration) \quad (5)$$

The *Physiology* mass parameter is defined as a function of the pressure vessel, single string subsystems in the Environmental Control and Life Support System (ECLSS), minimum crew accommodations, and mission duration. Because the internal atmospheric composition, pressure, and temperature are the driving factor for the crew's physiology, the pressure vessel skin structure, and passive thermal protection is considered to be part of the *Physiology* mass. The pressurized volume is determined by combining the minimum habitable volume and the internal subsystem component volume. The *Physics* parameter can be thought of as the mass to keep the entire spacecraft alive and the *Physiology* parameter is the mass needed to keep the crew alive.

A minimum functional design may not be a minimal optimized mass design. The minimum functionality is dependent upon a few fundamental parameters in any conceptual design. Changes in any one of these fundamental parameters will change the minimum functional design point. Thus, spacecraft designers must realize that choices made for subsystem technologies and materials can change the starting boundary condition of a minimum functionality design analysis. The fundamental parameters that are assumed constant in a minimum functional design are the number of crew, habitable volume, subsystem technology choices, baseline mission time, and spacecraft geometry and structural materials.

3.4.5 Safety Mass in Spacecraft Configurations

The mass associated with *Safety* is described as additions necessary to mitigate potential contingencies, reduction of failure modes through subsystem redundancy, and mass dedicated to singular components for increasing safety. Increases in FoS greater than 1.0 for structural components are allocated to an increase in *Safety*. The increase in structural mass also drives other subsystem mass such as increased Propulsion. The *Safety* mass of a spacecraft is calculated as a function of the following variables, components, and subsystems:

Safety = f(Factor of Safety, Fault Tolerance, Safety Components, Additional Propulsion, Abort Avionics, Additional Instrumentation, Additional Duration, Spacecraft Growth) (6)

Safety mass is defined as:

- <u>Factor of Safety</u> – Factor of Safety mass addition for structural components, tanks, lines, pressurant valves, etc... (Fractional mass of components above FoS = 1.0).
- <u>Fault Tolerance</u> – Additional subsystem redundant components or pathways to mitigate potential component failure modes to prevent loss of spacecraft functionality.
- <u>Safety Components</u> – Fire extinguishers, alarms, medical kit, special tools, guards and mass that has a singular purpose for providing safety and not contributing to the minimum functionality mission.
- <u>Additional Propulsion</u> – The amount of mass increase in the propulsion system due to the increase in safety mass; this includes additional propellant, larger engines, larger tanks, and plumbing. Included is the amount of additional propellant needed to correct an off-nominal event or “Anytime Return” scenario.

- Abort Avionics – Guidance, navigation, and control components that are designed to return the spacecraft to a safe flight configuration in the event of an off-nominal event.
- Instrumentation and Data Storage – Any data collected and stored that is not necessary for conducting the minimum functional mission including sensors, wiring, and gages.
- Additional Duration – Spacecraft operating time needed to correct an off-nominal event or condition such as an abort scenario for rescue.
- Subsystem Growth due to Safety – Spacecraft growth of other subsystems such as Thermal, Power, Primary Structure; due to an increase in the number of safety components and redundancy.

When assigning a function to one of the two non-negotiable mass parameters (Physics or Physiology), one can ask: "Is this component or subsystem string necessary to meet the mission objective if everything on the vehicle worked perfectly?" If the answer is no, then it either contributes to *Safety* or *Operability*. In the case of the primary or secondary structural components at a higher FoS than 1.0; the amount of mass that is allocated to *Safety* is the mass needed beyond a FoS = 1.0 design to meet the given nominal loading requirements. An increase in fault tolerance in the form of additional subsystem redundancy, addition of dissimilar redundancy or other mitigation strategies such as the addition of safety components is another contribution to the *Safety* mass of the spacecraft. The addition of abort capabilities is considered a *Safety* mass in the form of additional avionics, instrumentation, increased mission duration, and increases in spacecraft growth due the need for additional mass and volume. Dispersions from nominal values are a form of uncertainty and similar to the factor of safety are included with *Safety* mass.

Given a particular type of spacecraft geometry, an increase in *Safety* mass is not a linear increase in total spacecraft mass due to additional dependent burdens placed on subsystems such as Avionics, Power, Thermal, Structural, and Propulsion. The Propulsion subsystem must ultimately grow larger to accommodate additional mass in the form of larger engines, tanks, additional propellant, or a change in orbital trajectory to meet an "Anytime / Anywhere Return" scenario. Depending upon the capabilities of the Propulsion subsystem and the loading on the structures, a small increase in mass can increase the total spacecraft mass significantly.

3.4.6 Operability Mass in Spacecraft Configurations

The term *Operability* is defined in this methodology as the mass contribution of additional functions beyond the minimum functionality baseline that does not include *Safety*. *Operability* mass is expressed as:

$$Operability = f(Additional\ Functions\ for\ Mission\ Objectives,\ Enhanced\ User\ Interfaces) \quad (7)$$

Operability is the measure of the tasks that are planned to be completed in the mission. It includes mission specific tasks such as sample return, photography, observation, and EVA. *Operability* also includes enhanced user interface functions that make tasks more efficient or easier such as automation. *Operability* is mass for the spacecraft or mission to do tasks beyond the minimum functionality of transporting crew. It can be defined as mass that expands the minimum functionality mission to:

- Do mission specific tasks - (samples, science return, photography, observation, EVA);
- Make a task efficient;
- Make a task easier – (automation for crew tasks, spacecraft control); and
- Stay for a longer time.

One could characterize *Operability* as Mission Operations, but it does not encompass all of the mass that is typically associated with Mission Operations such as contingency operations. Much like the mass increase in *Safety*, an increase in *Operability* beyond a baseline minimum functional spacecraft configuration is not an equal increase in the total spacecraft mass due to the burden on other subsystems. Essentially the *Operability* mass is intended to capture those additions that make the spacecraft more capable rather than simply inherently safer.

3.4.7 Conceptual Design Tradespace

One of the reasons for quantifying the amount of mass due to *Safety* and *Operability* is to visualize the relationship between these parameters and the total spacecraft mass. Combining the three parameters into a three dimensional (3-D) tradespace of candidate spacecraft configurations allows the designer to understand how changes in subsystem redundancy through additional *Safety* or the addition of additional mission capability through *Operability* change the overall mass of the total spacecraft.

In order to generate a domain of spacecraft points, various spacecraft configurations of subsystem redundancy, FoS, safety components, and additional functions are calculated in a multidisciplinary Monte Carlo analysis. The collection of spacecraft configuration data points form a Pareto frontier that can be used for determining the most optimal contributions of *Safety* mass and *Operability* mass on the total spacecraft mass. Shown in Figure 9 is an illustration of the conceptual design tradespace. The three axes of the conceptual design tradespace are the Total Spacecraft mass (x), *Safety* mass (y), and *Operability* mass (z). The tradespace was created using spacecraft configurations of various levels of subsystem redundancy, number of *Safety* components, number of *Operability* components, and FoS. The outer boundary of the data points in the tradespace forms a Pareto frontier that can be used as input for the development of reduced order models for characterizing the relative relationship between *Safety*, *Operability*, and Total Spacecraft mass.

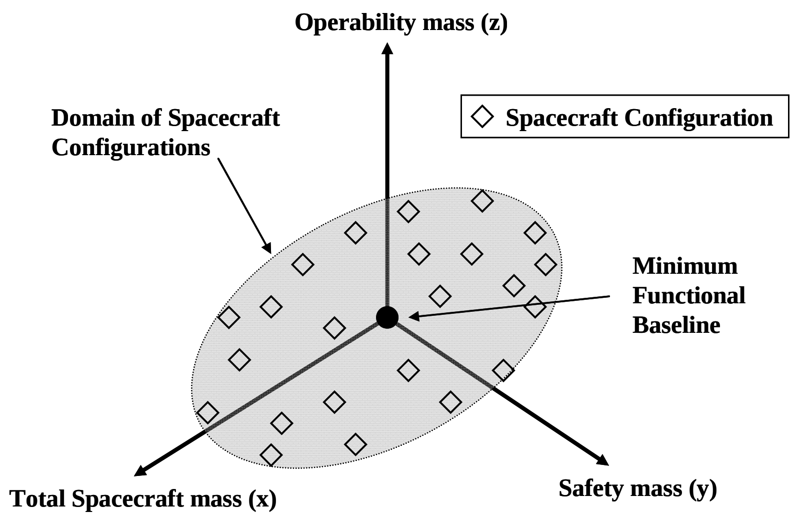


Figure 9: Conceptual design tradespace for spacecraft configurations.

Developing Pareto frontiers in this tradespace provides designers with the flexibility to conduct other trades. For example, if a metamodel of the Pareto frontier was created, designers could evaluate the impact upon total spacecraft mass rather quickly given an increase in *Safety* or

Operability mass of a specific configuration. Depending on the shape of the Pareto surface, there are many approaches for metamodeling that have different accuracies and methods for optimization (Jin *et al*., 2003; Wang and Shan, 2007; Simpson *et al*., 2004; Wilson *et al*., 2001). It is possible to compare Pareto frontiers from different analysis runs in a sensitivity analysis to determine relationships between the fundamental parameters and the overall spacecraft design. A metamodeling approach coupled with this design methodology is suggested as a tool for reducing the computational expense of examining a large number of spacecraft configurations.

The conceptual design tradespace presented here is intended to capture the mass relationships as a means of comparing between competing spacecraft configurations. It is recognized that other metrics such as performance, cost, and risk are important parameters of a spacecraft design, but in the context of this work, these metrics are left to the later stages of design as the configuration matures. The information developed in this conceptual design tradespace captures mass impacts of risk to be utilized as a starting point in future detailed design iterations.

3.4.8 Minimum Functionality Design Methodology Summary

The design methodology presented in this work builds upon previous approaches for minimum functionality design. The difference in this methodology compared to previous methods is that the Minimum Functionality design point is defined based on *Physics* and *Physiology* and does not include mass for FoS, contingencies, or minimum mission objectives. Using the values of *Safety* and *Operability* as mass add backs, various spacecraft configurations can be explored in a 3-D tradespace and developed into a reduced order model to understand the impacts of mass additions in *Safety* and *Operability* in relationship to the total spacecraft mass.

3.5 CONCEPTUAL LUNAR ASCENT MODULE PROGRAM

A mass and trajectory analysis code named the Conceptual Lunar Ascent Module Program was developed in MATLAB to calculate the minimum functionality baseline and to assess relative increases in *Safety* and *Operability* mass for various Lunar Ascent Module spacecraft configurations. This program is focused on the lunar ascent phase of the overall lunar architecture mission. The information developed in this analysis could be utilized by designers of other elements of the overall lunar architecture. The intent of the program was to determine how changes in subsystem mass affect the overall spacecraft mass required to return from the lunar surface. The program played a key role in developing this proposed methodology in order to understand how closely coupled the *Safety* and *Operability* mass is to the total spacecraft baseline mass. The CLAMP code approaches the development of a conceptual spacecraft design with a bottom-up design philosophy using individual components and heuristics matched to spacecraft functions. This approach can be challenging when little information is known about the exact mass and volume sizing of the technology components. But the advantage of this design methodology is that it forces designers to consider sensitivities in the lower levels at an early stage in order to understand the driving risks and uncertainties as the design is matured. Sensitivities of the low level components are captured in uncertainty bands and combined in a Monte Carlo analysis to determine the effect on the subsystem and total spacecraft mass.

The program relies on design teams to establish a minimum set of functions needed to transport the crew and keep them alive. It also serves as a tool for designers to evaluate changes in the design space variables beyond a minimum functionality baseline configuration and the relationships between Total Spacecraft, *Safety*, and *Operability* mass. The design teams ultimately drive the design process and the information generated with this program assists designers and teams with decision making. Other factors such as reliability and cost will always have an effect on design decisions; but the main reason for investigating this tradespace is to give subsystem designers visibility into the overall spacecraft design mass at an early stage of development. Shown in Figure 10 is a flowchart of the processes and subsystem modules of CLAMP.

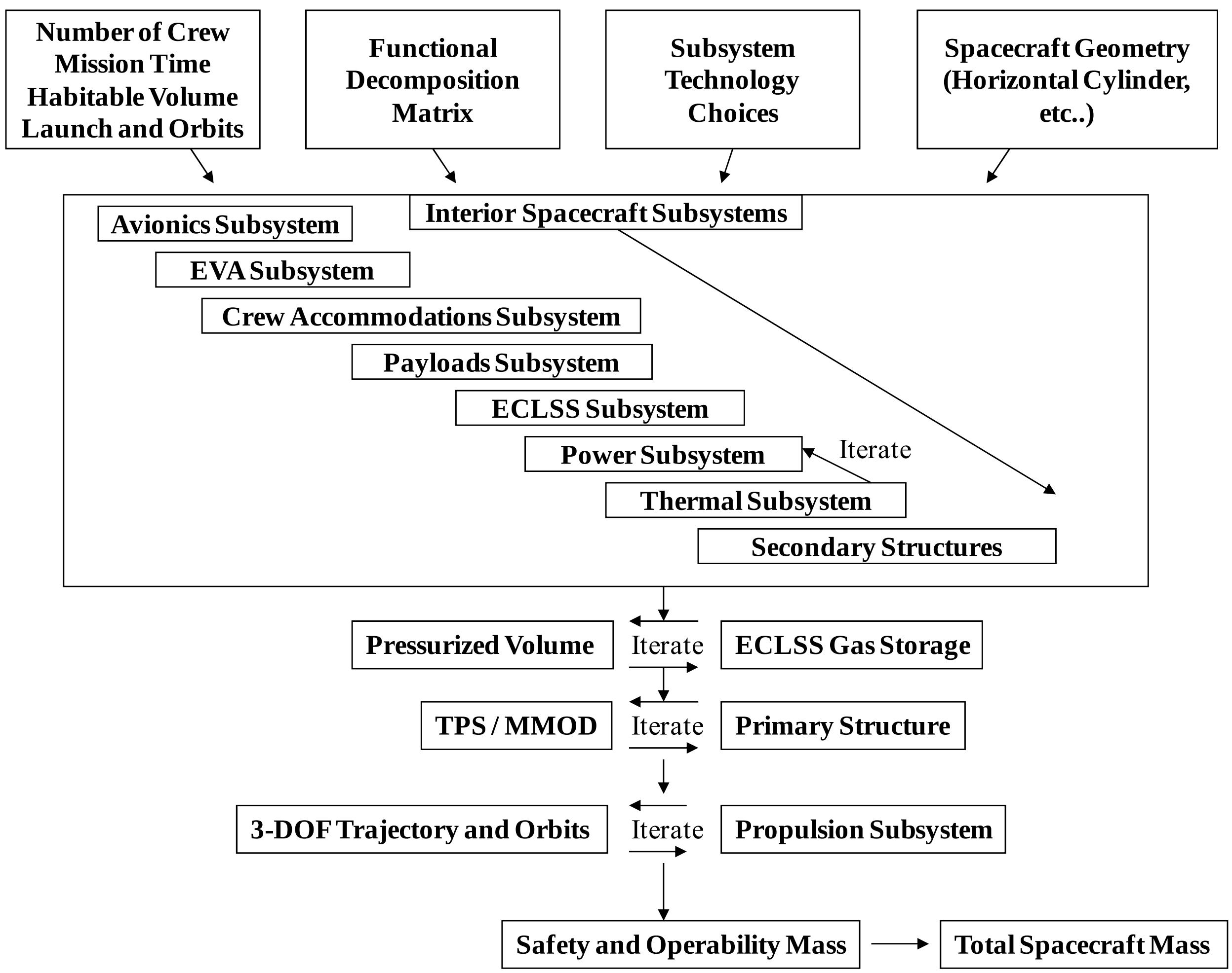


Figure 10: Flowchart of CLAMP subroutines.

3.5.1 CLAMP Inputs

The four boxes at the top of Figure 10 represent the necessary inputs for the analysis. The top level goals of the mission are captured in the number of crew, mission time, habitable volume, and trajectory and orbits. These four parameters are the key independent parameters of any spacecraft design. The functional decomposition matrix defines the low level spacecraft functions. The subsystem technologies are the components matched to the low level functions in the functional decomposition matrix and represent a LRU when possible in a bottom-up design philosophy. The choice of spacecraft geometry such as a horizontal cylinder, vertical cylinder, conic, or lifting body is another key input for determining the structural layout of the spacecraft. In this initial version of the code, the spacecraft configuration is assumed to be a horizontal cylinder module for the study of typical Lunar Ascent Module designs. The CLAMP code is flexible for adding other spacecraft geometries and could be updated in later versions.

3.5.2 Spacecraft Subsystem Sizing

The subsystem modules shown in Figure 10 calculate the mass of the individual interior subsystems based upon the input parameters and required functions. The choice of technologies matched to the low level functions is a key driver for mass and safety of the subsystems. Beginning with the Avionics subsystem, each individual subsystem mass is calculated using the technology choices and component redundancy. Because the Avionics subsystem is a large power and thermal demanding subsystem, it is the first module to be calculated followed by EVA, Crew Accommodations, Payloads, ECLSS, Power, Thermal, and Secondary Structures subsystems. This hierarchy of flow from one subsystem to another allows mass, power, and volume information from previous subsystem module to be passed to subsequent subsystem modules for input. The relationship between the Power and Thermal subsystem is iterated once the spacecraft thermal requirements have been calculated. The first step of the conceptual design process is to determine the interior subsystems component mass and volume in order to determine pressurized volume.

The flow process from one subsystem to another captures dependencies and sensitivities among the individual subsystems. In the example of the Avionics system, a small increase in mass has downstream effects on Power, Thermal, Pressurized Volume, Secondary Structure, Primary Structure, and Propulsion. Changes in one subsystem ultimately affect the sizing of other subsystems and the intent of the flow down approach is to capture the dependencies in order to provide designers with critical information about sizing of the individual subsystems. The process described here allows designers to evaluate subsystem trades to determine the sensitivities of the technology choices on the overall mass and safety of the spacecraft. For a single run, a common set of technologies are assumed for each of the subsystems throughout the analysis.

Once the interior spacecraft subsystems and components are defined, the pressurized volume module is used to calculate the required pressurized volume given a specified minimum habitable volume, the total interior subsystem volume, and a “packaging efficiency”. The use of a packaging efficiency takes into account the difficulty of neatly packaging all of the subsystem components and serves as a margin above the total interior subsystem volume. A simplified diagram of how total pressurized volume is derived is illustrated in Figure 11.

Using the calculated total pressurized volume, the outer diameter of the spacecraft is recalculated. In this example, the assumption is that in a cylindrical configuration as shown in Figure 11, the majority of the components are mounted to the cylindrical section and the interior diameter of the cylinder must grow larger to accommodate the change in volume. The amount of gaseous oxygen and leakage makeup gas for the ECLSS subsystem are determined from this pressurized volume. The CLAMP code calculates the storage tank size for the high pressure gas, lines, valves, and insulation. The mounting locations for the ECLSS gas storage are assumed to be external to the crew cabin.

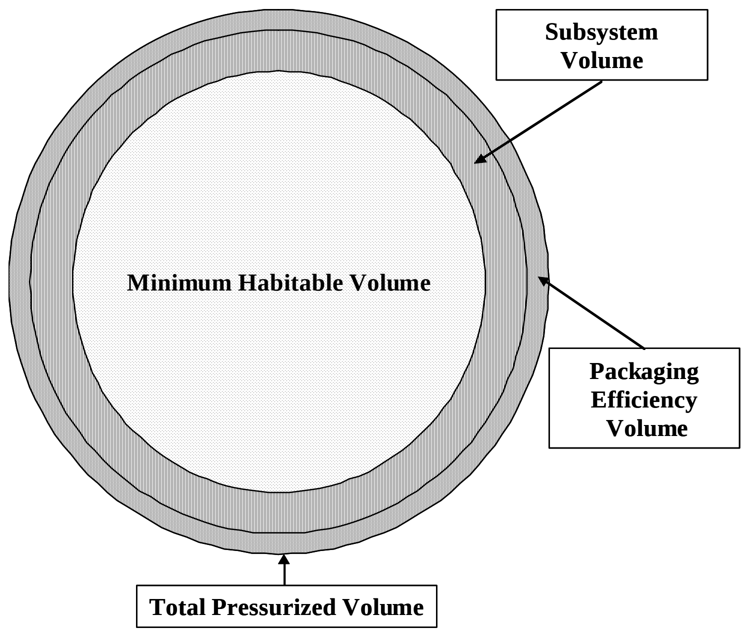


Figure 11: Derivation of total pressurized volume.

Using the calculated total pressurized volume, the outer diameter of the spacecraft is recalculated. In this example, the assumption is that in a cylindrical configuration as shown in Figure 11, the majority of the components are mounted to the cylindrical section and the interior diameter of the cylinder must grow larger to accommodate the change in volume. The amount of gaseous oxygen and leakage makeup gas for the ECLSS subsystem are determined from this pressurized volume. The CLAMP code calculates the storage tank size for the high pressure gas, lines, valves, and insulation. The mounting locations for the ECLSS gas storage are assumed to be external to the crew cabin.

The primary structure is the backbone of the spacecraft design and the choice of materials and configuration is a large driver in the uncertainty of the spacecraft mass. Because the primary structure greatly influences the overall spacecraft conceptual design, the mass estimating code develops a primary structure using the dimensions for total pressurized volume, subsystem loading, number of crew, docking dimensions, and structural materials. Shown in Figure 12 is a CAD representation of a horizontal cylinder primary structure for a proposed Lunar Ascent Module developed with the CLAMP code. Structural analysis in CLAMP begins with a preferred geometry such as a horizontal cylinder as shown in Figure 12. The pressure vessel skin is calculated based on the operating pressure of the cabin atmosphere and is supported by a series of outer hoops and stringers. The size of the hoops is a key driver in the mass of the primary structure and is calculated based on the subsystem loading on the walls of the structure.

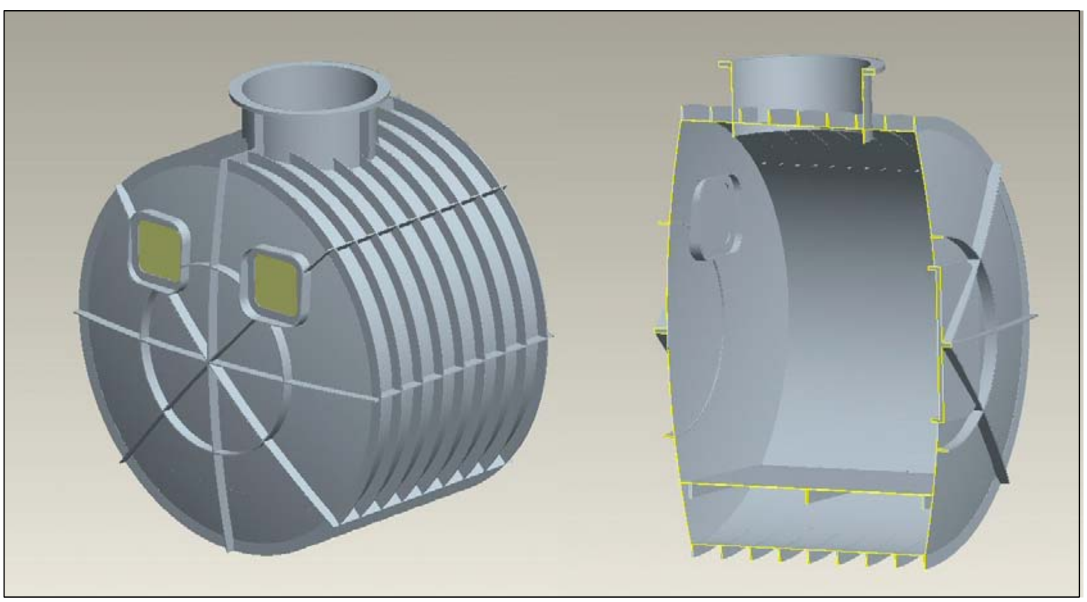


Figure 12: CAD model of the horizontal cylinder Lunar Ascent Module structure.

The propulsion subsystem consists of the main engine, Reaction Control System (RCS), tanks, lines, valves, and pressurant systems. The CLAMP code uses the mass of the entire spacecraft not including propulsion components as a "payload" for the iteration of the propulsion sizing. The main engine is sized according to a vacuum thrust to weight (T/W) ratio for the engine and the mass of the main engine is calculated using a linear equation of kg/kN of thrust based on heuristics. RCS thrusters are sized according to a given vacuum thrust in the same method used for the main engine. Engine parameters such as Specific Impulse (Isp) and Oxidizer to Fuel ratios (O/F) determine the amount of propellant mass required to fly a given trajectory.

The propulsion module performs a 3-DOF analysis of the spacecraft launch trajectory to verify the correct orbital insertion parameters. A loop for calculating propellant mass, sizing tanks and engines, and flying the spacecraft trajectory is iterated until the propellant mass converges. Shown in Figure 13 is an example of the 3-DOF trajectory for a Lunar Ascent Module based on an Apollo 15 type ascent. The orbital parameters for this analysis run form an elliptical orbit of approximately 20 km x 100 km.

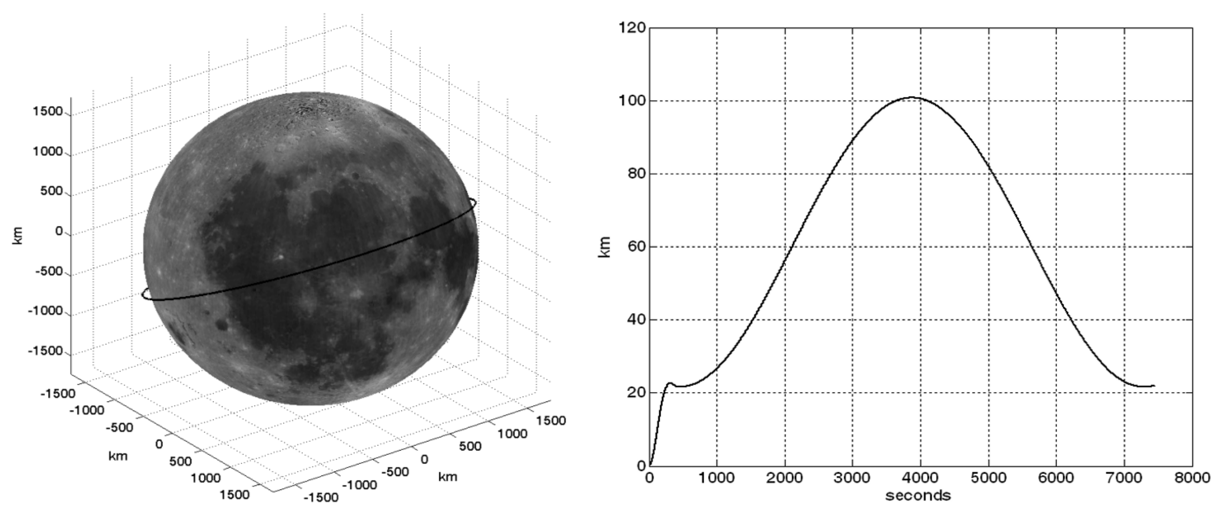


Figure 13: Lunar ascent trajectory analysis.

3.5.3 Total Spacecraft, Safety, and Operability Mass

The process for calculating the mass due to *Safety* and *Operability* begins with a minimum functional configuration. To determine the amount of *Safety* and *Operability* mass, three analysis runs are needed to determine the mass due to "add-backs" for a given configuration beyond the minimum functional baseline configuration. Shown in Figure 14 is a flow chart of the processes and the expressions for quantifying the minimum functionality baseline, *Safety*, and *Operability* mass of spacecraft configurations.

<u>Run (1)</u>
<u>Calculate Total Spacecraft Mass for the Minimum Functional Design Configuration</u>
FoS = 1.0, Single string subsystems at minimum number of functions
No Safety or Operability components
Minimum "best" mission time
Minimum Functionality Baseline Configuration Mass (Run 1)

↓

<u>Run (2)</u>
<u>Add back Safety</u>
Increase FoS to Design FoS (ex: 1.4, 1.51, etc..)
Include Safety Dedicated Components
Subsystem redundancy (single or dual redundant)
Minimum Functionality + Safety Mass (Run 2)

↓

<u>Run (3)</u>
<u>Add back Operability</u>
Increase mission time to maximum
Include Operability dedicated components
Subsystem redundancy (single or dual redundant)
Minimum Functionality + Safety Mass + Operability Mass (Run 3)

↓

<u>Minimum Functional Baseline Configuration (Physics & Physiology) Mass</u> = Total Spacecraft Mass (Run 1)
<u>Spacecraft Safety Mass</u> = (Run 2) – (Run 1)
<u>Spacecraft Operability Mass</u> = (Run 3) – (Run 2)

Figure 14: Flowchart for calculating Minimum Functional, *Safety*, and *Operability* mass.

The reason for three analysis runs is to determine the mass of the spacecraft configuration at various configurations of *Safety* and *Operability*. The first run is a spacecraft configuration that does not include any additional *Safety* or *Operability* components beyond the minimum "best" mission design timeline. The second run adds mass associated with improving safety such as FoS, safety components, contingency propellant, etc.., for any item in the spacecraft that is necessary for *Safety*. The third run determines the amount of *Operability* mass in the spacecraft where mass is added for components that are related to increasing *Operability* factors of the mission. In addition, the mission time is increased to the maximum expected length. To calculate the amount of mass that contributes to an increase in the *Safety*, the first run is subtracted from the second run. The same method is used to determine the amount of *Operability* mass where the second run is subtracted from the third run. This method provides insight into how much the total spacecraft mass increases with additions of mass due to *Safety* or *Operability*.

3.6 CONCEPTUAL SPACECRAFT ANALYSIS STUDIES

The following analysis studies demonstrate the use of the methodology in the design of a conceptual human spacecraft. The focus of these analyses is to study conceptual Lunar Ascent Module configurations to understand the relationship between mass add-backs due to *Safety* and *Operability* and the total spacecraft mass. Although the Lunar Ascent Module represents only one part of the overall lunar mission architecture, the results of these analyses would be included in the trade studies for the total lunar mission architecture. The top level goals for the Lunar Ascent Module conceptual design consist of the following:

- Return a crew of two from the lunar surface in a pressurized cabin,

- Launch from a location approximately at the lunar equator into an elliptical orbit, and
- Rendezvous with transport spacecraft.

The top level goals for the test studies are the same as Apollo and the design activity will use the Apollo Lunar Ascent Module as a comparison for verification of the CLAMP code results. The studies demonstrate the methods used to calculate *Safety* and *Operability* mass for various spacecraft configurations by predicting upper and lower bounds that the Apollo Lunar Ascent Module mass should fall within, ranging from the minimum functionality baseline to increasingly safer, more capable iterations.

3.6.1 Apollo Lunar Ascent Module Design Comparison

The CLAMP code used an input list of 235 variables in the development of various conceptual designs. The input list consisted of variables for habitable volume, spacecraft diameter, technology choices for components, metabolic rates, cabin atmosphere, FoS, structural materials, and launch parameters. The subsystems and components of the historical Apollo LEM were matched to a list of the lowest level functions of the spacecraft. Upon review of several sources including the: Apollo Experience Reports (Dietz *et al*., 1972; Shelton, 1975; Vernon, 1975; O'Brian and Woodfill, 1972; Kurten, 1975; Gillen *et al*., 1972; Campos, 1972; White, 1975; Weiss, 1973; Humphries and Taylor, 1973; Vaughan *et al*., 1972; Bennett, 1972; Ecord, 1972), Apollo Operations Handbook Lunar Module LM 10 and Subsequent Volume 1 Subsystems Data, LEM Design Criteria and Environments (Shreeves *et al*., 1963), Lunar Module Structures Handout LM-5, LEM mass property reports (Aeder *et al*., 1966a; Aeder *et al*., 1966b;

Aeder *et al*., 1966c), NASA Design Mass Properties II (Heineman, 1994), Apollo 15 Mission Report (NASA, 1971a), Apollo Stowage List Mission J-1 CM 112/LM-10 Apollo 15, and Apollo News Reference (NASA, 1972), the mass, volume, and power of the subsystems and components in the Lunar Ascent Module for lunar ascent were compiled as input variables. Where possible, input data based on heuristic textbook equations was used to demonstrate how the methodology could be useful for future conceptual designs with limited information about the components in the subsystems. Although the entire input list is too large to be shown here, a summary list of the sources for the input information is shown in Table 3. The Apollo Lunar Ascent Module serves as a very good example for validation of the methodology because of its design history in optimizing against stringent weight margins.

Table 3 Summary of Input Sources for Apollo LEM Comparison

Subsystem	Components	Source of Mass, Volume, and Power Information
Avionics	Communication	Apollo Experience Reports (Dietz *et al*., 1972); Design Mass Properties II (Heineman, 1994)
	GN&C	Apollo Experience Reports (Shelton, 1975; Vernon, 1975; Design Mass Properties II (Heineman, 1994)
	Instrumentation	Apollo Experience Reports (O'Brien and Woodfill, 1972); Design Mass Properties II (Heineman, 1994)
	Abort Avionics	Apollo Experience Reports (Kurten, 1975); Design Mass Properties II (Heineman, 1994)
EVA	Spacesuits	HSMAD (Larson *et al*., 1999); Apollo Stowage List Mission J-1 CM 112/LM-10
Crew Accommodations	Lighting	HSMAD (Larson *et al*., 1999); Apollo Stowage List Mission J-1 CM 112/LM-10
	Crew Restraints & Personal Storage	HSMAD (Larson *et al*., 1999); Apollo Stowage List Mission J-1 CM 112/LM-10
Payloads	Lunar Samples	Apollo 15 Mission Report [33]; Apollo Stowage List Mission J-1 CM 112/LM-10
	Tools, Equipment, and Photography	HSMAD (Larson *et al*., 1999); Apollo Stowage List Mission J-1 CM 112/LM-10
ECLSS	Carbon Dioxide Removal	BVAD (Hanford, 2004); Apollo Experience Reports (Gillen *et al*., 1972), Apollo News Reference (NASA, 1972)
	Metabolic Oxygen Storage	Apollo Experience Reports (Gillen *et al*., 1972); Apollo News Reference (NASA, 1972)
	Cabin Leak Rate, Humidity, Temperature	Apollo Operations Handbook Lunar Module LM 10 and Subsequent -Volume 1 Subsystems Data; BVAD (Hanford, 2004)

Table 3 Summary of Input Sources for Apollo LEM Comparison

Subsystem	Components	Source of Mass, Volume, and Power Information
Power	Batteries	HSMAD (Larson *et al*., 1999); Apollo Experience Reports (Campos, 1972)
	Distribution & Wiring	Apollo Experience Reports (Campos, 1972; White, 1975); Design Mass Properties II (Heineman, 1994)
	Controllers & Inverters	Apollo Experience Reports (Campos, 1972); Design Mass Properties II (Heineman, 1994)
Thermal	Coldplates, Coolant, Pumps & Valves	Apollo Experience Reports (Gillen *et al*., 1972); BVAD (Hanford, 2004)
	Sublimators	Apollo Experience Reports (Gillen *et al*., 1972); Tongue (1999)
Secondary Structures	Ratio	BVAD (Hanford, 2004); Design Mass Properties II (Heineman, 1994)
Primary Structure	Material, Pressurized Volume, Cabin Diameter, FoS	Apollo Experience Reports (Weiss, 1973); Apollo Operations Handbook Lunar Module LM 10 and Subsequent -Volume 1 Subsystems Data ;LEM Design Criteria (Ecord, 1972)
TPS / MMOD	Materials, Probabilities	Lunar Module Structures Handout LM-5; Apollo Operations Handbook Lunar Module LM 10 and Subsequent -Volume 1 Subsystems Data ; LEM Design Criteria (Shreeves *et al*., 1963); Structural Damage Prediction (Elfer, 1996)
Propulsion	Main Engine and RCS systems	Apollo Experience Reports (Humphries and Taylor, 1973; Vaughan *et al*., 1972; Bennett, 1972); Apollo Operations Handbook Lunar Module LM 10 and Subsequent -Volume 1 Subsystems Data
	Propellant and Pressurization Storage Tanks	Apollo Experience Reports (Humphries and Taylor, 1973; Vaughan *et al*., 1972; Bennett, 1972); Apollo Operations Handbook Lunar Module LM 10 and Subsequent -Volume 1 Subsystems Data

A Monte Carlo analysis of 5000 runs was conducted with input variables set at +/- 5% uncertainty in mass and volume. The range of uncertainty was chosen initially to determine how much the total spacecraft mass would vary based on a level of input uncertainty associated with the subsystem technology choices. In conceptual design, the variability of mass in technologies is often accounted through growth margins and the use of uncertainty in the input technologies and variables is intended to allow the user flexibility to generate concepts based on likely growth margins. Although +/- 5% is a very narrow range for conceptual design uncertainty, the intent of this comparison analysis is to determine the distribution in total spacecraft mass for a variety of spacecraft configurations.

3.6.2 Test Study Matrix

A matrix of analysis runs was developed to investigate the amount of subsystem and total spacecraft mass growth due to changes in FoS, *Safety*, and *Operability* at different levels of subsystem redundancy. Shown in Table 4 is a matrix of 11 analysis runs at different levels of FoS, subsystem redundancy, *Safety* components, and *Operability* components.

Table 4: Matrix of analysis studies.

		Minimum Functions	Maximum Mission Time	Factor of Safety	Additional Functions -Safety Components	Additional Functions - Operability Components	Single String Subsystems	Redundant Subsystems	Dual Redundant Subsystems
Code Verification	Apollo Baseline Configuration	X	X	X	X	X	X	X	
Run 1	Minimum Functional Design Point	X							
Run 2	Minimum Functional Design at Maximum Mission Time	X	X						
Run 3	Minimum Functional Design with Time, Factor of Safety and Single String Subsystems	X	X	X			X		
Run 4	Minimum Functional Design with Time, Factor of Safety, Addn'l Safety Components, and Single String Subsystems	X	X	X	X		X		
Run 5	Single String Subsystems with Time, Safety, Operability, and Factor of Safety	X	X	X	X	X	X		
Run 6	Minimum Functional Design with Time, Factor of Safety, and Redundant Subsystems	X	X	X				X	
Run 7	Minimum Functional Design with Time, Factor of Safety, Addn'l Safety Components, and Redundant Subsystems	X	X	X	X			X	
Run 8	Redundant Subsystems with Time, Safety, Operability, and Factor of Safety	X	X	X	X	X		X	
Run 9	Dual Redundant Subsystems with Time, Safety, Operability, and Factor of Safety	X	X	X	X	X			X
Run 10	Tradespace Development of Conceptual Design Configurations	X	X	X	X	X	X	X	X

To verify the accuracy of the CLAMP code when compared to heuristic data, the nominal “as flown” configuration of the Apollo 15 Lunar Ascent Module is compared with the results of

CLAMP. For each analysis run, a Monte Carlos set of 5000 runs was completed with an input uniform random uncertainty of +/- 5%. The range of spacecraft configurations in this analysis matrix are varied starting from a minimum functional design (lowest mass) to a dual redundant subsystem configuration (highest mass). These boundary conditions represent the minimum and maximum anticipated mass of a spacecraft based on habitable volume, number of crew, geometry, structural materials, trajectory, and orbital parameters.

3.6.3 *Safety* and *Operability* Mass Fractions

In each of the analysis runs, the amount of *Safety* and *Operability* mass is determined. Calculating the *Safety* and *Operability* mass as a percentage of the total spacecraft mass provides another metric for understanding the mass fractions dedicated to the four fundamental values. Although *Safety* and *Operability* are defined in the context of mass in this analysis, it is important to understand how an increase in reliability through mass addition such as increased subsystem redundancy relates to the total spacecraft mass.

3.6.4 Tradespace of Conceptual Design Configurations

The relationship between *Safety*, *Operability*, and total spacecraft mass can be visualized in a 3-D tradespace of various spacecraft subsystem component configurations. The three axes of the conceptual design tradespace are the Total Spacecraft mass (x), *Safety* mass (y), and *Operability* mass (z). The tradespace is developed by varying the level of subsystem redundancy, number of *Safety* components, number of *Operability* components, and Factor of Safety for a given spacecraft configuration of habitable volume, number of crew, mission time, spacecraft geometry, structural materials, and orbital parameters. A Monte Carlo analysis of 5000 runs was used to generate the tradespace points.

3.7 CONCEPTUAL SPACECRAFT ANALYSIS STUDY RESULTS

3.7.1 Apollo Lunar Ascent Module Verification

Using the input information given in Table 4; the CLAMP code developed a conceptual Lunar Ascent Module design based on the Apollo 15 Lunar Ascent Module configuration. The total gross weight of the Apollo 15 Lunar Ascent Module at lunar liftoff was calculated to be 10,927 lbs. based on information gathered from the literature sources and including crew. Uncertainty limits of +/- 5% were applied in the analysis to determine the distribution range of the total spacecraft mass and the accuracy of the code when compared to heuristic data. Shown in Figure 15 is a comparison of the upper and lower limit 95% confidence bands of the subsystems compared to the mass of the Lunar Ascent Module of Apollo 15.

In each subsystem, the nominal mass values of an Apollo 15 configuration were within upper and lower limits for 95% confidence. This analysis demonstrated that the CLAMP code could recreate a human spacecraft design for a mission that matched known heuristics within 95% confidence, therefore providing a calibration case for the following parametric analysis runs across various levels of subsystem redundancy and spacecraft configurations.

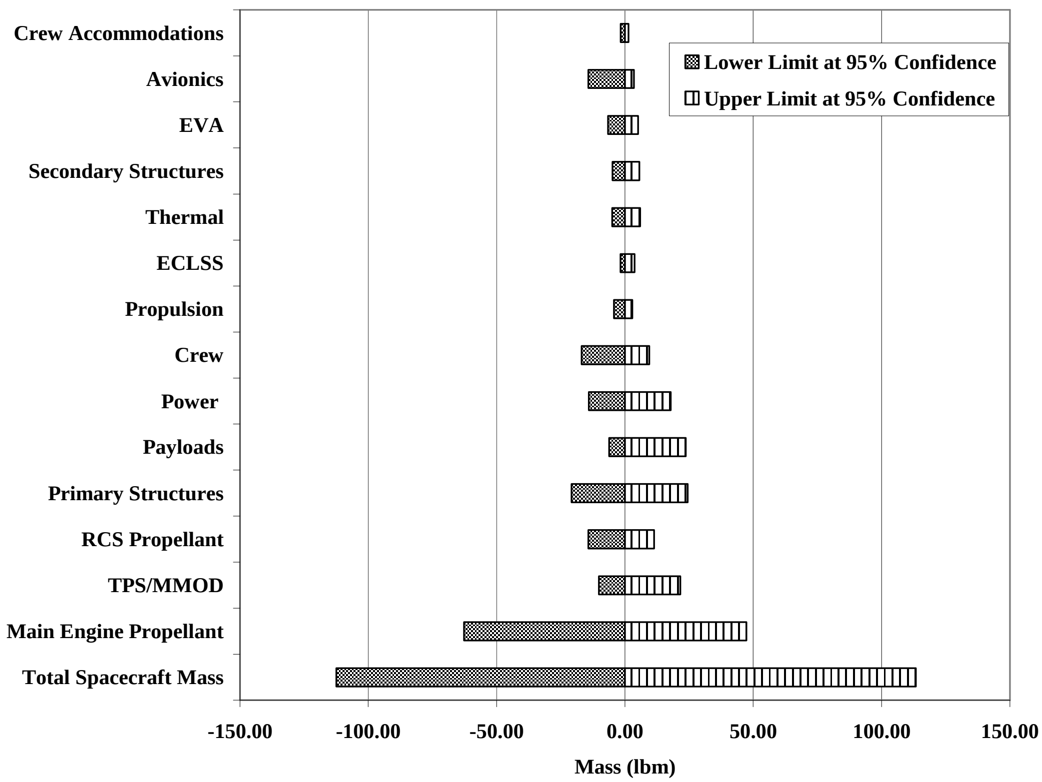


Figure 15: Comparison of subsystem confidence limits to Apollo 15 subsystem mass.

3.7.2 Subsystem Mass in Various Spacecraft Configurations

Shown in Figure 16 are the subsystem mass values for the various spacecraft configuration runs in the analysis study matrix. Beginning with the minimum functional baseline configuration at a total mass of 5210 lbs, the mass of this design point is approximately half of the nominal as flown Apollo 15 Lunar Ascent Module mass. It is recognized that while this is theoretically a technically feasible design option, it not likely to be considered programmatically acceptable due to the high risk it entails. In this run, the minimum mission time was assumed to be 1.37 hours till rendezvous. This establishes a lower bound for the spacecraft mass needed to complete the mission assuming no hardware failures or operational contingencies. The next run evaluated the minimum functional design at the maximum design time of 12.45 hours. The increase in mission time of approximately 11 hours for the short mission increased the total spacecraft mass by 461 lbs. The additional mass was due to the additional tank mass, power, and structure needed for a longer time interval before docking. The mission duration time was held constant at 12.45 hours for the remainder of the analysis runs. The third configuration run added in FoS to the minimum functional design at maximum design time. The increase of 334 lbs above the previous run was due to the addition of FoS of 1.4 in the structure and 1.5 in high pressure tanks, and propulsion storage tanks. The third configuration run loosely corresponds to a minimum functional design point according to a ZBV approach not including additional mission objective mass. The mass difference between the lowest minimum functional design and a comparable ZBV configuration is 795 lbs for an Apollo Lunar Ascent Module configuration.

The remainder of the configuration runs added FoS, *Safety* components, and *Operability* components for either single string, redundant, or dual redundant subsystem components. The maximum mass of approximately 25,000 lbs. represents the upper boundary condition of dual redundancy in the subsystem components. Compared to the minimum functional, high-risk baseline configuration, the all dual redundant spacecraft mass is approximately 5 times larger. One also should note that this configuration would not be a programmatically acceptable spacecraft configuration due to excess mass, rather it defines the maximum boundary condition.

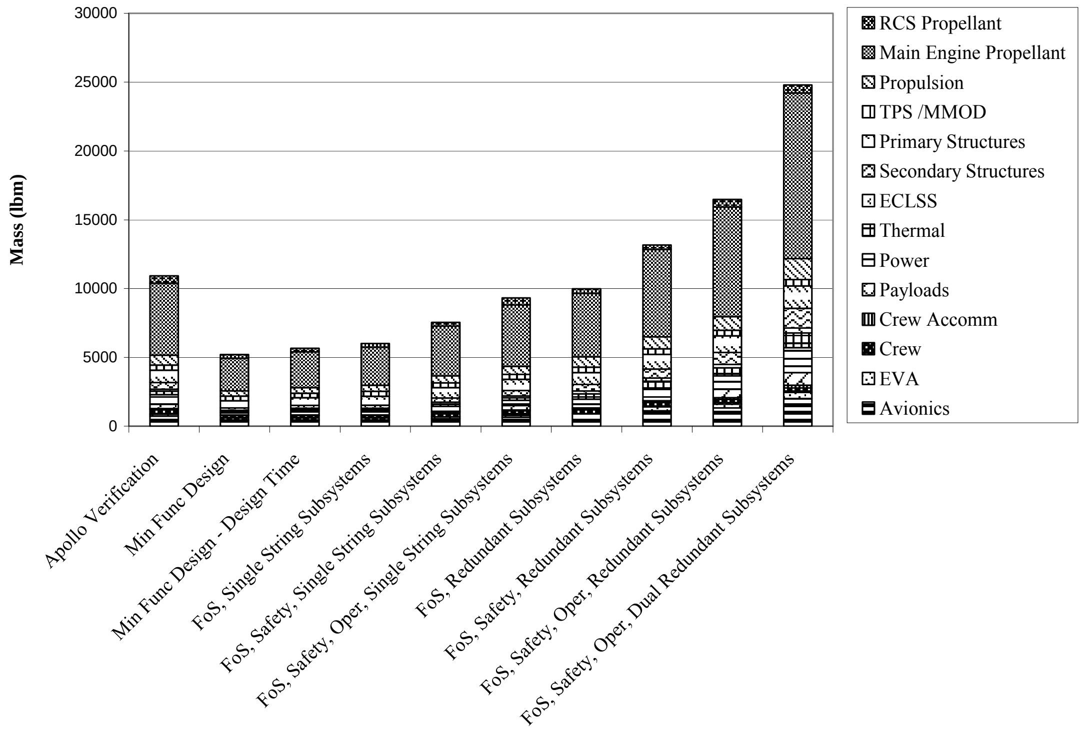


Figure 16: Subsystem mass for various configurations of an Apollo Lunar Ascent Module.

An interesting finding common to all of the analysis runs is the fraction of propulsion mass needed for the spacecraft to reach the orbital insertion parameters. In all cases, the sum of propulsion, main engine propellant, and RCS propellant mass fraction of the total spacecraft mass was calculated to be approximately 58%. As the mass of the spacecraft grew larger, the propulsion mass could be easily predicted with this percentage. However, one should also note that the propulsion system was sized for a particular launch trajectory and orbital parameters. A change in the launch trajectory and orbital parameters will change the propulsion mass fraction. The mass fraction of the propulsion system is a function of the technology selections and, in this example, the hypergolic propulsion system utilized by Apollo was a significant mass of the Lunar Ascent Module. Other types of propulsion systems such as liquid oxygen and methane will produce a different propulsion mass fraction. Trades between technology choices are left to the design teams.

As expected, the increase from all single string subsystems to all redundant subsystems increased the total spacecraft mass by 7154 lbs. This is a 75% increase from the single string subsystem configuration that included FoS, *Safety*, and *Operability* due to added subsystem component redundancy. The previous results should not come as a surprise to most spacecraft designers; but the takeaway lesson is that a 1 lb increase due to a *Safety* addition increased the overall spacecraft mass 3.85 lbs. Because of the multidisciplinary nature of a spacecraft, a designer needs to understand the impacts of a small increase in mass in one subsystem and the relationship to the whole spacecraft during the early conceptual design phase when so many subsystem designs are in flux.

3.7.3 *Physics*, *Physiology*, *Safety*, and *Operability* Mass Fractions

Increasing subsystem redundancy and adding components to mitigate failure modes and contingencies is intended to improve reliability through increased fault tolerance. The amount of *Safety* mass in the vehicle can become the significant driver in the spacecraft if unneeded complexity is built into the design. Shown in Figure 17 is a chart of mass fractions of the total spacecraft mass grouped into the values of Minimum Functionality, *Safety*, and *Operability*. A significant finding of this analysis is that approximately 28% of the total spacecraft mass of an Apollo Lunar Ascent Module was dedicated solely to *Safety*. The amount of *Operability* in additional time, payload, and mission objectives is approximately 24% of the total spacecraft mass. The amount of spacecraft mass necessary for *Physics* and *Physiology* was 48% of the total spacecraft mass. The most surprising result of this analysis is that the addition of *Safety* components, FoS, and *Operability* accounted for approximately half of the mass of the total spacecraft to meet reliability and mission objectives.

In a minimum functional design configuration, the entire mass of the vehicle is dedicated 100% to *Physics* and *Physiology*. As *Safety* and *Operability* components are added to increase reliability, the mass fraction of *Physics* and *Physiology* decrease while the *Safety* and *Operability* mass fractions increase or decrease. For example, when the *Operability* mass is added to a single string subsystem configuration, the mass fraction for the *Safety* decreased. In the redundant subsystem designs, the amount of *Safety* is the significant mass driver with a range of 40-50% of total spacecraft mass depending upon the addition of *Safety* or *Operability* components.

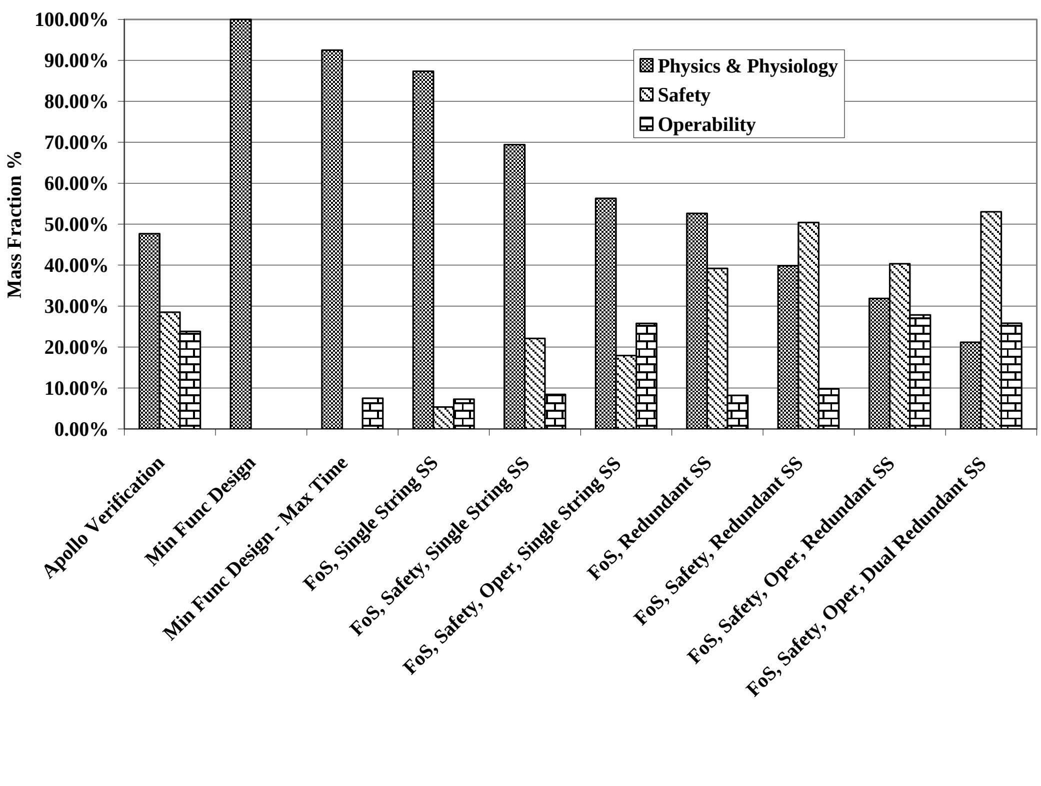


Figure 17: Mass fractions of *Physics* & *Physiology*, *Safety*, and *Operability*.

3.7.4 Tradespace of Conceptual Spacecraft Design Configurations

Because spacecraft designers may not have adequate time or resources to study many different configurations of proposed conceptual spacecraft in depth, the methodology presented in this work could be used in the development of metamodels based on the objective tradespace. Shown in Figure 18 are the results of a Monte Carlo analysis for a run of 5000 spacecraft configurations. Because the Propulsion system sizing was optimized for specified orbital parameters, the relationship between *Safety*, *Operability*, and total spacecraft mass form a 3-D surface that is approximately planer. The use of a plane as a metamodel could be easily developed by designers for calculating total spacecraft mass based on given *Safety* and *Operability* mass. The number of points required to generate the plane could be significantly reduced from the number shown in the example. The important lesson of this exercise is to demonstrate the utility of this simplified methodology in early conceptual design.

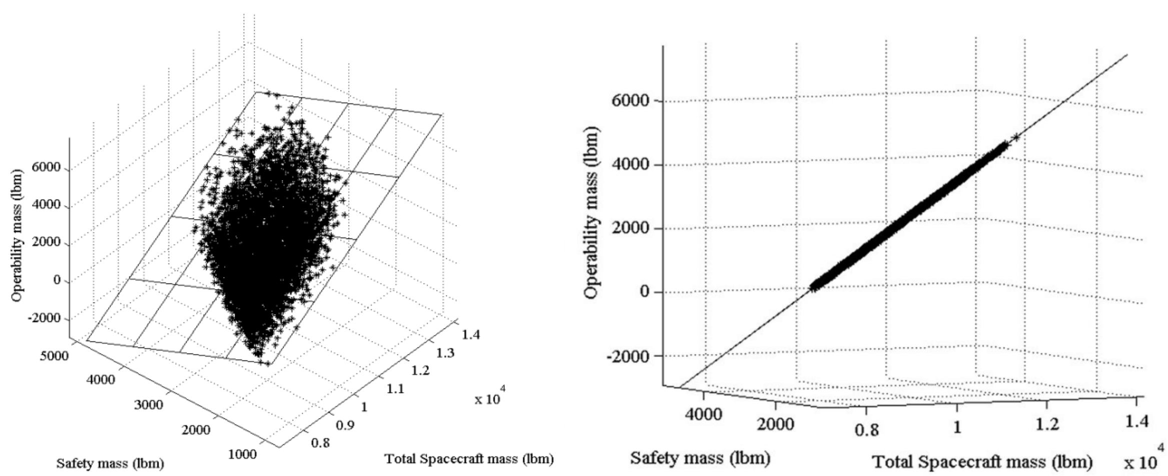


Figure 18: Tradespace of Total Spacecraft, *Safety*, and *Operability* mass.

For a set of 5000 data points, the equation of the plane of spacecraft configuration representing an Apollo type Lunar Ascent Module with an Apollo 15 ascent trajectory was calculated as:

$$0.5774\,(x) - 0.5771(y) - 0.5775(z) = 3029.9 \tag{8}$$

Where, x = Total Spacecraft mass (lbm), y = *Safety* mass (lbm), and z = *Operability* mass (lbm). As mentioned previously, if one of the fundamental input parameters of crew size, spacecraft geometry, materials, orbital parameters, or technology selection is changed, a new tradespace must be developed. Further work will explore the sensitivities of changes in these parameters on the spacecraft design. The projection of the points on the Total Spacecraft and *Safety* mass plane is shown in Figure 19.

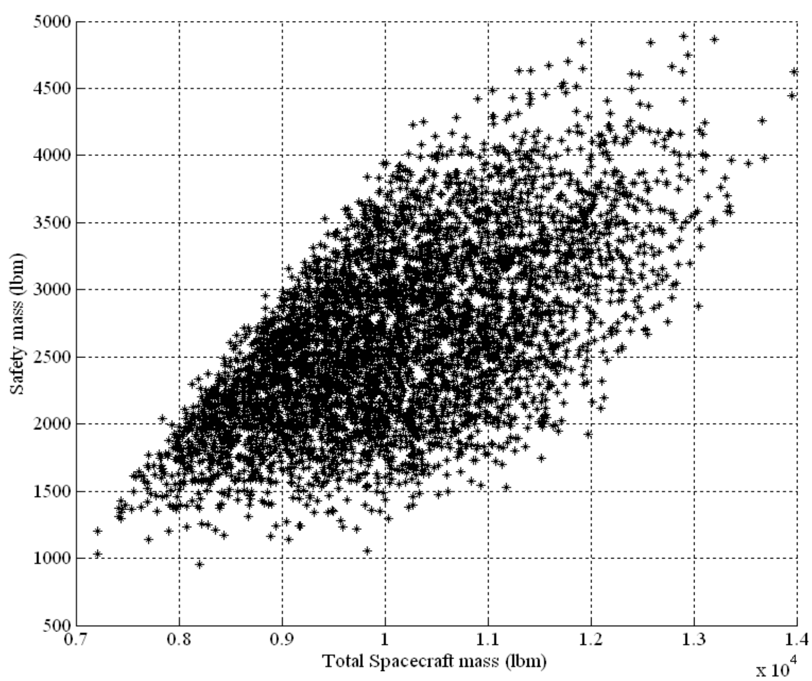


Figure 19: Total Spacecraft mass vs. *Safety* mass.

Observed in Figure 19, the data in the tradespace also offer many more opportunities for investigating the overall mass based on small changes in *Safety* or *Operability*. Judging whether a spacecraft meets detailed safety requirements is the next step in the design process, and obtaining a thorough understanding of how hardware additions relate to the overall spacecraft mass can assist with key trade decisions. The methodology presented here is intended to help designers and program decision makers understand the associated mass impact of making a spacecraft safer or better beyond what is needed to meet the mission objectives. The development of this conceptual level tradespace lends itself well for the use of figures of merit as a metric for quantifying relative increases in *Safety* or *Operability* in the early stages of design. Future work is aimed at extending this design methodology by defining quantifiable figures of merit for *Safety* and *Operability*.

3.8 CONCLUSIONS

A minimum functional spacecraft design approach is a method that establishes a clear "minimum" baseline configuration before the addition of mass to improve safety, performance, or meet other mission objectives. In previous minimum functionality design approaches, the baseline configuration varied according to how the starting "minimum" configuration was defined. In approaches such as NASA's Altair Minimum Functionality Design and Orion's Zero Based Design, the minimum functionality vehicle configuration was already assumed to include factors of safety in the structure and redundant subsystems beyond what was absolutely necessary to fly a baseline mission and perform a set of minimum mission objectives.

The difference between the previous approaches and the methodology outlined here is that a lower boundary spacecraft configuration is more clearly defined, as it is based on the minimum *Physics* and *Physiology* parameters required for a given reference mission. Because the *Physics* and *Physiology* values are "non-negotiable" requirements, a challenging aspect of spacecraft design is deciding on what constitutes sufficient safety and reliability, and then trading this goal against increased subsystem redundancy or additional FoS needed to achieve it. Adding mission capabilities beyond *Physics*, *Physiology*, and *Safety* is defined in terms of *Operability*. The degree of *Operability* required in a spacecraft is highly dependent upon the mission objectives, but designers must be aware that making small changes to achieve additional functionality can greatly affect the total mass of the spacecraft because of the burden placed on other subsystems to accommodate the upgrade. In the conceptual design phase of human spacecraft development, the central goal is to determine feasible design solutions that accommodate flexibility while simultaneously reducing uncertainties due to integration challenges and various risks that must be mitigated as the design concept matures. In order to make well informed decisions at this stage that will affect the success of the final design, the relationships between the various multidisciplinary aspects must be well understood.

The development of the CLAMP mass analysis code was shown to be a useful tool that demonstrates the design methodology and enables the comparison of different spacecraft configurations with various levels of *Safety* and *Operability* mass. One interesting finding from this design analysis was that the mass fraction of the Apollo Lunar Ascent Module that was dedicated to Safety was approximately 28% of the total spacecraft, and this was considered to be a relatively high-risk design. The results of the Monte Carlo analysis of spacecraft subsystem configurations were combined in a tradespace to visualize the relationship between *Safety*, *Operability*, and total mass for optional spacecraft configurations. Because the Propulsion subsystem was optimized for the given trajectory and orbital parameters, a plane of spacecraft configurations was formed that would enable a user to create a simple expression to define the relationship between total spacecraft mass, *Safety* mass, and *Operability* mass, across a large number of design options.

The minimum functional design methodology presented in this research is intended to provide human spacecraft designers with a systematic outline for determining a baseline mass and then quantifying *Safety* and *Operability* driven mass additions by trade study. The result of the methodology establishes a tradespace of configurations that visually represent the relationship between total spacecraft, *Safety*, and *Operability* mass fractions. It is hoped that this minimum functional baseline design methodology will generate interest among the spacecraft design community and serve as a useful guideline in the development of future human spacecraft.

3.8.1 Acknowledgements

This work would not have been possible without the contributions of the William F. Marlar Memorial Foundation for providing partial funding of this research and Human Spaceflight research in general at the University of Colorado in Boulder. Special recognition is given to the Aerospace Engineering Sciences Department, the CU LunarMARS graduate projects team, the CU Bioastronautics Research Group, the Lunar Module Design class of spring 2007, and the Space Habitat Design classes of fall 2007 and fall 2008.

CHAPTER 4

EFFECTIVE INTEGRATION OF RAPIDLY RECONFIGURABLE PROTOTYPING INTO MINIMUM FUNCTIONAL DESIGN FOR THE DEVELOPMENT OF CONCEPTUAL HUMAN SPACECRAFT

4.1 ABSTRACT

A full-scale mockup of a Lunar Lander habitat was constructed based on the dimensions defined in NASA's 2005 Exploration Systems Architecture Study. An initial goal of the project was to establish methods and procedures for constructing a rapidly reconfigurable engineering prototype while concurrently using the analogue as a means of developing systems-level requirements based on anticipated operational concepts. The use of cost effective materials in the mockup provided a simplified approach for construction of system concepts and readily allowed subsequent configuration changes. The application of rapid prototyping provides a means of incorporating a hands-on 'human in the loop' component to spacecraft system design. This effort was originally intended to be used to help evaluate vehicle configuration options, determine subsystem mass and volume budgets, reduce risk factors and derive requirements before the Lunar Lander preliminary design review. The lessons learned in the initial prototyping activities were developed into a conceptual design process based on a minimum functionality design methodology. This process allows designers to fully investigate the design configuration before the development of CAD models and higher fidelity models such as CFD and FEA.

4.2 INTRODUCTION

The term "rapidly reconfigurable" prototyping used in this context refers to the flexibility and adaptability of quickly changing the geometry of a proposed spacecraft design in real time to investigate concepts and issues. Unlike other physical prototypes that are constructed using plywood or foam-core, the materials used in this activity were sufficiently sturdy and adaptable to changing subsystem hardware layouts in a cost effective manner. In the course of this work, the overall spacecraft configuration was changed many times. Each change to the overall design was on the order of a few minutes to a couple of hours to rebuild and reconfigure the prototype. This rapidly reconfigurable prototyping activity was focused on reducing risk, investigating human factors, and determining subsystem integration and layout before the use of CAD design and high fidelity analyses.

This chapter describes the prototyping research activities at the University of Colorado from the fall of 2006 until the spring of 2009. Many lessons were learned as a result of this prototyping activity and one of the significant aspects of rapidly reconfigurable prototyping is that it integrates well into a minimum functionality design methodology. The early research efforts beginning in 2006 have led to a project based design curriculum for human spacecraft conceptual design at the University of Colorado. Two conference papers resulted from this work (Higdon and Klaus, 2008; Klaus and Higdon, 2009).

The many lessons learned from the prototyping work shaped the development of a conceptual design process based on minimum functionality design and risk based design. This process was developed after the prototyping efforts and serves as an easy to use roadmap for future spacecraft conceptual design exploration.

4.2.1 Rationale for Rapidly Reconfigurable Engineering Prototypes

A central theme in the use of rapidly reconfigurable prototypes during the conceptual development phase is to reduce the overall vehicle costs through the investigation of many factors in the spacecraft design. The impact of early program decisions on the overall development cost of a spacecraft is well documented (Huang, 2005; Hammond, 1999; Bell *et al.*, 1995). As the vehicle matures from concept to preliminary design, unexpected requirement drivers increasingly add to overall program costs (NASA, 1995). The use of prototypes has been suggested as a starting point in the development of small, high-risk projects by Mulenburg and Gundo (2004), who stated that "*despite creating beautiful three dimensional models, and detailed computer drawings that can consume hundreds of engineering hours, the resulting designs can be extremely difficult to make, requiring many changes that add to the cost and schedule*". In light of the desire to keep program costs down, as well as to produce a safe, reliable human-rated spacecraft, incorporating rapidly reconfigurable physical prototyping throughout the design flow can be used to identify and analyze key system drivers, vehicle interfaces and technology updates early in the program, when their modifications result in cost savings, rather than later, when changes tend to create cost overruns.

Another advantage of physical prototyping is use of a design aid termed a "Boundary Object", which is intended to facilitate communication across different disciplines involved in the design process. In a research study of adaptive design using system representations, Dare *et al*. (2004) defined Boundary Objects as "*representations of knowledge that can improve communication and understanding between groups or organizations with different lexicons and cognitive foundations*". The use of a rapidly reconfigurable physical (full-scale) prototype facilitates communication between multidisciplinary groups and helps to resolve discrepancies before a design configuration is chosen. Other studies into how prototypes support engineering design decisions have been conducted by Brereton and McGarry (2000), who concluded that physical objects help to influence design thinking by helping relate previous knowledge and experiences to the current needs. The advantage of this approach is that having a physical (full-scale) system that everyone can see and touch reduces the possibility of misinterpretation where Computer Aided Design (CAD) drawings or other computer representations may be viewed incompletely or in diverse ways by different members of the design team.

The use of rapidly reconfigurable prototyping assists in conducting multiple, low cost trade studies in a short time frame and helps to identify and optimize key variables during the conceptual design phase. Research conducted by Thomke *et al*. (1998) on the modes of experimentation highlighted the use of a "serial experimentation process" in which rapid learning occurs between successive iterations. This type of experimentation process produces results at the lowest cost, but at a "medium" speed compared to parallel experimentation. The downstream cost savings, however, become particularly important in the design of human spacecraft due to the unique one-off nature of the business, where a little more time and effort up front pays off substantially in the long run.

4.2.2 Objectives of Rapidly Reconfigurable Prototyping

Incorporating standard principles of systems engineering with the use of a full-scale, rapid prototype mockup and analytical processes has demonstrated a dynamic design approach that builds the spacecraft around the mission objectives and operational needs of the crew. However, as the research progressed, it became obvious that a minimum functionality design process was needed to account for incremental mass changes in the subsystems. Due to complex integration factors, small changes in one subsystem can become amplified through multiple other subsystems; thus disproportionately impacting the overall vehicle mass. In order to understand integrated mass and reliability drivers in the complete vehicle, a minimum functionality design methodology is proposed as a starting point before prototyping to study the relationships between system and subsystem designs and the resulting changes to the overall vehicle design.

In order to simultaneously minimize overall development time and cost, as well as operational risk and cost, a minimum functionality design methodology coupled with the use of physical prototyping and computer modeling can be employed to establish a Multidisciplinary Design Optimization scheme tailored for human spacecraft conceptual design. During the conceptual design of the Apollo Lunar Module, the major factors that drove the LM mass during the design phase involved meeting reliability and mission operational requirements, and configuration definition (Kelly, 1999). The design process presented in this work could be used to develop and iterate requirements down to the subsystem level during the conceptual design phase. In turn, the early prototypes offer a means of CAD model validation via a sanity check using the physical mockup. While it may be well recognized that prototypes enable a direct 'human in the loop' component to a design, the focus of this research is on the development of an integrated, end-to-end process that utilizes physical prototypes in conjunction with a minimum functionality design methodology for cost effective and time efficient development of human spacecraft. This low cost and readily reconfigurable approach provides a competitive advantage for government and industry seeking to develop and validate interfaces for new systems and technologies for human spacecraft, as well as a unique educational tool for academic purposes.

4.2.3 Background of Rapidly Reconfigurable Prototyping

The first ideas for constructing a prototype of the Lunar Habitat Module were investigated in the ASEN 5158 Space Habitat Design class in the fall of 2006. A group of students first conceived ideas for the Lunar Module Habitat based on the recently published NASA ESAS report. Several small scale models and CAD models were generated in this class to investigate the notion of building a full scale prototype for further investigation. Shown in Figure 20 is a CAD model developed for the ASEN 5158 class for studying the Lander Habitat design.

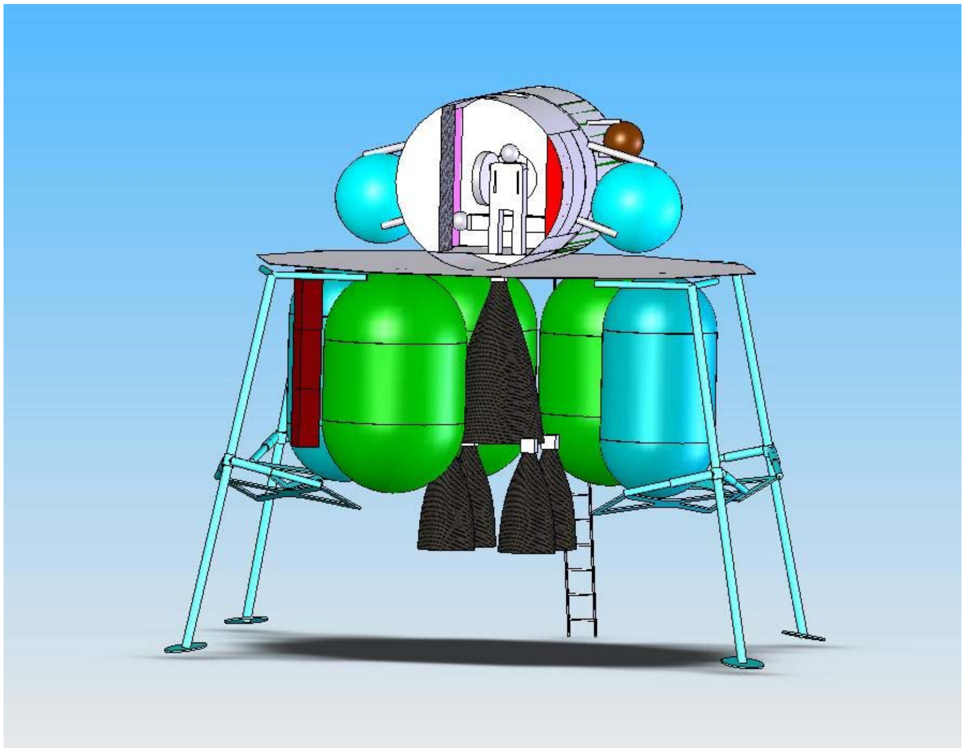


Figure 20: ASEN 5158 Space Habitat Design, fall 2006 design concept.

The initial development of concepts in the fall of 2006 led to the investigation of where to build a full scale mockup and how to build the mockup. The initial design effort was led by Dan Baca, who gathered team members for beginning the fabrication as a voluntary activity. In a stroke of luck, a large room located in the Discovery Learning Center near the Engineering Center was unoccupied and was the perfect location for showcasing the activity in this early design stage and generating interest among the engineering students. Shown in Figure 21 is a form factor skeleton with Dan Baca (on left) and myself. The room was a perfect fit for our needs and one can see that the top of the form factor (approximately 10 feet in diameter) was almost touching the hanging lights; which were eventually moved upwards to allow clearance.

Figure 21: An early form factor Lunar Habitat skeleton.

4.2.4 Goals of Rapidly Reconfigurable Prototyping

A rapidly reconfigurable prototyping process is intended to follow a minimum functionality design methodology where a baseline minimum functional vehicle configuration is chosen for additional study of human factors and integration of proposed technology choices. The efforts of the prototyping activity can be conducted in parallel with other analysis efforts such as CAD, but the risk with developing CAD models too early is that based on the lessons learned from rapid prototyping and the study of human factors, the model will change. Using information gathered from the prototyping activity and a human factors standpoint, the minimum functionality baseline could be iterated to further define the spacecraft configuration based on crew needs and subsystem integration challenges.

Because the prototypes were easy to reconfigure, it readily accommodated incremental design changes throughout the research efforts. Using test subjects in simulated spacesuits, various tasks such as egress through hatches were studied to assess operational feasibility related to human-vehicle interfaces. This application demonstrated a means of incorporating a hands-on ‘human in the loop’ component to spacecraft system design and the information was used to validate previous mass and volume budgets to determine which critical drivers should be further optimized. The advantage of using a full-scale prototype in the system-level design process is that it provides information from a firsthand perspective that can be difficult to achieve using CAD models alone. As a result of the human spacecraft prototyping activity, several other projects were developed. The activity that was once voluntary has been fully developed into a project based design curriculum. Although this was not a primary objective in the initial phases, its usefulness for teaching students the fundamentals of spacecraft development has proved to be very beneficial from an academic standpoint.

4.3 LUNAR MODULE PROTOTYPING RESEARCH

The initial prototyping activity focused on the construction of an ESAS design configuration for the LSAM habitation and ascent module. As NASA was in the early stages of development for the Lunar Lander, personnel at the Johnson Space Center suggested that a key need within NASA was determining the minimum mass ascent stage for lunar ascent. The prototyping activity conducted in the Discovery Learning Center was based on an ESAS Minimum Ascent Module and Habitation Module configuration.

The follow on prototyping activity was conducted approximately a year after the start of the initial activity and focused on the ESAS Minimum Ascent Module configuration. Because of funding constraints the prototype project was relocated to a much smaller lab in the Engineering Center in the Aerospace Engineering Sciences wing. Because of limited room to construct the entire prototype, the decision was made to strictly focus on the Ascent Module only. In hindsight, the “downsizing” of the prototyping activity to a much smaller and focused scale allowed more information to be gathered about the spacecraft configuration, where a true minimum functionality approach was investigated.

4.3.1 ESAS LSAM Configuration

Using the ESAS guidelines specified in paragraph 4.2.5.1.3, LSAM Configuration Trades, the LSAM Configuration Concept 1 design was used as a point of departure for the initial prototyping activity. The emphasis of this configuration was to minimize the overall size for the Ascent Module (NASA, 2005). Shown in Figure 22 is the layout for the LSAM configuration for the Ascent Module, the Surface Living Module and the Airlock.

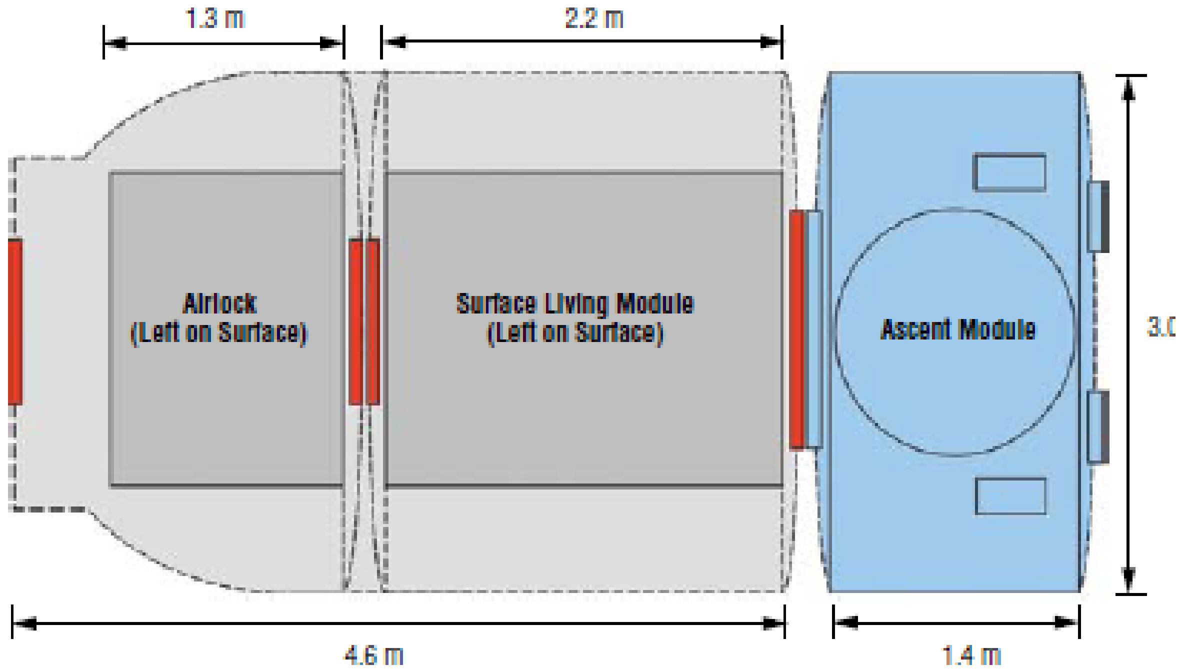


Figure 22: ESAS LSAM Configuration Concept 1 with Ascent Module (NASA, 2005).

The challenge with this configuration was determining the minimum size of the Ascent Module in relationship to the Habitation Module. In addition, a hatch and release mechanism would need to be designed to allow the lateral release of the Ascent Module from the Surface Living Module. A docking adaptor would be placed on the top of the Ascent Module for docking with the CEV. Other issues with this design configuration such as an off thrust axis docking location with CEV would drive additional mass in the structure to overcome torque loads during EDS propulsion. Even with the many issues surrounding the layout of the modules, the recommendations from JSC helped to guide the focus of the prototyping activity toward minimizing the Ascent Module and evaluating the design from a human factors and operational standpoint. Other ESAS concepts utilized a combined Living Module and Ascent stage with a separable airlock that would be left on the surface. These concepts were not investigated because of the focus to minimize the Ascent Module configuration. Shown in Figure 23 is an illustration of the early LSAM concept with a combined Ascent Module, Living Module, and Airlock.

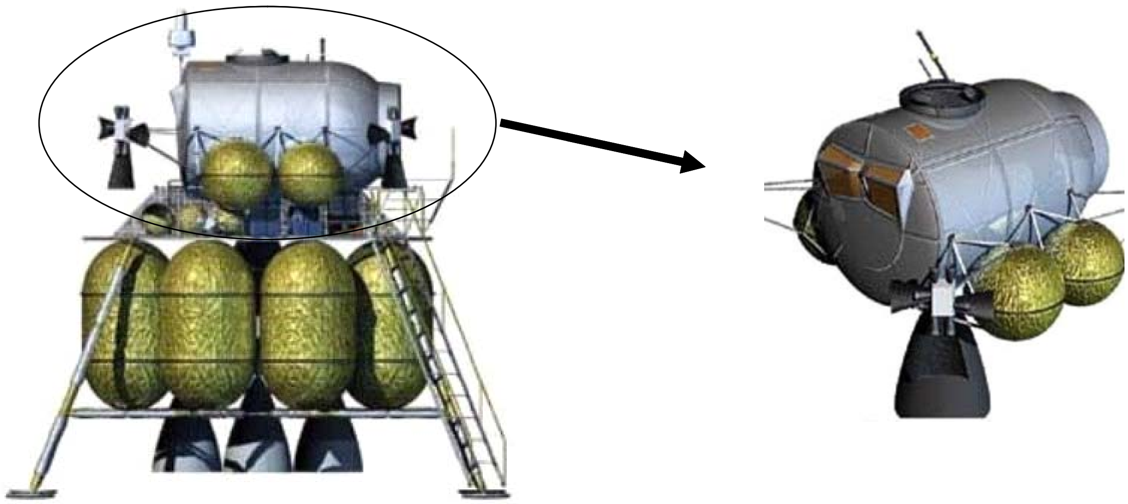


Figure 23: LSAM Lander and Ascent Module (NASA, 2005).

4.3.2 ESAS Minimum Ascent Module Configuration

The prototyping activities for the ESAS Minimum Ascent Module configuration followed the initial prototyping of the outer skeleton prototyping of the ESAS Ascent Module and Habitation Module configuration. This effort was focused primarily on the interior subsystem components and layouts for a minimum functionality Ascent Module. Single string subsystems and minimum crew accommodations formed the baseline for the minimum functionality prototype. Components such as propulsion, tanks, and exterior systems were not included in the prototyping effort due to lack of available space for building the full scale prototype. Shown in Figure 24 is a concept developed in the ASEN 5158 Space Habitat Design Project, fall 2007.

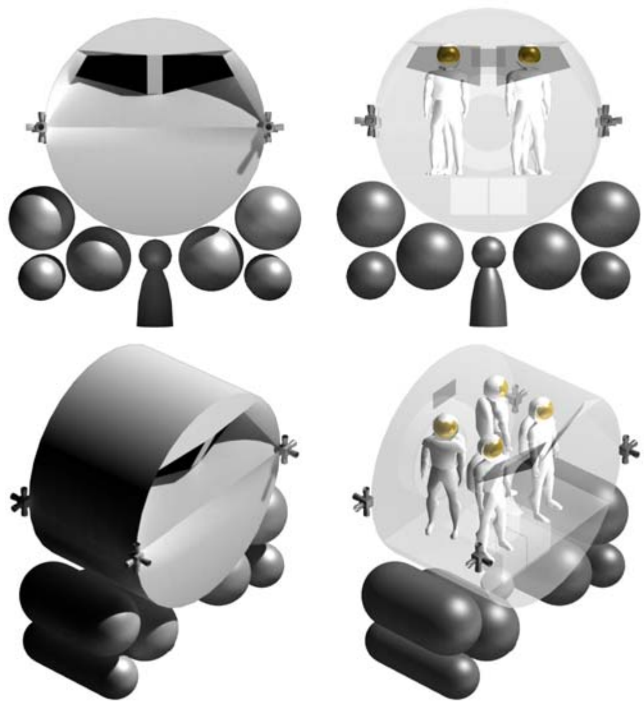


Figure 24: Minimum Functionality Ascent Module concept from ASEN 5158 Space Habitat Design Project, fall 2007.

4.3.3 Prototyping the Minimum Ascent and Habitation Module

A physical full scale mockup of the LSAM Lunar Lander's Ascent Module, Habitation Module, and Airlock was kicked off with the construction of a form-factor outer structure based on NASA's Exploration Systems Architecture Study baseline configuration (NASA, 2005). The use of various construction methods and materials was iteratively attempted over the course of the first months to determine fabrication techniques that would allow the mockup to be easily reconfigured for future design modifications.

The effort began with defining mission objectives and identifying anticipated crew operations in order to derive initial volumetric and mass requirements for the Lunar Lander based on a functional decomposition. Next, the initial prototype was constructed using these top level requirements to define an external form factor of the proposed vehicle. A design goal throughout the activity was defining requirements for a minimal Ascent Module mass. At the same time, the potential for extensibility of the Habitat Module hardware remaining on the moon for use as part of the Lunar Outpost was considered through the design of universal hatches and reusable docking mechanisms. A major driver of the Lunar Lander mockup construction was the division of two separate modules in order to study trade factors for minimizing the Ascent Module (flight deck) mass and optimizing surface access airlock operations. The overarching goal was to maximize habitable volume and mass remaining on the moon for future use, while minimizing the Ascent Module mass needed to launch and rendezvous with the crew module.

Beginning in the fall of 2006, the outer shape of the LSAM Modules was being constructed to determine construction materials and methods. The first attempts at constructing a 10 foot diameter cylindrical shape with PVC were not very successful. Shown earlier in Figure 21 was the first skeleton constructed with PVC. The difficulty with PVC is that it cannot hold lateral loads in the skeletal members. An alternative to PVC was needed, but the use of plywood would be heavy and very difficult to reconfigure. After multiple visits to hardware stores in the area, a non-traditional solution was proposed. The use of ½ inch aluminum electrical conduit was a cheap and lightweight alternative to the use of PVC. Shown in Figure 25 is the upper section of the LSAM prototype that utilized aluminum electrical conduit for the structural members.

Figure 25: Upper section of LSAM Prototype with electrical conduit as structure.

The conduit was easily formed by hand and was sturdy enough to hold the weight of the structure. In typical applications of conduit, the sections are joined at an electrical box. However for this application, something that would resemble more of a "tinker toy" than a hard connection to an electrical box was required. Joints between structural members had to be strong enough to carry the weight and easily reconfigurable. Thus, a non-traditional method was adopted. Shown in Figure 26 are the joints used in the prototyping.

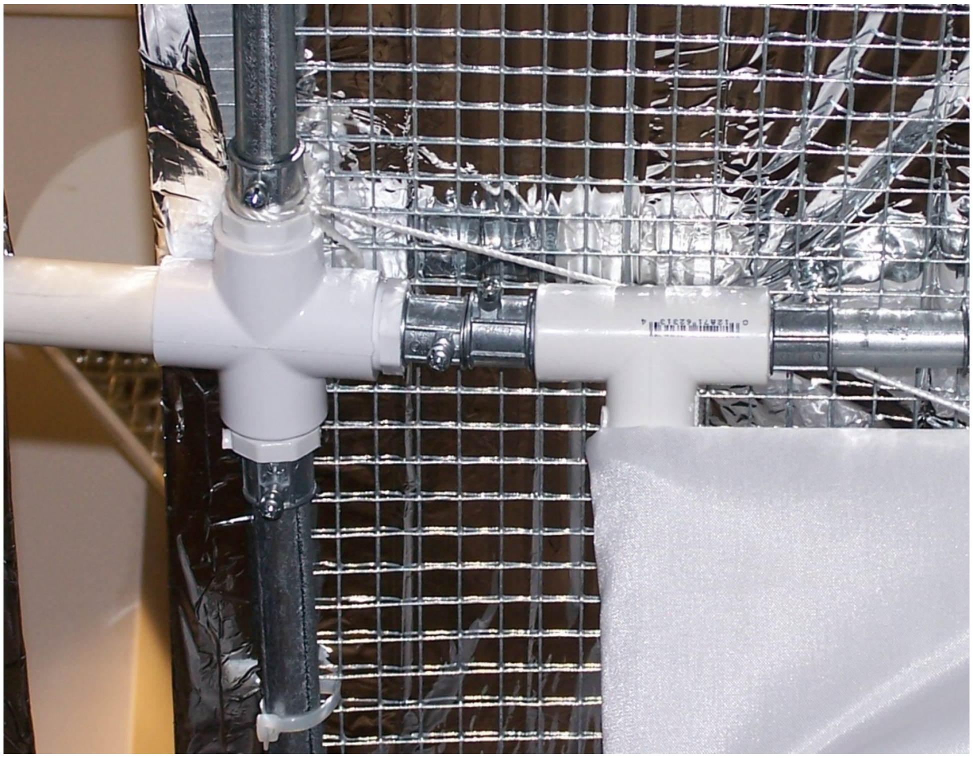


Figure 26: Typical joints used for prototyping.

As observed in Figure 26, the ends of the electrical conduit are fastened to fittings that would normally thread into an electrical box. In this application, the threads of the electrical conduit fittings were cross threaded into PVC threaded adapters for a ½ inch pipe to a ¾ inch PVC. Because of the difference between pipe threads and the electrical fitting threads, the threads were cross threaded; but the interference made the joint strong enough to carry the loads. Also shown in the figure are PVC "T" joints and cross joints that could be attached to ¾ inch PVC pipe. A wire mesh was used to simulate the OML panels where a layer of aluminized Mylar was taped to the ends of the panel to prevent cuts from sharp edged wires. The panels were zip tied to the electrical conduit structural members and were easily removable by cutting the ties and replacing. The wire mesh was affectionately called "chicken wire" but this author disagrees with this term and prefers the term “wire mesh fabric.” Overall, the materials used in this prototype were very cost effective, simple, and easily reconfigurable.

In January 2007, a full skeleton of the LSAM configuration was constructed as shown in Figure 27. This form factor skeleton was the correct shape of the ESAS LSAM baseline and formed the basis for the project level class, ASEN 5519 Lunar Module Design, in spring 2007 to fully construct the LSAM module prototype and further develop the facility.

Figure 27: LSAM skeleton in January 2007 with simulated spacesuits.

Participants in the project level class were: Dr. David Klaus, Dan Baca, Bruce Davis, Kevin Eberhart, Lisa Geschwill, Kevin Higdon, Laurren Kanner, Ryan Kobrick, Danielle Massey, Sean O'Dell, and Courtney Wright. The goals of this class were to:

- Construct a full-scale prototype of Lunar Lander modules;
- Study human to vehicle interfaces, and incorporate latest changes to design concepts (volumetric, mass, geometry, etc.);
- Use results from mockup activity to derive requirements for subsystem development;
- Develop mass analysis computational models;
- Identify Con-Ops requirements;
- Conduct outreach activities; and
- Develop funding sources for future research topics.

Over the course of the semester, the prototype was reconfigured multiple times and developed into a fully enclosed structure. The specific accomplishments of the ASEN 5519 Lunar Module Design class included:

- Constructed a full-scale (reconfigurable) mockup of the exterior form factor of the Lunar Lander;
- Reconfigured and rebuilt the Ascent and Habitation modules multiple times;
- Developed a first draft of mass analysis computational models;
- Identified top level Con-Ops requirements using NASA's 181 things to do on the Moon;

- Conducted six K-12 outreach activities; and
- Secured continued funding through William F. Marlar Memorial Trust.

In addition to the accomplishments of constructing the prototype and identifying top level Con-Ops requirements, the class received the following recognition:

- 3rd Place in AIAA Region V Student Paper Conference in April 2007;
- Competed in a national design competition for development of a Lunar Outpost analogue to be built in Hawaii with the Pacific International Space Center for Exploration Systems (PISCES) where Dr. Klaus, Jonathan Metts, and Bruce Davis represented CU in the final competition and awarded 2nd place in the competition; and
- Channel 4 CBS Television Interview.

Figure 28: Channel 4 CBS Denver television interview.

At the end of the spring 2007 semester, the initial construction phase of the exterior shell was completed and detailed studies were focused on hatch design and dust mitigation strategies. The overall configuration of the mockup was a 10-foot diameter shell with a re-configurable floor and an exterior covering of Mylar attached to modular, wire mesh backing panels as shown in Figs. 29 and 30. The entire effort to construct the form factor skeleton was the collaboration of many students who volunteered additional time to assist with the construction of the prototype.

Figure 29: ESAS Split Ascent and Habitation Module configuration front view.

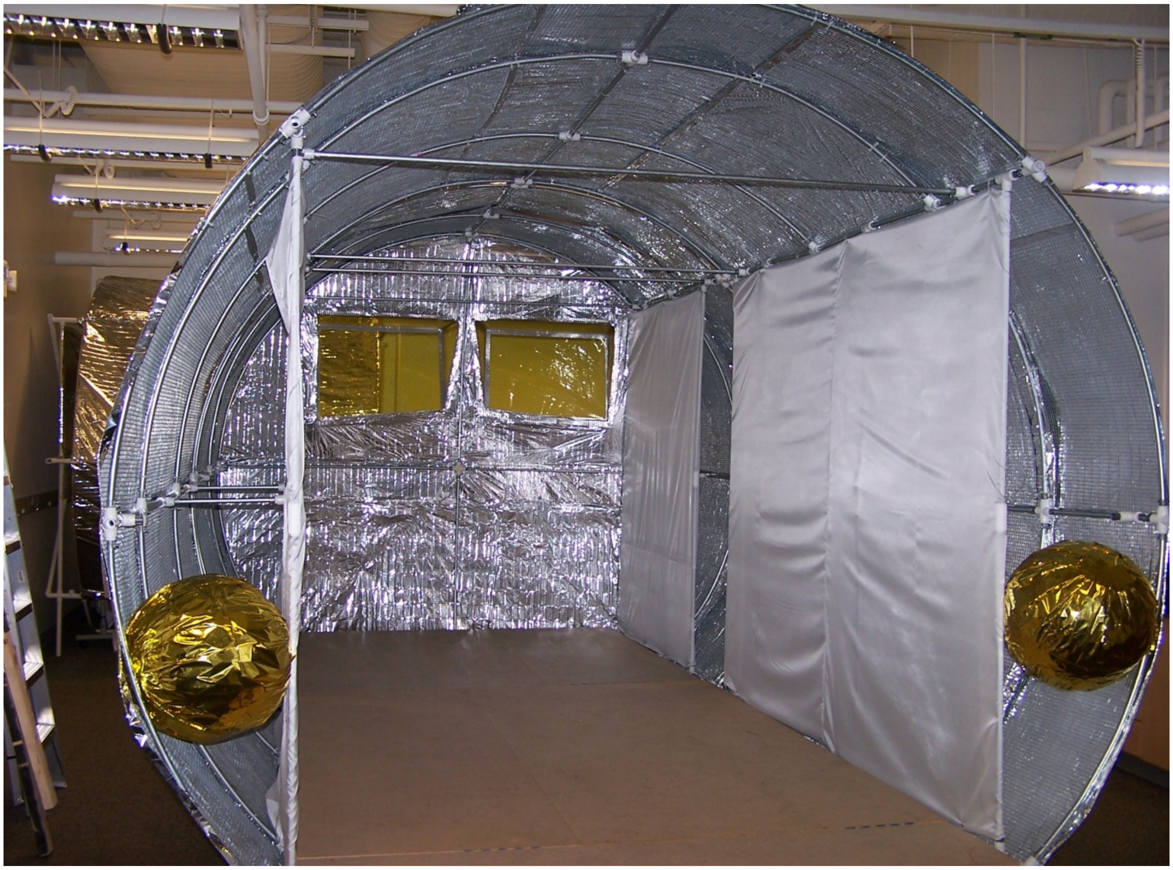


Figure 30: ESAS Split Ascent and Habitation Module configuration interior view.

4.3.4 Prototyping the Minimum Ascent Module

During the summer of 2007, the prototyping activity for the Lunar Module moved from the Discovery Learning Center to a much smaller lab in the Engineering Center. This focus of the new lab was to house the prototype for a minimum functionality Lunar Ascent Module. During the fall 2007 Space Habitat Design class, the semester project topics were tailored toward the minimum functionality Lunar Ascent Module spacecraft configurations. The prototype of the Lunar Ascent Module in August 2007 is shown in Figs. 31 and 32. The top and lower portions of the cylinder were removed because of height limitations.

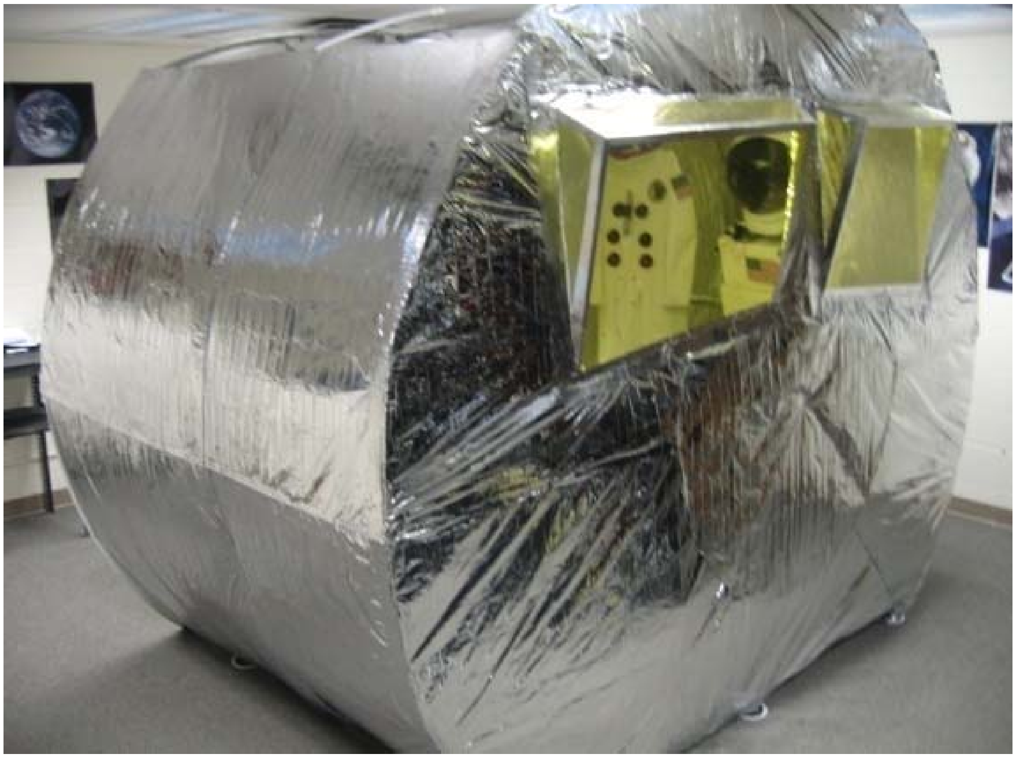


Figure 31: Lunar Ascent Module prototype in August 2007 front view.

Shown in Figure 32 is a form factor prototype for a hatch design. The size of the hatch could be reconfigured and test subjects (students) in simulated spacesuits were used to determine the minimum size. In addition to investigating human factors and hatch configurations, the specific accomplishments during this semester included:

- Determined the minimum mass required for lunar ascent based on physics;
- Developed a “top-down” heuristic mass breakdown of subsystems;
- Created a “bottom-up” mass based on available subsystem technologies;
- Identified key drivers for future study; and
- Developed propulsion and trajectory simulations for future studies.

Figure 32: Lunar Ascent Module prototype in August 2007 hatch view.

Shown in Figure 33 are examples of the propulsion and trajectory simulation tools developed in MATLAB to analyze propulsion requirements for an equatorial orbit and a series of plane changes from a polar landing site to an equatorial orbit.

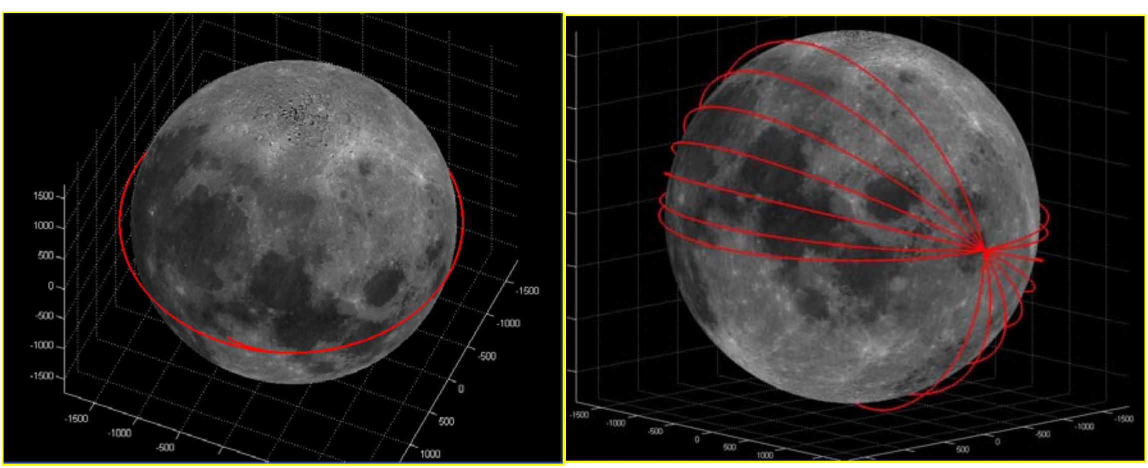


Figure 33: Propulsion and trajectory simulations of lunar ascent and plane change.

The research conducted in the fall of 2007 led to a more focused approach for determining how to couple the prototyping activities with a minimum functional design approach. The last part of the prototyping effort was the most significant because many lessons were learned about the minimum functionality design approach and how to determine subsystem mass and volume through a bottom-up philosophy. Starting with the top level goals of a minimum functionality mission, all of the subsystem functions were revisited in order to develop the baseline minimum functional Lunar Ascent Module configuration. Shown in Figure 34 is a diagram of the top level functional decomposition of the Lunar Ascent Module.

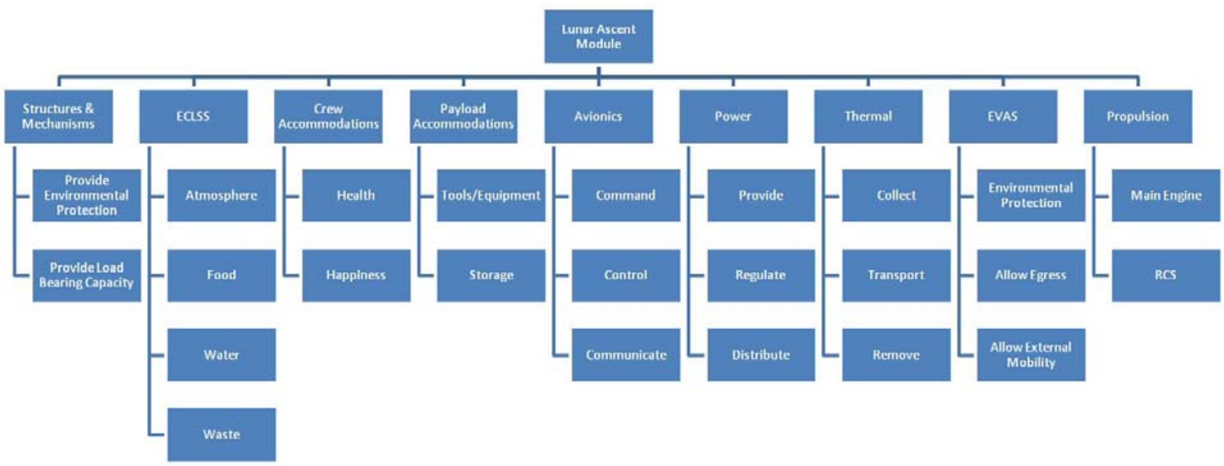


Figure 34: Top level functional decomposition of Lunar Ascent Module.

Shown in Figures. 35 – 40 are the functional decomposition diagrams for Avionics, Structures and Mechanisms, ECLSS, Thermal, Crew Accommodations, and Payload subsystems. Although the entire decomposition matrix is too long to be listed here, a detailed decomposition matrix is listed in Appendix C. The following diagrams show how the various subsystems were decomposed into lower level functions.

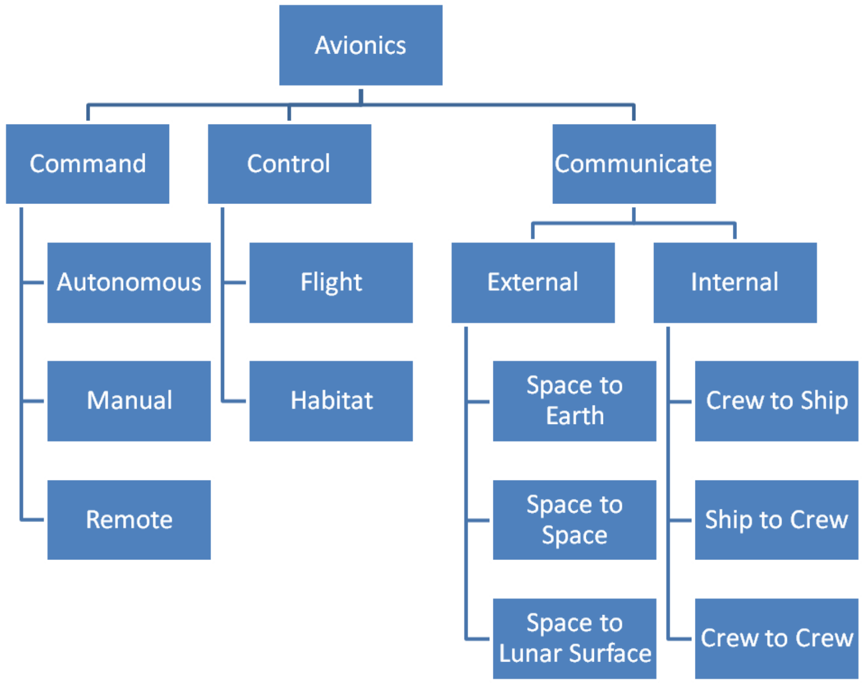


Figure 35: Avionics subsystem functional diagram.

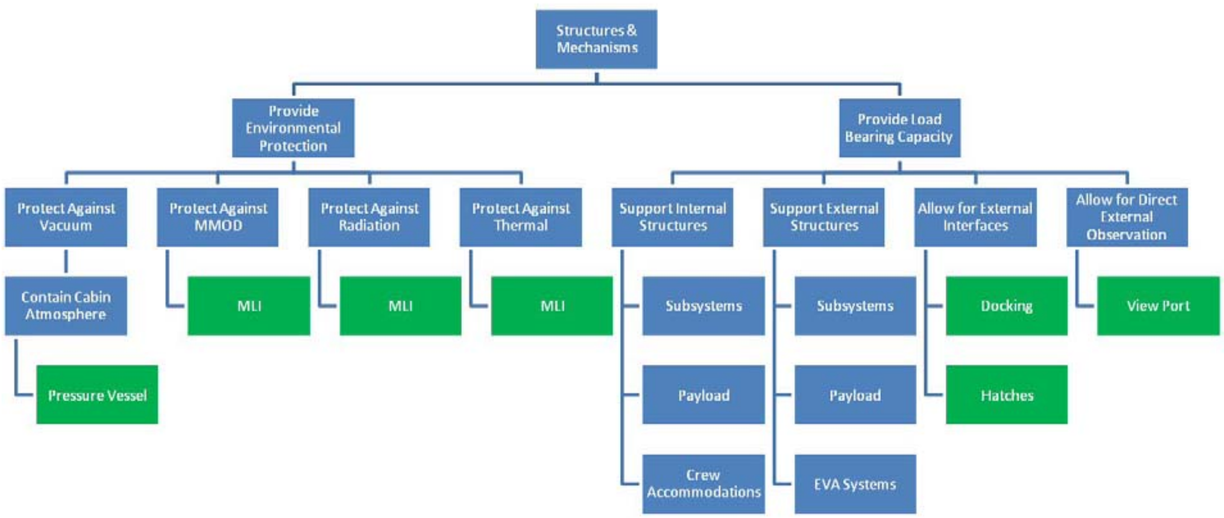


Figure 36: Structures and Mechanisms subsystem functional diagram.

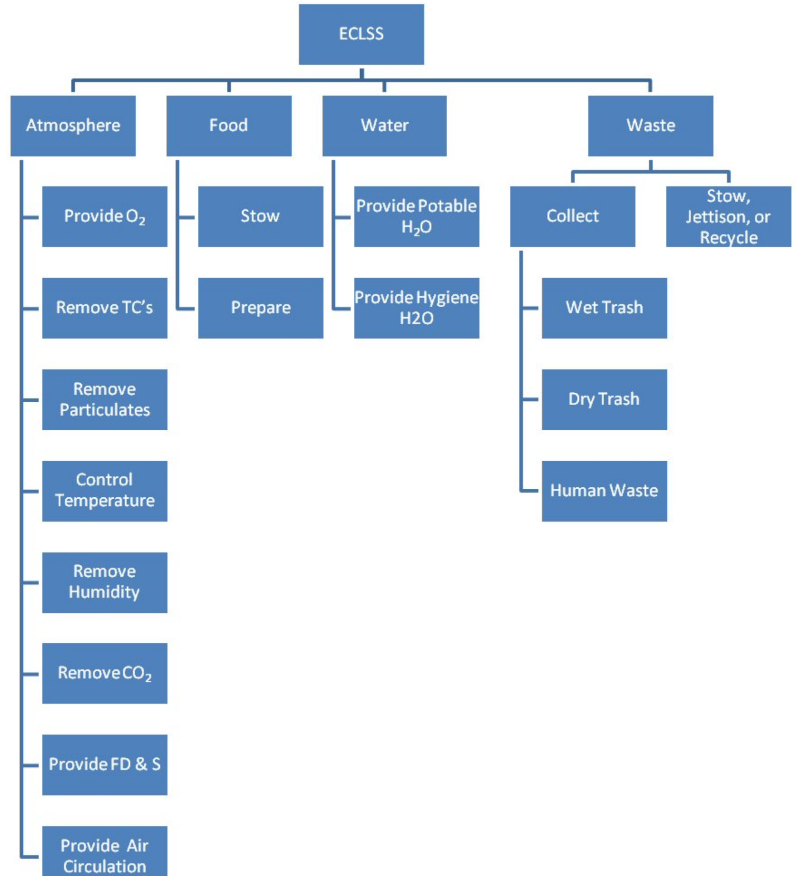


Figure 37: ECLSS subsystem functional diagram.

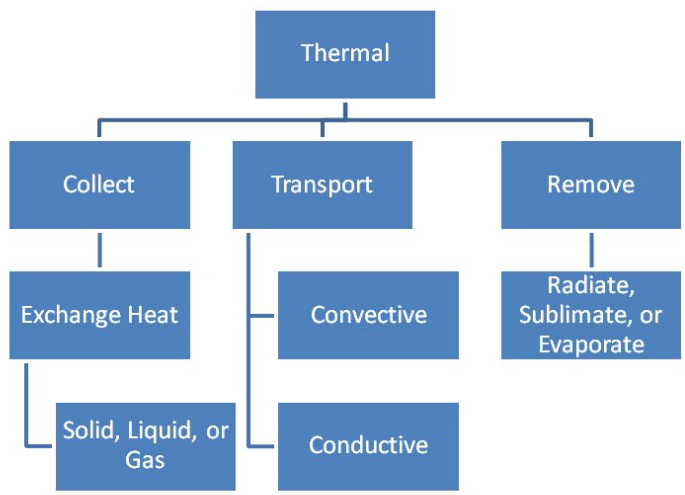


Figure 38: Thermal subsystem functional diagram.

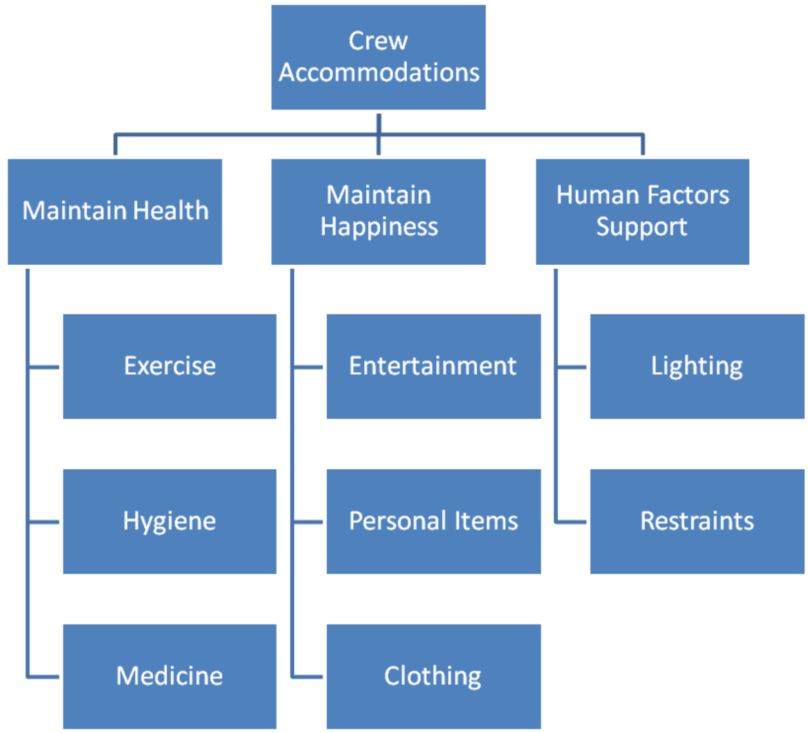


Figure 39: Crew Accommodations subsystem functional diagram.

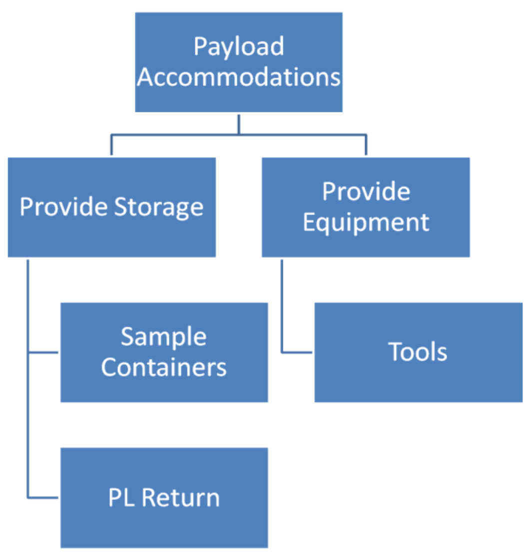


Figure 40: Payload Accommodations subsystem functional diagram.

Based on the minimum number of functions, subsystems were sized according to heuristics using mass and volume information given in Human Spacecraft Mission Analysis and Design (Larson *et al.*, 1999) and Advanced Life Support Baseline Values and Assumptions Document (Hanford, 2004). Potential technologies were identified to perform the low level functions. For each of the individual components, a series of specification sheets were developed to assist in the sizing of the component prototypes. The specification sheet contained as much information as available in the open literature and heuristic volumetric equations were used. If sufficient information about mass and volume was not available, a cubic volume was used as a placeholder. Shown in Figure 41 is a list of the information captured in the specification sheets.

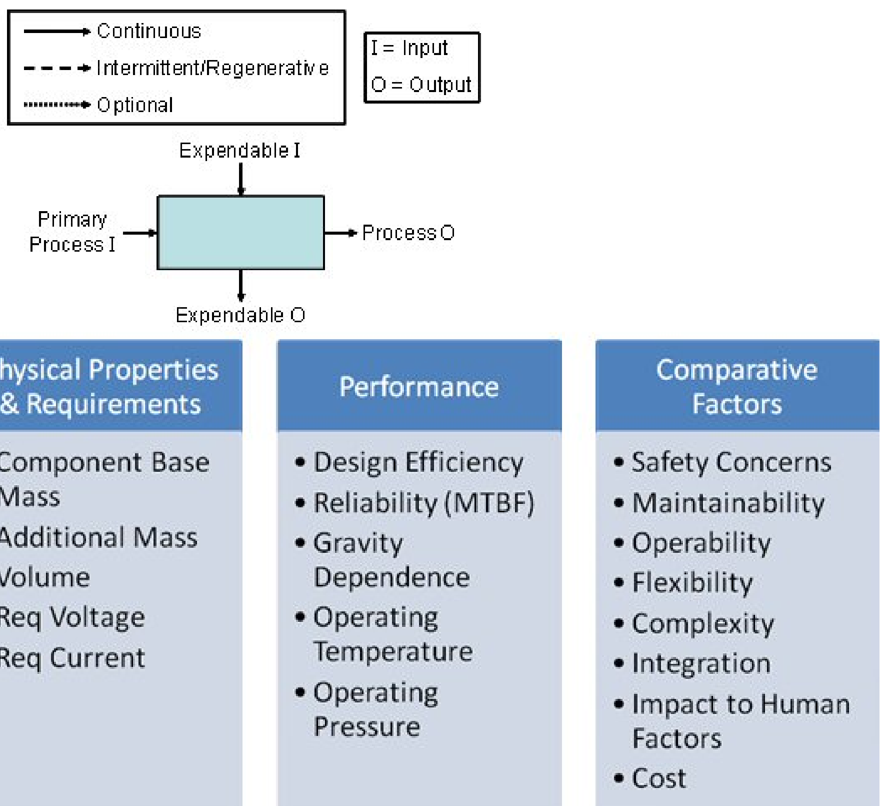
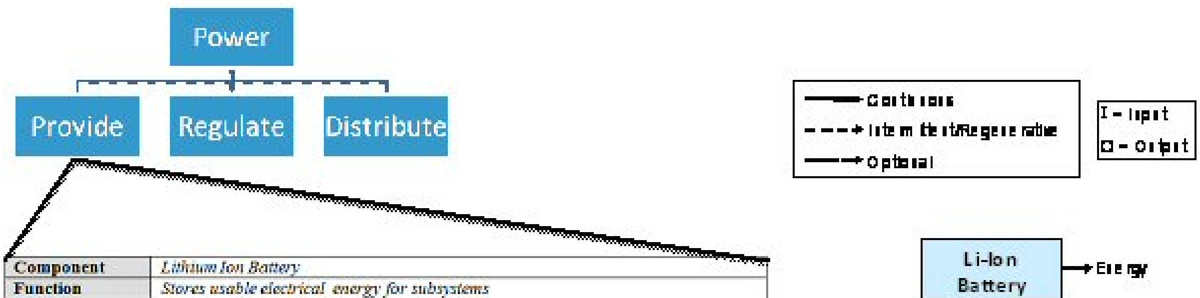


Figure 41: Information documented in specification sheets.

Shown in Figure 42 is an example of one function linked to a specification sheet. In this example, a Lithium Ion battery is chosen as a technology candidate for conducting the function "Provide Power." The specification sheet captured as much information as possible to provide a reasonable estimate for a prototype battery.



Component	*Lithium Ion Battery*
Function	*Stores usable electrical energy for subsystems*
Mode	*Active*
Description	*Chemical reaction results in current flow via a cathode and electrolyte. One type of lithium cell is a lithium-thionyl chloride cell. A liquid mixture of thionyl chloride ($SOCl_2$) and lithium tetrachloroaluminate ($LiAlCl_4$) acts as the cathode and electrolyte respectively. A porous carbon material serves as a cathode current collector which receives electrons from the external circuit.*
Technology	*Cathode and electrolyte. Porous carbon material acts as the cathode current collector.*
Inputs (mass / energy)	
Outputs (mass / energy)	*123 W-hr/kg at 20 °C and 85 W-hr/kg at -20 °C*
Component Base Mass, kg	*750W-hr/kg (theoretical)* *123 W-hr/kg at 20 °C and 85 W-hr/kg at -20 °C* *~7.7kg (~17 lbm) (Mars Rover)*
Additional Mass, kg	*Wiring*
Voltage, V	*28V (3.6 V closed circuit voltage)*
Current, A	*25 A-hr*
Volume, m^3	*3870 cm^3 (.014 ft^3) (BVAD)* *4195 cm^3 (.0148 ft^3) (Mars Rover)*
Geometric Approx.	*16" x 4" x 4" (Mars Rover)*
Design Efficiency (%)	
Operating Temperature, K	*-20 °C to 20 °C*

Figure 42: Example of specification sheet linked to a function.

Central to the sizing of the subsystem components were the physiological needs of the crew during the lunar ascent. Shown in Figure 43 are the physiological requirements for a crew of 4 for the Lunar Ascent Module.

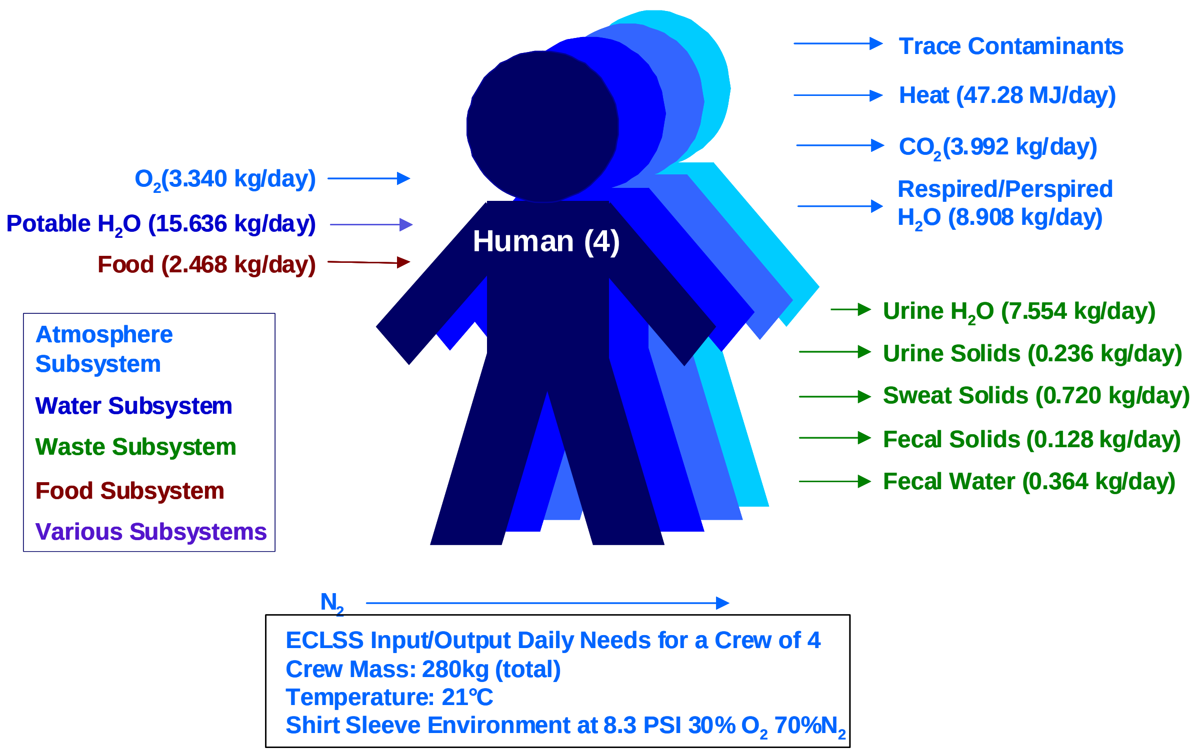


Figure 43: Physiological requirements for a crew of four.

Prototypes for the following interior subsystem components were constructed and installed in the minimum functionality Lunar Ascent Module baseline configuration:

- Avionics – guidance and control computers, transponder, docking radar, communications, data storage, flight deck, heads up display, and flight controls:

- ECLSS – carbon dioxide removal, air ducts, oxygen and makeup gas supply, humidity removal, particulate removal, and umbilical interfaces;
- Crew Accommodations – food, water, personal hygiene, tools and equipment, lighting, waste management, and stowage;
- Thermal – cold plates, pumps, and coolant loops;
- Power – batteries, inverters, power controllers;
- Secondary Structures;
- Hatches and docking tunnel geometry; and
- Payloads – sample return containers.

Shown in Figure 44 is the final configuration of the minimum functionality Lunar Ascent Module. The cardboard boxes represent subsystem volumes for the various components. Also prototyped were the cabin air ducts as well as a flight control display that allowed re-configurability. The rear bulkhead of the prototype was removable (not shown in Figure 44) to allow easy access to the interior for installing large components. A "hoop" on the ceiling represented the area of a docking tunnel. Each component was labeled according to its particular subsystem and function according to the functional decomposition matrix. At the conclusion of this prototyping effort, it was obvious that an ESAS sized minimum functionality Lunar Ascent Module could be further optimized for habitable volume. Although four crewmembers could comfortably stand in the "square cube" of the habitable volume, much volume was not being utilized in the additional spaces. The next step in the prototyping activity would have likely been a reduction in the habitable volume and repackaging of subsystem components. Because the focus of this prototyping activity was to determine the usefulness of rapidly reconfigurable prototype in a minimum functional design, it succeeded in establishing the usefulness of physical prototyping as a first step in the conceptual design process. The role of human factors is a very important part in analyzing various subsystem integration and layout configurations.

Figure 44: Minimum functionality Lunar Ascent Module with interior components.

4.3.5 Summary of Lunar Module Prototyping Research

Because the focus of the prototyping activity was not to design the entire Lunar Ascent Module, this research served a purpose in demonstrating the usefulness of physical prototyping as a first step of defining human factors early in conceptual design. The significant aspects of the prototyping activities included the following:

- Establishing a "Boundary Object" to increase communication between designers with various backgrounds;
- Determining subsystem components using a bottom up philosophy;
- Understanding the limitations of subsystem integration and layout;
- Exploring human factors such as reach, access, egress, and proper configuration of subsystems;
- Providing a "test-bed" for simulated suit activities;
- Understanding human and vehicle interactions as a precursor to analysis and trade studies;
- Assisting with the design of Secondary Structures;
- Pre-validating CAD design activities;
- Developing design requirements; and
- Reducing design and configuration uncertainty.

The amount of time to construct a full scale mockup was very small compared to the same amount of time required to develop detailed CAD models. In this authors experience with many thousand hours of CAD modeling, the CAD model can only provide the designer with limited parametric information such as mass, interference, and rudimentary layout. The issue with relying completely on CAD is that it does not provide sufficient information about other aspects of the design such as human factors, manufacturing, and integration. The assembly might seem to be straight forward in the CAD, but the part may be either difficult to manufacture or install on the vehicle due to tight clearances, tolerance stack up, or other physical factors. The use of rapidly reconfigurable physical prototyping coupled with a minimum functionality design methodology and risk based design approach is an efficient method for quantifying uncertainty and uncovering many issues in the early stages of conceptual design.

4.4 CONCLUSIONS

The use of prototypes in conceptual design is not a new approach. However its use has been limited in aerospace applications since CAD was introduced as an engineering tool. Although CAD is less expensive than building prototypes, there is much information in a CAD model that is not easily recognized when attempting to assemble and manufacture components for subsystems. One limitation of CAD is the study of human factors in the early stages. Volumetric studies of human interaction provide some indication of how humans will interact with hardware but this does not always uncover many of the unknowns in human factors until hardware is built and tested.

The purpose of studying rapidly reconfigurable prototypes was to learn what can be gained and the limitations. The design configurations chosen were based on ESAS Lunar Module concepts and provided a start in understanding how the many complex subsystems should interact. Because of the initial prototyping activity, the development of the minimum functional design methodology became much easier. It was through hands on experience with building prototype hardware using a bottom up philosophy that many of the key design issues in a minimum functional design were discovered. One of the limitations to prototyping is that it is focused on a single spacecraft configuration and the study of many different configurations can take time. However, when the prototyping activity utilizes results from the Minimum Functional design methodology, the investigation of many more configurations is possible and the rapidly reconfigurable prototype serves as a validation of the analysis assumptions.

The lessons learned in the prototyping activity of the Lunar Module were key in developing the human spacecraft conceptual design process that couples minimum functionality and prototyping in a risk based design approach. Although the prototyping activity described here did not fully explore the entire conceptual design process, it demonstrated how minimum functionality can be coupled to a physical prototype as a precursor to detailed analyses, requirements development, and risk reduction.

After the conclusion of the Lunar Ascent Module prototyping activity in the spring of 2009, the next phase of spacecraft prototyping began developing subsystem layouts and configurations for the Commercial Crew Development program. Although this author was not involved in the prototyping activities beyond 2009, the processes, techniques, and materials were recycled for other spacecraft applications. The collaborative work of many participants has grown from a seemingly small spacecraft design activity into a curriculum for training future aerospace explorers.

4.4.1 Acknowledgements

This work would not have been possible without the contributions of the William F. Marlar Memorial Foundation for providing partial funding of this research and Human

Spaceflight research at the University of Colorado in Boulder. Additional funding was provided by the Engineering Excellence Fund of the School of Engineering at the University of Colorado.

Special recognition should be given to the Dean's Office of the School of Engineering for allowing the first Lunar Habitat prototype to be constructed in the Discovery Learning Center. In addition this effort would not have succeeded without the support of the Aerospace Engineering Sciences Department, the CU LunarMARS team, the CU Bioastronautics team, the Lunar Module Design class of spring 2007, the Space Habitat Design classes of fall 2007 and fall 2008, as well as Dr. David Klaus, Dan Baca, Bruce Davis, Lisa Geschwill, Laurren Kanner, Ryan Kobrick, Danielle Massey, Sean O'Dell, Courtney Wright, Miranda Mesloh, Brad Greybeck, Kevin Eberhart, Jennifer Mindock, Brock Kowalchuk, and Joe Tanner.

CHAPTER 5

A SYSTEMATIC PROCESS FOR EVALUATING SAFETY AND OPERATIONAL FUNCTIONALITY IN CONCEPTUAL HUMAN SPACECRAFT

5.1 ABSTRACT

A systematic process has been developed for evaluating safety and operational capability in conceptual human spacecraft designs. The process begins with a Minimum Functionality baseline configuration and focuses on mitigating hazards through the addition of similar or dissimilar redundancy and is defined in this context as *Safety*. Likewise, operational capability, is added in the form of additional mission time and components for performing mission objectives, and is defined in this context as *Operability*. A matrix of spacecraft configurations is defined with varying levels of *Safety* and *Operability* in order to designate locations of design configurations in the objective tradespace. The grid of spacecraft configurations maps the objective tradespace to the input design space and provides designers with a complete view of potential spacecraft configurations within the tradespace. A series of zones are designated within the tradespace domain to narrow the focus of design concepts to regions of interest for further investigation. The process can be utilized throughout the conceptual design to continually assess the impacts of design changes on the overall spacecraft. Three design configurations of Lunar Ascent Modules were used as case studies to demonstrate and validate the mass addition and tradespace exploration methodology.

5.2 INTRODUCTION

The goal of a conceptual design process is to investigate and develop feasible solutions to a given problem. The conceptual design phase is characterized by a large amount of uncertainty due to many unknown factors. Conceptual designs are developed using the creative skills of designers and design teams and the conceptual design phase can last from months to years; depending upon the type of problem under investigation. In the example of government contracts, conceptual design trade studies are conducted within months to propose candidate solutions to contract proposals.

The conceptual design phase is critical for determining the success or failure of a proposed design. There are many challenges associated with developing a human spacecraft to perform the operational aspects of a mission and bring the crew home safe. A central problem in the early stages of development is due to uncertainty and designs are initially based on heuristic data or designer experience until further analysis or testing is available. In today’s economic environment, additional testing or analysis may not be feasible in the early stages of development. Although this approach is typical in human spacecraft development, it also introduces potential uncertainty and issues in the conceptual design that might not be discovered until later in development. However, the risks associated with any conceptual design will always be present until hardware is built and flown.

Given that the conceptual design is dependent upon the creativity and experience of the designers and design teams, the processes used to develop the initial designs are often unstructured and focused on solving one particular issue at a time. This approach works to solve issues in a sequential fashion, but also creates the possibility of overlooking larger problems at a systems level. Because the subsystems in a human spacecraft are highly coupled, the overall relationships between the subsystems must be evaluated during each step of the design process.

The difficulty faced by designers is that the relationships between the discipline specific areas and the effects upon the safety posture of the spacecraft may not be fully defined or understood by all members of the design team. The assumptions made during the conceptual design phase are often carried over into preliminary design where designers struggle to mature the designs with limited information. The overall systems level relationships between the subsystems should be known prior to preliminary design such that changes in the design will not significantly affect other subsystems in the form of redesign. After the conceptual design reaches a level of maturity, the preliminary design begins and system level requirements are matured based on the knowledge gained during conceptual design.

5.2.1 Process for Tradespace Development and Exploration

The process described in this work develops a conceptual design tradespace for evaluating concepts based on the levels of *Safety* and *Operability* in a spacecraft configuration. In this context, *Safety* is defined as the addition of mass and functionality beyond a "Minimum Functional" baseline configuration that provides hazard mitigation through similar or dissimilar redundancy or safety dedicated components. *Operability* is added functionality dedicated to perform mission specific activities beyond transporting the crew and keeping them alive for a given minimum mission time. The relationship between *Safety*, *Operability*, and total spacecraft mass can be visualized in a tradespace such that design changes made at the subsystem level can be evaluated at the systems level. The tradespace provides the design team with an overall picture of the conceptual spacecraft design such that designers can identify areas for increasing safety and reducing risk while maintaining requirements for mass. Although this approach is focused on early conceptual development, the process can be utilized as the design matures into preliminary design.

Building upon the Minimum Functionality design methodology presented in Chapter 3, this chapter fully explores the conceptual design tradespace in order to explain how *Safety* and *Operability* configurations in the design space can be mapped to the objective tradespace. The minimum functionality design approach has been successfully utilized by the Altair Lunar Lander in the LDAC cycles and the Orion program before System Design Review. However, the previous approaches took many months of investigation and focused primarily on optimizing single design configurations.

The novelty of this process is the methods used to characterize spacecraft conceptual design configurations in terms of *Safety* and *Operability.* The advantage of this approach is that a tradespace of potential design configurations can be developed quickly such that design teams can focus on the most critical elements of the spacecraft. This process also structures the design process in a way that requires designers to focus on the lowest levels of the spacecraft in order to reduce and quantify uncertainty associated with component mass. The philosophy of this approach is different from traditional single design optimization methods because it trades the impacts of *Safety* and *Operability* at an earlier stage in the conceptual design phase.

5.3 BACKGROUND

According to NASA NPR 8705.2B, Human Rating Requirements for Space Systems, a human rated system is described as one that:

> *"Accommodates human needs, effectively utilizes human capabilities, controls hazards with sufficient certainty to be considered safe for human operations, and provides to the maximum possible extent practical, the capability to safety recover the crew from hazardous situations"* (NASA, 2009).

In summary, human rating a spacecraft is the process of developing and maintaining a spacecraft with a focus on the safety of the crew and mission success. The NASA procedural requirements in NPR 8705.2B are applicable to all stages of the spacecraft life cycle process. The reduction of risk and uncertainty is key driver for increasing safety and mission success and continues throughout flight operations. The technical requirements for human rating are specified in Chapter 3 of NASA NPR 8705.2B. In paragraph 3.2.2, failure tolerance in systems is described as:

> *"The space system shall provide failure tolerance to catastrophic events (minimum of one fault tolerant), with the specific level of failure tolerance (one, two, or more) and implementation (similar or dissimilar redundancy) derived from an integrated design and safety analysis. Failure of primary structure, structural failure of pressure vessel walls, and failure of pressurized lines are excepted from the failure tolerance requirement provided the potentially catastrophic failures are controlled through a defined process in which approved standards and margins are implemented that account for the absence of failure tolerance"* (NASA, 2009).

The need to design failure tolerance into the system is a defining requirement for human spacecraft in order to mitigate potential hazards and contingencies. However, the rationale for this requirement also points to the need to choose how and where failure tolerance should be applied. Failure tolerance is a frequently used term to identify minimum acceptable redundancy, but it may also be used to describe the number of similar or dissimilar systems, cross strapping or functional interrelationships that maintain minimum acceptable performance despite failures or additional features that mitigate the failure (NASA, 2009). In addition, reliability is the key driver in the determination of safety and redundancy; without sufficient reliability the spacecraft does not meet the intent of the requirement. Failure tolerance in the spacecraft is intended as a first line of defense against potential hazards. Safety equipment such as fire extinguishers, launch and entry pressure suits, and launch abort systems are not considered part of the failure tolerance requirement (NASA, 2009). The NASA human rating technical requirements provide the overall guidelines for defining what is considered failure tolerance in a human spacecraft, but the decision of applying failure tolerance in the system is dependent upon the designers and decision makers.

5.3.1 NASA Best Practices Guidelines

In the NASA document, Design, Development, Test, and Evaluation (DDT&E) Considerations for Safe and Reliable Human Rated Spacecraft Systems a set of guidelines for how to approach the development of a system using a risk based design philosophy are presented. In these guidelines, a "three pronged" approach for assessing safety and reliability is used to *"specify safety and reliability requirements through a triad of fault tolerance, bounding failure probability and adhering to proven practices and standards"* (Miller *et al*., 2008). The authors also caution that levying a two fault tolerant requirement at lower levels may introduce system complexities that may be inappropriate since migration can occur at higher levels where the systems interact. These guidelines also specify steps for conceiving the right system by *"thoroughly exploring risks from the top down and using a risk based design loop to iterate the operations concept, design, and requirements until the system meets minimum mission objectives at minimum complexity and is achievable within constraints"* (Miller et al., 2008).

The steps for conceiving a system begin with a minimum set of functions necessary to achieve the mission objectives and a simple conceptual design is developed. Elements are added to the simple system that may reduce performance but increase reliability to meet safety needs. The additional "legs" add to system fault tolerance. Utilizing this information, Functional Failure Mode and Effects Analysis and Fault Tree Analysis along with an integrating technique such as Event Sequence diagrams are used to identify risk drivers (Miller *et al*., 2008). The approach described by Miller *et al*. (2008) utilizes a single point design and adds functionality in a sequential fashion by evaluating the risk and safety impacts to the design. Cost impacts are included in the design decisions because the early activities can commit over 50% of the total project costs (Miller *et al*., 2008). Miller *et al*. (2008) also point out that while safety is the primary goal; the designers should "*make the design work first, make it safe and reliable, and then assure it is affordable*."

5.3.2 Altair and Orion Risk Approaches

The Altair Lunar Lander program office was the first to utilize a minimum functionality approach for conceptual design. After the minimum functional conceptual design had been developed in the first Lunar Design and Analysis Cycle LDAC-1, the following design cycle LDAC-2 added mass to the configuration in order to reduce LOC risk. The third design cycle LDAC-3 utilized the previous design configuration to reduce LOM risk and address global access capability (Dorris, 2008). The building block approach of Altair occurred over several months and two design cycles (Dorris, 2008).

In addition to the methods utilized by Altair for buying down risk through mass additions, the Orion program utilized a method called "Risk Balancing" in order to provide an assessment of the safety and reliability of the minimum functional vehicle (Hu *et al*., 2008). The Orion spacecraft was revised due to mass concerns before System Design Review.

5.3.3 Apollo Reliability Analysis

The Apollo program utilized Failure Modes and Effects Analysis to determine possible modes of failure and the effects of failures on mission objectives and crew safety. The areas of potential risk such as single point failures were studied and solutions identified. Where fault tolerance or backups to functions was not practical, extensive test programs were conducted for qualification and flight certification together with rigorous configuration control and quality assurance to minimize the risk (Sperber, 1973). All single point failures were not completely eliminated. An example of a single point risk was the Lunar Ascent Module main engine. If this engine failed to operate, the astronauts would be stranded on the surface of the moon. The focus of the Apollo reliability process was to drive down risk as much as possible.

5.3.4 Previous Risk Approaches and Operability

Throughout the development of human spacecraft, the process for driving down risk and increasing safety in the vehicle is one that is iterated between design teams and reliability experts. The methods used for "buying down" risk such as observed with the Altair Lunar Lander are one approach for balancing the safety requirements of the spacecraft against the mass requirements. In all of the approaches, the level of *Operability* in the spacecraft is dependent upon first achieving the safety and mass objectives. The NASA DDT&E guidelines, (Miller *et al*., 2008) point to accepting reduced performance in order to meet the safety requirements. The balancing of all the competing objectives in a spacecraft design is one that is challenging for designers and decision makers. A comprehensive view of all competing objectives is necessary to make informed decisions.

5.3.5 Conceptual Design Variables

There are many competing variables and parameters in the design of a human spacecraft. The challenge that designers face are how to best balance all of the competing objectives. In many cases, time does not permit a detailed examination of all possible solutions. During the conceptual design of human spacecraft, many variables must be balanced simultaneously and this is part of the challenge of driving out uncertainty when so many variables are coupled. Shown in Figure 45 is an illustration of the many variables and parameters that must be balanced in the conceptual design process.

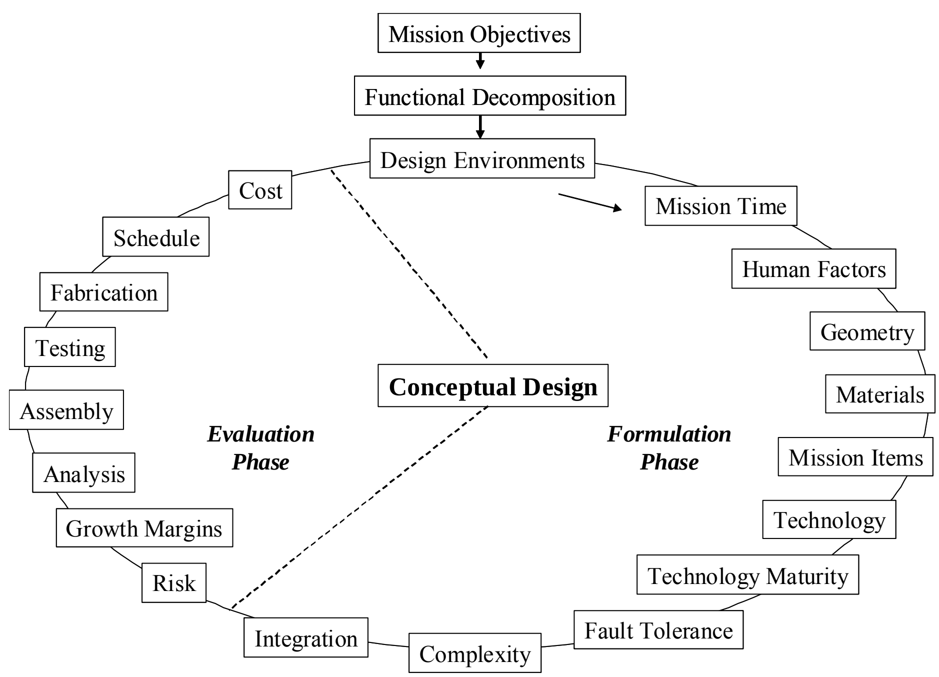


Figure 45: Conceptual design variables and parameters.

As shown in Figure 45, the conceptual design process consists of two parts, the formulation phase and the evaluation phase. In the formulation phase of developing concepts, ideas are generated for investigation in the evaluation phase. The formulation of a conceptual design begins with the design environments and progresses through the other variables that will define the configuration of the proposed solution. The evaluation phase is where candidate solutions are quantified and judged for potential issues. For example, a proposed concept will be evaluated to determine levels of risk, potential growth margins, and analyses to be performed, how the assembly fits together as a whole, the amount of testing required, issues with fabrication and potential schedule and cost. The figure does not contain every possible area that is evaluated in conceptual design; but serves to illustrate the highly coupled nature of the design process. As concepts are developed, the conceptual design process is iterated to determine the most feasible solutions. In the early stages, many variables are unknown or must be derived from examination, iteration, or heuristics. The challenge in conceptual design is to start with all of the unknowns and develop the concept into a feasible solution for further investigation in preliminary design.

5.4 MASS ADDITION PROCESS FOR EVALUATING SAFETY AND OPERABILITY

The mass addition process described in this work was developed to assist designers and decision makers with making informed decisions about conceptual spacecraft design configurations. Building upon the fundamental approach of adding mass beyond a minimum functional baseline as proposed by Altair and Orion, this process expands upon the previous approaches to develop a systematic method for investigating mass additions in order to provide an overall view of the spacecraft objective tradespace. The mass addition process is based upon a Minimum Functionality design methodology and is focused on providing additional *Safety* and *Operability* in the spacecraft functions. The most challenging part of spacecraft design is balancing all of the competing mass additions and considering alternatives or trades that will enable the spacecraft to meet the requirements for mass, safety, and performance. The advantage of using this process as an early design tool is that more knowledge is gained by evaluating the potential spacecraft configurations earlier in the conceptual design process.

5.4.1 Mass Addition Process

The mass addition process starts with the list of functions of a minimum functionality baseline configuration, the functions are mapped to candidate technology choices and criticalities are assigned to the functions. The criticalities form the basis for evaluating the mass additions and should be carefully peer reviewed by the design teams and the risk analysis experts. Shown in Figure 46 is a flow diagram of the mass addition process used for tradespace exploration.

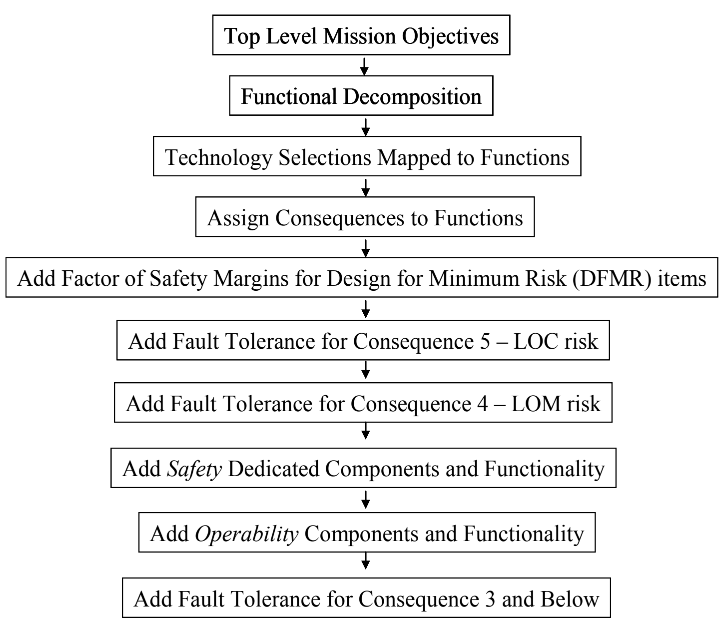


Figure 46: Minimum Functionality mass addition process overview.

5.4.2 Functional Decomposition

Using the methods presented in Chapter 3 for decomposing top level mission objectives of the spacecraft into lower level functions, the mass addition process starts with a list of functions for a minimum functionality baseline configuration. These functions are typically associated with a particular subsystem. Shown in Figure 47 is a decomposition of subsystems and associated high level functions.

Human Spacecraft Subsystems

Avionics	**Crew Accommodations**	**Payloads**	**Environmental Control and Life Support System (ECLSS)**	**Power**	**Thermal**	**Structures**	**Propulsion**	**EVA**
Provide Command and Data Handling	Support Human Factors	Return Science Payload	Maintain Atmosphere	Store Power	Collect Heat	Provide Load Bearing Capability	Provide Main Propulsion	Environmental Protection
Provide Navigation	Provide Lighting	Store Tools and Equipment	Provide Food and Water	Distribute Power	Transport Heat	Provide Environmental Protection	Provide Flight Control	Allow External Mobility
Provide Guidance and Control	Provide Restraints and Handholds	Store Photography	Manage Waste	Regulate Power	Remove Heat	Support Subsystems		
Store Spacecraft Data	Maintain Health			Provide Grounding		Provide External Interfaces		
Monitor Vehicle Subsystems	Store Operational Supplies			Provide Overload Protection		Provide Direct External Observation		
Provide Communications				Provide Shielding		Support TPS and MMOD		

Figure 47: Human spacecraft subsystems and high level functions.

The higher level functions shown in Figure 46 are further decomposed into lower level functions and mapped to potential technologies. Shown in Table 5 are the functions of the Avionics subsystem mapped to candidate technologies.

Table 5: Avionics subsystem functions mapped to technology choices.

Avionics		
High Level Function	**Low Level Function**	**Technologies**
Provide Command and Data Handling	Sense Subsystem Commands	Data bus network boxes
Provide Command and Data Handling	Process/Amplify Subsystem Commands	Master event controllers
Provide Command and Data Handling	Send Commands to Subsystems	Sensor communication wiring
Provide Navigation	Sense Spacecraft Position	IMU
Provide Navigation	Sense Spacecraft Position	Navigation base
Provide Navigation	Sense Spacecraft Position	Navigation power
Provide Navigation	Sense Spacecraft Position	Star tracker
Provide Navigation	Convert Analog to Digital Navigation Inputs	Navigation analog to digital converter
Provide Navigation	Output Navigation to Guidance	Multiplexer demultiplexers
Provide Navigation	Calculate Spacecraft Guidance	Flight control computer
Provide Navigation	Communicate with Instrumentation	Instrumentation wiring
Provide Navigation	Display Spacecraft Navigation	Crew displays navigation
Provide Guidance and Control	Input Human Flight Controls	Flight joystick
Provide Guidance and Control	Input Human Flight Controls	Flight translation joystick
Provide Guidance and Control	Input Human Navigation to Computer	Computer keyboard
Provide Guidance and Control	Output Spacecraft Control to Propulsion	Control propulsion
Provide Guidance and Control	Display Spacecraft Control	Crew displays control
Provide Guidance and Control	Sense Spacecraft Abort Position	Abort navigation
Provide Guidance and Control	Sense Spacecraft Abort Velocity Inputs	Abort control
Provide Guidance and Control	Calculate Spacecraft Abort Trajectory	Abort guidance
Provide Guidance and Control	Input Abort Commands	Abort input
Provide Guidance and Control	Provide Rendezvous Guidance	Rendezvous radar
Store Spacecraft Data	Store Spacecraft Data	Data storage
Monitor Vehicle Subsystems	Monitor Subsystem Data	Health monitoring computer
Monitor Vehicle Subsystems	Monitor ECS Subsystems	Instrumentation sensors ECLSS
Monitor Vehicle Subsystems	Monitor Prop Subsystems	Instrumentation sensors Propulsion
Monitor Vehicle Subsystems	Monitor Prop Subsystems	Instrumentation sensors RCS
Monitor Vehicle Subsystems	Monitor Crew Accommodations	Instrumentation sensors CA
Monitor Vehicle Subsystems	Monitor Payload Subsystem	Instrumentation sensors Payload
Monitor Vehicle Subsystems	Monitor Power Subsystem	Instrumentation sensors Power
Monitor Vehicle Subsystems	Monitor Communication Subsystem	Instrumentation sensors Communication
Monitor Vehicle Subsystems	Monitor Command and Data Handling (C&DH)	Instrumentation sensors CDH
Monitor Vehicle Subsystems	Monitor Health Monitoring Subsystem	Instrumentation sensors Health
Monitor Vehicle Subsystems	Monitor Flight Control Subsystem	Instrumentation sensors Flight
Monitor Vehicle Subsystems	Monitor Thermal Subsystem	Instrumentation sensors Thermal
Monitor Vehicle Subsystems	Monitor Structures Subsystem	Instrumentation sensors Structural
Monitor Vehicle Subsystems	Manual Control Spacecraft Subsystems	Crew displays subsystems
Provide Communications	Communicate with Earth ground station	Long range transceiver
Provide Communications	Communicate with Earth ground station	Long range amplifier
Provide Communications	Communicate with Earth ground station	Long range antenna
Provide Communications	Communicate with Earth ground station	Long range steerable antenna
Provide Communications	Communicate with Earth ground station	Long range data processor
Provide Communications	Communicate with relay satellites	Short range transceiver
Provide Communications	Communicate with relay satellites	Short range antenna
Provide Communications	Communicate with relay satellites	Short range data processor
Provide Communications	Communicate between suited crewmembers	Interior voice communication

Listed in Table 5, the Avionics low level functions are derived from the high level functions in Figure 47. Within the low level functions, one or more technologies may be used to perform the function. A detailed listing of all the high level functions mapped to low level functions for the other subsystems is listed in Appendix C.

5.4.3 Assigning Consequences to Functions

If a reliability approach is not used to quantify risk and safety, a qualitative method is used to judge the levels of risk. A typical qualitative risk approach matrix is the combination of consequence of an event (or failure) occurring and the likelihood of the event. Based on the levels of consequence and likelihood, a risk score is assigned (NASA, 2007a). For the process described in this work, the low level functions are assigned a consequence number based on the severity of the loss of the function. In the conceptual design phase when the exact configuration and integration has yet to be determined, a functional approach is one method used to identify components that contain the highest risk. The definition of consequence is sometimes defined differently by many organizations and the definitions should be clearly defined (NASA, 2007a). The typical scale for consequence is from 1 to 5 with 5 being the highest in severity of the event such as Loss of Crew. Similarly for likelihood, the typical scales are from 1 to 5 with 5 being the most likely that an event will occur. The scale of consequences used in this process was derived from the NASA Constellation program risk guidelines.

- Consequence 5 – Loss of life or permanently disabling injury and top level requirements not achievable with existing engineering capabilities; Loss of Crew.
- Consequence 4 – Severe injury or illness requiring extended medical treatment or major impact to requirements, design margins or loss of mission objectives; Loss of Mission.

- Consequence 3 – Injury, illness, or incapacitation requiring emergency treatment or moderate impact to requirements, design margins or mission objectives.
- Consequence 2 – Injury requiring first aid treatment, moderate crew discomfort or minor impact to requirements, design margins or mission objectives.
- Consequence 1 – Minor injury not requiring first aid treatment, minor crew discomfort or negligible impact to requirements, design margins or mission objectives.

Using the consequences scale, all of the low level functions in the spacecraft are mapped to a consequence based on the severity of the loss of the function. In the example of a lunar ascent module, the loss of the main engine is considered a consequence 5 because it is a single point failure for transporting the crew to Low Lunar Orbit.

5.4.4 Adding Failure Tolerance and Factor of Safety Margins

The addition of failure tolerance to mitigate the likelihood of a failure is required by NASA for a human rated spacecraft. The specific methods used to add failure tolerance include similar or dissimilar redundancy, cross strapping or functional interrelationships. In the early conceptual design phase, the operational aspects of mitigating failures is not always known or fully defined and the design teams are challenged with designing failure tolerance into the system. The easiest approach is to consider failure (or fault) tolerance across all the components in the system. However, this is not usually the correct method of adding failure tolerance and can lead to an over designed system. In order to understand where failure tolerance should be added, the functions are evaluated according to their consequence rating in order to reduce the likelihood of a failure. In the mass addition process described in this work, the low level functions associated with consequence 5 and 4 are investigated for increasing failure tolerance from a minimum functional baseline of 0 Failure Tolerance (FT) to 1 FT. The components in each function are evaluated based on the likelihood of failure and assigned similar or dissimilar redundancy. Cross strapping and functional interrelationships are not considered in this mass addition process because of the immaturity in the design operations concepts. The functions are assumed to be independent and common cause failure effects are not considered. The effects of common cause failure are best left to Probabilistic Reliability Assessments after the design has reached a level of maturity.

In some components of the spacecraft, a failure tolerance strategy for mitigating failures is not feasible because of mass and volume constraints. NASA defines structures and pressure vessels in this category where the components are design according to accepted standards and margins. The typical method used by designers in the early stages of design is to assign factors of safety for the loading on the structure or pressure vessels. A common acronym used to describe the failure tolerance strategy of these components is Design for Minimum Risk (DFMR).

5.4.5 Adding Safety and Operability Functions

The functions associated with *Safety* and *Operability* are considered bonus functions beyond the minimum functionality baseline configuration. As mentioned earlier, NASA does not consider additional safety functionality to contribute to failure tolerance unless the additional safety mitigates the potential hazard as a first line of defense. In this context, *Safety* functions are used to mitigate the effects of a hazard and safely protect the crew. The most common used example of *Safety* is a launch abort system. The launch abort system is not considered in the failure tolerance strategy unless it prevents the occurrence of the failure (NASA, 2009).

The functions associated with *Operability* are functions that allow the crew to perform specific aspects of the mission or to make the duties more efficient. A loss of *Operability* in this process is not associated with Loss of Mission because the minimum functionality design approach assumes that the primary objective of the mission is transporting crew safely. However, depending upon the top level objectives, it can be argued that a loss of *Operability* will affect Loss of Mission because of the loss of mission objectives. This debate is best left to the design teams to decide the importance of *Operability* in the context of the mission objectives. The success or failure of mission objectives will be covered in later reliability assessments and will be addressed as the design matures.

5.4.6 Tradespace of Total Spacecraft, Safety, and Operability Mass

The tradespace of Total Spacecraft, *Safety*, and *Operability* mass was introduced in Chapter 3. The mass addition process described in this chapter explores the tradespace in greater detail to provide an overall view of possible spacecraft configurations based on varying levels of failure tolerance, *Safety* functions, and *Operability* functions. It was shown earlier that the relationship between *Safety*, *Operability*, and Total Spacecraft mass form a 3-D surface that is approximately planer. The use of a plane as a metamodel could be easily developed by designers for calculating total spacecraft mass based on given levels of *Safety* and *Operability* mass. Shown in Figure 48 is the tradespace of Total Spacecraft, *Safety*, and *Operability* mass.

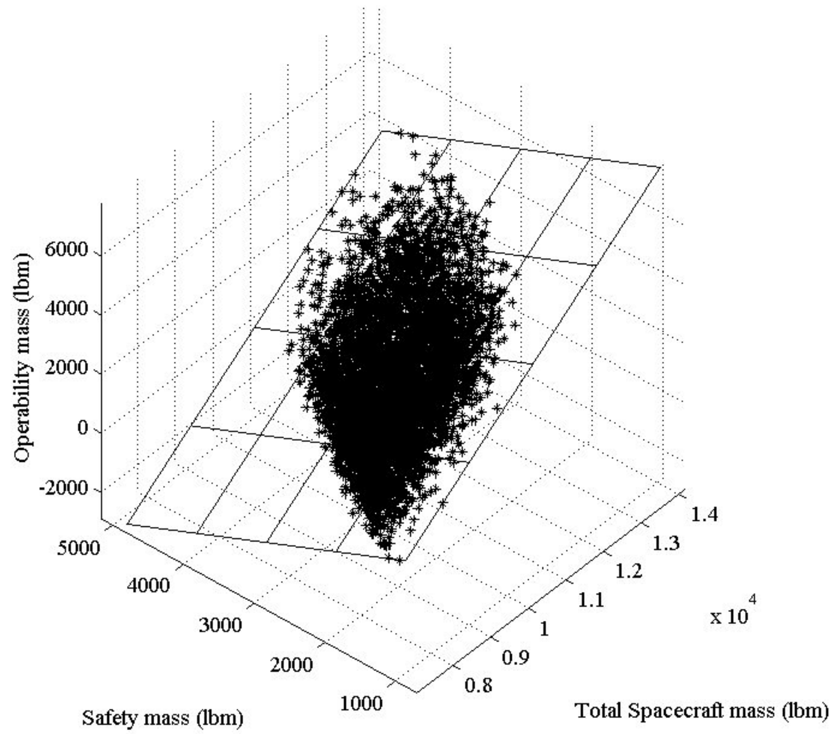


Figure 48: Tradespace of Total Spacecraft, *Safety*, and *Operability* mass.

5.4.7 Developing Nodes in the Tradespace Domain

In order to understand how the design space input variables map to the objective tradespace, a series of "nodes" within the tradespace are used to provide information about the level of *Safety* and *Operability* in the spacecraft designs. Each point in Figure 48 represents a specific spacecraft configuration and in the earlier analysis, the configurations were varied according to levels of fault tolerance, *Safety* components and *Operability* components in order to derive an overall view of the tradespace. Although the earlier analysis in Chapter 3 demonstrated the possibilities of the tradespace, the methods used to generate the points in the tradespace needed to be refined in order to determine regions that mapped to the design input variables.

A set of 40 nodes at various spacecraft configurations of *Safety* and *Operability* is used to create a grid in the tradespace domain. Each node is a different spacecraft configuration with a different level of fault tolerance, *Safety*, and *Operability*. A set of 10 *Safety* levels and 4 *Operability* levels are used in the 10×4 matrix of nodes. Listed in Table 6 are the *Safety* levels and listed in Table 7 are the *Operability* levels.

Table 6: *Safety* levels in mass addition process.

Safety Level	
Level	**Mass Addition**
1	Minimum Functionality
2	Factor of Safety Addition
3	Consequence 5 Failure Tolerance (1FT) Addition
4	Consequence 4 Failure Tolerance (1FT) Addition
5	Safety Functionality Addition
6	Consequence 3 and Below (1FT) Addition
7	Consequence 5 Failure Tolerance (2FT) Addition
8	Consequence 4 Failure Tolerance (2FT) Addition
9	Safety Functionality Addition
10	Consequence 3 and Below (2FT) Addition

Table 7: Operability levels in mass addition process.

Operability Level	
Level	**Mass Addition**
1	Minimum Functionality
2	Mission Time Addition
3	Operability Components and Functions Addition
4	Double Operability and Mission Time Addition

As shown in Table 6, the *Safety* levels are varying according to the mass addition of components to the Minimum Functional baseline configuration. An important note to clarify in this process is that each level builds upon the previous one. A *Safety* level of 3 contains the Minimum Functionality baseline with Factors of Safety and functions designated as consequence 5 with one additional component for failure tolerance. As the levels increase, the *Safety* mass increases. The same approach is used in Table 7 where *Operability* levels increase with additional *Operability* mass. The divisions between the levels were chosen to provide the design team visibility as to how the addition of failure tolerance and *Operability* corresponded to total spacecraft mass. The levels shown here are for demonstration of the process and to provide an example of how the tradespace can be mapped for greater understanding of spacecraft configurations. The 10 levels of Safety and the 4 levels of Operability are combined in a 10 × 4 matrix for the individual nodes in the tradespace. Shown in Table 8 is a map of the nodes used to define the grid in the tradespace.

Table 8: Node map for tradespace.

		Operability Level			
	Node Number	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition
Safety Level	Minimum Functionality	1	11	21	31
	Factor of Safety Addition	2	12	22	32
	Consequence 5 Failure Tolerance (1FT) Addition	3	13	23	33
	Consequence 4 Failure Tolerance (1FT) Addition	4	14	24	34
	Safety Functionality Addition	5	15	25	35
	Consequence 3 and Lower (1FT) Addition	6	16	26	36
	Consequence 5 Failure Tolerance (2FT) Addition	7	17	27	37
	Consequence 4 Failure Tolerance (2FT) Addition	8	18	28	38
	Safety Functionality Addition	9	19	29	39
	Consequence 3 and Lower (2FT) Addition	10	20	30	40

A spacecraft configuration is calculated for each of the individual nodes and plotted in the tradespace of Total Spacecraft, *Safety*, and *Operability* mass. Shown in Figure 49 is the tradespace domain with the node configurations.

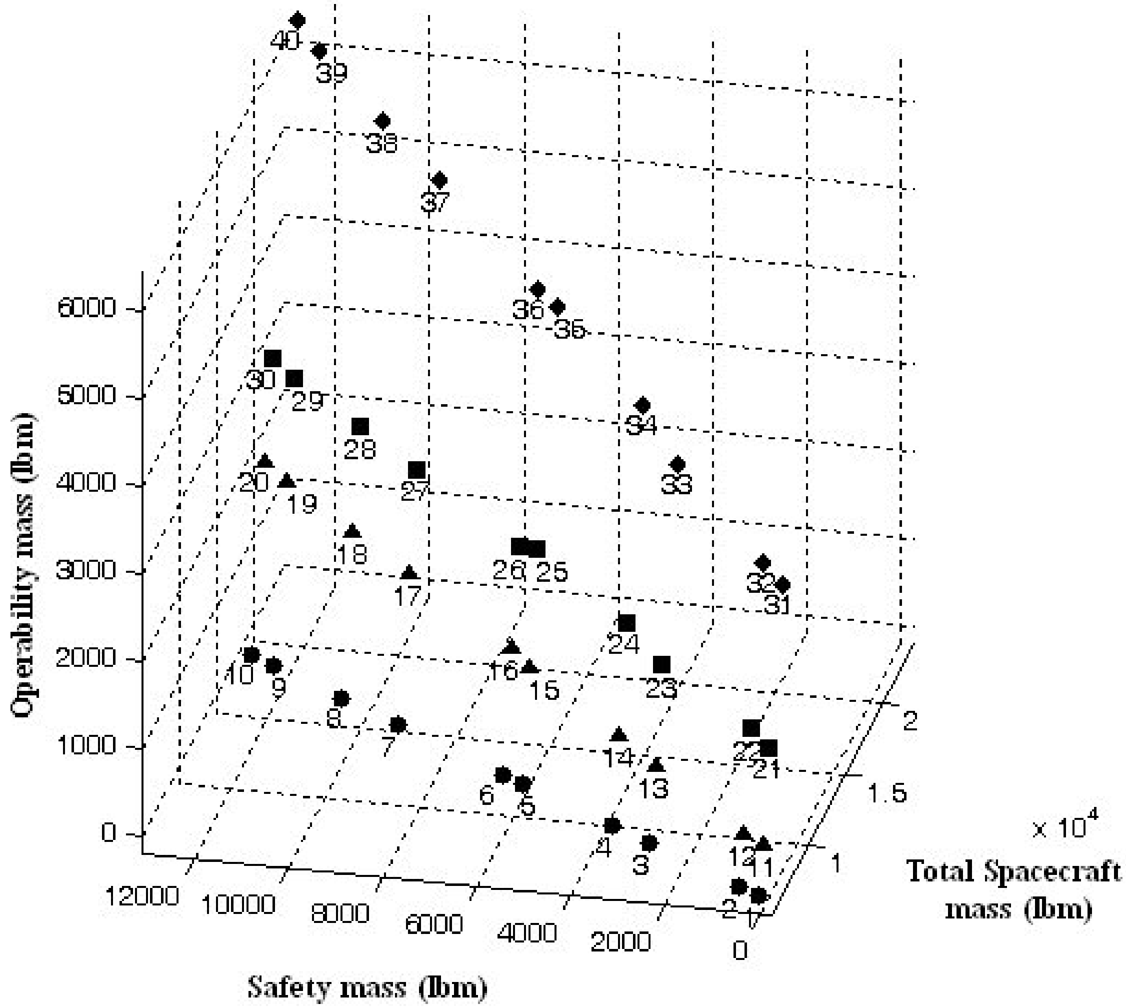


Figure 49: Node locations in the tradespace domain.

The node locations in the tradespace domain identify zones for various *Safety* and *Operability* levels; where, x = Total Spacecraft mass (lbm), y = *Safety* mass (lbm), and z = *Operability* mass (lbm). The nodes lie on a plane within the domain and provide a means for developing a grid that maps the input design configurations to the tradespace domain. The differences between the nodes allow the user to distinguish between various levels. Projecting the nodes on the *Safety* vs. *Operability* mass plane and creating Iso lines of total spacecraft mass presents a simple method of viewing the entire tradespace and understanding the various impacts of mass additions. Shown in Figure 50 is the *Safety* and *Operability* mass addition chart.

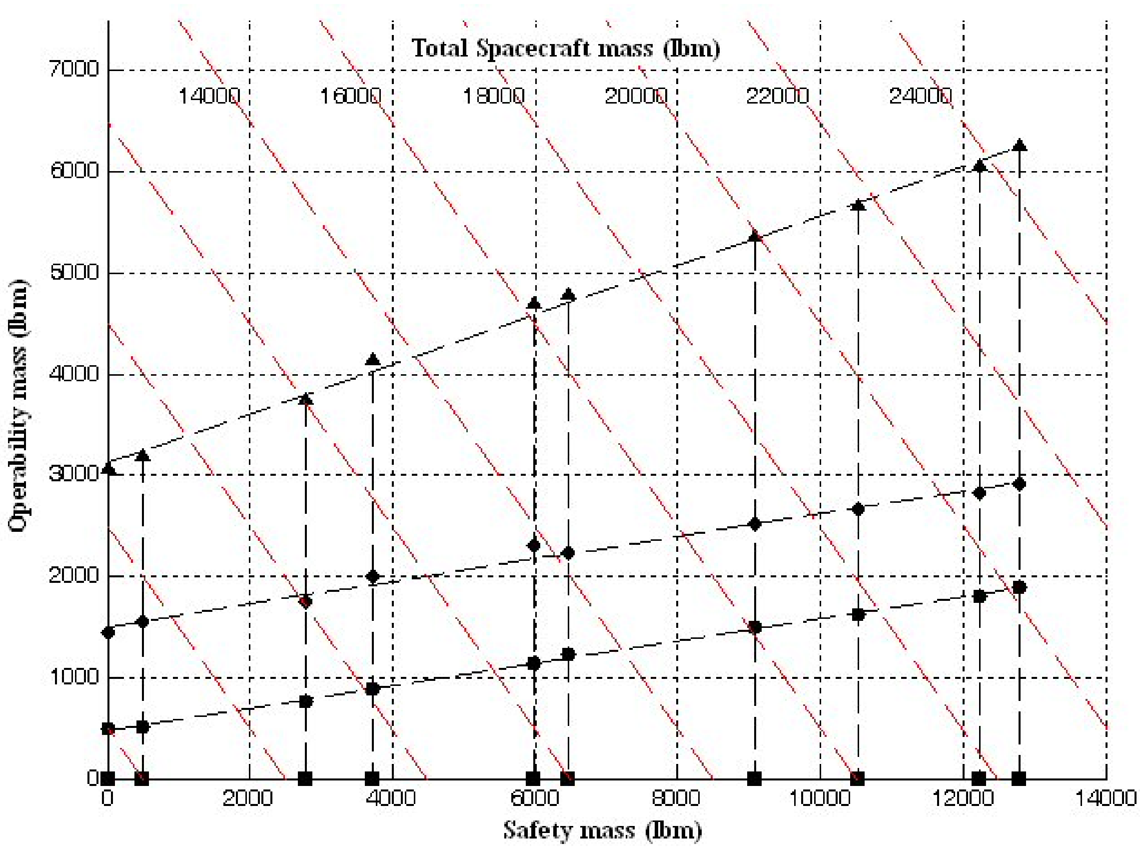


Figure 50: *Safety* and *Operability* mass addition chart.

The *Safety* and *Operability* mass addition chart provides a grid of the tradespace domain such that a user can determine the levels of *Safety* and *Operability* in relation to the total spacecraft mass. The origin is the Minimum Functionality baseline configuration and *Safety* and *Operability* mass is added to determine total spacecraft mass. Within the nodes, a grid has been developed to identify zones of *Safety* and *Operability*. The lines of *Operability* levels are best fit linear regression lines between the points. The chart provides a complete view of all the potential spacecraft configurations from a Minimum Functional baseline to a fully two failure tolerant spacecraft configuration with the maximum expected level of *Operability*.

5.4.8 Conceptual Lunar Ascent Module Program

The previous examples of the mass addition process were derived using the Conceptual Lunar Ascent Module Program. This program was developed in MATLAB to calculate the minimum functionality baseline and to assess relative increases in *Safety* and *Operability* mass for various Lunar Ascent Module spacecraft configurations. The program played a key role in the development of the mass addition process. Shown in Figure 51 is a flowchart of the CLAMP subroutines. A detailed summary of the CLAMP program logic is provided in Appendix D.

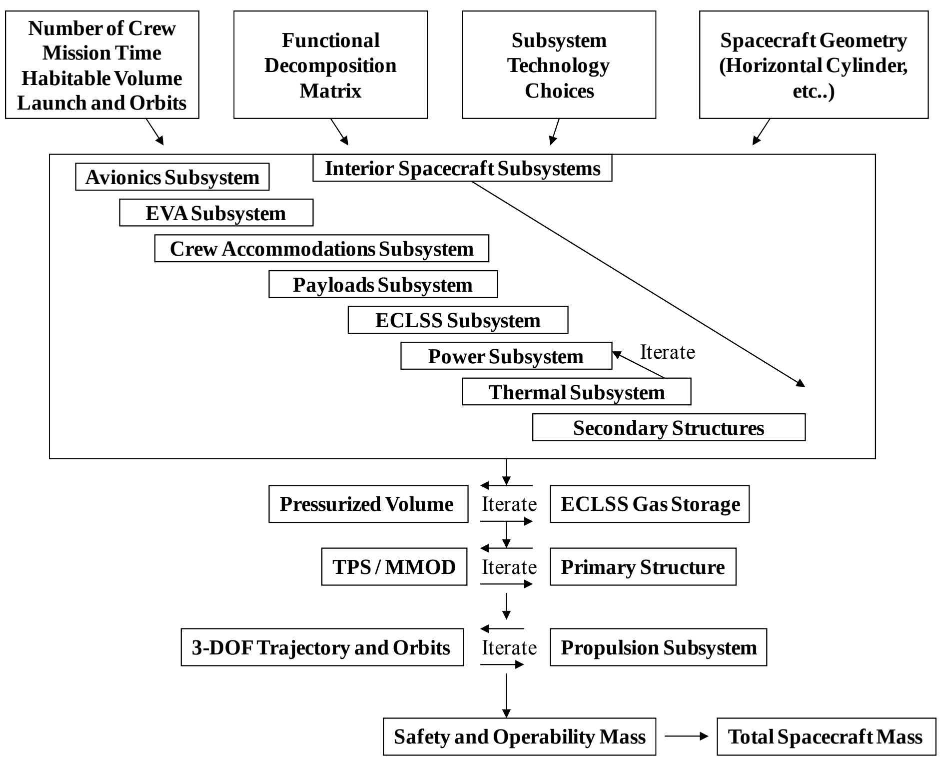


Figure 51: Flowchart of CLAMP subroutines.

5.4.9 Mass Addition Process Summary

The process used for adding *Safety* and *Operability* mass beyond a Minimum Functional baseline configuration is based on varying the levels of failure tolerance and *Safety* and *Operability* components. The process is structured such that functions are decomposed into lower levels where technologies are assigned to conduct the function. The functions are assigned Consequence values based on the severity of the loss of the function. The Consequence levels are based on industry and government approaches for qualitatively evaluating risk. The mass addition process identifies the highest priority Consequence functions for increasing fault tolerance through similar or dissimilar redundancy and steps through the spacecraft functions to create spacecraft configurations at various levels.

Increased functionality due to the addition of *Safety* or *Operability* components does not contribute to the failure tolerance levels. The mass addition process steps through the potential mass additions in *Safety* or *Operability* to develop a complete view of the tradespace domain of likely spacecraft configurations using a set of predetermined nodes. The nodes are derived from a combination of *Safety* and *Operability* levels and combined into a matrix for spacecraft configuration analysis. The final result of the process is a *Safety* and *Operability* mass addition chart that provides designers with a complete view of the tradespace. The chart is intended to assist designers and decision makers with information such that decisions can be made for revising functions and the conceptual design. The process is not intended to be a single pass recipe for quantifying fault tolerance. It is meant as an early design tool that can provide guidance about the conceptual design as concepts are investigated and matured.

5.5 ANALYSIS STUDIES

A series of analysis studies was conducted to verify the mass addition process and explore the tradespace domain. Three configurations of Lunar Ascent module designs were used in the studies:

- Apollo 15 Lunar Ascent Module configuration:
    - Baseline mass of 10,927 lbm at lunar ascent,
    - Pressurized volume of 235 ft^3,
    - 2 Crewmembers,
    - Sample return mass of 255 lbm, and
    - Mission time of 12.36 hours.
- Apollo One Man Lunar Ascent Module configuration:
    - Baseline mass of 8,068 lbm at lunar ascent,
    - Pressurized volume of 117 ft^3,
    - 1 Crewmember,
    - ½ sample return mass of 127.5 lbm, and
    - Mission time of 12.36 hours,
- ESAS Lunar Ascent Module configuration
    - Baseline mass of 22,534 lbm at lunar ascent,
    - Pressurized volume of 1136 ft^3,
    - 4 Crewmembers,
    - No sample return, and
    - Mission time of 3 hours.

The analysis studies were focused on investigating the mass addition process and the resulting tradespace of the three Lunar Ascent Modules. Graphs of the mass addition process and tradespaces were developed to demonstrate the process. The intent was to provide designers with information that could be utilized for decision making.

5.5.1 Lunar Module Configurations

The Apollo Lunar Ascent Module main propulsion system utilized nitrogen tetraoxide (oxidizer) and an equal mixture of hydrazine and unsymmetrical dimethylhydrazine (fuel) at an oxidizer to fuel ratio of 1.6 to 1 (Humphries and Taylor, 1973). The Apollo One Man configuration utilized many of the same components as the Apollo baseline configuration with the exception of one crewmember, half the pressurized volume, and half the sample return capability. Other redundant components necessary for two crewmembers such as flight controls and displays were removed from the Apollo One Man configuration.

The ESAS Lunar Ascent Module configuration was derived from information listed in the ESAS report on page 166 for the reference point of departure design. The ESAS derived configuration utilized liquid oxygen (oxidizer) and methane (fuel) at an oxidizer to fuel ratio of 3.4 to 1 for the main propulsion system.

The main difference between the Apollo and ESAS configurations other than number of crewmembers, propulsion system, and pressurized volume is that Apollo utilized a cabin atmosphere of 100% Oxygen at 5 psia. The ESAS configuration used a two gas cabin atmosphere of 30% Oxygen at 9.5 psia. The makeup gas for the ESAS configurations was assumed to be Nitrogen.

Because of the limited amount of information related to subsystem components for the ESAS configuration, the Apollo component values were used to derive assumptions for the likely mass of the ESAS components. A listing of the ESAS components used in this analysis is provided in Appendix G.

The Apollo baseline configuration used in the studies was compared to the flight mass values for Apollo 15. The derived ESAS configuration was compared to the published subsystem estimated values in the ESAS report (NASA, 2005). Both configurations were verified within 95% confidence of the published mass values.

5.5.2 Mass Addition Process

The mass addition process was utilized for the three Lunar Ascent Modules to generate a domain of nodes that quantified *Safety* and *Operability* levels in the design. The use of three different configurations provided insight as to the mass increases due to the number of crewmembers, pressurized volume, and propulsion system. The mass addition process utilized in this analysis study assumed that all components dedicated to the functions increased with redundancy. Dissimilar redundancy was not considered in the analysis. A one failure tolerant system contained two similar components or strings to perform the function. *Safety* dedicated components not included in the minimum functional baseline were added during the mass addition process. These components were added after the Consequence 4 addition for both the one failure tolerant and two failure tolerance steps. A set of 40 nodes was used to map the *Safety* and *Operability* levels in the tradespace domain as described in Table 8. Shown in Figure 52 is the process used for mass additions in the analysis study

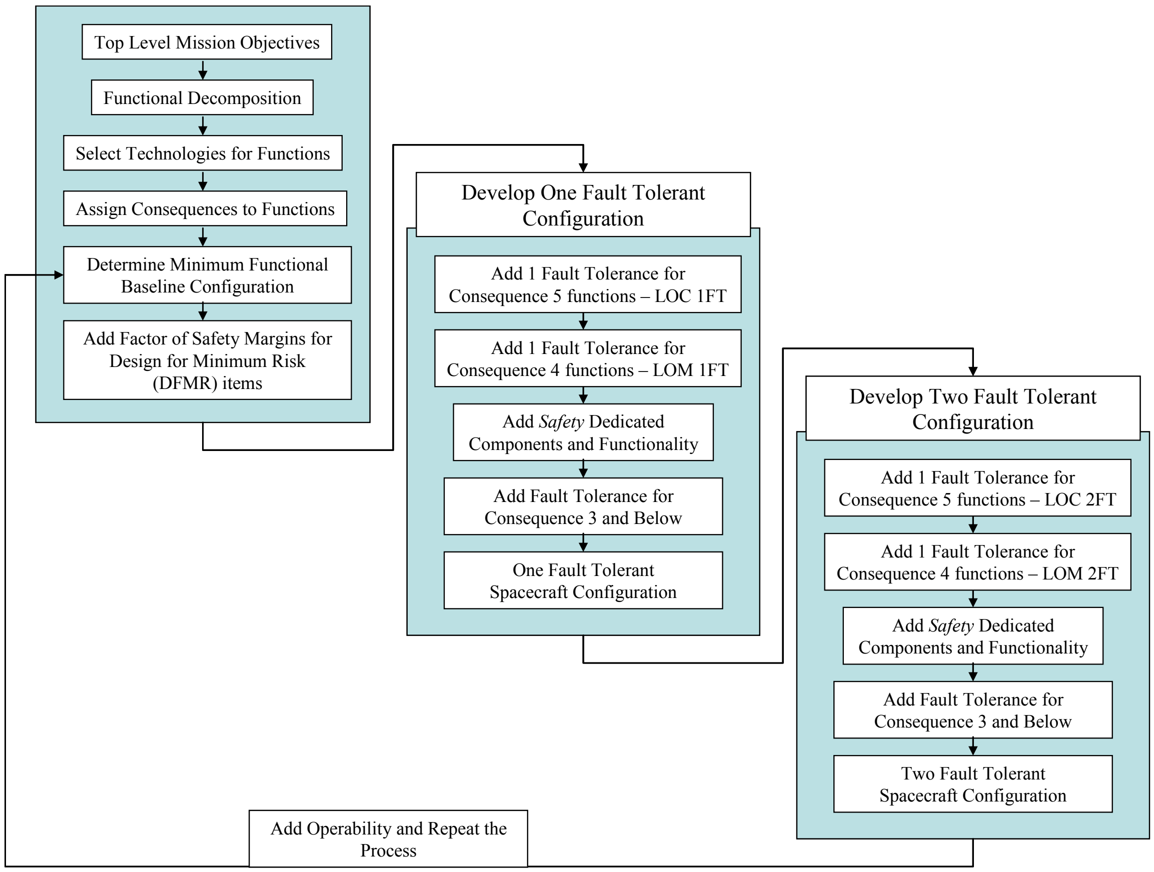


Figure 52: Mass addition process used in analysis studies.

The mass addition process starts with a Minimum Functional baseline configuration based upon the functional decomposition and technology selections. The first step in the process is to add Factors of Safety for DFMR items. The Factors of Safety are carried throughout the remaining steps of the process. The first grouping of mass additions is focused on adding mass in order to develop a one failure tolerant spacecraft configuration. The second grouping of mass additions is focused on the addition of components to develop a two failure tolerant spacecraft configuration. The process is repeated with the addition of *Operability* components. The most important functions in the configuration are assigned Consequence 5, LOC and 4, LOM and are listed in Table 9 - 13.

Table 9: Avionics subsystem functions with assigned consequences.

Function	Consequence
Avionics	
Sense Subsystem Commands	5
Process/Amplify Subsystem Commands	5
Send Commands to Subsystems	5
Monitor Prop Subsystems	5
Sense Spacecraft Position Nav Inputs	5
Convert Analog to Digital Nav Inputs	5
Output Navigation to Guidance	5
Calculate Spacecraft Guidance	5
Communicate with Instrumentation	5
Input Human Flight Controls	5
Input Human Navigation to Computer	5
Output Spacecraft Control to Propulsion	5
Display Spacecraft Control	5
Display Spacecraft Navigation	5
Communicate with Earth ground station	4
Communicate with relay satellites	4
Sense Spacecraft Position Nav Inputs	4
Manual Control Spacecraft Subsystems	4
Provide Rendezvous Guidance	4

Table 10: Power and Thermal subsystem functions with assigned consequences.

Function	Consequence
Power	
Provide Power	5
Distribute Power	5
Regulate Power	5
Thermal	
Collect Heat	4
Transport Heat	4
Remove Heat	4

Table 11: ECLSS subsystem functions with assigned consequences.

Function	Consequence
ECLSS	
Remove Carbon Dioxide	5
Provide Metabolic Oxygen	5
Store High Pressure Oxygen	5
Transport Oxygen	5
Control Oxygen Flow	5
Provide Cabin Air Control Logic	5
Store High Pressure Oxygen	4
Store High Pressure Makeup Gas	4
Transport Makeup Gas	4
Control Makeup Gas Flow	4
Circulate Air	4
Provide Air Pressure Sensor	4
Provide Cabin Air Control Logic	4
Provide Potable Water	4

Table 12: Structures subsystem functions with assigned consequences.

Function	Consequence
Structures	
Provide protection from vacuum	5
Provide MMOD/TPS protection	5
Provide Load Bearing Capability	5
Provide Forward Windows	5
Support Docking Mechanism for Lunar Ascent Module to CEV	4
Provide Ingress/Egress for Lunar Ascent Module to CEV	4
Provide Docking Mechanism for Lunar Habitat to Lunar Ascent Modu	4
Provide Docking Viewing Ports	4
Secondary Structures	
Support Internal Subsystems and Components	4

Table 13: Propulsion subsystem functions with assigned consequences.

Function	Consequence
Propulsion	
Control Main Engine Fuel	5
Control Main Engine Oxidizer	5
Control ME Pressurant - Fuel	5
Control ME Pressurant - Oxidizer	5
Control RCS Fuel	5
Control RCS Oxidizer	5
Control RCS Propellant	5
Provide Main Engine	5
Provide Main Engine Fuel	5
Provide Main Engine Oxidizer	5
Provide Main Engine Pressurant	5
Provide RCS Pressurant	5
Provide RCS Propellant	5
Provide RCS thrusters	5
Store Main Engine Propellant	5
Store RCS Propellant	5
Support Main Engine Components	5
Transport Main Engine Propellant	5
Transport RCS Pressurant	5
Transport RCS Propellant	5
Insulate Main Engine Propellant	4
Insulate RCS Propellant	4

A majority of the functions in the spacecraft were assigned either Consequence 4 or 5. This approach is likely a very conservative estimate and can be refined in later iterations. The purpose of assigning so many of the functions at a high level was to investigate the mass impacts of increasing redundancy. Shown in the results, the addition of redundancy is a significant driver of total spacecraft mass.

5.5.3 Tradespace Monte Carlo Analysis

Using the information generated in the mass addition process, seven tradespace zones were investigated. The zones represent various levels of *Safety* and *Operability* spacecraft configurations. For each zone, a Monte Carlo analysis of 2000 runs was completed. The points within the zones were used to develop a distribution of spacecraft configurations based on the levels of *Safety* or *Operability*. In each zone, a starting and ending configuration was used to vary the redundancy and addition of components. The seven tradespace zones are listed in Table 14. Shown in Figure 53 is a visual representation of the tradespace zones.

Table 14: Starting and ending configurations for tradespace zones.

Zone	Starting Configuration	Ending Configuration
1	Minimum Functionality at minimum mission time (Node 1)	Factor of Safety Addition at Nominal Mission Time (Node 12)
2	Factor of Safety addition at Nominal Mission Time (Node 12)	Consequence 5 (LOC) - 1 FT addition at Nominal Mission Time (Node 13)
3	Consequence 5 (LOC) - 1 FT addition at Nominal Mission Time (Node 13)	Consequence 4 (LOM) - 1 FT addition with Operability components at Nominal Mission Time (Node 24)
4	Consequence 4 (LOM) - 1 FT addition with Operability components at Nominal Mission Time (Node 24)	All components -1FT addition with Operability components at Nominal Mission Time (Node 26)
5	All components -1FT addition with Operability components at Nominal Mission Time (Node 26)	Consequence 5 (LOC) - 2 FT addition with Operability components at Nominal Mission Time (Node 27)
6	Consequence 5 (LOC) - 2 FT addition with Operability components at Nominal Mission Time (Node 27)	Consequence 4 (LOM) - 2 FT addition with Operability components at Nominal Mission Time (Node 28)
7	Consequence 4 (LOM) - 2 FT addition with Operability components at Nominal Mission Time (Node 28)	All components -2FT addition with double Operability components at double Nominal Mission Time (Node 26)

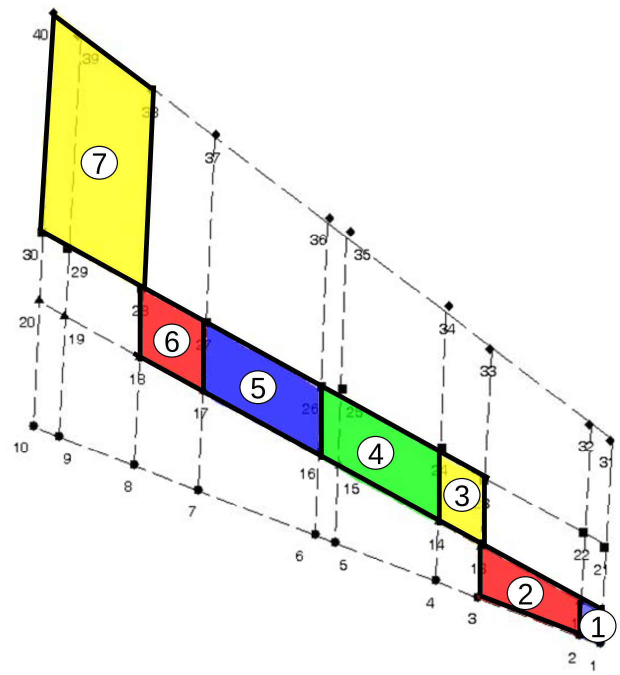


Figure 53: Tradespace zones in relationship to mass addition nodes.

The purpose of focusing on specific areas within the trad space is intended to simulate the steps for how mass would be added to the minimum functionality design. The most likely areas of a future spacecraft design are zones 4, 5, and 6. Focusing on these zones bounds the region of interest in the tradespace domain. The Total Spacecraft, *Safety* and *Operability* mass average and standard deviation were calculated from the distribution of points in each zone.

5.5.4 Safety vs. Operability Mass Addition Charts

Using the information generated in the mass addition process, a set of charts for the three Lunar Ascent Module configurations was developed. These charts represent the overall results of the analysis and provide a point of departure for future studies.

5.6 RESULTS

5.6.1 Apollo and ESAS Baseline Comparisons

The Apollo and ESAS baseline configurations used in the analysis were verified against published mass values. Using an uncertainty of +/- 5% with the component input values, a Monte Carlo analysis of 2000 runs was completed to generate a distribution of subsystem mass. The standard deviation of the subsystem mass was used to calculate confidence levels of 95%. Shown in Figure 54 are the 95% confidence levels compared to the Apollo nominal values. For all subsystems, the upper and lower limits bounded the nominal Apollo subsystem values. This verification provided confidence that the input values used in the analysis were matching the Apollo heritage values.

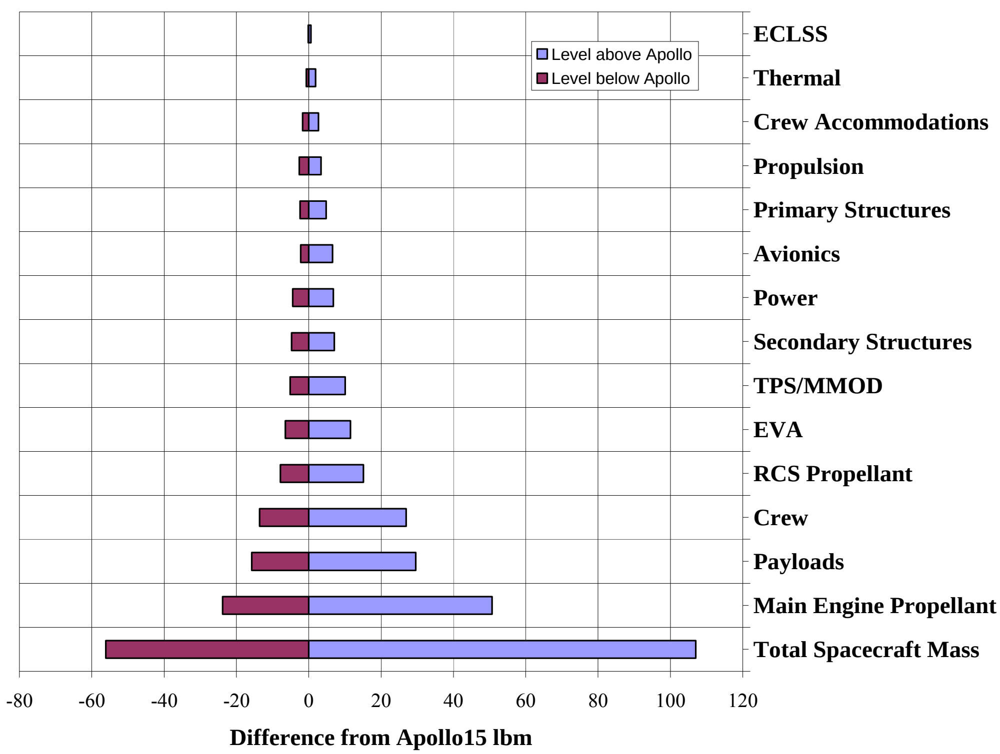


Figure 54: Comparison of 95% confidence limits to Apollo 15 subsystems.

The same approach was used to verify the ESAS input configuration. The values for the ESAS Lunar Ascent Module were derived from the point of departure reference design as listed in the ESAS report on page 166 (NASA, 2005). Shown in Figure 55 are the 95% confidence levels compared to the published ESAS values.

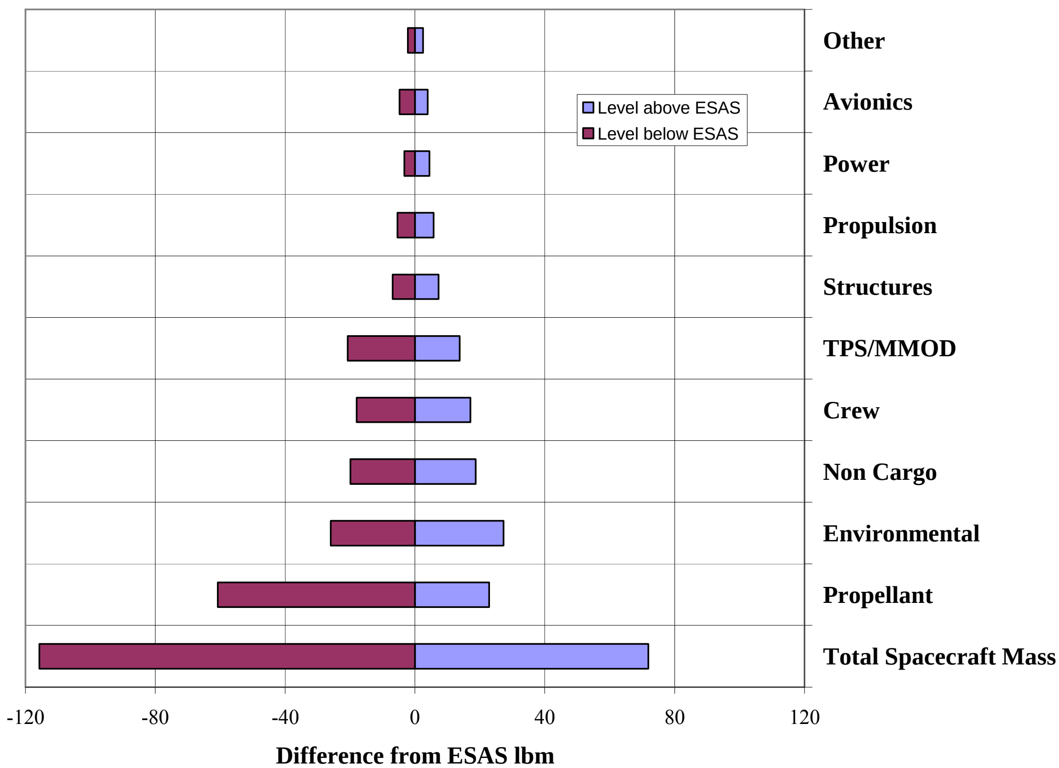


Figure 55: Comparison of 95% confidence limits to ESAS subsystems.

Using estimated values for subsystem component mass, the ESAS input values compared very well to the published values. This step in the analysis process is very important to establish the correct configurations of the subsystems in order to investigate the mass additions due to *Safety* and *Operability*.

5.6.2 Mass Addition Process Results

The mass addition process was conducted for all three Lunar Ascent Module configurations. The results of this analysis are shown as points in the Total Spacecraft, *Safety*, and *Operability* tradespace. Shown in Figure 56 are the results of the Apollo mass addition process.

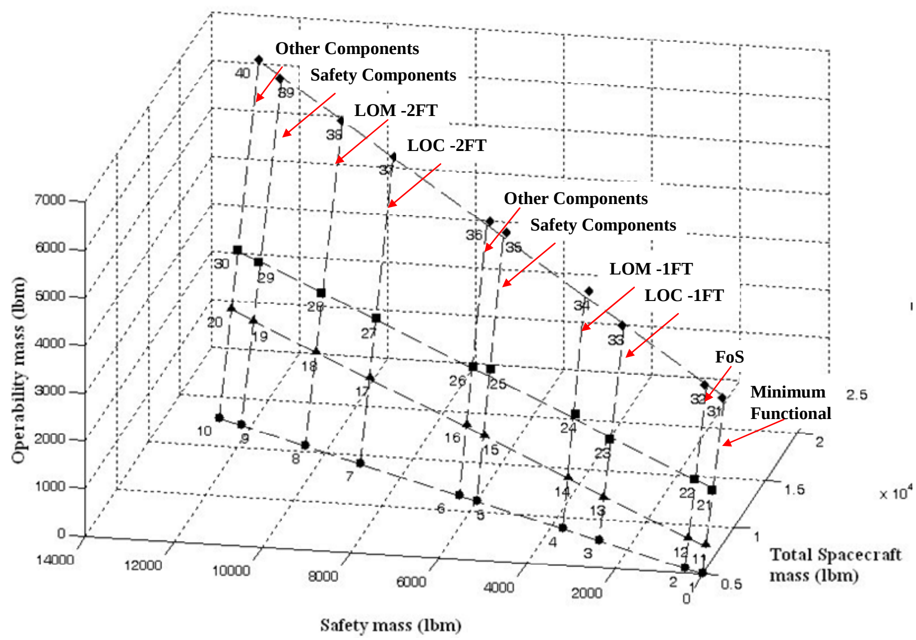


Figure 56: Mass addition process results for Apollo configuration.

Shown in Figure 56 are the 40 node points along with grid lines to visualize the levels of *Safety* and *Operability* in the tradespace. The vertical grid lines represent levels of *Safety* and have been labeled for the reader. As the *Safety* levels increases (example: nodes 1-10), the mass increases greatly. For a two failure tolerant spacecraft, the *Safety* mass is approximately 13,000 lbm. As mentioned earlier, this analysis assumed that all components in the spacecraft were increased in redundancy. This analysis was a first attempt at demonstrating the process and provides the designers and teams the ability to quantify the impacts of additional redundancy on the spacecraft. Shown in Fig 57 is the mass addition process for the Apollo One Man configuration.

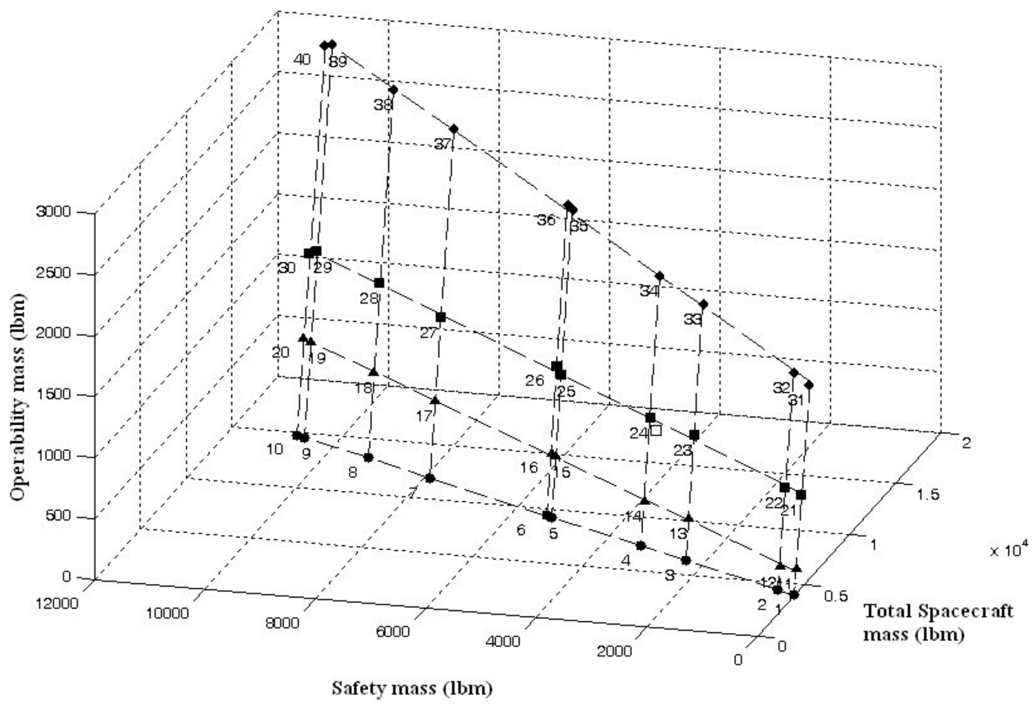


Figure 57: Mass addition process results for Apollo One Man configuration.

Comparing the Apollo One Man configuration to the Apollo configuration reveals a few significant differences. The most notable difference is that the *Operability* mass is approximately half of the Apollo configuration. The addition of components with a Consequence less than 4 (between nodes 5 and 6) did not have a significant impact on the spacecraft *Safety* mass. The total two failure tolerant spacecraft *Safety* mass is similar to the

Apollo nominal configuration with a mass of approximately 11,000 lbs. Comparing the differences between the two graphs allows designers to determine very quickly where mass drivers are located. As this example illustrated, the mass addition process is intended to provide information that can be generated very quickly in order to assist with design decisions. Shown in Figure 58 are the results for the ESAS configuration.

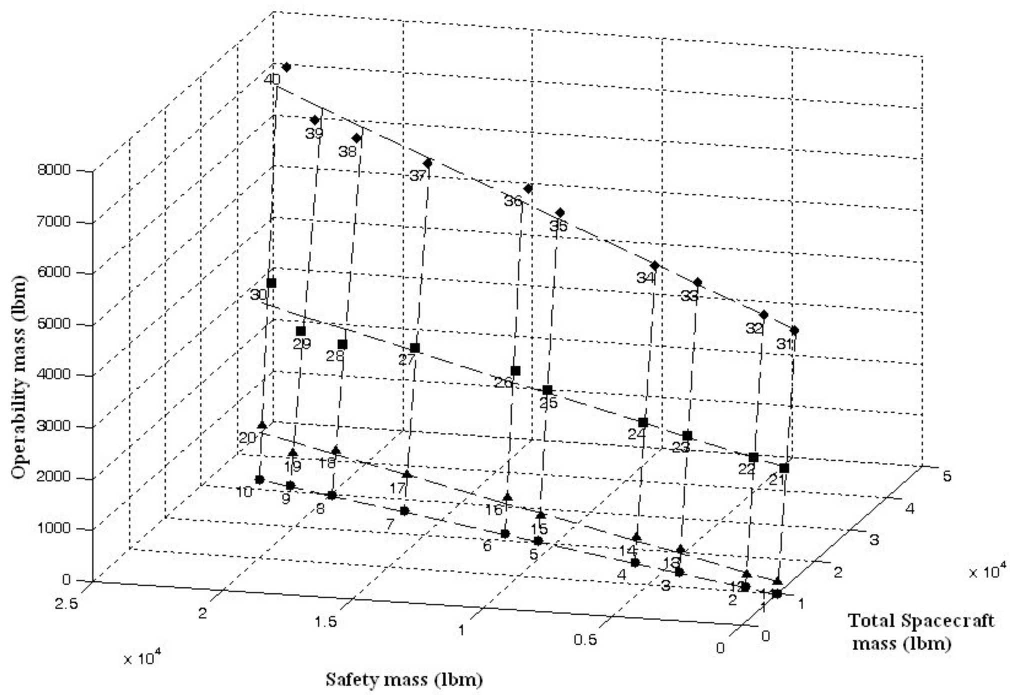


Figure 58: Mass addition process results for the ESAS configuration.

As expected, the ESAS results for *Safety* are approximately twice the Apollo values. A two failure tolerance spacecraft will have approximately 23,000 lbm of *Safety* mass. The *Operability* levels are comparable to the Apollo configuration. However, the *Safety* mass is the key driver in this configuration. Tables of the subsystem mass for the node locations of the three

Lunar Ascent Module configurations are listed in Appendix H. The mass addition process analysis results for the Lunar Ascent Modules revealed similarities and differences among the designs. The most notable difference between the Apollo and ESAS configuration is the doubling of the *Safety* mass. The *Operability* levels were comparable and this is likely due to the difference in mission time between the Apollo and ESAS configurations. The advantage of this process is that it provides a quick view of the potential mass additions related to *Safety* and *Operability*.

5.6.3 Tradespace Monte Carlo Results

Shown in Figs. 59 – 61 are the results of the tradespace Monte Carlo analysis for the three Lunar Ascent Module configurations.

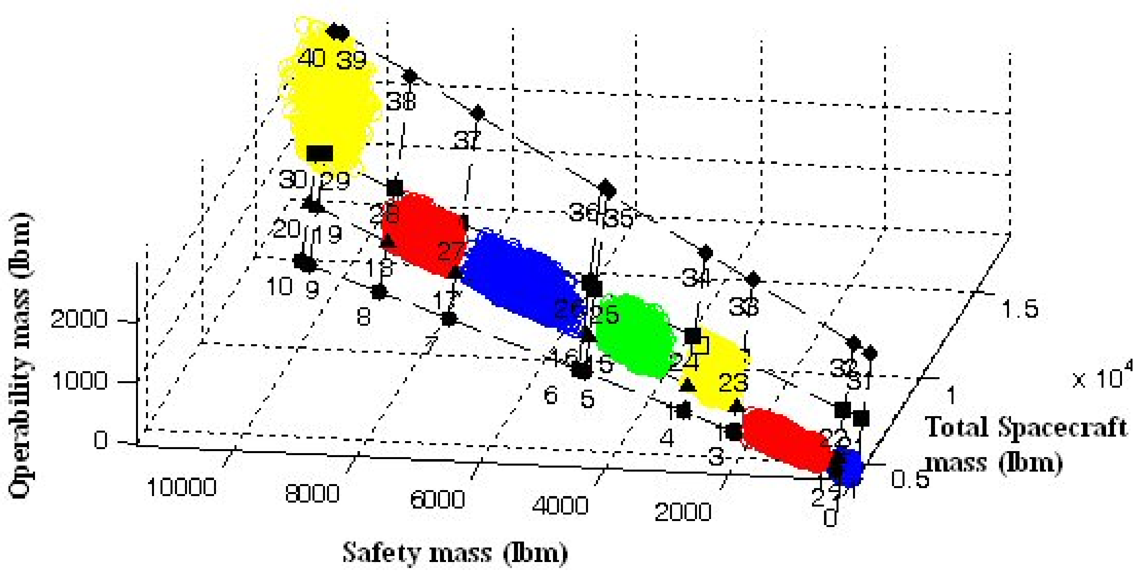


Figure 59: Tradespace Monte Carlo results for Apollo One Man configuration.

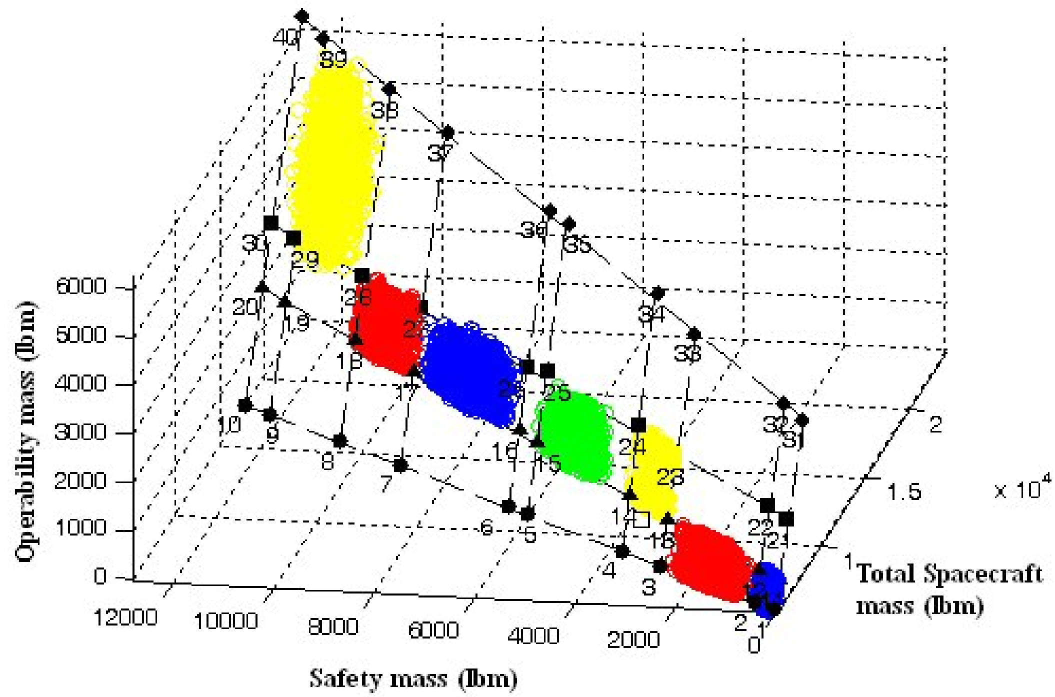


Figure 60: Tradespace Monte Carlo results for Apollo configuration.

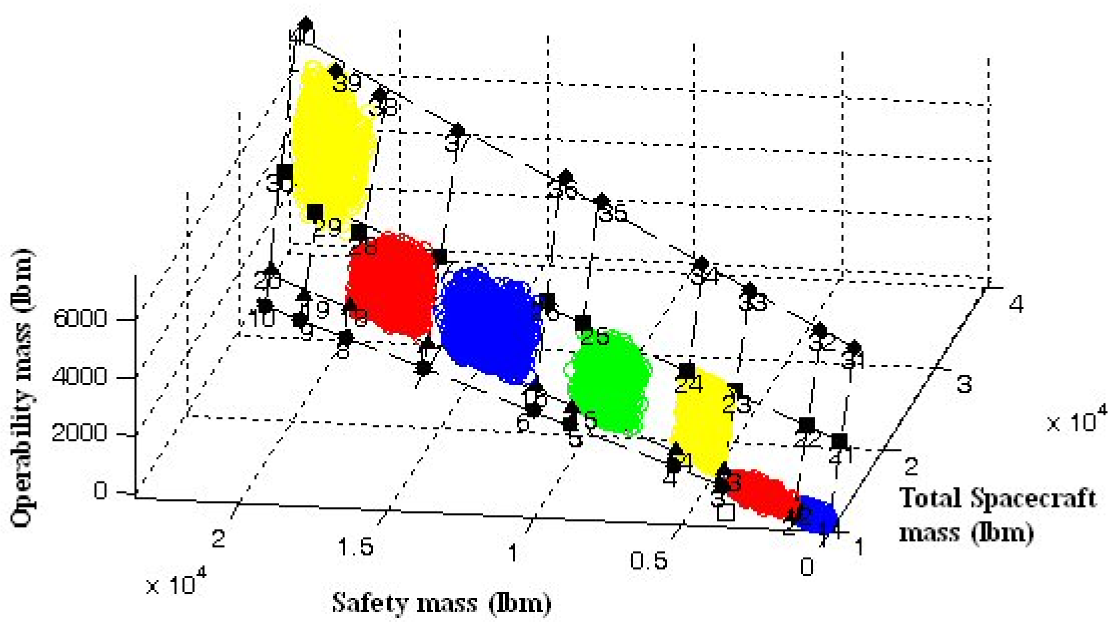


Figure 61: Tradespace Monte Carlo results for ESAS configuration.

Listed in Table 15 are the results of the tradespace Monte Carlo analysis for each of the three Lunar Ascent Module configurations.

Table 15: Tradespace Monte Carlo analysis results.

Apollo Configuration						
Zone	**Average Total Spacecraft mass (lbm)**	**Tot S/C Standard Deviation**	**Average *Safety* mass (lbm)**	***Safety* Standard Deviation**	**Average *Operability* mass (lbm)**	***Operability* Standard Deviation**
1	5647.5	175.8	261.2	91.1	259.3	152.3
2	7061.1	368.3	1604.4	295.8	329.7	195.9
3	9732.2	411.9	3243.2	184.1	1362.0	352.3
4	11665.8	397.9	5130.7	283.9	1408.1	246.9
5	14583.4	508.5	7751.6	424.7	1704.8	252.0
6	16914.5	385.8	9873.2	267.2	1914.3	255.4
7	20839.3	751.8	11614.3	298.6	4098.0	668.0
Apollo One Man Configuration						
Zone	**Average Total Spacecraft mass (lbm)**	**Tot S/C Standard Deviation**	**Average *Safety* mass (lbm)**	***Safety* Standard Deviation**	**Average *Operability* mass (lbm)**	***Operability* Standard Deviation**
1	4250.5	80.9	170.3	57.2	93.1	59.6
2	5439.7	286.7	1326.7	270.7	125.9	76.1
3	7498.1	272.4	2864.1	180.0	646.9	195.3
4	9062.7	269.8	4423.5	217.8	652.1	146.5
5	11362.1	436.2	6614.9	395.1	760.1	148.0
6	13374.4	289.6	8538.3	241.7	849.0	147.6
7	16786.2	431.3	10709.5	248.6	2089.6	330.1
ESAS Configuration						
Zone	**Average Total Spacecraft mass (lbm)**	**Tot S/C Standard Deviation**	**Average *Safety* mass (lbm)**	***Safety* Standard Deviation**	**Average *Operability* mass (lbm)**	***Operability* Standard Deviation**
1	10772.8	270.6	725.5	260.5	129.2	87.4
2	12928.5	470.2	2849.6	453.8	160.8	106.9
3	16915.6	674.7	5533.6	409.0	1463.9	524.6
4	20775.1	685.7	9401.4	460.4	1455.7	491.5
5	25924.0	988.2	14349.1	844.0	1656.8	504.7
6	29940.0	876.1	18302.0	704.4	1719.9	504.7
7	36265.1	959.1	21666.2	490.4	4680.8	736.6

Investigating the differences among the three configurations reveals a few unexpected differences. The first notable difference is between the Apollo and Apollo One Man configurations where the average total spacecraft mass difference in zone 1 is approximately 1400 lbs. The reduction in pressurized volume and one crewmember did not have as significant impact as expected where the difference in the Apollo configuration and the ESAS configuration was approximately doubled for the addition of two crewmembers. In addition to the total spacecraft mass differences, the increase in *Safety* mass was the leading driver in total spacecraft mass. Shown in Figure 62 are the *Safety* mass fractions for the three Lunar Ascent Module configurations according to the tradespace zone.

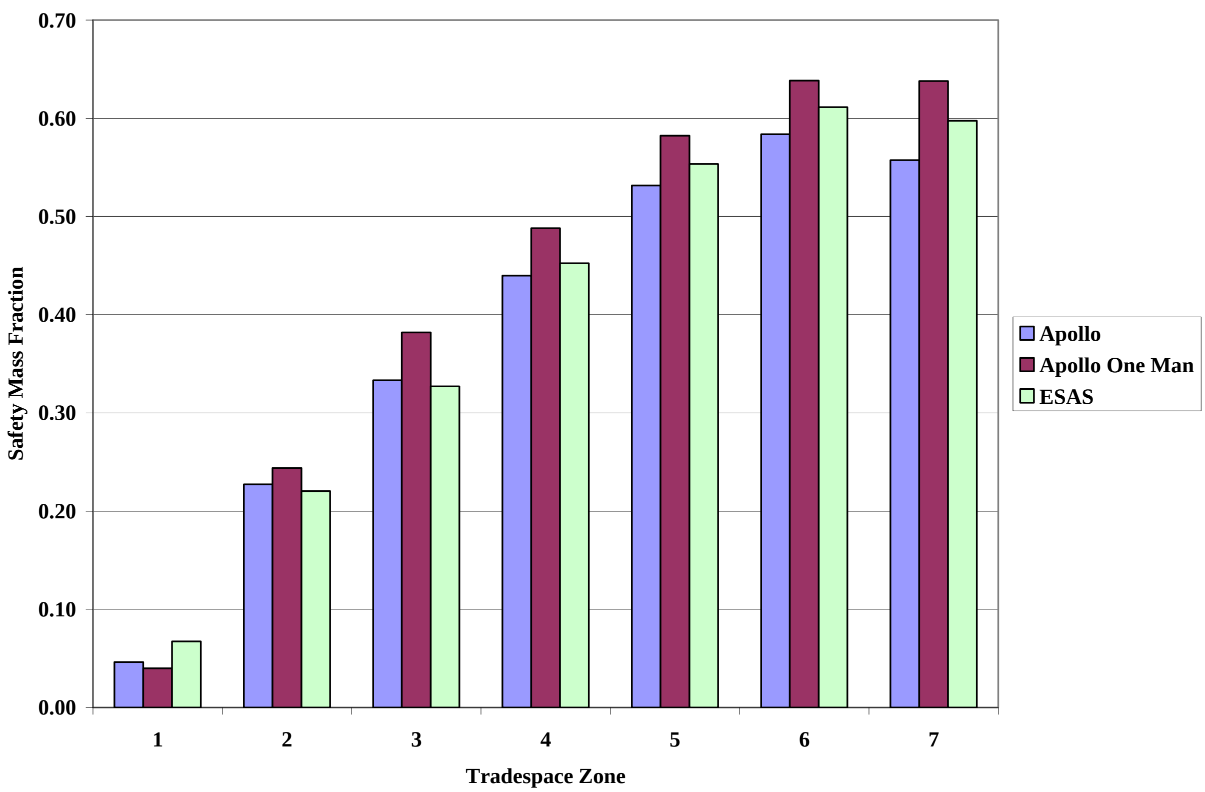


Figure 62: Safety mass fractions in tradespace zones.

The results in Figure 62 show a linear increase in *Safety* mass fraction as the spacecraft configuration adds *Safety* through redundancy or additional components. Tradespace zone 5 is where the spacecraft configuration exceeds a *Safety* mass fraction of 50%. According to this analysis, a fully one failure tolerance spacecraft with two failure tolerance for Consequence 5, LOC would approach 50% of the spacecraft mass. This is the most significant outcome of the analysis and one that designers should keep in mind when developing concepts.

5.6.4 Safety vs. Operability Mass Addition Charts

The following *Safety* and *Operability* charts summarize the mass addition process and the tradespace Monte Carlo analysis. The Minimum Functionality design methodology is the foundation of this approach and provides a method for designers to investigate potential spacecraft configurations based on the levels of *Safety* and *Operability*. Shown in Figs. 63 - 65 is the resulting *Safety* vs. *Operability* charts for the three Lunar Ascent Module configurations.

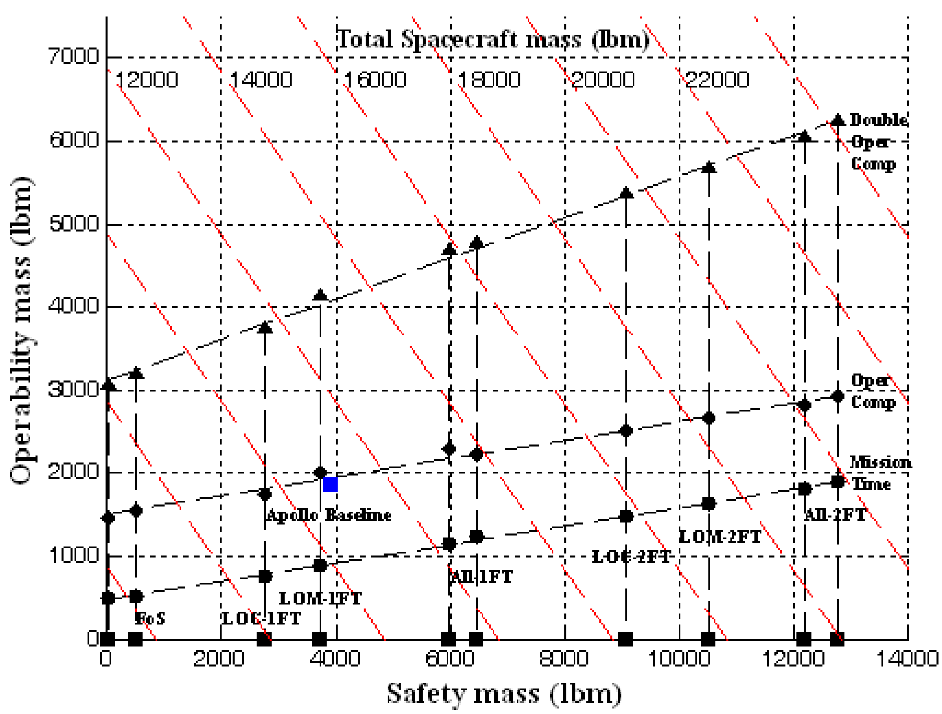


Figure 63: *Safety* vs. *Operability* mass addition chart for Apollo configuration.

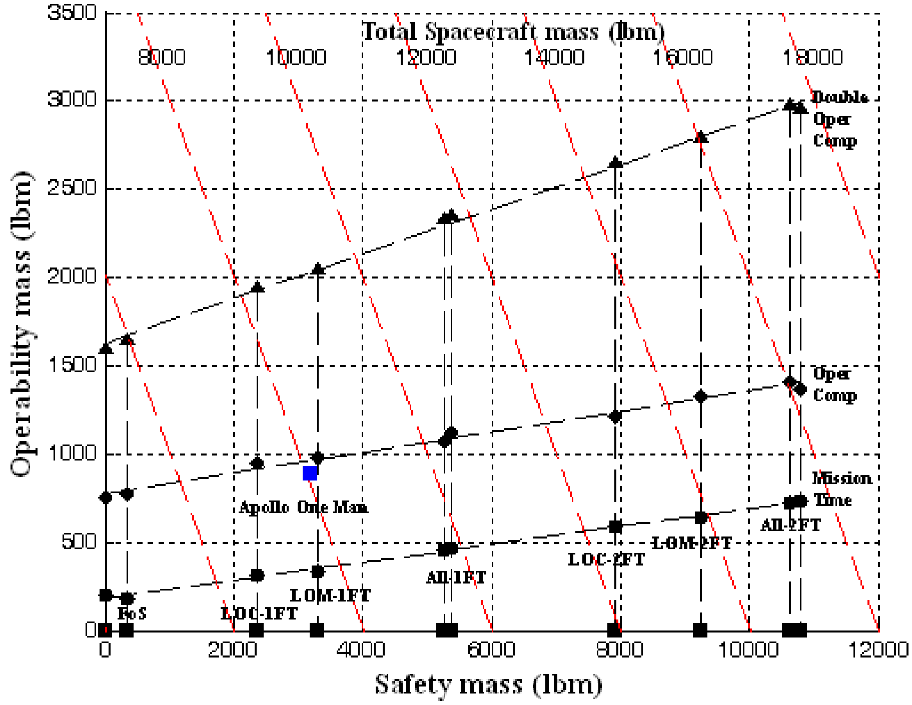


Figure 64: *Safety* vs. *Operability* mass addition chart for Apollo One Man configuration.

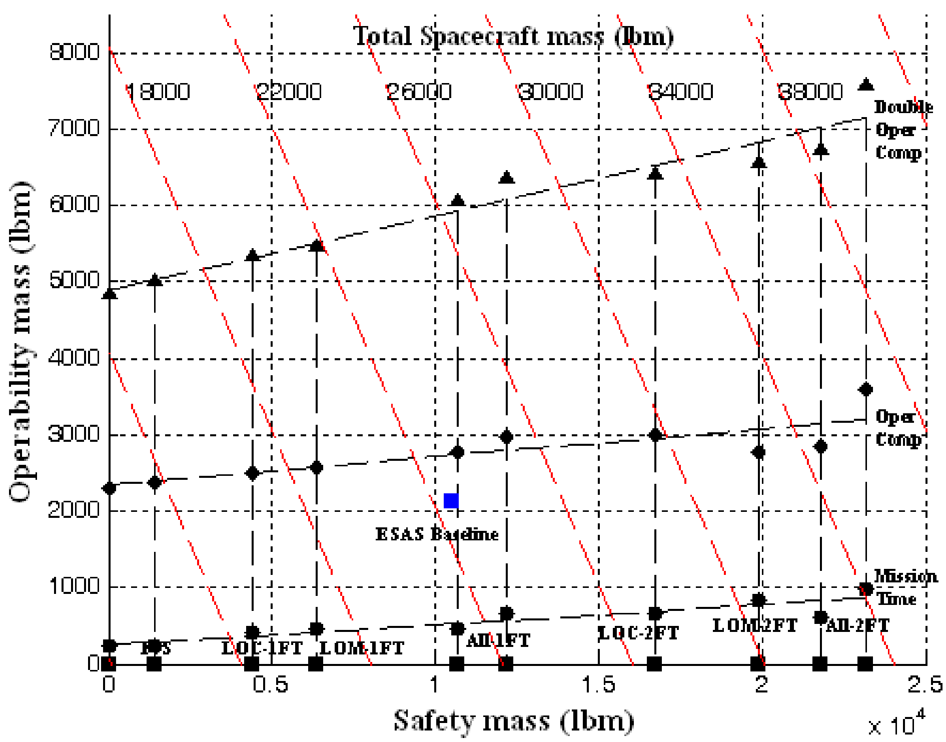


Figure 65: *Safety* vs. *Operability* mass addition chart for ESAS configuration.

The charts in Figs. 63 – 65 provide a complete view of the conceptual design tradespace from a minimum functional configuration beginning at the origin. The dashed lines represent levels of *Safety* and *Operability*. The major levels for *Safety* have been identified according to Consequence and failure tolerance. The major levels for *Operability* have been identified according to the addition of *Operability* to the spacecraft configuration. One will notice the red dashed lines that represent the total spacecraft mass and where the lines intersect with the various *Safety* and *Operability* levels. If a target goal of total spacecraft mass is an upper limiting boundary, the user can easily identify the maximum level of *Safety* and *Operability* possible for the designated spacecraft mass. Developing the chart does not require a large amount of time because a minimum number of node configurations could be determined and the results plotted in the format. As the spacecraft configuration matures, the mass addition process utilized with the *Safety* and *Operability* charts is an efficient method for understanding the entire spacecraft configuration tradespace.

The baseline values for the Apollo, Apollo One Man, and ESAS configurations are plotted as blue squares in the charts. The Apollo and Apollo One Man baseline values are very close to the LOM 1FT Safety level and this is because many of the components in the spacecraft configuration were only redundant for critical functions. The Apollo configurations were not a full one failure tolerant system in all the subsystems and the addition of contingency propellant for *Safety* is a significant contributor to the level. The ESAS baseline is closer to a one failure tolerant configuration and many of the components in the baseline configuration were assumed to be redundant. The *Operability* level of the ESAS baseline is below the full operational level because not all operability functions were included in the baseline.

5.7 CONCLUSIONS

This chapter focused on a systematic process for evaluating safety and operational capability in conceptual human spacecraft designs. The mass addition process is based upon a Minimum Functionality design methodology such that mass additions to the minimum functionality configuration are in the form of increasing *Safety* through additional redundancy or *Operability* components. The specifics of how mass is added to the minimum functionality configuration is largely dependent upon the design teams. However, the design teams must be aware that small changes in redundancy can have larger consequences on the spacecraft as a whole and the mass addition process described here provides an overall view of the likely spacecraft configurations.

The mass addition process begins with a minimum functional baseline configuration and a matrix of spacecraft configurations is defined with varying levels of *Safety* and *Operability* in order to designate node locations in the objective tradespace. The grid of spacecraft configurations maps the objective tradespace to the input design space and provides designers with a complete view of potential spacecraft configurations. A series of zones are designated within the tradespace domain to narrow the focus of design concepts to regions of interest for further investigation. The process can be utilized throughout the conceptual design to continually assess the impacts of design changes on the spacecraft.

Three design configurations of Lunar Ascent Modules were used as case studies to demonstrate and validate the methodology. These configurations included the nominal Apollo Lunar Ascent Module, a reduced Apollo one-man configuration and an ESAS derived configuration. The purpose of using three concepts was to demonstrate the mass addition process for different types of spacecraft designs.

Because so many unknowns exist in the early conceptual design phase due to factors such as unknown integration, complexity, and uncertainty in requirements, it can be a challenging time for correctly estimating the mass of a spacecraft. However, a bottom up philosophy as described in this work forces designers to consider the low level functions in the configuration where candidate technologies are chosen and evaluated. As part of the evaluation process, the overall vehicle configuration must be considered in addition to subsystem discipline specific trade studies so that designers can understand the impact of decisions on the spacecraft configuration. Incorrect assumptions in the conceptual design phase can lead to redesign or reduced performance.

Although mass is a key driver in any spacecraft design, the spacecraft must be safe and perform the intended mission. The mass addition and tradespace exploration process described is one method that could be coupled with reliability analysis or other figures of merit for deciding how to improve the spacecraft configuration. A difficulty in the early conceptual design phase is the limited amount of reliability data. Common needs for designers are methods that assess the other characteristics of *Safety* in the spacecraft to provide an indication of increased reliability. This work is the focus of the next chapter and will use the information developed here as a case study to demonstrate the methodology.

5.7.1 Acknowledgements

This work would not have been possible without the educational support from Lockheed Martin Corporation and Sierra Nevada Corporation. The experience of maturing conceptual designs and designing flight hardware for the Orion program played a key role in the development of this methodology.

CHAPTER 6

FIGURES OF MERIT FOR TRADING CONCEPTUAL HUMAN SPACECRAFT CONFIGURATIONS

6.1 ABSTRACT

Two figures of merit, the Safety Index and Operability Index were developed to provide designers with a scoring system that evaluates the overall spacecraft configuration. The figures of merit are not intended to replace reliability based analysis methods, but a method to provide designers with a complementary approach for evaluating the safety and operational aspects of a spacecraft configuration with limited knowledge of the reliability of the components. Based upon previous methods for functional analysis of early spacecraft conceptual designs, the Safety Index scores the entire spacecraft based upon consequence of loss of function, complexity of subsystem components, similar and dissimilar redundancy, and potential hazards. Items such as Design for Minimum Risk are included to provide an overall spacecraft safety score. The Operability Index scores are based upon the mission time, crew size, payloads, and operational functionality and provide a measure of the spacecraft configuration against top level objectives. The Safety Index and Operability Index can be combined into a Total Spacecraft score such that comparisons can be made against competing spacecraft types. Three case studies of human spacecraft Lunar Ascent Module designs were used to demonstrate and validate the scoring methodologies.

6.2 INTRODUCTION

The purpose of this chapter is to introduce two figures of merit for evaluating *Safety* and *Operability* in conceptual human spacecraft configurations. Although mass is an important part of a spacecraft design, other factors such as complexity, failure modes, similar and dissimilar redundancy, and operational capabilities are just a few of the characteristics that need to be considered when evaluating a spacecraft design. In many cases, the focus of design is to reduce mass and using this approach alone can introduce additional complexity or result in optimizing the spacecraft too early before other issues are discovered. The figures of merit developed in this chapter will provide designers with another method for evaluating parameters that are not always easily quantifiable, but must be considered as the spacecraft design matures.

The figures of merit described in this work are not intended to replace reliability analyses. An attempt was made to determine a scoring method that correlated with reliability. The exact correlation between the figure of merit scores and reliability is very difficult to determine without reliability values for the components. However, increases in component redundancy in the figure of merit scores point to an increase in reliability if certain assumptions are made for the reliabilities of the functions and spacecraft components. Because of the large number of assumptions needed to compare reliability to the figure of merit scores, it was decided that additional work is needed to fully understand how to structure the figure of merit scores with predicted reliability in the spacecraft components. The figures of merit described in this work provide a foundation for later research, but do not include reliability.

The two figures of merit described in this work are named the Safety Index and Operability Index and are intended to be utilized by spacecraft designers in the early stages of conceptual design for investigating spacecraft configurations. As mentioned in Chapter 5, many variables and parameters must be traded when developing conceptual designs and the combination of the many parts is the challenge. Other than mass, an additional scoring method is needed to provide a means of comparison between concepts. The methods presented here are a preliminary look into the early developmental stages of a project when many of the variables are uncertain and ideas are being formulated. The figures of merit are intended to "point" the designers in the correct direction for iterating and maturing a spacecraft design.

6.3 BACKGROUND

The main purpose of using a figure of merit is to quantify aspects of a design for decision making. A figure of merit can be used to compare risks (Miller *et al*., 2008), represent launch vehicle reliability (Hassan and Crossley, 2002), and score spacecraft components (Thunnissen, 2004). As mentioned by Miller *et al*. (2008), a figure of merit is suggested as a technique for comparing risk but requires people and judgment to balance the risks. A figure of merit should consider not only local effects, but the system as a whole.

NASA's Systems Engineering Handbook (NASA, 1995) provides guidelines on figures of merit used as a multi-objective selection rule. The alternatives of a system are scored in quantifiable terms of how well it achieves each objective. Combining the multi-objective scores yields an overall figure merit for the alternative (NASA, 1995). The method suggested by the NASA guidelines (NASA, 1995) is to linearly combine scores for each of the objectives into a weighted sum of the scores. The choice of weighting factors can be assigned a prior or determined using Multi Attribute Utility Theory (MAUT) or other techniques such as Analytic Hierarchy Process (AHP) (NASA, 1995). One should also be aware that weight choices can influence the figure of merit score and overlook potential best alternatives (NASA, 1995).

Safety is typically scored using risk and reliability analyses. The most common approaches in early design development and Phase-A spacecraft activities are Functional Failure Modes and Effects Analysis (FFMEA) and preliminary hazard analysis (Miller *et al*, 2008). Physics based models are combined with the FFMEA and hazards to understand critical failure modes (Miller *et al*., 2008). This approach is useful for identifying hardware issues and directing the design as it matures. In this early stage, accurate reliability values for all of the spacecraft hardware may not be available and allocated reliability goals can be used to estimate system reliability (FAA, 2005). Monte Carlo simulation and sensitivity analyses are recommended by the FAA and NASA to understand failures in operational scenarios. The typical method for quantifying safety in the early stage of development is through the use of risk and reliability analyses that rely on heuristic or expert data.

Operability, used in this context, is the functionality required to perform specific mission objectives and is typically associated with high level mission objectives. NASA defines these mission objectives as destination, purpose, stay time, crew size and support requirements (Miller *et al*., 2008). Although these requirements are high level mission objectives, these parameters are often traded in order to meet performance objectives. As a spacecraft design matures, the *Operability* of the spacecraft is typically associated with performance parameters and is dependent upon many other spacecraft parameters.

The figures of merit, the Safety Index and Operability Index, are scoring methodologies built upon a Minimum Functional design methodology. Based on typical risk and reliability approaches used in the early design phase, the two figures of merit provides designers and decision makers with a scoring method that evaluates the whole spacecraft configuration and quantifies the relative differences between competing design concepts.

6.4 DERIVATION OF FIGURES OF MERIT

Any figure of merit used for relative comparison of design concepts should include as many quantifiable parameters as possible in order to develop a composite "score". Although mass and safety are key aspects to the design, the operational aspects of a mission should be considered at the same time because the three parameters are coupled in the spacecraft configuration. A decrease in *Operability* mass is an increase in *Safety* mass for a given total spacecraft mass.

A challenge in the early stage of development is determining the parameters that contribute the most to the overall design. Using a minimum functionality design methodology where low level functions are coupled to potential technology solutions, a designer can trade and evaluate potential configurations based on the known information. This information includes:

- Consequence of loss of function,
- Number of components required to perform the functions,
- Potential failure causes and hazards associated with technology selections,
- TRL of technology selections,
- Similar and dissimilar redundancy in low level functions,
- Design for Minimum Risk components,
- Factors of Safety margins,
- Mission time,
- Number of crew,
- Payloads and operational equipment,

- Basic understanding of spacecraft integration, and
- Mass estimate of the functions and spacecraft configuration.

The previous information is a list of the main quantifiable parameters in a spacecraft configuration and used in the derivation of the figures of merit for *Safety* and *Operability*. Although the above list is focused on the components of the spacecraft, the intent is to utilize known quantitative information about the configuration in order to score the overall design. The figures of merit provide a complete score of all the components and functions in the spacecraft.

6.4.1 Safety Index Derivation

The Safety Index figure of merit is a score of the complexity, margins, failure tolerance, safety components, and hazards in the spacecraft. Although the Safety Index is not directly correlated to reliability, the key assumption is that increasing redundancy through similar or dissimilar redundancy will increase the failure tolerance of the spacecraft in order to mitigate potential hazards. Another key assumption is that future reliability and safety analyses will be completed when the number of components and redundancy are clearly defined for a selected spacecraft design. An advantage of the Safety Index is that it provides an early indication of potential issues that will be examined in later detailed analyses. The Safety Index evaluates the typical areas in risk based approaches and is calculated based on the following parameters:

- Consequence of loss of the function
- Number of components needed to perform the function

- Average redundancy of components in the function
- Failure causes or hazards associated with components in the function
- Use of dissimilar redundancy or safety components in the function

The consequence of loss of function is a key parameter in understanding which functions are the highest contributors to safety of the spacecraft. The number of components needed to perform a function is an indicator of the complexity within the function. A high number of components in a function increase the likelihood of potential failure of the function. The redundancy of components in a function reduces the likelihood of failure of the function, but also must be traded against the number of components required to perform the function. The potential hazards associated with a function are the key events that could cause failure of the function. Dissimilar redundancy or additional safety components increase the chances of mitigating a failure through additional means. In typical reliability methods, the addition of safety components does not add to the reliability of the spacecraft only if the use prevents the hazard from occurring. Based on these key parameters and the information available to designers in the early stage of development, the Safety Index is a scoring method that is a precursor to detailed risk and reliability methods. Shown in Figure 66 is the process for calculating the Safety Index.

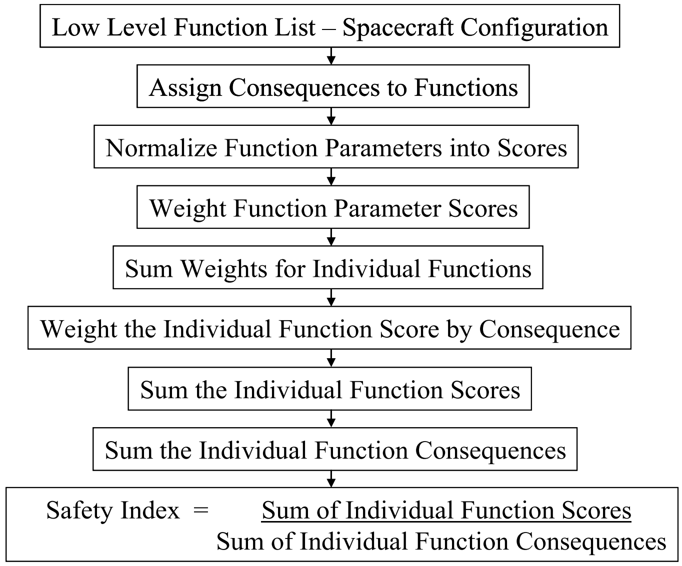


Figure 66: Process for calculating the Safety Index.

The Safety Index is based upon the string of components that perform a function. Shown in Figure 67 is an illustration of how the functions are related to the components in a string.

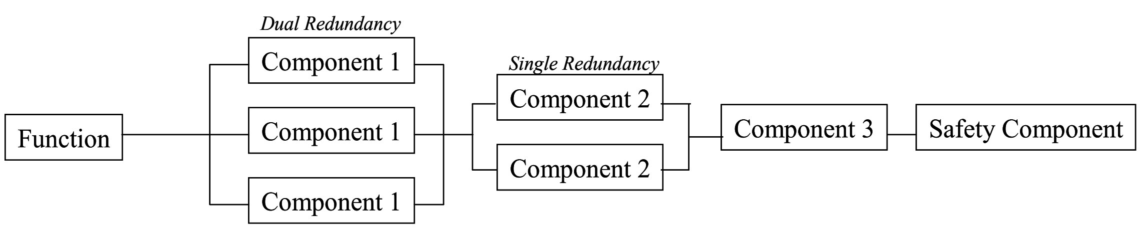


Figure 67: Component string dedicated to a function.

6.4.1.1 Assigning Consequences

A typical qualitative risk approach combines the consequence of an event (or failure) and the likelihood of the event. Based on the levels of consequence and likelihood, a risk score is assigned (NASA, 2007a). For the Safety Index, the low level functions are assigned a consequence value based on the severity of the loss of the function. The scale of consequences used in this process was derived from the NASA Constellation program risk guidelines and are listed as follows:

- Consequence 5 – Loss of life or permanently disabling injury and top level requirements not achievable with existing engineering capabilities; Loss of Crew.
- Consequence 4 – Severe injury or illness requiring extended medical treatment or major impact to requirements, design margins or loss of mission objectives; Loss of Mission.
- Consequence 3 – Injury, illness, or incapacitation requiring emergency treatment or moderate impact to requirements, design margins or mission objectives.
- Consequence 2 – Injury requiring first aid treatment, moderate crew discomfort or minor impact to requirements, design margins or mission objectives.
- Consequence 1 – Minor injury not requiring first aid treatment, minor crew discomfort or negligible impact to requirements, design margins or mission objectives.

6.4.1.2 Safety Index Equations

Given a set of functions and parameters that characterize the configuration of the spacecraft, the individual function parameters of number of components, average redundancy, failure causes (hazards), and safety components are normalized into scores for the individual functions. The number of components is scored from 1-10 based on the number of components in the function. Because the distribution of functions can vary depending upon how the low level functions are defined, the normalized component score is based upon the maximum number of components in an individual function. The maximum number of components in a single function is determined by identifying the function with the largest number of components. The normalized score for the number of components in a single function is calculated using the following expression:

$$Score_{Components} = \left[\left[\left(1 - \left(\frac{C_{function}}{C_{\max imum}}\right)\right)\right] \times 10\right] + 1 \tag{9}$$

Where $C_{function}$ is the number of individual components in a function and $C_{maximum}$ is the largest number of components in any of the functions. The scoring for redundancy is determined using the average redundancy in the function. Because not all components have the same amount of redundant components in a function string, the use of average redundancy is an indicator of the redundancy level of the components that perform the function. The average redundancy is calculated by dividing the number of single type components by the total number of components in the function. The average redundancy in the function is scored from 1-10 based on a range of redundancy from 1-3. Single redundancy in a function is a score of 3.35 while dual redundancy is a score of 10. The normalized redundancy score is calculated as:

$$Score_{\mathrm{Re}\,dundancy} = \left(\left(\left(\frac{R_{Average} - 1}{2}\right)^2\right) \times 9\right) + 1 \tag{10}$$

For components that do not have redundant aspects such as primary structure and pressure vessels, the factor of safety is used in place of average redundancy. In this approach, a factor of safety of 3 is considered to be the highest factor of safety for DFMR components.

Although it is possible to assign a larger factor of safety, the largest expected factor of safety would be used to adjust the scoring.

The scoring for failure causes in a function is based upon the number of identified hazards or failure causes. For example, an external mounted electronic box component would be subjected to loading, extreme temperature, Micrometeoroids and Orbital Debris (MMOD), and contamination that could cause a failure. Other causes such as manufacturing flaws, design flaws, electrical failure, and software could contribute to a failure in the component. To determine how well a function relates to the total number of potential hazards, a listing of all the hazards to the spacecraft is compiled. A list of potential hazards could include electrical failure, mechanical failure (wear), stress and structural loading, contamination, temperature, manufacturing flaws, design flaws, duty cycle, MMOD, and software. The total number of potential hazards to the spacecraft is determined and used in the normalized scoring of hazards. The normalized hazard score is calculated as:

$$Score_{Hazards} = \left[\left[1 - \left(\frac{H_{Function} / R_{Average}}{H_{Maximum}}\right)\right] \times 10\right] + 1 \qquad (11)$$

Where $H_{Function}$ is the number of hazards that are related to the individual function and $H_{Maximum}$ is the number of all the potential hazards to the spacecraft. The number of hazards is divided by the average redundancy to account for redundant aspects in components that could mitigate a potential failure. The scoring for additional safety components is a "bonus" score for the individual function and is 0 or 10 based upon the addition of safety components.

$$Score_{Safety\ Components} = 0\ or\ 10 \qquad (12)$$

6.4.1.3 Safety Index Weight Factors

The weight factors are an important part of the scoring method. For the examples used in this work, the following weight factors were chosen based on importance and the sensitivity of the weighting factors will be demonstrated in the analysis.

- $\textit{Weight Factor}_{Redundancy} = 3.475$
- $\textit{Weight Factor}_{Components} = 2.475$
- $\textit{Weight Factor}_{Safety\ Components} = 2.5$
- $\textit{Weight Factor}_{Hazards} = 1.55$

The expressions used to weight the parameter scores are:

$$\textit{Weighted Components} = \textit{Score}_{Components} \times \textit{Weight Factor}_{Components} \tag{13}$$

$$\textit{Weighted Redundancy} = \textit{Score}_{Redundancy} \times \textit{Weight Factor}_{Redundancy} \tag{14}$$

$$\textit{Weighted Hazards} = \textit{Score}_{Hazards} \times \textit{Weight Factor}_{Hazards} \tag{15}$$

$$\textit{Weighted Safety Components} = \textit{Score}_{Safety\ Components} \times \textit{Weight Factor}_{Safety\ Components} \tag{16}$$

After the parameter scores have been weighted, the function score is the sum of the weighted scores. This score represents the sum and weighting of the parameter scores and does not include weighting for consequence of loss of function.

$$\textit{Function Score} = \textit{Sum of Weighted Parameter Scores} \tag{17}$$

The function scores are weighted according to the consequence value of the function to develop a total weighted function score.

$$\textit{Total Weighted Function Score} = \textit{Function Score} \times \textit{Consequence} \quad (18)$$

6.4.1.4 Safety Index

The Safety Index of the spacecraft is determined by dividing the sum of the total weighted function scores by the sum of the consequence values. The Safety Index has a scale from 8-100 and is calculated as follows:

$$\textit{Safety Index} = \textit{Sum of Total Weighted Function Scores} / \textit{Sum of Consequences} \quad (19)$$

The Safety Index is a comprehensive scoring of all the functions in the spacecraft. The figure of merit provides designers with a method for evaluating not only redundant components but also DFMR items and safety additional components. Utilizing the Safety Index not only allows comparison between spacecraft of similar type, but also spacecraft of different types of configurations. The examples that follow will demonstrate the application of the Safety Index for simple and complex functionality.

6.4.2 Operability Index Derivation

The Operability Index is a score of the operational aspects of the spacecraft configuration and is based upon the preferences of the decision makers more than the design team. The Operability Index is a much simpler scoring method because it uses parameters that are easier to quantify and interpret. For the purposes of this derivation, four fundamental parameters have been chosen for the Operability Index. In future studies, the Operability Index could be expanded to include other parameters beyond what is described here. The four fundamental parameters chosen for evaluation are:

- Mission time
- Number of *Operability* functions,
- Number of Crew, and
- Payload mass.

The four parameters were chosen based on typical top level objectives for a human spacecraft. The mission time is a key driving aspect of the design and is often related to total spacecraft mass. The number of *Operability* functions is a measure of the additional functionality of the spacecraft beyond a minimum functional design. The number of crew is also a top level objective that is often traded for mass savings, but the ability of the spacecraft to carry additional crew is an increase in operational capability. The payload mass is a key parameter based upon the operational aspects of a mission. An increase in payload mass is associated with an increase in operational functionality and the capability of the spacecraft.

6.4.2.1 Operability Index Equations

The equations for the Operability Index are normalized according to target goals. In the first parameter, mission time, the mission time is normalized according to maximum mission duration. The maximum mission time represents a goal of the maximum possible time, but for the purposes described here it is an upper boundary condition. The normalized time is scaled from 0 – 10 and is calculated with the following expression:

$$Score_{Time} = \left(\frac{Time_{Nominal}}{Time_{Maximum}} \right) \times 10 \tag{20}$$

The number of crew is normalized to a maximum number of crew and is calculated with the following expression:

$$Score_{Crew} = \left(\frac{Crew_{Number}}{Crew_{Maximum}} \right) \times 10 \tag{21}$$

The amount of payload mass is normalized to a maximum desired payload and is calculated with the following expression:

$$Score_{Payload} = \left(\frac{Payload_{Mass}}{Payload_{Maximum}} \right) \times 10 \tag{22}$$

The number of *Operability* functions in the spacecraft is normalized to the total number of function in the spacecraft and is calculated with the following expression:

$$Score_{Operability\_Functions} = \left(\frac{Functions_{Operability}}{Functions_{Total}} \right) \times 10 \tag{23}$$

6.4.2.2 Operability Index Weight Factors

For the examples used in this work, the following weight factors were chosen based on importance and the sensitivity of the weighting factors is shown in the analysis.

- $Weight\ Factor_{Time} = 4$
- $Weight\ Factor_{Crew} = 3$
- $Weight\ Factor_{Payload} = 2$
- $Weight\ Factor_{Operability\ Functions} = 1$

The expressions used to weight the parameter scores are:

$$Weighted\ Time = Score_{Time} \times Weight\ Factor_{Time} \tag{24}$$

$$Weighted\ Crew = Score_{Crew} \times Weight\ Factor_{Crew} \tag{25}$$

$$Weighted\ Payload = Score_{Payload} \times Weight\ Factor_{Payload} \tag{26}$$

$$Weighted\ Operability\ Functions = Score_{Operability\ Functions} \times Weight\ Factor_{Operability\ Functions} \tag{27}$$

6.4.2.3 Operability Index

The Operability Index score is a measure of the operational aspects of the spacecraft. The score is dependent upon the target goals for each parameter and the weighting factors. The Operability Index has a scale from 0-100 and is calculated as follows:

$$Operability\ Index = Sum\ of\ Weighted\ scores \tag{28}$$

6.4.3 Total Spacecraft Scoring

The Safety Index and Operability Index scores can be combined into a Total Spacecraft score to be visualized in a tradespace. The weighting of the scores is dependent upon the user preferences and for this example, the Safety Index score is weighted twice the Operability Index score. The Total Spacecraft score is scaled from 0 – 100 and is calculated with the following expression:

$$Score_{Total\_Spacecraft} = \left(\frac{\left(Score_{Safety\_Index} \times 2\right) + Score_{Operability\_Index}}{3} \right) \tag{29}$$

6.5 ANALYSIS STUDIES

The following analysis studies demonstrate the Safety Index, Operability Index, and Total Spacecraft score using simple examples and case studies. As mentioned in Chapter 5, a figure of merit scoring is necessary to evaluate spacecraft parameters other than mass. The Lunar Ascent Module case studies are used to demonstrate the scoring methodology for a human spacecraft configuration.

6.5.1 Weight Factors

The weight factors for the Safety Index and the Operability Index were examined to determine the sensitivity of each weight factor on the overall scores. A simple list of 10 functions was used to demonstrate the scoring methods and the sensitivities of the weight factors. The number of components, redundancy, hazards, and safety components in each function were generated randomly to simulate a likely spacecraft configuration with a limited number of functions. A Monte Carlo analysis of 10,000 runs was completed to evaluate the distribution of the scores. The Safety Index weight factors were determined through iteration such that the score was not sensitive to a single parameter. The Operability Index weight factors were established based on ranking of the parameters. Distribution ranges and average score values for the Safety Index and Operability Index were determined.

6.5.2 Safety Index Comparisons

A baseline example of the Safety Index using a set of 10 functions was created to investigate the relative changes in the scores due to increases in overall failure tolerance and number of components. The baseline example simulated a typical spacecraft configuration with a smaller set of functions. The values for consequence, components, total number of components, hazards, and safety components were randomly generated for the set of functions. A minimum functional, one failure tolerant, and two failure tolerant configurations were used to quantify the relative differences in scores due to increased failure tolerance. A simple comparison of changes in Safety Index due to component level changes was conducted to investigate the resolution of the Safety Index scale. The simple examples demonstrated the sensitivity of the Safety Index to increases in failure tolerance and number of components.

6.5.3 Operability Index Comparisons

A set of four comparisons were conducted to investigate the relative difference in Operability Index score for small increments in each of the key parameters. A baseline configuration was established for comparison with incremental increases in time, crewmembers, payload, and Operability functions. The comparisons demonstrated the sensitivity of the Operability Index to each of the parameters.

6.5.4 Lunar Ascent Module Scoring

The three Lunar Ascent Module case studies discussed in Chapter 5 were used to investigate and demonstrate the scoring methodology. The three spacecraft types of, Apollo nominal, Apollo One Man, and ESAS point of departure were scored and compared. The first part of this analysis focused on the scoring of the node configurations for each of the three spacecraft types. Shown in Table 16 is the spacecraft configuration nodes used in the scoring process.

Table 16: Spacecraft configuration nodes for Safety Index and Operability Index scoring.

		Operability Level			
	Node Number	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition
Safety Level	Minimum Functionality	1	11	21	31
	Factor of Safety Addition	2	12	22	32
	Consequence 5 Failure Tolerance (1FT) Addition	3	13	23	33
	Consequence 4 Failure Tolerance (1FT) Addition	4	14	24	34
	Safety Functionality Addition	5	15	25	35
	Consequence 3 and Lower (1FT) Addition	6	16	26	36
	Consequence 5 Failure Tolerance (2FT) Addition	7	17	27	37
	Consequence 4 Failure Tolerance (2FT) Addition	8	18	28	38
	Safety Functionality Addition	9	19	29	39
	Consequence 3 and Lower (2FT) Addition	10	20	30	40

In addition to the node configuration scoring, the tradespace results described in Chapter 5 were scored using the Safety Index and Operability Index. The scores were combined into a Total Spacecraft score and plotted in a three dimensional tradespace. The average and standard deviation of the scores in each tradespace zone were calculated and compared. The scoring of the Monte Carlo analysis results provided information about relative differences in the tradespace. The seven tradespace zones are listed in Table 17. Shown in Figure 68 is a visual representation of the tradespace zones.

Table 17: Starting and ending configurations for tradespace zones.

Zone	Starting Configuration	Ending Configuration
1	Minimum Functionality at minimum mission time (Node 1)	Factor of Safety Addition at Nominal Mission Time (Node 12)
2	Factor of Safety addition at Nominal Mission Time (Node 12)	Consequence 5 (LOC) - 1 FT addition at Nominal Mission Time (Node 13)
3	Consequence 5 (LOC) - 1 FT addition at Nominal Mission Time (Node 13)	Consequence 4 (LOM) - 1 FT addition with Operability components at Nominal Mission Time (Node 24)
4	Consequence 4 (LOM) - 1 FT addition with Operability components at Nominal Mission Time (Node 24)	All components -1FT addition with Operability components at Nominal Mission Time (Node 26)
5	All components -1FT addition with Operability components at Nominal Mission Time (Node 26)	Consequence 5 (LOC) - 2 FT addition with Operability components at Nominal Mission Time (Node 27)
6	Consequence 5 (LOC) - 2 FT addition with Operability components at Nominal Mission Time (Node 27)	Consequence 4 (LOM) - 2 FT addition with Operability components at Nominal Mission Time (Node 28)
7	Consequence 4 (LOM) - 2 FT addition with Operability components at Nominal Mission Time (Node 28)	All components -2FT addition with double Operability components at double Nominal Mission Time (Node 26)

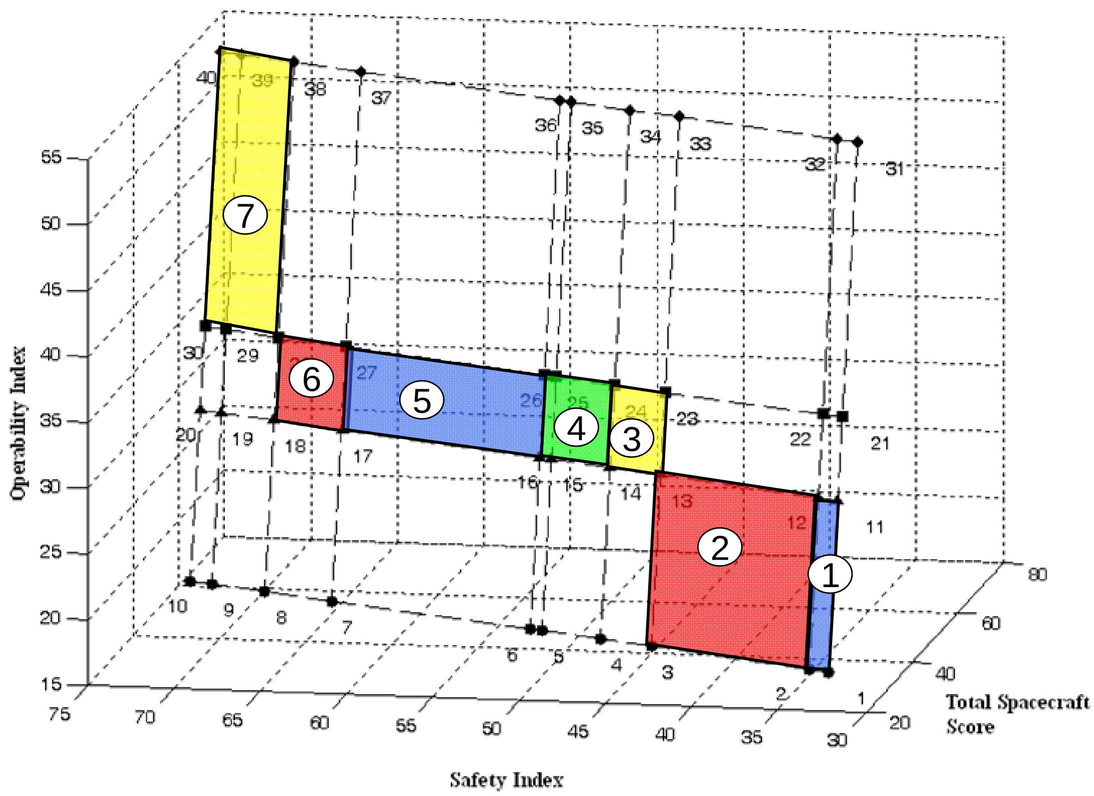


Figure 68: Zones for Safety Index, Operability Index, and Total Spacecraft Score.

6.6 RESULTS

6.6.1 Weight Factors

A simple example of the Safety Index scoring system was created to demonstrate the methods and investigate sensitivities. Shown in Table 18 is an example of the Safety Index scoring process where the values for consequence, components, total number of components, hazards, and safety components were randomly generated.

Table 18: Baseline example of Safety Index scoring.

Function	Consequence	Components	Total Number of Components	Average Redundancy	Hazards	Safety Components	Normalized Components	Normalized Redundancy	Normalized Hazards	Normalized Safety Components	Weighted Components	Weighted Redundancy	Weighted Hazards	Weighted Safety Components	Function Score	Total Weighted Function Score
1	3	8	20	2.50	1	0	3.00	6.06	10.56	0.00	7.43	21.07	16.37	0.00	44.86	134.58
2	3	3	5	1.67	2	0	8.00	2.00	9.56	0.00	19.80	6.95	14.82	0.00	41.57	124.70
3	2	4	10	2.50	7	1	7.00	6.06	8.32	10.00	17.33	21.07	12.90	25.00	76.29	152.58
4	3	5	7	1.40	10	0	6.00	1.36	3.93	0.00	14.85	4.73	6.09	0.00	25.67	77.00
5	4	6	16	2.67	7	1	5.00	7.25	8.41	10.00	12.38	25.19	13.04	25.00	75.61	302.43
6	4	6	7	1.17	3	1	5.00	1.06	8.34	10.00	12.38	3.69	12.93	25.00	54.00	215.99
7	5	2	6	3.00	5	0	9.00	10.00	9.43	0.00	22.28	34.75	14.62	0.00	71.65	358.23
8	4	6	12	2.00	1	0	5.00	3.25	10.30	0.00	12.38	11.29	15.97	0.00	39.63	158.54
9	4	6	11	1.83	4	1	5.00	2.56	9.04	10.00	12.38	8.90	14.01	25.00	60.29	241.14
10	4	2	3	1.50	4	0	9.00	1.56	8.20	0.00	22.28	5.43	12.71	0.00	40.41	161.66

Weight Factors			
2.475	3.475	1.550	2.500

Sum of Total Weighted Function Scores	1926.86
Sum of Consequences	36.00

Safety Index Score **53.52**

The weight factors for the Safety Index and Operability Index were evaluated to determine the sensitivity of the weight factors on the overall score. Through a process of iteration with the Monte Carlo analysis, the weight factors were adjusted such that one parameter was not a significant contributor to the overall score. Shown in Figure 69 is a distribution of Safety Index scores generated with a Monte Carlo analysis run of 10,000 points. For each individual score, a set of 10 functions contained random generated values for consequence, components, total number of components, hazards, and safety components; this approach was the same as described in Table 18. The range of the scoring was between 29.3 and 76.9 with an average of 52.7.

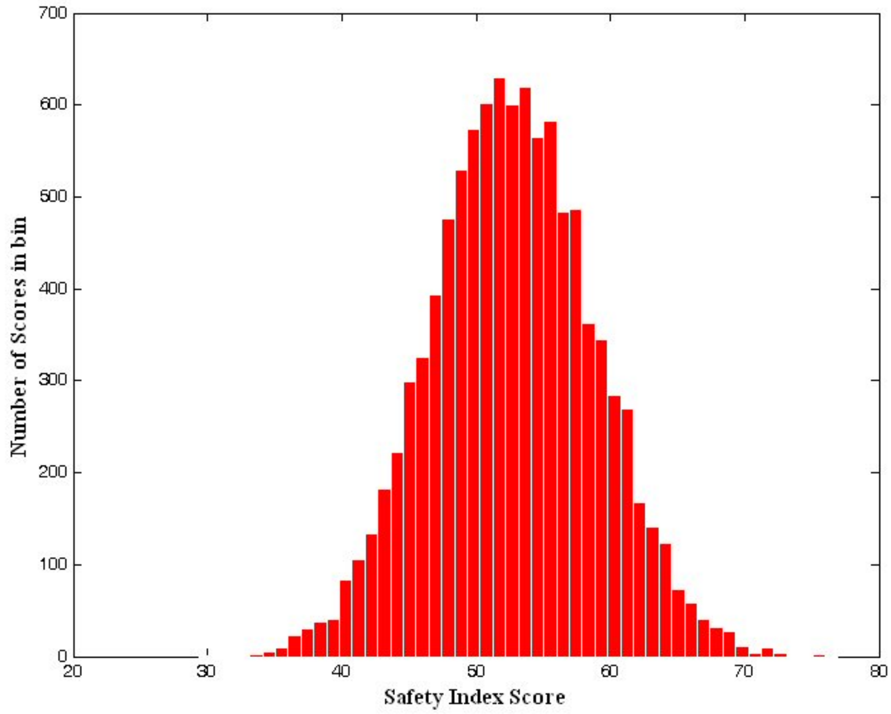


Figure 69: Distribution of Safety Index scores.

To determine the sensitivity of the weight factors, the Monte Carlo analysis was iterated to determine the weight factors. Shown in Table 19 are the results of the Monte Carlo analysis.

Table 19: Sensitivity of Safety Index weight factors.

Weight Factor Redundancy	3.475	0	3.475	3.475	3.475
Weight Factor Components	2.475	2.475	0	2.475	2.475
Weight Factor Hazards	1.55	1.55	1.55	0	1.55
Weight Factor Safety Components	2.5	2.5	2.5	2.5	0
Safety Index Average Score	**52.74**	**38.70**	**38.93**	**40.40**	**40.20**
	Difference	**-14.04**	**-13.81**	**-12.34**	**-12.54**

After a trial and error process, the final weight factors were decided based on the relative sensitivity and importance of the parameter. As the weight factors were set to zero, the Safety Index average scores decreased as expected. The difference relative to the original Safety Index score was the deciding variable in the determination of the weight factors. The differences among the analyses were judged to be adequate because the result demonstrated that all differences were very close to being equal.

The sensitivity of the weight factors for the Operability Index was examined in the same manner as the Safety Index. Because the Operability Index evaluates key operational parameters of the spacecraft configuration, the following parameters were randomly varied in a Monte Carlo analysis of 10,000 runs:

- Time – 0 to 36 hrs
- Crew – 1 to 4 crewmembers
- Payload – 0 to 1000 lbs
- Operability Functions – 0 to 20% of total spacecraft functions

Shown in Figure 70 is the distribution of Operability Index scores based on the previous parameters. The weight factors used in this analysis were:

- Time weight factor = 4,
- Crew weight factor = 3,
- Payloads weight factor = 2, and
- Operability Functions = 1.

The distribution of Operability Scores resulted in an average score of 49.53 with a maximum score of 90.9 and a minimum score of 9.57. The sensitivity of the weight factors is listed in Table 20.

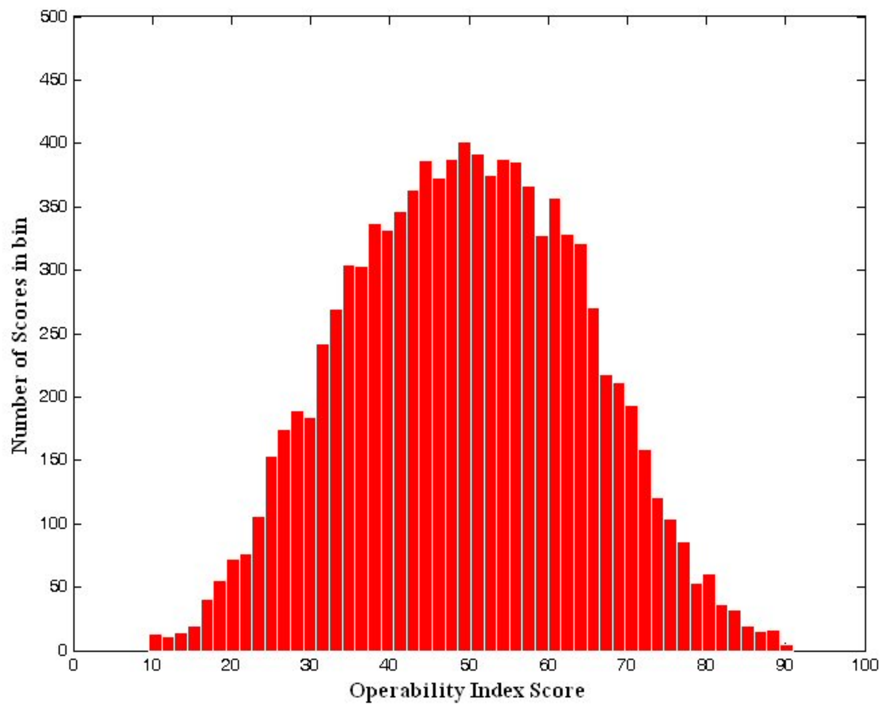


Figure 70: Distribution of Operability Index scores.

Table 20: Sensitivity of Operability Index weight factors.

Weight Factor Time	4	0	4	4	4
Weight Factor Crew	3	3	0	3	3
Weight Factor Payloads	2	2	2	0	2
Weight Factor Operability Functions	1	1	1	1	0
Operability Index Average Score	**49.53**	**29.67**	**30.85**	**39.59**	**48.47**
Difference		**-19.86**	**-18.68**	**-9.94**	**-1.06**

For the Operability Index, the weight factors were not adjusted from their importance ranking. The results of the sensitivity analysis show a similar sensitivity for Time and Crew. This result was judged to be adequate because increasing Time and Crew is a large operational feature of a spacecraft. The sensitivity to payload is approximately half of the previous two parameters and was judged to be appropriately ranked. The addition of operability functions was not a significant driver in the overall score and is due to the low weight factor.

6.6.2 Safety Index Comparisons

The resolution of the Safety Index to small changes in the spacecraft configuration is an important characteristic of the figure of merit. Using the baseline example in Table 18, a set of comparisons were investigated to determine the sensitivity of the Safety Index to changes in overall redundancy and total number of components. Shown in Table 21, the yellow columns represent the changes to the baseline example of Table 18. For a minimum functionality configuration, the functions are single string without redundancy and safety components. The Safety Index score for the minimum functional configuration is 29.44 compared to the baseline score of 53.52. The low score of the minimum functional configuration represents a lower boundary for the Safety Index and correlates well with the Monte Carlo analysis lower boundary of 29.3.

Table 21: Baseline example revised to a Minimum Functionality configuration.

Function	Consequence	Components	Total Number of Components	Average Redundancy	Hazards	Safety Components	Normalized Components	Normalized Redundancy	Normalized Hazards	Normalized Safety Components	Weighted Components	Weighted Redundancy	Weighted Hazards	Weighted Safety Components	Function Score	Total Weighted Function Score
1	3	8	8	1.00	1	0	3.00	1.00	9.90	0.00	7.43	3.48	15.35	0.00	26.25	78.74
2	3	3	3	1.00	2	0	8.00	1.00	8.60	0.00	19.80	3.48	13.33	0.00	36.61	109.82
3	2	4	4	1.00	7	0	7.00	1.00	4.30	0.00	17.33	3.48	6.67	0.00	27.47	54.93
4	3	5	5	1.00	10	0	6.00	1.00	1.10	0.00	14.85	3.48	1.71	0.00	20.03	60.09
5	4	6	6	1.00	7	0	5.00	1.00	4.10	0.00	12.38	3.48	6.36	0.00	22.21	88.82
6	4	6	6	1.00	3	0	5.00	1.00	7.90	0.00	12.38	3.48	12.25	0.00	28.10	112.38
7	5	2	2	1.00	5	0	9.00	1.00	6.30	0.00	22.28	3.48	9.77	0.00	35.52	177.58
8	4	6	6	1.00	1	0	5.00	1.00	9.60	0.00	12.38	3.48	14.88	0.00	30.73	122.92
9	4	6	6	1.00	4	0	5.00	1.00	7.40	0.00	12.38	3.48	11.47	0.00	27.32	109.28
10	4	2	2	1.00	4	0	9.00	1.00	6.80	0.00	22.28	3.48	10.54	0.00	36.29	145.16

Weight Factors			
2.475	3.475	1.550	2.500

Sum of Total Weighted Function Scores	1059.71
Sum of Consequences	36.00

Safety Index Score **29.44**

The same approach was used to calculate the Safety Index for a one failure tolerant configuration and a two failure tolerant configuration. The one failure tolerant configuration resulted in a score of 50.29 and the two failure tolerant configuration resulted in a score of 74.85. The difference between the minimum functionality configuration and a one failure tolerant configuration was 20.85 and the difference between a one failure tolerant configuration and a two failure tolerant configuration was 24.56. This exercise demonstrated the ability of the Safety Index to distinguish between failure tolerances, but the remaining comparison will investigate if the Safety Index can distinguish between smaller changes in the number of components.

Using the baseline example as shown in Table 18, the fourth function with a consequence of 3 and an average redundancy of 1.4 was used to evaluate the resolution of the Safety Index to smaller changes. A reduction in the number of components from 7 to 5 resulted in a decrease of 0.48. Shown in Table 22 is the revised baseline example.

Table 22: Baseline example with revised number of components in function 4.

Function	Consequence	Components	Total Number of Components	Average Redundancy	Hazards	Safety Components	Normalized Components	Normalized Redundancy	Normalized Hazards	Normalized Safety Components	Weighted Components	Weighted Redundancy	Weighted Hazards	Weighted Safety Components	Function Score	Total Weighted Function Score
1	3	8	20	2.50	1	0	3.00	6.06	10.56	0.00	7.43	21.07	16.37	0.00	44.86	134.58
2	3	3	5	1.67	2	0	8.00	2.00	9.56	0.00	19.80	6.95	14.82	0.00	41.57	124.70
3	2	4	10	2.50	7	1	7.00	6.06	8.32	10.00	17.33	21.07	12.90	25.00	76.29	152.58
4	3	5	5	1.00	10	0	6.00	1.00	1.00	0.00	14.85	3.48	1.55	0.00	19.88	59.63
5	4	6	16	2.67	7	1	5.00	7.25	8.41	10.00	12.38	25.19	13.04	25.00	75.61	302.43
6	4	6	7	1.17	3	1	5.00	1.06	8.34	10.00	12.38	3.69	12.93	25.00	54.00	215.99
7	5	2	6	3.00	5	0	9.00	10.00	9.43	0.00	22.28	34.75	14.62	0.00	71.65	358.23
8	4	6	12	2.00	1	0	5.00	3.25	10.30	0.00	12.38	11.29	15.97	0.00	39.63	158.54
9	4	6	11	1.83	4	1	5.00	2.56	9.04	10.00	12.38	8.90	14.01	25.00	60.29	241.14
10	4	2	3	1.50	4	0	9.00	1.56	8.20	0.00	22.28	5.43	12.71	0.00	40.41	161.66

Weight Factors			
2.475	3.475	1.550	2.500

Sum of Total Weighted Function Scores	1909.48
Sum of Consequences	36.00

Safety Index Score **53.04**

The previous analysis demonstrated that small changes in components may not have a large impact on the overall Safety Index score. However, if the example is continued and the consequence value is changed from 3 to 4, the Safety Index decreases to 52.14, which is a change of 1.38. Designers should be aware that small changes in the Safety Index could indicate significant changes in the spacecraft configuration.

6.6.3 Operability Index Comparisons

Four comparisons were conducted to determine the sensitivity of the Operability Index to small changes in time, crew, payload, and *Operability* functions. A baseline score was established using the following values:

- Time = 18 hours,
- Crew = 2 crewmembers,
- Payload = 500 lbs, and
- Operability functions = 10% of total spacecraft functions.

Shown in Table 23 are the results of the Operability Index comparisons. The first comparison increased the time from 18 hours to 20 hours. This change increased the score 2.22 from the baseline score of 46.00. For the second comparison, the increase in one crewmember is a significant increase of 7.50. The third comparison increased the payload mass by 100 lbs and resulted in a score increase of 2.00. The final comparison increased the percentage of *Operability* functions and did not increase the score significantly. The sensitivities demonstrated that small changes in time, crew, and payload were easily observed in the Operability Index score. The increase in *Operability* functions is not as significant. The primary reason for the small increase is due to the weight factor for *Operability* functions. In future analyses, an increase in the weighting of the *Operability* functions is needed. The overall results of the comparisons demonstrate the ability of the Operability Index to capture incremental changes in spacecraft *Operability*.

Table 23: Operability Index comparisons.

	Time	Crew	Payload	Operability Functions	Normalized Time	Normalized Crew	Normalized Payload	Normalized Operability Functions	Weighted Time	Weighted Crew	Weighted Payload	Weighted Operability Functions	Operability Index	Difference from Baseline
Baseline	18	2	500	10.00%	5.00	5.00	5.00	1.00	20.00	15.00	10.00	1.00	46.00	
Baseline + 2 hours	20	2	500	10.00%	5.56	5.00	5.00	1.00	22.22	15.00	10.00	1.00	48.22	2.22
Baseline + 1 crewmember	18	3	500	10.00%	5.00	7.50	5.00	1.00	20.00	22.50	10.00	1.00	53.50	7.50
Baseline + 100 lbs payload	18	2	600	10.00%	5.00	5.00	6.00	1.00	20.00	15.00	12.00	1.00	48.00	2.00
Baseline + 5% Operability Functions	18	2	500	15.00%	5.00	5.00	5.00	1.50	20.00	15.00	10.00	1.50	46.50	0.50

6.6.4 Lunar Ascent Module Scoring

The three Lunar Ascent Module case studies as presented in Chapter 5 were used to investigate the scoring methodology. The three configurations, Apollo nominal, Apollo One Man, and ESAS point of departure were scored for Safety Index, Operability Index, and Total Spacecraft score. Shown in Figs. 71 -73 are the Safety Index and Operability Index results for the Apollo, Apollo One Man, and ESAS configurations. Listed in Tables 25 -26 are the scores for Safety Index, Operability Index, and Total Spacecraft at the node locations.

Observing the results between the Apollo, Apollo One Man and ESAS configurations shows a range of Safety Index between 33 and 74. Many of the Safety Index levels are comparable among the three configurations. A one failure tolerance configuration among the three types reveals a safety score of approximately 52. The reason for the continuity is due to the number of components and functions among the three. The component list for the Apollo One Man is similar to the Apollo configuration and the ESAS configuration was derived from the Apollo configuration.

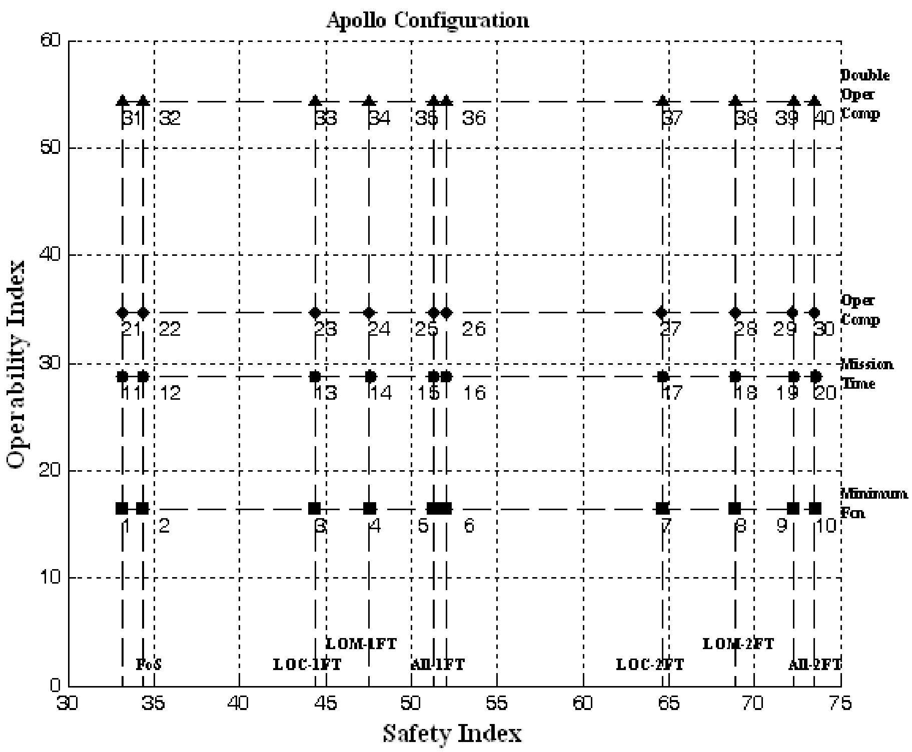


Figure 71: Safety Index and Operability Index scores for the Apollo configuration.

Table 24: Safety Index and Operability Index scores for the Apollo configuration.

Safety Index Scores at Node Locations Apollo Configuration	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition
	Safety Index Score				**Operability Index Score**				**Total Spacecraft Score**			
Minimum Functionality	33.16	33.16	33.13	33.16	16.48	28.73	34.67	54.34	27.60	31.69	33.64	40.22
Factor of Safety Addition	34.39	34.39	34.36	34.39	16.48	28.73	34.67	54.34	28.42	32.50	34.46	41.04
Consequence 5 Failure Tolerance (1FT) Addition	44.38	44.38	44.36	44.39	16.48	28.73	34.67	54.34	35.08	39.16	41.13	47.71
Consequence 4 Failure Tolerance (1FT) Addition	47.62	47.62	47.51	47.55	16.48	28.73	34.67	54.34	37.24	41.32	43.23	49.81
Safety Functionality Addition	51.34	51.34	51.26	51.29	16.48	28.73	34.67	54.34	39.72	43.80	45.73	52.31
Consequence 3 and Lower (1FT) Addition	52.06	52.06	52.01	52.05	16.48	28.73	34.67	54.34	40.20	44.29	46.23	52.81
Consequence 5 Failure Tolerance (2FT) Addition	64.63	64.63	64.58	64.62	16.48	28.73	34.67	54.34	48.58	52.66	54.61	61.19
Consequence 4 Failure Tolerance (2FT) Addition	68.91	68.91	68.84	68.89	16.48	28.73	34.67	54.34	51.43	55.51	57.45	64.04
Safety Functionality Addition	72.27	72.27	72.21	72.25	16.48	28.73	34.67	54.34	53.67	57.76	59.70	66.28
Consequence 3 and Lower (2FT) Addition	73.59	73.59	73.45	73.51	16.48	28.73	34.67	54.34	54.55	58.64	60.52	67.12

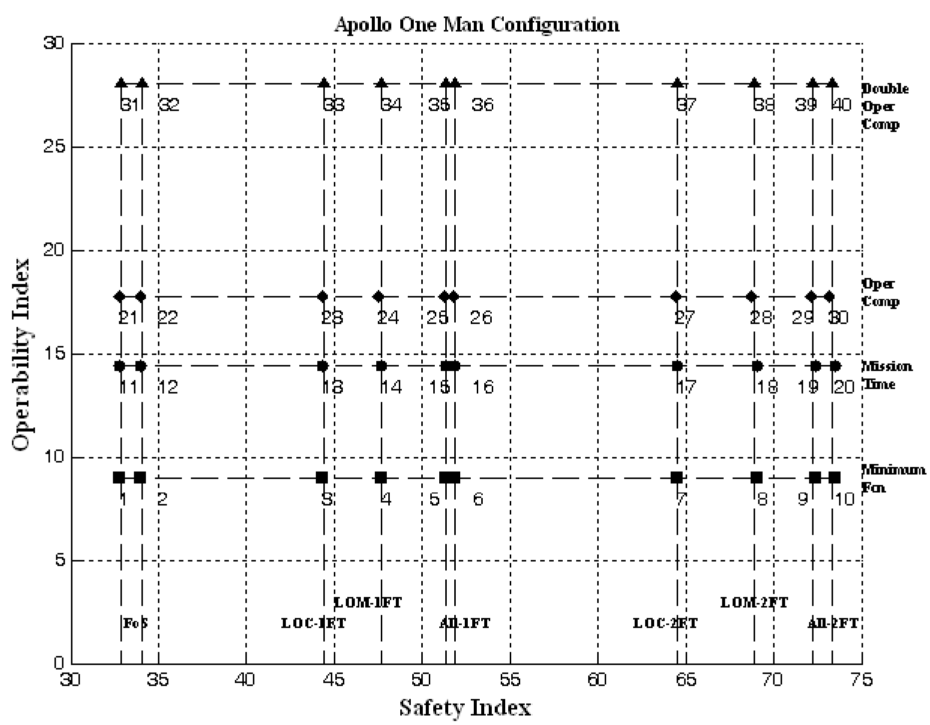


Figure 72: Safety Index and Operability Index for the Apollo One Man configuration.

Table 25: Safety Index and Operability Index for the Apollo One Man configuration.

Safety Index Scores at Node Locations Apollo One Man Configuration	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition
	Safety Index Score				**Operability Index Score**				**Total Spacecraft Score**			
Minimum Functionality	32.79	32.79	32.76	32.83	8.98	14.37	17.75	28.01	24.85	26.65	27.76	31.23
Factor of Safety Addition	34.02	34.02	33.98	34.06	8.98	14.37	17.75	28.01	25.67	27.47	28.57	32.04
Consequence 5 Failure Tolerance (1FT) Addition	44.32	44.32	44.31	44.38	8.98	14.37	17.75	28.01	32.54	34.34	35.45	38.92
Consequence 4 Failure Tolerance (1FT) Addition	47.66	47.66	47.54	47.64	8.98	14.37	17.75	28.01	34.76	36.56	37.61	41.10
Safety Functionality Addition	51.38	51.38	51.28	51.36	8.98	14.37	17.75	28.01	37.24	39.04	40.11	43.58
Consequence 3 and Lower (1FT) Addition	51.90	51.90	51.81	51.89	8.98	14.37	17.75	28.01	37.59	39.39	40.45	43.93
Consequence 5 Failure Tolerance (2FT) Addition	64.50	64.50	64.41	64.50	8.98	14.37	17.75	28.01	46.00	47.79	48.86	52.33
Consequence 4 Failure Tolerance (2FT) Addition	69.02	69.02	68.75	68.86	8.98	14.37	17.75	28.01	49.01	50.81	51.75	55.24
Safety Functionality Addition	72.40	72.40	72.13	72.24	8.98	14.37	17.75	28.01	51.26	53.06	54.00	57.50
Consequence 3 and Lower (2FT) Addition	73.47	73.47	73.19	73.31	8.98	14.37	17.75	28.01	51.97	53.77	54.71	58.21

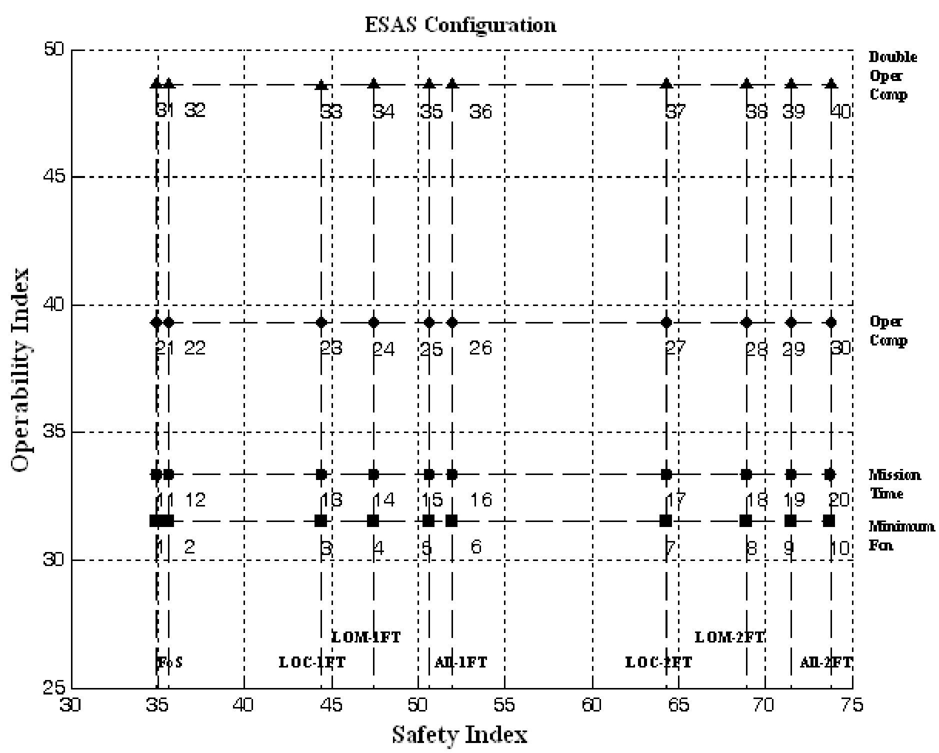


Figure 73: Safety Index and Operability Index scores for the ESAS configuration.

Table 26: Safety Index and Operability Index scores for the ESAS configuration.

Safety Index Scores at Node Locations ESAS Configuration	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition	Minimum Functionality	Mission Time Addition	Operability Components and Functions Addition	Double Operability and Mission Time Addition
	Safety Index Score				**Operability Index Score**				**Total Spacecraft Score**			
Minimum Functionality	34.94	34.94	34.94	34.94	31.53	33.39	39.31	48.61	33.81	34.43	36.40	39.50
Factor of Safety Addition	35.65	35.65	35.65	35.65	31.53	33.39	39.31	48.61	34.27	34.89	36.87	39.97
Consequence 5 Failure Tolerance (1FT) Addition	44.41	44.41	44.41	44.41	31.51	33.37	39.30	48.59	40.11	40.73	42.71	45.81
Consequence 4 Failure Tolerance (1FT) Addition	47.48	47.48	47.48	47.48	31.50	33.36	39.29	48.60	42.16	42.77	44.75	47.85
Safety Functionality Addition	50.67	50.67	50.67	50.67	31.50	33.36	39.29	48.59	44.28	44.90	46.88	49.98
Consequence 3 and Lower (1FT) Addition	52.01	52.01	52.01	52.01	31.52	33.37	39.31	48.61	45.18	45.80	47.78	50.88
Consequence 5 Failure Tolerance (2FT) Addition	64.30	64.30	64.30	64.30	31.51	33.37	39.30	48.60	53.37	53.99	55.97	59.07
Consequence 4 Failure Tolerance (2FT) Addition	68.89	68.89	68.89	68.89	31.51	33.36	39.30	48.60	56.43	57.05	59.03	62.13
Safety Functionality Addition	71.53	71.53	71.53	71.53	31.50	33.36	39.29	48.60	58.19	58.81	60.78	63.88
Consequence 3 and Lower (2FT) Addition	73.71	73.71	73.77	73.77	31.51	33.37	39.30	48.60	59.64	60.26	62.28	65.38

The differences among the three types of spacecraft are the *Operability* levels. Using the Apollo configuration as a baseline, the Operability for the ESAS minimum functionality configuration is almost twice as high as the Apollo configuration and this is due to the increased number of crew. However, because the ESAS configuration did not carry a payload and the mission time was approximately half of the Apollo configuration, the Operability Index score is comparable with Apollo scoring 34.67 and ESAS scoring 39.30 for a nominal *Operability* level. The results suggest that an ESAS configuration with two additional crewmembers, no payload, and 25% of the time is approximately equivalent to a nominal Apollo configuration for *Operability*. The Operability Index scores for the Apollo One Man configuration are approximately half of the nominal Apollo configuration. One should also keep in mind the scoring weights used for time, crew, and payload when evaluating the differences. Shown in Figs. 74 – 76 are the results of the Monte Carlo analysis for the tradespace zones.

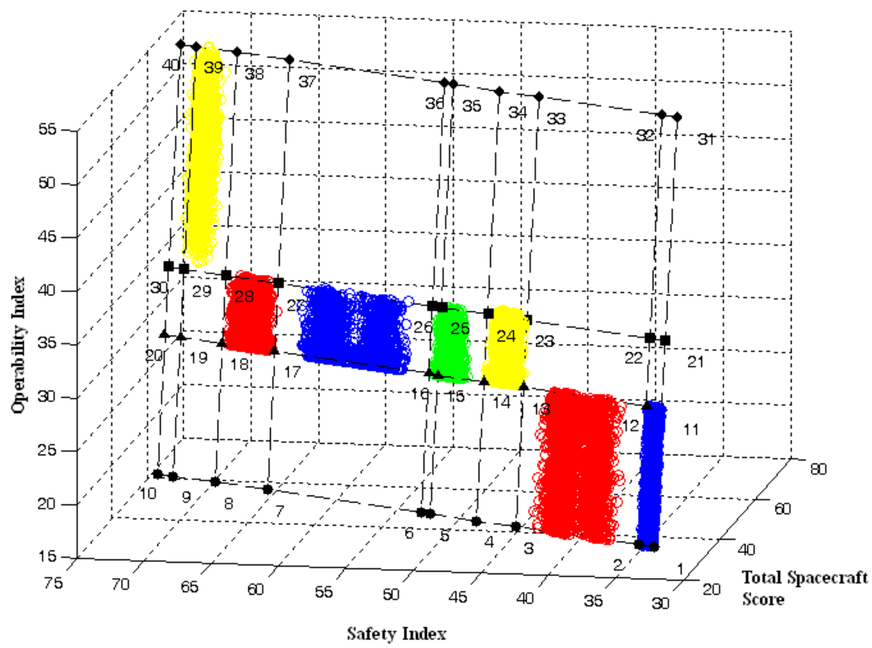


Figure 74: Safety and Operability Index tradespace for the Apollo configuration.

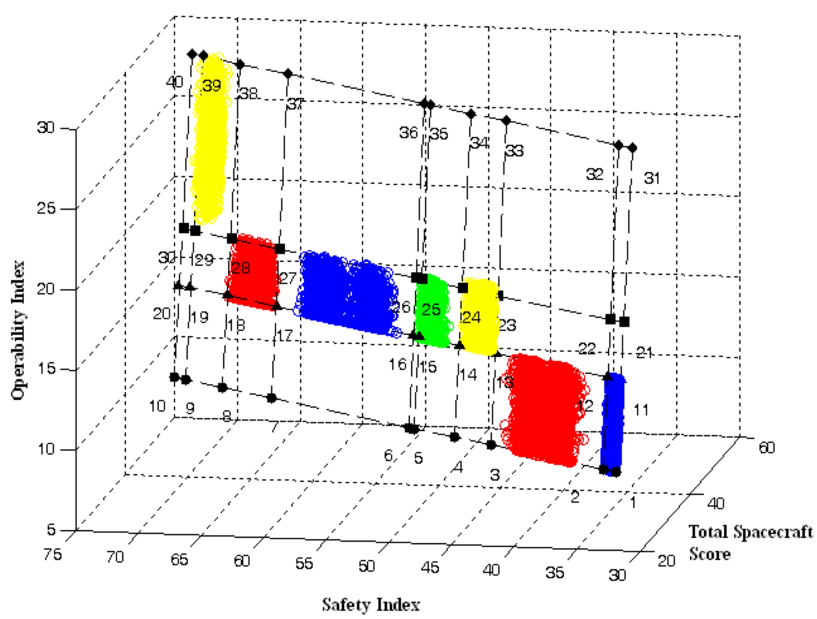


Figure 75: Safety and Operability Index tradespace for the Apollo One Man configuration.

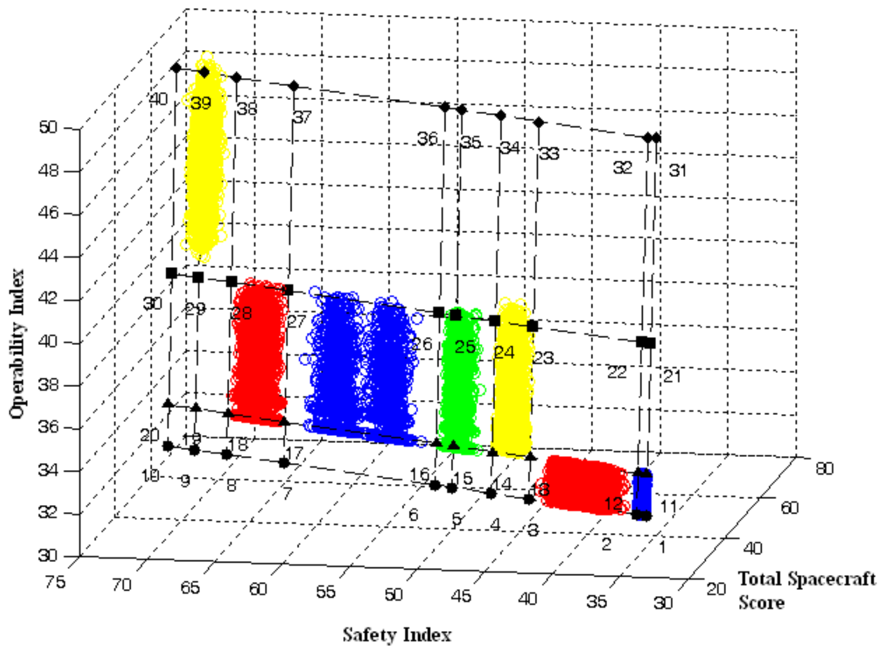


Figure 76: Safety and Operability Index tradespace for the ESAS configuration.

Listed in Table 27 are the results of the Monte Carlo tradespace analysis. The average and standard deviation of the Safety Index, Operability Index, and Total Spacecraft scores are calculated by zone. As shown in Table 27, the scores for the Safety Index by zone are comparable between the three spacecraft types. The difference in the Total Spacecraft score is due to the differences in Operability Index. A comparison of the Total Spacecraft scores between the three spacecraft types shows that the ESAS configuration scored slightly higher in the majority of the zones. The major driver for the ESAS score was the increased number of crewmembers. If the nominal mission time had been increased for ESAS, then the Operability Index would have outranked the Apollo configuration. The scoring weights for the Operability Index are critical for evaluating designs and comparing parameters such as crew, time, and payload.

Table 27: Safety Index, Operability Index, and Total Spacecraft scores by tradespace zone.

Apollo Configuration						
Zone	Average Total Spacecraft Score	Tot S/C Score Standard Deviation	Average Safety Index	Safety Index Standard Deviation	Average Operability Index	Operability Index Standard Deviation
1	30.0	1.2	33.7	0.2	22.6	3.5
2	34.0	1.5	39.7	1.5	22.6	3.6
3	40.9	0.7	46.0	0.5	30.8	1.8
4	43.8	0.6	50.5	0.4	30.4	1.7
5	49.0	1.6	58.3	2.2	30.4	1.7
6	54.6	0.8	66.7	0.8	30.4	1.7
7	62.5	1.4	71.2	0.4	45.0	4.2
Apollo One Man Configuration						
Zone	Average Total Spacecraft Score	Tot S/C Score Standard Deviation	Average Safety Index	Safety Index Standard Deviation	Average Operability Index	Operability Index Standard Deviation
1	26.1	0.5	33.3	0.2	11.6	1.6
2	30.3	1.1	39.6	1.5	11.7	1.6
3	35.8	0.5	45.9	0.6	15.7	1.0
4	38.7	0.4	50.3	0.4	15.5	0.9
5	43.9	1.5	58.1	2.2	15.4	0.9
6	49.6	0.6	66.7	0.8	15.4	0.9
7	55.2	0.8	71.2	0.4	23.3	2.1
ESAS Configuration						
Zone	Average Total Spacecraft Score	Tot S/C Score Standard Deviation	Average Safety Index	Safety Index Standard Deviation	Average Operability Index	Operability Index Standard Deviation
1	34.3	0.2	35.3	0.1	32.5	0.5
2	37.5	1.0	40.0	1.4	32.5	0.5
3	42.5	0.7	46.0	0.5	35.6	1.7
4	45.3	0.6	50.3	0.4	35.1	1.7
5	50.4	1.6	58.1	2.2	35.0	1.7
6	56.1	0.8	66.6	0.8	35.1	1.7
7	62.4	0.7	71.3	0.4	44.6	1.8

6.6.5 Advantages and Limitations

The weighting factors of a figure of merit are critical to the outcome of the score and a bias toward a single objective can overlook potential solutions. The comparison of the three types of Lunar Ascent Modules revealed similarities and differences among the scores that were driven by weighting factors and functionality. The advantage of this approach is that it allows designers to simultaneously compare *Safety* and *Operability* without a bias toward mass. The Safety Index is not focused entirely on high consequence functions and scores the entire spacecraft including Design for Minimum Risk components. The Operability Index provides a measure of the spacecraft *Operability* to top level mission objectives and can be used to compare different types of spacecraft configurations. The two scores can be combined into a Total Spacecraft score for relative comparisons.

The limitations of the figures of merit are primarily due to the resolution of the Safety Index scale. Small changes at the component level may not be as noticeable as large changes in subsystem failure tolerance. However, this approach is intended to provide designers with a quantifiable value such that configuration details can be explored prior to detailed risk and reliability analysis of a preferred spacecraft configuration.

6.7 CONCLUSIONS

The two figures of merit, the Safety Index and Operability Index are methods that can be utilized by designers and decision makers to investigate spacecraft configurations prior to detailed risk and reliability analyses. The Safety Index points to an increase in failure tolerance and includes parameters of consequence of loss of function, complexity, similar and dissimilar redundancy, hazards, safety equipment, and Design for Minimum Risk items. The Operability Index provides a measure of the level of operational functionality in the spacecraft configuration and can be used to judge if a candidate configuration will meet top level requirements. The Safety Index and Operability Index scoring methodologies are based upon quantitative information and can be easily reproduced. The weighting factors for the Safety Index were specified according to sensitivities in the scoring such that one parameter did not influence the overall score. The weight factors for the Operability Index were chosen based upon preferences in the design. The results demonstrated that the weighting factors for the Operability Index were the likely drivers for the magnitude of the scores and influence the outcome. The advantage of the approach is that *Safety* and *Operability* are scored simultaneously and the two indices can be combined into a Total Spacecraft score for relative comparison among competing designs.

6.7.1 Acknowledgements

This work would not have been possible without the educational support from Lockheed Martin Corporation and Sierra Nevada Corporation. The experience of maturing conceptual designs and designing flight hardware for the Orion and Dream Chaser programs played a key role in the development of this methodology. Special recognition is given to Dr. John Turner for helpful feedback and advice concerning the development of the figures of merit.

CHAPTER 7

DISCUSSION

7.1 SUMMARY OF RESEARCH

This section provides a summary of the research conducted in this dissertation. Chapter 2 reviewed the literature on Systems Engineering, Conceptual Design, Multidisciplinary Design Optimization, Risk Based Design, Minimum Functionality Design, and Lunar Spacecraft Development. Chapter 3 presented the Minimum Functionality design methodology for determining *Safety* and *Operability* mass in human spacecraft. The three remaining chapters explored the Minimum Functionality design methodology in greater detail. Chapter 4 described the research activities and development of rapidly reconfigurable prototypes of conceptual human spacecraft. Chapter 5 explored the Minimum Functionality design methodology tradespace and developed a mass addition process focused on the early stages of conceptual design. Finally, Chapter 6 introduced two figures of merit for evaluating spacecraft configurations based on the levels of *Safety* and *Operability*. Sections 7.1.1 to 7.1.5 provide a detailed summary of Chapters 2 through 6.

The methods described in Chapters 3 – 6 are combined in a conceptual design process that couples with typical approaches used within NASA and industry. The minimum functionality design methodology serves as a pre-analysis activity to gather information and reduce uncertainty before detailed modeling and analysis.

7.1.1 Summary of Chapter 2

Chapter 2 reviewed literature topics on spacecraft design and development. Beginning with the fundamentals of Systems Engineering, the literature review progressed through the many subject areas related to the design, development, and maturation of human spacecraft. The final part of Chapter 2 summarized previous lunar spacecraft programs to provide the reader with background information related to the case studies in the following chapters.

It is commonly known that the conceptual design phase is one of the most important phases for determining the costs of a program and yet the processes used in conceptual design are often unstructured. Systems Engineering is a methodology widely used by many aerospace organizations in the design, development, and operation of human spacecraft. The discipline of Systems Engineering emerged from technical projects after World War II and was utilized in the Apollo program. Many organizations view Systems Engineering differently and many definitions are not consistent. However, the practice of Systems Engineering is well established in industry and will continue to be utilized in the development of future human spacecraft.

The challenge during conceptual design is obtaining knowledge in order to develop feasible solutions. The use of "Boundary Objects" as representations of knowledge is one method to communicate and present information. Physical prototyping is a form of a Boundary Object and has been historically used to explore concepts and gain knowledge in the early stages of development. The use of CAD models has largely replaced physical prototyping because CAD models can be used to represent geometry and concepts and is easily updated. However, one limitation of CAD is difficulties associated with evaluating human factors of a spacecraft design. Virtual reality is being used to simulate physical aspects of a design in a digital world.

The uncertainty in the conceptual design phase can lead to mass growth in later phases of development. Mass growth during conceptual design is a common problem and has been observed on many successful and unsuccessful spacecraft programs. The typical approach for dealing with uncertainty is to use standard mass growth allowance tables to predict mass in the later phases of development.

Multidisciplinary Design Optimization is an area of research that has largely been constrained to the research community. The introduction of MDO in the early 1990's showed great promise for solving and optimizing complex engineering problems. However, because of the complexities of linking discipline specific codes, the use of MDO has not become widespread in the aerospace industry and is due to the computational expense of coupling high fidelity models and the time required to generate optimized solutions. Researchers have recognized this limitation and developed surrogate "Metamodels" or models of a model to reduce the computational burden. The development and visualization of metamodels is an active area of research in the field of MDO.

A recent approach proposed by NASA for the development of human spacecraft is Risk Based Design. In Risk Based Design, the goal is to couple the rapid design activities with risk analysis in order to evaluate the safety and guide the design as it matures. The difficulty with this approach as observed by some researchers is the lack of complete reliability knowledge and the speed at which the design progresses in the early phases. Reliability analysis such as Probabilistic Reliability Analysis requires detailed configuration and reliability information in order to quantify safety in a proposed design. In the early stages of conceptual design, many factors such as the particular type of technology used in subsystems, the probability of failure in components, and the exact layout and integration of the subsystem components may not be fully defined. Other reliability analysis methods such as FMEA, FTA, and ETA are typically used to quantify reliability in the early stages of conceptual design.

Minimum functionality design is an approach where a baseline configuration is defined before trading other factors in a spacecraft design. Minimum functionality gained attention due to its use by the Altair Lunar Lander Project Office. The design approach for minimum functionality in the development of the Altair Lunar Lander started with a single point baseline design point of departure for cost and risk trades in later design cycles. The minimum functionality approach was also used in the Orion Zero Baseline Vehicle to reduce mass prior to SDR. According to NASA's Design, Development, Test, and Evaluation Considerations for Safe and Reliable Human Rated Spacecraft Systems, a minimum functional design is the simplest, most robust, and highest performance design option as the starting point for assessing fault tolerance. The exact definition of minimum functionality is interpreted differently among various groups.

Human Rating Requirements for Space Systems are defined in NASA NPR 8705.2B. In this document, the human rating certification process, certification requirements, and technical requirements for human rating are specified. The set of human rating requirements is likely the minimum set of requirements that must be satisfied in order to achieve human rating certification.

A human lunar spacecraft, commonly known as a Lunar Lander or Lunar Habitat is very different from other spacecraft designs such as capsules and lifting bodies. Although there are many similarities to conventional LEO operational spacecraft, a lunar spacecraft must operate in a different set of environments on the surface of the moon with a larger set of requirements.

The goal of the Apollo program was to put a man on the moon and return him safely to Earth. The most remarkable aspect of this achievement is that the LM evolved from a conceptual idea to operational hardware on the lunar surface in less than 9 years. The major factors that drove LM mass during the preliminary design phase were reliability requirements, mission operational requirements, and configuration definition. The proposed design objectives of the Constellation Altair Lunar Lander (formerly known as LSAM) were more complex than the Apollo LM. The Altair Lunar Lander was designed to carry a crew of four, be able to land at any location on the moon, and remain on the surface for up to 2 weeks.

The Constellation program suffered many challenges and was officially cancelled in June 2011. At the current time, the direction of human exploration is being reexamined and the focus of commercial partners to provide services to the International Space Station along with the current development of the Orion Multi Purpose Crew Vehicle and Space Launch System (SLS) as an exploration vehicle is a step in the right direction toward achieving spaceflight beyond LEO. Due to the cancellation of Constellation, no plans for Lunar Lander development are currently proposed by NASA; but it is hoped that the development of the Orion and SLS programs will eventually spur a new Lunar Lander program to continue the legacy of the Constellation Altair program.

7.1.2 Summary of Chapter 3

A systematic methodology was presented for defining a minimum functionality baseline configuration of a human spacecraft. In order to estimate a lower bound for the spacecraft mass, a set of essential functions are coupled to single string subsystems with zero fault tolerance. This minimum functionality baseline is defined to meet the physical requirements needed to transport the crew to the target destination and ensure their physiological needs are met; but without margin, redundancy or factor of safety. This constitutes a set of 'non-negotiable' requirements based on fundamental parameters derived from *Physics* and *Physiology*. By definition, this represents a technically feasible solution, but results in the 'highest risk' design. Mass additions beyond the minimum functional configuration are allocated to increase *Safety* through redundancy, fault tolerance or factor of safety, or to increase *Operability* through additional mission functionality or improved human-system interfaces, and are determined by risk analyses and design trade studies. The methodology was used to analyze a range of lunar ascent stage spacecraft configurations and a process was developed to allow systematic estimation of mass for the specified spacecraft subsystems. The modeled results are verified by comparison to actual subsystem mass of the Apollo Lunar Module ascent stage.

A mass and trajectory analysis code named the Conceptual Lunar Ascent Module Program was developed in MATLAB to calculate the minimum functionality baseline and to assess relative increases in *Safety* and *Operability* mass for various Lunar Ascent Module spacecraft configurations. The program is focused on the lunar ascent phase of the overall lunar architecture mission. The intent of the program was to determine how changes in subsystem mass affect the overall spacecraft mass required to return from the lunar surface. The CLAMP code approaches the development of a conceptual spacecraft design in a bottom-up design philosophy using individual components and heuristics matched to spacecraft functions. The program played a key role in developing the proposed methodology in order to understand how closely coupled the *Safety* and *Operability* mass is to the total spacecraft baseline mass.

The Minimum Functionality design methodology establishes a tradespace of configurations that visually represent the relationship between total spacecraft, *Safety*, and

Operability mass. It is hoped that this minimum functional baseline design methodology will generate interest among the spacecraft design community and serve as a useful guideline in the development of future human spacecraft.

7.1.3 Summary of Chapter 4

This chapter describes the prototyping research activities at the University of Colorado from the fall of 2006 until the spring of 2009. Many lessons were learned as a result of this prototyping activity and one of the significant aspects of rapidly reconfigurable prototyping is that it integrates well into a minimum functionality design methodology. The early research efforts beginning in 2006 have led to a project based design curriculum for human spacecraft conceptual design at the University of Colorado.

A full-scale mockup of a Lunar Lander habitat was constructed based on the dimensions defined in NASA’s Exploration Systems Architecture Study. An initial goal of the project was to establish methods and procedures for constructing a rapidly reconfigurable engineering prototype while concurrently using the analogue as a means of developing systems-level requirements based on anticipated operational concepts. The use of cost effective materials in the mockup provided a simplified approach for construction of system concepts and readily allowed subsequent configuration changes. The application of rapid prototyping provides a means of incorporating a hands-on ‘human in the loop’ component to spacecraft system design. This effort was originally intended to be used to help evaluate vehicle configuration options, determine subsystem mass and volume budgets, reduce risk factors and derive requirements before the Lunar Lander preliminary design review. The lessons learned in the initial prototyping activities were developed into a conceptual design process based on a minimum functionality design methodology. This process allows designers to fully investigate the design configuration before the development of CAD models and higher fidelity models such as CFD and FEA.

One purpose of studying rapidly reconfigurable prototyping was to learn what can be gained in the approach and the limitations. The design configurations constructed were based on ESAS Lunar Module concepts and provided a point of departure for understanding how the subsystems interact. The experience of building prototype hardware helped to shape the minimum functional design methodology. One of the limitations to prototyping is that it focuses on a single spacecraft configuration and the study of many different configurations takes additional time. However, when the prototyping activity is coupled with the Minimum Functional Design Methodology, the investigation of many more configurations is possible and the rapidly reconfigurable prototype served as validation of the analysis assumptions.

After the conclusion of the Lunar Ascent Module prototyping activity in the spring of 2009, the next phase of spacecraft prototyping began developing subsystem layouts and configurations for the Commercial Crew Development program. Although this author was not involved in the prototyping activities beyond 2009, the processes, techniques, and materials were recycled for other spacecraft applications. The collaborative work of many participants has grown from a seemingly small spacecraft design activity into a curriculum for training future aerospace explorers.

7.1.4 Summary of Chapter 5

In the context of human spacecraft, the conceptual design phase is critical for determining the success or failure of a proposed design. There are many challenges associated with developing a human spacecraft to perform the operational aspects of a mission and bring the crew home safe. A central problem in the early stages of development is due to uncertainty and designs are initially based on heuristic data or designer experience until further analysis or testing is available. In today's economic environment, additional testing or analysis may not be feasible in the early stages of development. Although this approach is typical in human spacecraft development, it also introduces potential uncertainty and issues in the conceptual design that might not be discovered until much later in development. However, the risks associated with any conceptual design will always be present until hardware is built and flown. The goal of a conceptual design process is to investigate and develop feasible solutions to a given problem. Conceptual designs are developed using the creative skills of designers and design teams.

The process described in this chapter introduces conceptual design tradespace methods for evaluating concepts based on the levels of *Safety* and *Operability* in a spacecraft configuration. The tradespace provides the design team with an overall picture of the conceptual spacecraft design configurations so that designers can identify areas for increasing safety and reducing risk while maintaining requirements for mass. Although this approach is focused on early conceptual development, the process can be used as the design matures into preliminary design.

Because many unknowns exist due to integration, complexity, and uncertainty in requirements, it can be a challenging time for correctly estimating the mass of a spacecraft.

However, a bottom up philosophy forces designers to consider the low level functions in the design where candidate technologies are chosen and evaluated. As part of the evaluation process, the overall vehicle configuration must be considered in addition to subsystem discipline specific trade studies so that designers can understand the impact of decisions on the spacecraft configuration. Incorrect assumptions in the conceptual design phase can lead to redesign or reduced performance.

Three designs of Lunar Ascent Modules were used as case studies to demonstrate and validate the methodology. These configurations included the nominal Apollo Lunar Ascent Module, a reduced Apollo One Man configuration and an ESAS derived configuration. The purpose of using the three concepts was to demonstrate and explore the mass addition process for different types of spacecraft designs.

Although mass is a key driver in any spacecraft design, the spacecraft must be safe and perform the intended mission. The mass addition and tradespace exploration process described in this work is one method that could be coupled with reliability analysis or other figures of merit for improving the spacecraft design.

7.1.5 Summary of Chapter 6

Two figures of merit, the Safety Index and Operability Index were derived to provide designers with a scoring system that evaluates the entire spacecraft configuration. The figures of merit are not intended to replace typical reliability based analysis methods; but provide designers with a complementary method for evaluating the safety and operational aspects of a spacecraft configuration. Based upon traditional methods for functional analysis of early spacecraft conceptual designs, the Safety Index scores the entire spacecraft upon consequence of loss of function, complexity of subsystem components, similar and dissimilar redundancy, and potential hazards. Components that are categorized as Design for Minimum Risk are included in the overall spacecraft Safety Index. The Operability Index scores are based upon the mission time, crew size, payloads, and operational functionality and provide a measure of the spacecraft configuration against top level mission objectives. The Safety Index and Operability Index can be combined into a Total Spacecraft score such that comparisons can be made against competing spacecraft types. Three case studies of human spacecraft designs were used to demonstrate and validate the scoring methodologies.

The Safety Index and Operability Index scoring methodologies are based upon quantitative information and can be easily reproduced. The weighting factors for the Safety Index were specified according to sensitivities in the scoring such that one parameter did not influence the overall score. The weight factors for the Operability Index were chosen based upon preferences in the design. The results demonstrated that the weighting factors for the Operability Index were the likely drivers for the magnitude of the scores and influenced the outcome. The advantage of the approach is that *Safety* and *Operability* are scored simultaneously and the two indices can be combined into a Total Spacecraft score for relative comparison among competing designs.

7.2 A CONCEPTUAL DESIGN PROCESS UTILIZING MINIMUM FUNCTIONALITY AND RAPIDLY RECONFIGURABLE PROTOTYPING

The design and analysis methods described in Chapters 3 - 6 can be combined into a conceptual design process for exploring and maturing candidate designs. The foundation of the approach is the Minimum Functional design methodology that establishes a baseline configuration for trading *Safety* and *Operability*. These methods serve as a pre-analysis step in the conceptual design process to identify candidate solutions before significant resources are dedicated to high fidelity modeling such as CAD, CFD, and FEA. Shown in Figure 77 is a diagram of the conceptual design process.

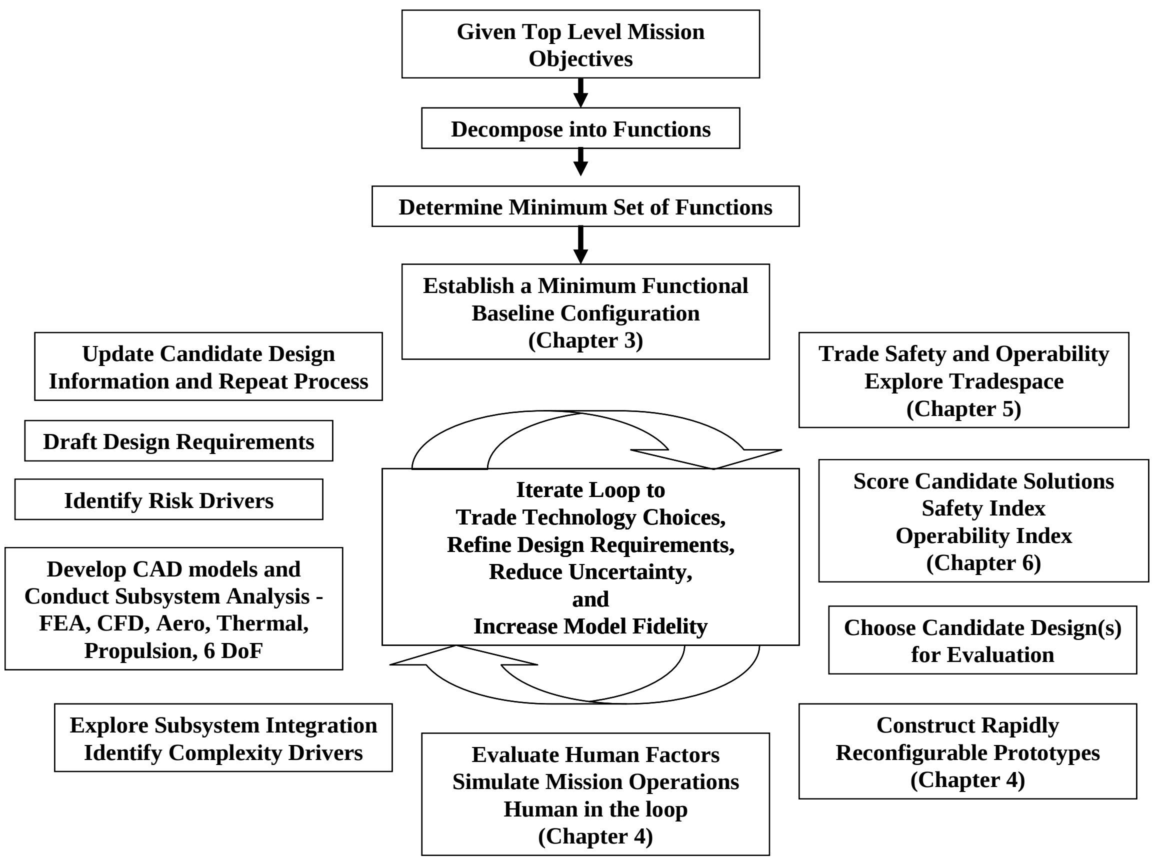


Figure 77: The Minimum Functionality Conceptual Design Process.

The conceptual design process begins with a set of given top level mission objectives. These objectives are decomposed into functions for the activities the spacecraft has to perform in order to accomplish the mission. Within the overall set of functions, a set of minimum functions can be derived to establish a minimum functional baseline configuration. The minimum functional baseline configuration is established according to the guidelines in Chapter 3. After the spacecraft geometry has been chosen and the minimum functionality baseline established, the next step in the process is to trade *Safety* and *Operability* using the guidelines described in Chapter 5 for mass additions and tradespace exploration. Based upon the results of the tradespace, the next step scores candidate solutions according to the figures of merit described in Chapter 6. This step allows designers to focus on the most preferred spacecraft configuration for further examination.

Using the information gathered from the Minimum Functionality design analysis, a spacecraft configuration is chosen for further investigation using physical prototyping. The particular spacecraft configuration could be a minimum functionality baseline or representative of a configuration with added *Safety* components or *Operability* components. A rapidly reconfigurable prototype is constructed to further study the conceptual design as it relates to system and subsystem interfaces, layout, habitability, and human factors. Much information can be gathered during this activity that will provide valuable information to designers before the development of CAD models. The initial engineering prototypes should be based on the specified volumetric layout for subsystems. This activity forms an experimental basis for determining the form and fit of the subsystem interfaces with the overall vehicle as well as the crew. Because the prototype is designed to be easy and inexpensive to reconfigure, numerous parameters can be varied quickly to determine optimum layouts for subsystems, interfaces, and human machine interactions. Manufacturability and maintenance requirements of the system and subsystems could be independently investigated. This prototype now serves as a Boundary Object for the spacecraft design configuration and is intended to facilitate communication between multi-disciplinary design groups working on the project. It serves as a means of enhancing knowledge transfer regarding subsystem interfaces, vehicle layout, mass and volume analyses, and operational needs, including accessibility for performing maintenance tasks, which often results in underestimated operational impacts (Russell and Klaus, 2007).

In order to define the total vehicle mass required as a function of the full mission timeline, consumables and expendables needed by subsystems such as Crew Accommodations, Power, Environmental Control and Life Support System and Extravehicular Activity must be re-evaluated, along with any supporting mass required for enabling the science and exploration objectives. Using the baseline vehicle mass budgets previously analyzed in the minimum functionality analysis, these additional mission duration- dependent consumable and expendable mass requirements are added to the prototype to simulate mission operations.

Using the prototyping activity to study human interaction and subsystem layouts can be used to lead or supplement concurrent CAD efforts. As the prototype develops in fidelity, detailed CAD modeling will parallel the construction to develop digital prototypes of the spacecraft anchored to the physical Boundary Object. Components that are not easily modeled (or necessary) in the physical mockup could be included in the CAD, such as radiation protection, passive thermal control and micrometeoroid protection on the outer shell of the spacecraft. Mass and center of gravity calculations of the vehicle structure would include all actual materials and components, and be analyzed during the CAD phase. After completion of a 3-D CAD model, an FEA model evaluates the spacecraft mass based on material choices and verifies structural integrity of the spacecraft during all operational profiles from pre-flight testing to end of mission. Including standard factors of safety to the structural analysis will identify areas within the structure that need iteration to optimize mass while maintaining safety requirements of the structure and interfaces. The FEA analysis also serves as a verification of the physical prototyping activity.

The final step in the iterative design loop is the development of design requirements for the system and subsystems. Using the previous analyses, critical design drivers can be identified for further optimization. As the fidelity of the design increases, the potential failure modes in the spacecraft can be identified and their integrated effects analyzed. From this collective information, trade studies into risk reduction versus performance optimization can be conducted in a risk based design approach.

The first round of the process concludes with the updating of the candidate design information into the original minimum functionality baseline configuration. The entire process is repeated such that areas of uncertainty can be further explored. The second and following rounds use the previous information to increase model fidelity and trade additional technology choices.

7.2.1 Summary of Conceptual Design Process

The conceptual design process identifies and explores issues before significant cost and effort is incurred in later stages of the design. Alternative concepts within the design are readily explored using the methods described in Chapters 3 - 6 and requirements are derived based on information gathered through analysis and demonstration in the prototyping activity instead of solely relying on expert opinion and heuristics.

Given a set of top level objectives, a functional decomposition is developed to identify the functions required for the mission. These top level functions are used to develop and explore concept vehicle architectures. In the example of a Lunar Mission, there are many systems that must interact together to make the mission successful (i.e. Launch Vehicles, Spacecraft, Landers, etc.). After the baseline architecture has been chosen, a minimum set of functions are identified to establish a Minimum Functionality baseline configuration for trading *Safety* and *Operability* using the methods described in Chapter 5 and 6. Because human spacecraft are highly coupled systems, small changes in one subsystem can disproportionately affect the mass and performance of the entire spacecraft. This in turn impacts mission operations, risk, reliability, and other factors. The use of a Minimum Functionality design approach coupled with rapidly reconfigurable physical prototypes provides a cost effective method for including a 'human in the loop' at the earliest stages of the conceptual design. Using data gathered from a physical prototype that serves as a Boundary Object, models can concurrently be used to analyze vehicle layout and performance characteristics, and allow requirements to be refined as the design process incrementally progresses. As the iteration loops progress, the fidelity of the models will increase and the design requirements will become increasingly better defined. The emphasis on physical prototyping in the process design loop provides the opportunity for assessing requirements and configuration options and serves as a means of uncovering design issues early in the program when alternative solutions are still relatively easy and inexpensive to implement.

As diminishing returns from the iterations are reached, the conceptual design process will conclude with various detailed, high fidelity analytical models in place and a physical engineering prototype that can serve as a Boundary Object for future work. The concurrently developed mockup provides a physical anchor for the detailed CAD drawings of the spacecraft, related structural and thermal analyses, mission operations evaluations, risk and reliability indices, and draft requirements. The amount of information gathered as a result of this conceptual development process is aimed at reducing development costs of the spacecraft development process as it moves forward into the preliminary design phase. A key premise of this concept is to gather as much 'hands on' operational and detailed analytical information about the spacecraft design as possible, in the most cost effective manner practical, well before the actual flight hardware design is locked for testing and qualification.

7.3 FUTURE RESEARCH

Based on the results of this dissertation, the most promising areas of future research are related to scoring spacecraft design configurations using the Safety Index and Operability Index. The two indices are coupled in a spacecraft configuration and should be evaluated simultaneously when comparing different concepts. The research presented in this dissertation introduces the figures of merit and how they are used in scoring candidate solutions. A more detailed study of the two figures of merit is recommended for future research.

7.3.1 Safety Index

Of the two figures of merit, the Safety Index was the most complicated to develop. Because the accepted approach for quantifying safety in the aerospace industry is based upon reliability methods, the introduction of another figure of merit is likely to be received with skepticism until proven through detailed analysis. As the Safety Index was being derived using classical reliability based approaches, a comparison between spacecraft reliability and Safety Index scores was attempted but was not fully explored because the information required was more than the scope of this dissertation. However, if certain assumptions are made concerning the reliability allocation of components within the spacecraft, the Safety Index score can be tuned to provide trends in reliability. Additional research is needed to fully examine how the Safety Index can be correlated to reliability. The Safety Index is not intended to replace reliability based analyses, but provide designers and decision makers with a scoring system that can be easily adopted in order to predict reliability in a spacecraft.

7.3.2 Operability Index

Unlike the Safety Index, the Operability Index is not an easily defined figure of merit. The Operability Index is a measure of the spacecraft configuration against top level mission objectives that can change due to other factors such as mass, time, and safety. The weighting factors for the Operability Index were the most significant contributors to the overall score and more detailed research is necessary to determine user preferences in a spacecraft. The weighting factors used in this dissertation was an example of what decision makers might prefer and the scoring results reflected the preferences. Additional research for the Operability Index would focus on developing realistic target goals for the normalization of the parameters and methods used to weight the decisions properly. The use of an Analytic Hierarchy Process is suggested as a starting point for maturing the Operability Index. The *Operability* of a spacecraft is typically quantified last in the spacecraft design, but as observed in the results of this dissertation are an important contributor to the overall spacecraft design and should be considered simultaneously with other parameters. The Operability Index coupled with the Safety Index can provide designers with an overall scoring methodology for evaluating and trading early conceptual designs.

7.4 CONCLUSIONS

The research efforts in this dissertation focused on reducing uncertainty in the conceptual design phase through a process of establishing a minimum functionality baseline before trading *Safety* and *Operability* in a spacecraft configuration. The challenge in any spacecraft development process is how to combine the many parts into a working design that complies with many requirements for top level mission objectives, safety, and mission success. The design methodologies presented in this dissertation provides designers and decision makers with a larger view of candidate design concepts. The following lists of items are significant contributions to the field of human spacecraft development:

- Defined the four fundamental parameters in all human spacecraft as *Physics*, *Physiology*, *Safety*, and *Operability* (Chapter 3);
- Defined the minimum functional baseline configuration based on *Physics* and *Physiology* (Chapter 3);
- Used rapidly reconfigurable prototypes for the study of human factors before detailed CAD and analysis modeling (Chapter 4);
- Developed tradespace exploration methods for quantifying levels of *Safety* and *Operability* (Chapter 5);

- Developed a mass addition process to evaluate different types of spacecraft configurations instead of solely focusing on single point optimization (Chapter 5);
- Created two figures of merit, the Safety Index and the Operability Index for trading spacecraft configurations (Chapter 6);
- Developed a conceptual design process based on a minimum functionality and bottom up philosophy that couples easily with common approaches in human spacecraft conceptual design (Chapter 7); and
- Created a Conceptual Lunar Ascent Module Program for quantifying Minimum Functionality, *Safety*, and *Operability* in spacecraft configurations.

This work established a definition for a minimum functional design baseline and is the first to group the fundamental mass parameters of a human spacecraft in the categories of *Physics*, *Physiology*, *Safety*, and *Operability*. The minimum functional baseline configuration defined in this work is different from previous approaches because it eliminates the bias toward a minimum set of requirements. The amount of *Safety* in the spacecraft is the mass dedicated to safety through similar or dissimilar redundancy, safety components, margins, and dispersions. The amount of *Operability* in the spacecraft is the mass used to perform mission objectives and make functions easier or efficient. Because human spacecraft are highly coupled systems, the introduction of mass in one subsystem has downstream effects on other subsystems that are not easily recognized by designers and the use of rapidly reconfigurable prototypes allows designers and multidisciplinary teams to utilize Boundary Objects as a means of communication for maturing design concepts. The mass addition process coupled with the minimum functionality approach creates a tradespace of potential spacecraft configurations and provides designers with an overall view of how various levels of *Safety* or *Operability* additions will affect the overall spacecraft mass. The information provided in the mass addition tradespace can reveal subsystem issues and bound the overall mass of the spacecraft such that future mass growth allowances can be allocated. The figures of merit provide a simplified method for evaluating the spacecraft as a whole instead of focusing solely on criticality functions that could overlook subtle dependencies in other functions. The Safety Index is based upon quantifiable information and uses parameters that are typically evaluated in early reliability analysis. The advantage of the Safety Index is that it couples the various parameters into a complete score; which is different from typical reliability based approaches that focus on limited areas of the spacecraft design. Although the Safety Index is not intended to replace conventional reliability methods, its purpose is to provide the designer with another tool such that comparisons of competing design concepts can be evaluated in a rapid design process. The Operability Index is a measure of the design configuration against top level mission objectives. In typical spacecraft development, the performance is often quantified as a resulting objective of the spacecraft design. In this process, the Operability Index is evaluated simultaneously with the Safety Index and the total spacecraft to provide a complete score of the design for future trading.

The previous methods can be combined into an overall conceptual design process that couples easily with typical industry approaches to human spacecraft development. The use of minimum functionality as a precursor to more conventional approaches allows the spacecraft configuration to take shape before time consuming detailed CAD and higher fidelity analyses.

The overall conclusion of this dissertation is that human spacecraft are challenging vehicles to design, develop, and operate. The decisions made in the conceptual design phase are critical to the success of the program and uncertainty can lead to unnecessary redesign in later phases. It is hoped that the information presented in this dissertation will provide designers with a different perspective on how to approach the problem of conceptual design in the development of future human spacecraft.

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APPENDIX A

PUBLICATIONS RESULTING FROM THIS DISSERTATION

JOURNAL PUBLICATIONS

Higdon, K. P. and Klaus, D. M. (2012). A Minimum Functionality Design Methodology for Determining Safety and Operability Mass in Human Spacecraft. *Journal of Spacecraft and Rockets* (accepted February 5, 2012)

Manuscripts in preparation:

Higdon, K. P. and Klaus, D. M. A Systematic Process for Evaluating Safety and Operational Functionality in Conceptual Human Spacecraft.

Higdon, K. P. and Klaus, D. M. Figures of Merit for Trading Conceptual Human Spacecraft Configurations.

CONFERENCE PUBLICATIONS

Higdon, K. P. and Klaus, D. M. (2008). Effective Integration of Rapid Prototyping into Multidisciplinary Design Optimization for the Development of Human Spacecraft. *11th ASCE Aerospace Division International Conference (Earth and Space 2008).* 3-6 March 2008. Long Beach, California.

Klaus, D. M. and Higdon, K. P. (2009). Academic Principles of Human Space Habitat Design. Paper 2009-01-2547. *Society of Automotive Engineers 39th International Conference on Environmental Systems*. 12-16 July 2009. Savannah, Georgia.

APPENDIX B

LUNAR ASCENT MODULE DESIGN GUIDELINES

The following sections present the top level mission objective, ground rules and assumptions, and definitions of a Lunar Ascent Module human spacecraft conceptual design.

B.1 TOP LEVEL MISSION OBJECTIVES

1. The Lunar Ascent Module shall transport a crew from the lunar surface to an elliptical lunar orbit of approximately 20 x 100km.

B.2 GROUND RULES AND ASSUMPTIONS

1. The Lunar Ascent Module is pressurized and encloses the crew during ascent, rendezvous, and docking.

2. The Lunar Ascent Module minimum functionality baseline is the functionality required to meet the physical and physiological requirements of the lunar ascent flight.

3. The Lunar Ascent Module minimum functional baseline configuration assumes all systems perform without failures and does not address emergencies, contingencies, failures, safety, or comfort.

4. The Lunar Ascent Module baseline should include subsystems and components that are of minimal mass and volume.

5. The Lunar Ascent Module assumed minimum flight operations time is 1.37 hours.

6. The Lunar Ascent Module baseline subsystem components default location will be located outside the habitable volume unless a need is shown to include the subsystem components inside the pressurized volume.

7. The Lunar Ascent Module will be designed with two hatches such that one is used for ingress/egress with a transfer orbiting spacecraft and a second for ingress/egress to the lunar surface, an airlock, or a habitation module.

8. The outer mold line of the Lunar Ascent Module is configuration independent.

9. The Lunar Ascent Module windows are designed for viewing of descent, landing, ascent, and rendezvous with orbiting spacecraft.

10. The Lunar Ascent Module flight control system will include components used for descent, landing, and ascent.

11. The Lunar Ascent Module main engine propulsion system will be designed for single use.

12. The Lunar Ascent Module will utilize its own Reaction Control System for ascent flight. The descent flight RCS is assumed to be a separate system and located on the Descent Module.

13. The Lunar Ascent Module minimum functional structure design is based on maximum acceleration loads during Earth Launch Ascent, Trans Lunar Injection, Lunar Descent, Lunar Landing, and Lunar Ascent.

14. The Lunar Ascent Module internal subsystems shall be capable of operating in a vacuum environment.

B.3 DEFINITIONS

Assumptions - Variables, equations, or procedures used to simplify complex problems and uncertainties.

Configuration – The arrangement of subsystem components within a spacecraft design.

Contingency – A scenario where off-nominal operations are required, usually due to a failure and includes degraded performance, dealt with by operational workaround or redundant systems.

Factor of Safety – Design factors or margin used to account for uncertainty in structural loading.

Functional Decomposition – The process of decomposing high level mission objectives into lower level functions to meet the Mission Goals.

Ground Rules – The top level constraints for the proposed research or development program used to create boundary conditions and limits for the design architecture.

Margin – Design factors used to account for uncertainty in deterministic variables and parameters.

Minimum Functionality – The lowest achievable baseline configuration of a human spacecraft based upon the parameters of habitable volume, crew size, material selection, propulsion system, and required trajectories using single string subsystems without margins or accounting for contingencies.

Mission Goals – A set of objectives given by stakeholders.

Operability – The means within a spacecraft to achieve top level mission objectives other than transporting crew and mitigating contingencies.

Redundancy – The use of similar or dissimilar means of achieving a given functional requirement.

Reliability – probability of a given device functioning within expected environmental conditions.

Risk - The combination of consequence and likelihood of component failure, subsystem failure, loss of mission, or loss of crew; a quantitative assessment of scenarios that will prevent the mission from meeting requirements.

Safety - The means within a spacecraft design for mitigating and preventing contingencies in order to prevent harm to the crew or the mission objectives.

Solutions – Technology choices used to perform tasks required for low level functions.

APPENDIX C

LUNAR ASCENT MODULE FUNCTIONAL DECOMPOSITION TABLES

The following tables list the derived functions for a Lunar Ascent Module based upon the top level mission objectives and ground rules and assumptions. The functional decomposition list was a precursor to the development of the CLAMP code and provided guidance in the development of the minimum functionality prototyping efforts. The tables are grouped by subsystem beginning with Primary Structures to provide an overall view of the derived functionality of a Lunar Ascent Module.

Table C.1: Primary Structure subsystem functional decomposition.

Top Level Functional Objectives (#.0)	**Second Level (#.#.#)**	**Third Level (#.#.#.#)**
Structures		
1.1 Provide Environmental Protection	**1.1.1 Provide protection from vacuum**	1.1.1.1 Provide Pressure Vessel
	1.1.2 Provide MMOD/TPS protection	
1.2 Provide Load Bearing Capability	**1.2.1 Provide Structural Support for Environmental Loads**	1.2.1.1 Provide Structural Support for Internal Pressure Loads
		1.2.1.2 Provide Structural Support for Launch Loads
		1.2.1.3 Provide Structural Support for TLI Loads
		1.2.1.4 Provide Structural Support for Landing Loads
		1.2.1.5 Provide Structural Support for Docked Loads
1.3 Support External Subsystems and Components	**1.3.1 Provide Structural Support for External ECLSS Components**	1.3.1.1 Support High Pressure Oxygen Tanks
		1.3.1.2 Support High Pressure External Oxygen Lines
		1.3.1.3 Support High Pressure Makeup Gas Tanks
		1.3.1.4 Support High Pressure Makeup Gas Exterior Lines and Valves
		1.3.1.5 Support Interface between External and Internal transfer piping

Table C.1: Primary Structure subsystem functional decomposition (continued).

Top Level Functional Objectives (#.0)	**Second Level (#.#.#)**	**Third Level (#.#.#.#)**
Structures		
	1.3.2 Provide Structural Support for External Crew Accomodations Components	1.3.2.1 Provide Structural Support for External Lighting
	1.3.3 Provide Structural Support for Exterior Payload Equipment	
	1.3.4 Provide Structural Support for Exterior Communications Equipment	
	1.3.5 Provide Structural Support for Exterior Power Equipment	
	1.3.6 Provide Structural Support for Exterior Command and Data Handling Equipment	
	1.3.7 Provide Structural Support for Exterior Health Monitoring Equipment	
	1.3.8 Provide structural Support for Exterior Flight Navigation and Control Equipment	
	1.3.9 Provide Structural Support for Exterior Thermal System Equipment	1.3.9.1 Support Radiator
		1.3.9.2 Support External Piping
		1.3.9.3 Support External Pumps
		1.3.9.4 Support Thermal Fluid Interfaces
		1.3.9.5 Support Valves
		1.3.9.6 Support Heat Exchanger
	1.3.10 Provide Structural Support for Propulsion System	1.3.10.1 Support Ascent Engine
		1.3.10.2 Support Oxidizer Tank
		1.3.10.3 Support Fuel Tank
		1.3.10.4 Support Pressurant Tank
		1.3.10.5 Support RCS system
		1.3.10.6 Support RCS piping / valves
		1.3.10.7 Support Oxidizer Lines
		1.3.10.8 Support Fuel Lines
		1.3.10.9 Support Pressurant Lines
	1.3.11 Provide EVA Mounts	1.3.11.1 Provide EVA Mounts
1.4 Support Internal Subsystems and Components	**1.4.1 Provide Structural Support for ECLSS**	1.4.1.1 Support Air Return Vent
		1.4.1.2 Support Trace Contaminant Removal Components
		1.4.1.3 Support Particulate Removal Components
		1.4.1.4 Support Cabin Fan

Table C.1: Primary Structure subsystem functional decomposition (continued).

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Structures		
		1.4.1.5 Support Bypass Valve
		1.4.1.6 Support Humidity Separator
		1.4.1.7 Support Humidity Waste Storage
		1.4.1.8 Support Humidity Waste Transport
		1.4.1.9 Support CO2 Removal Components
		1.4.1.10 Support Air Flow Mix
		1.4.1.11 Support Air Output Vent
		1.4.1.12 Support Air Circulation Ductwork
		1.4.1.13 Provide Mounting Location for Cabin Air Pressure Sensor
		1.4.1.14 Provide Mounting Location for Cabin Air Temperature Sensor
		1.4.1.15 Support Food Storage Container
		1.4.1.16 Support Food Preparation Components
		1.4.1.17 Support Potable Water Container
		1.4.1.18 Support Hygiene Water Container
		1.4.1.19 Support Urine Collection Container
		1.4.1.20 Support Fecal Solid Collection Container
		1.4.1.21 Support Liquid Waste Container
		1.4.1.22 Support Solid Waste Container
		1.4.1.23 Support Liquid Waste Jettison Transport and Control
		1.4.1.24 Support Fire Detection Sensor
		1.4.1.25 Support Fire Suppression Equipment
	1.4.2 Provide Structural Support for Crew Accommodations	1.4.2.1 Support Overhead Lighting
		1.4.2.2 Support Side Lighting
		1.4.2.3 Support Commander Restraints
		1.4.2.4 Support Pilot Restraints
		1.4.2.5 Support Mission Specialist Restraints
		1.4.2.6 Support Handholds
	1.4.3 Provide Structural Support for Payloads & Storage	1.4.3.1 Support Personal Storage
		1.4.3.2 Support Hygiene Storage
		1.4.3.3 Support Medical Kit Storage
		1.4.3.4 Support Science Payload Storage
		1.4.3.5 Support General Payload Storage
		1.4.3.6 Support Tools and Equipment Storage
		1.4.3.7 Support Space Suit Storage
		1.4.3.8 Support Consumable Equipment Storage
		1.4.3.9 Support Additional Clothing Storage
	1.4.4 Provide Structural Support for Communication Systems	

Table C.1: Primary Structure subsystem functional decomposition (continued).

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Structures		
	1.4.5 Provide Structural Support for Power Systems	1.4.5.1 Support Power Source
		1.4.5.2 Support Power Distribution
		1.4.5.3 Support Power Regulation
		1.4.5.4 Support Grounding Interfaces
		1.4.5.5 Support Overload Protection Cabinet
	1.4.6 Provide Structural Support for Command and Data Handling Systems	
	1.4.7 Provide Structural Support for Health Monitoring Systems	
	1.4.8 Provide Structural Support for Flight Navigation and Control Systems	
	1.4.9 Provide Structural Support for Thermal Systems Equipment	1.4.9.1 Support Internal Thermal System Coldplates
		1.4.9.2 Support Internal Thermal System Fluid Transport
		1.4.9.3 Support Internal Thermal System Fluid Control
		1.4.9.4 Support Internal Thermal System Fluid Pumps
	1.4.10 Provide Structural Support for Propulsion Equipment	
1.5 Provide External Interfaces	**1.5.1 Support Docking Mechanism for Lunar Ascent Module to CEV**	1.5.1.1 Provide Lunar Ascent Module with Interface to LIDS system
	1.5.2 Provide Ingress/Egress for Lunar Ascent Module to CEV	1.5.2.1 Provide Ingress / Egress for Microgravity Environment with LIDS Interface
	1.5.3 Provide Docking Mechanism for Lunar Habitat to Lunar Ascent Module	1.5.3.1 Provide Docking / Undocking Interface between Modules
		1.5.3.2 Provide Mechanical Decoupling between Modules
	1.5.4 Provide Ingress/Egress for Lunar Habitat to Lunar Ascent Module	1.5.4.1 Provide Ingress/Egress for 1/6g Environment
1.6 Provide Direct External Observation	**1.6.1 Provide Forward Windows**	1.6.1.1 Provide Look Angles for Descent / Ascent Flight Control
		1.6.1.2 Provide Look Angles for Landing
	1.6.2 Provide Docking Viewing Ports	1.6.2.1 Provide Look Angles for Docking
	1.6.3 Provide Aft Viewing Ports	1.6.3.1 Provide Look Angles for Aft

Table C.2: Crew Accommodations subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Crew Accommodations		
3.1 Support Human Factors	**3.1.1 Provide Cabin Lighting**	3.1.1.1 Provide Light Overhead
		3.1.1.2 Provide Light for Sides of Interior
	3.1.2 Provide Panel Lighting	3.1.2.1 Provide Light Subsystem Instrument Panels
		3.1.2.2 Provide Flight Instrument Panel Lighting
	3.1.3 Provide Light for Exterior Viewing	3.1.3.1 Provide Light Forward Viewing
		3.1.3.2 Provide Light for S/C Docking
		3.1.3.3 Provide Light for Aft Viewport
	3.1.4 Provide Restraints	3.1.4.1 Restrain Commander
		3.1.4.2 Restrain Pilot
		3.1.4.3 Restrain Mission Specialist(s)
	3.1.5 Provide Handholds	3.1.5.1 Provide Handholds for Hatch
		3.1.5.2 Provide Handholds for Interior
		3.1.5.3 Provide Handholds for Exterior
3.2 Maintain Happiness	**3.2.1 Provide Personal Storage**	
	3.2.2 Provide Additional Clothing Storage	
	3.2.3 Provide Entertainment	
3.3 Maintain Health	**3.3.1 Provide Hygiene Supplies**	
	3.3.2 Provide Medical Kit	
	3.3.3 Provide Exercise Capability	
	3.3.4 Provide Sleep Accommodations	
	3.3.5 Provide Operational Supplies	

Table C.3: Payloads subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Payloads		
4.1 Store Payloads	**4.1.1 Provide Science Payload Storage**	4.1.1.1 Provide Sample Storage (lunar materials)
	4.1.2 Provide General Payload Storage	
4.2 Store Tools and Equipment	**4.2.1 Provide Tools**	4.2.1.1 Provide Hand Tools Storage
	4.2.2 Provide Consumable Equipment	4.2.1.1 Provide Extra LiOH Canister Storage
		4.2.1.2 Provide Extra Trace Contaminant Filter Storage
		4.2.1.3 Provide Extra Particulate Filter Storage
		4.2.1.4 Provide Additional Equipment Storage
	4.2.3 Provide Space Suit Storage	4.2.3.1 Provide Helmet Storage
		4.2.3.2 Provide Suit / Gloves Storage
		4.2.3.3 Provide Umbilical Storage

Table C.4: EVA subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
EVA		
12.1 Environmental Protection	**12.1.1 Provide Space Suit**	
12.2 Allow External Mobility	**12.2.1 Provide Mobility Aids**	
	12.2.2 Provide Umbilical	

Table C.5: ECLSS subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#) (i.e. Tech Choice)
Environmental Control and Life Support System (ECLSS)		
2.1 Control Atmosphere	**2.1.1 Remove Carbon Dioxide**	
	2.1.2 Provide Metabolic Oxygen	2.1.2.1 Store High Pressure Oxygen
		2.1.2.2 Transport Oxygen
		2.1.2.3 Control Oxygen Flow
		2.1.2.4 Measure pp Oxygen Level in Cabin Air
	2.1.3 Provide Makeup / Buffer Gas	2.1.3.1 Store High Pressure Makeup Gas
		2.1.3.2 Transport Makeup Gas
		2.1.3.3 Control Makeup Gas Flow
	2.1.4 Remove Trace Contaminants	2.1.4.1 Provide Filter
	2.1.5 Remove Particulates	2.1.5.1 Provide Filter
	2.1.6 Remove Humidity	2.1.6.1 Provide Humidity Separator
		2.1.6.2 Condense Cabin Air
		2.1.6.3 Transport Humidity to Waste Container
	2.1.7 Control Air Temperature	2.1.7.1 Provide Air Circ Bypass
		2.1.7.2 Measure Air Temperature
	2.1.8 Provide Air Circulation	2.1.8.1 Circulate Air
		2.1.8.2 Return Air
		2.1.8.3 Output Air
		2.1.8.4 Direct Air Flow
		2.1.8.5 Mix Air Circ Streams
	2.1.9 Control Cabin Air Pressure	2.1.9.1 Provide Air Pressure Sensor
		2.1.9.2 Provide Cabin Air Control Logic
	2.1.10 Provide Fire Detection and Suppression	
2.2 Provide Food	**2.2.1 Store Food**	2.2.1.1 Prevent Food Contamination
		2.2.1.2 Control Food Storage Temperature
	2.2.2 Prepare Food	2.2.2.1 Heat Food
		2.2.2.2 Handle Food
2.3 Provide Water	**2.3.1 Provide Potable Water**	2.3.1.1 Store Potable Water
		2.3.1.2 Dispense Potable Water
	2.3.2 Provide Hygiene Water	2.3.2.1 Store Hygiene Water
		2.3.2.2 Dispense Hygiene Water
2.4 Manage Waste	**2.4.1 Collect Urine**	2.4.1.1 Provide Urine Bags
		2.4.1.2 Collect Urine Waste
	2.4.2 Collect Fecal Solids	2.4.2.1 Provide Garments
		2.4.2.2 Control Odor
		2.4.2.3 WCS supplies
		2.4.2.4 Collect Garments
	2.4.3 Collect Liquid Waste	2.4.3.1 Collect Humidity Waste
		2.4.3.2 Collect Liquid Waste
	2.4.4 Collect Solid Waste	2.4.4.1 Collect Solid Waste
	2.4.5 Jettison Liquid Waste	2.4.5.1 Transport Liquid Waste
		2.4.5.2 Control Liquid Waste Flow

Table C.6: Communications subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Communication (Radio Frequencies)		
6.1 Provide voice/data communication	**6.1.1 Communicate with Earth ground station (External to Ascent Module)**	6.1.1.1 Transmit voice communication
		6.1.1.2 Receive voice communication
	6.1.2 Communicate with relay satellites (External to Ascent Module)	6.1.2.1 Transmit voice communication
		6.1.2.2 Receive voice communication
	6.1.3 Communicate between Suited Crewmembers	6.1.3.1 Transmit voice communication
		6.1.3.2 Receive voice communication
		6.2.1.1 Uplink data
		6.2.1.2 Downlink data
		6.2.2.1 Uplink data
		6.2.2.2 Downlink data
		6.2.3.1 Uplink data
		6.2.3.2 Downlink data

Table C.7: Command and Data Handling subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Command & Data Handling (Computer Control)		
7.1 Sense Subsystem Commands	**7.1.1 Accept ECLSS Commands**	7.1.1.1 Output commands from ECLSS
		7.1.1.2 Input commands to C&DH
	7.1.2 Accept Crew Accommodations Commands	7.1.2.1 Output commands from Crew Accommodations
		7.1.2.2 Input commands to C&DH
	7.1.3 Accept Payload Commands	7.1.3.1 Output commands from Payload
		7.1.3.2 Input commands to C&DH
	7.1.4 Accept Power Commands	7.1.4.1 Output commands from Power
		7.1.4.2 Input commands to C&DH
	7.1.5 Accept Communication Commands	7.1.5.1 Output commands from Communication
		7.1.5.2 Input commands to C&DH
	7.1.6 Accept C&DH Commands	7.1.6.1 Output commands from C&DH
		7.1.6.2 Input commands to C&DH
	7.1.7 Accept Health Monitoring Commands	7.1.7.1 Output commands from Health Monitoring
		7.1.7.2 Input commands to C&DH
	7.1.8 Accept Flight Control Commands	7.1.8.1 Output commands from Flight Control
		7.1.8.2 Input commands to C&DH
	7.1.9 Accept Thermal Commands	7.1.9.1 Output commands from Thermal System
		7.1.9.2 Input commands to C&DH
	7.1.10 Accept Propulsion Commands	7.1.10.1 Output commands from Propulsion
		7.1.10.2 Input commands to C&DH
	7.1.11 Accept Structures Commands	7.1.11.1 Output commands from Structures
		7.1.11.2 Input commands to C&DH

Table C.7: Command and Data Handling subsystem functional decomposition (continued).

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Command & Data Handling (Computer Control)		
7.2 Process/Amplify Subsystem Commands	**7.2.1 Check input commands for errors**	7.2.1.1 Check bit count
		7.2.1.2 Verify system will not fail with command implimented
	7.2.2 Prioritize commands	7.2.2.1 Categorize commands
		7.2.2.2 Execute commands
7.3 Send Commands to Subsystems	**7.3.1 Send Commands to ECLSS Subsystem**	7.3.1.1 Output commands from C&DH
		7.3.1.2 Input commands to ECLSS
	7.3.2 Send Commands to Crew Accommodations Subsystem	7.3.2.1 Output commands from C&DH
		7.1.2.2 Input commands to Crew Accommodations
	7.3.3 Send Commands to Payload Subsystem	7.3.3.1 Output commands from C&DH
		7.3.3.2 Input commands to Payloads
	7.3.4 Send Commands to Power Subsystem	7.3.4.1 Output commands from C&DH
		7.3.4.2 Input commands to Power
	7.3.5 Send Commands to Communication Subsystem	7.3.5.1 Output commands from C&DH
		7.3.5.2 Input commands to Communications
	7.3.6 Send Commands to C&DH Subsystem	7.3.6.1 Output commands from C&DH
		7.3.6.2 Input commands to C&DH
	7.3.7 Send Commands to Health Monitoring Subsystem	7.3.7.1 Output commands from C&DH
		7.3.7.2 Input commands to Health Monitoring
	7.3.8 Send Commands to Flight Control Subsystem	7.3.8.1 Output commands from C&DH
		7.3.8.2 Input commands to Flight Control
	7.3.9 Send Commands to Thermal Subsystem	7.3.9.1 Output commands from C&DH
		7.3.9.2 Input commands to Thermal
	7.3.10 Send Commands to Propulsion Subsystem	7.3.10.1 Output commands from C&DH
		7.3.10.2 Input commands to Propulsion
	7.3.11 Send Commands to Structures Subsystem	7.3.11.1 Output commands from C&DH
		7.3.11.2 Input commands to Structures
7.4 Store Spacecraft Data	**7.4.1 Input Health Monitoring Data**	7.4.1.1 Input ECLSS sensor data
		7.4.1.2 Input Crew Accommodations sensor data
		7.4.1.3 Input Payloads sensor data
		7.4.1.4 Input Power sensor data
		7.4.1.5 Input Communications sensor data
		7.4.1.6 Input C&DH sensor data
		7.4.1.7 Input Health Monitoring sensor data
		7.4.1.8 Input Flight Control sensor data
		7.4.1.9 Input Thermal sensor data
		7.4.1.10 Input Propulsion sensor data
		7.4.1.11 Input Structures sensor data
	7.4.2 Store Health Monitoring Data	7.4.2.1 Store ECLSS sensor data
		7.4.2.2 Store Crew Accommodation sensor data
		7.4.2.3 Store Payloads sensor data
		7.4.2.4 Store Power sensor data
		7.4.2.5 Store Communications sensor data
		7.4.2.6 Store C&DH sensor data
		7.4.2.7 Store Health Monitoring sensor data
		7.4.2.8 Store Flight Control sensor data
		7.4.2.9 Store Thermal sensor data
		7.4.2.10 Store Propulsion sensor data
		7.4.2.11 Store Structures sensor data

Table C.8: Health monitoring subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Health Monitoring (Sensor System)		
8.1 Monitor Vehicle Subsystems	8.1.1 Monitor Subsystem Data	
	8.1.2 Report Status to Ground	
	8.1.3 Report Status to Crew	
8.2 Sense Vehicle Health Status	8.2.1 Monitor ECLSS	
	8.2.2 Monitor Crew Accommodations	
	8.2.3 Monitor Payload Subsystem	
	8.2.4 Monitor Power Subsystem	
	8.2.5 Monitor Communication Subsystem	
	8.2.6 Monitor Command and Data Handling (C&DH) Subsystem	
	8.2.7 Monitor Health Monitoring Subsystem	
	8.2.8 Monitor Flight Control Subsystem	
	8.2.9 Monitor Thermal Subsystem	
	8.2.10 Monitor Propulsion Subsystem	
	8.2.11 Monitor Structures Subsystem	

Table C.9: Flight Control and Navigation subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Flight Control & Nav (Avionics)		
9.1 Provide Spacecraft Navigation	9.1.1 Sense Spacecraft Position Nav Inputs	
	9.1.2 Convert Analog to Digital Nav Inputs	
	9.1.3 Output Navigation to Guidance	
9.2 Provide Spacecraft Guidance (PGNS)	9.2.1 Calculate Spacecraft Guidance	
	9.2.2 Interface with Instrumentation	
9.3 Provide Spacecraft Control (CES)	9.3.1 Input Human Flight Control	
	9.3.2 Input Human Navigation to Computer	
	9.3.3 Output Spacecraft Control to Propulsion	
	9.3.4 Display Spacecraft Control	
	9.3.5 Display Spacecraft Navigation	
	9.3.6 Manual Control Spacecraft Subsystems	
9.4 Provide Abort Guidance (AGS)	9.4.1 Sense Spacecraft Abort Position	
	9.4.2 Sense Spacecraft Abort Velocity Inputs	
	9.4.3 Calculate Spacecraft Abort Trajectory	
	9.4.4 Input Abort Commands	
9.5 Provide Rendezvous Guidance		
9.6 Provide Inertial Reference		
9.7 Provide Additional Avionics		

Table C.10: Power subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Power		
5.1 Provide Power	5.1.1 Provide Power from Internal Source	5.1.1.1 Generate Power
		5.1.1.2 Store Power
5.2 Distribute Power	5.2.1 Distribute Power to ECLSS	5.2.1.1 Distribute Power to Cabin Fan
		5.2.1.2 Distribute Power to Humidity Separator
	5.2.2 Distribute Power to Crew Accommodations	
	5.2.3 Distribute Power to Payload Subsystem	
	5.2.4 Distribute Power to Power Subsystem	
	5.2.5 Distribute Power to Communication Subsystem	
	5.2.6 Distribute Power to Command and Data Handling Subsystem	
	5.2.7 Distribute Power to Health Monitoring Subsystem	
	5.2.8 Distribute Power to Flight Control Subsystem	
	5.2.9 Distribute Power to Thermal Subsystem	
	5.2.10 Distribute Power to Propulsion Subsystem	
	5.2.11 Distribute Power to Structures Subsystem	
5.3 Regulate Power	5.3.1 Regulate Power to ECLSS	
	5.3.2 Regulate Power to Crew Accommodations	
	5.3.3 Regulate Power to Payload Subsystem	
	5.3.4 Regulate Power to Power Subsystem	
	5.3.5 Regulate Power to Communication Subsystem	
	5.3.6 Regulate Power to Command and Data Handling Subsystem	
	5.3.7 Regulate Power to Health Monitoring Subsystem	
	5.3.8 Regulate Power to Flight Control Subsystem	
	5.3.9 Regulate Power to Thermal Subsystem	
	5.3.10 Regulate Power to Propulsion Subsystem	
	5.3.11 Regulate Power to Structures Subsystem	
5.4 Provide Grounding	5.4.1 Ground ECLSS	
	5.4.2 Ground Crew Accommodations	
	5.4.3 Ground Payload Subsystem	
	5.4.4 Ground Power Subsystem	
	5.4.5 Ground Communication Subsystem	
	5.4.6 Ground Command and Data Handling Subsystem	
	5.4.7 Ground Health Monitoring Subsystem	
	5.4.8 Ground Flight Control Subsystem	
	5.4.9 Ground Thermal Subsystem	
	5.4.10 Ground Propulsion Subsystem	
	5.4.11 Ground Structures Subsystem	

Table C.10: Power subsystem functional decomposition (continued).

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Power		
5.5 Provide Shielding	5.5.1 Shield ECLSS wiring	
	5.5.2 Shield Crew Accommodations wiring	
	5.5.3 Shield Payload Subsystem wiring	
	5.5.4 Shield Power Subsystem wiring	
	5.5.5 Shield Communication Subsystem wiring	
	5.5.6 Shield Command and Data Handling Subsystem wiring	
	5.5.7 Shield Health Monitoring Subsystem wiring	
	5.5.8 Shield Flight Control Subsystem wiring	
	5.5.9 Shield Thermal Subsystem wiring	
	5.5.10 Shield Propulsion Subsystem wiring	
	5.5.11 Shield Structures Subsystem wiring	
5.6 Provide Overload Protection	5.6.1 Overload Protection for ECLSS	
	5.6.2 Overload Protection for Crew Accommodations	
	5.6.3 Overload Protection for Payload Subsystem	
	5.6.4 Overload Protection for Power Subsystem	
	5.6.5 Overload Protection for Communication Subsystem	
	5.6.6 Overload Protection for Command and Data Handling Subsystem	
	5.6.7 Overload Protection for Health Monitoring Subsystem	
	5.6.8 Overload Protection for Flight Control Subsystem	
	5.6.9 Overload Protection for Thermal Subsystem	
	5.6.10 Overload Protection for Propulsion Subsystem	
	5.6.11 Overload Protection for Structures Subsystem	

Table C.11: Thermal subsystem functional decomposition.

Top Level Functional Objectives (#.#)	Second Level (#.#.#)	Third Level (#.#.#.#)
Thermal		
10.1 Collect Heat	10.1.1 Collect Heat from ECLSS Subsystem	
	10.1.2 Collect Heat from Crew Accommodations Subsystem	
	10.1.3 Collect Heat from Payload Subsystem	
	10.1.4 Collect Heat from Power Subsystem	
	10.1.5 Collect Heat from Communication Subsystem	
	10.1.6 Collect Heat from Command and Data Handling Subsystem	
	10.1.7 Collect Heat from Health Monitoring Subsystem	
	10.1.8 Collect Heat from Flight Control Subsystem	
	10.1.9 Collect Heat from Thermal Subsystem	
	10.1.10 Collect Heat from Propulsion Subsystem	
	10.1.11 Collect Heat from Structures Subsystem	
10.2 Transport Heat	10.2.1 Transport Heat from ECLSS Subsystem	
	10.2.2 Transport Heat from Crew Accommodations Subsystem	
	10.2.3 Transport Heat from Payload Subsystem	
	10.2.4 Transport Heat from Power Subsystem	
	10.2.5 Transport Heat from Communication Subsystem	
	10.2.6 Transport Heat from Command and Data Handling Subsystem	
	10.2.7 Transport Heat from Health Monitoring Subsystem	
	10.2.8 Transport Heat from Flight Control Subsystem	
	10.2.9 Transport Heat from Thermal Subsystem	
	10.2.10 Transport Heat from Propulsion Subsystem	
	10.2.11 Transport Heat from Structures Subsystem	
10.3 Remove Heat	10.3.1 Remove Heat From Spacecraft	

Table C.12: Propulsion subsystem functional decomposition.

Top Level Functional Objectives (#.0)	Second Level (#.#.#)	Third Level (#.#.#.#)
Propulsion		
11.1 Provide Main Ascent Engine Propulsion	**11.1.1 Provide Main Engine Fuel**	
	11.1.2 Provide Main Engine Oxidizer	
	11.1.3 Provide Main Engine Pressurant	
	11.1.4 Control Main Engine Pressurant	11.1.4.1 Control ME Pressurant - Fuel
		11.1.4.2 Control ME Pressurant - Oxidizer
	11.1.5 Transport Pressurant	
	11.1.6 Provide Main Engine	
	11.1.7 Store Main Engine Fuel	
	11.1.8 Store Main Engine Oxidizer	
	11.1.9 Transport Main Engine Propellant	
	11.1.10 Control Main Engine Propellant	11.1.10.1 Control Main Engine Fuel
		11.1.10.2 Control Main Engine Oxidizer
	11.1.11 Provide Contingency Propellant	
	11.1.12 Support Main Engine Components	
	11.1.13 Insulate Propellant	
	11.1.14 Control Propellant Bypass	
11.2 Provide Flight Control (RCS)	**11.2.1 Provide RCS Fuel**	
	11.2.2 Provide RCS Oxidizer	
	11.2.3 Provide RCS Pressurant	
	11.2.4 Control RCS Pressurant	
	11.2.5 Transport RCS Pressurant	
	11.2.6 Provide RCS Engine	
	11.2.7 Store RCS Fuel	
	11.2.8 Store RCS Oxidizer	
	11.2.9 Transport RCS Propellant	11.2.9.1 Filter RCS propellant
	11.2.10 Control RCS Propellant	11.2.10.1 Control RCS Fuel
		11.2.10.2 Control RCS Oxidizer
	11.2.11 Provide RCS Contingency Propellant	
	11.2.12 Support RCS Components	
	11.2.13 Insulate RCS Propellant	
	11.2.14 Heat RCS lines	11.2.14.1 Heat RCS Fuel Lines
		11.2.14.2 Heat RCS Oxidizer Lines

APPENDIX D

CONCEPTUAL LUNAR ASCENT MODULE PROGRAM

D.1 INTRODUCTION

The Conceptual Lunar Ascent Module Program was developed to analyze and study various configurations of a Lunar Ascent Spacecraft. The code is a combination of heuristics, textbook equations, and 3-degree of freedom trajectory analysis. It is an end-to-end program that quantifies total spacecraft mass and subsystem configurations with a set of given input variables and parameters. Shown in Figure D.1 is a flowchart of the CLAMP code.

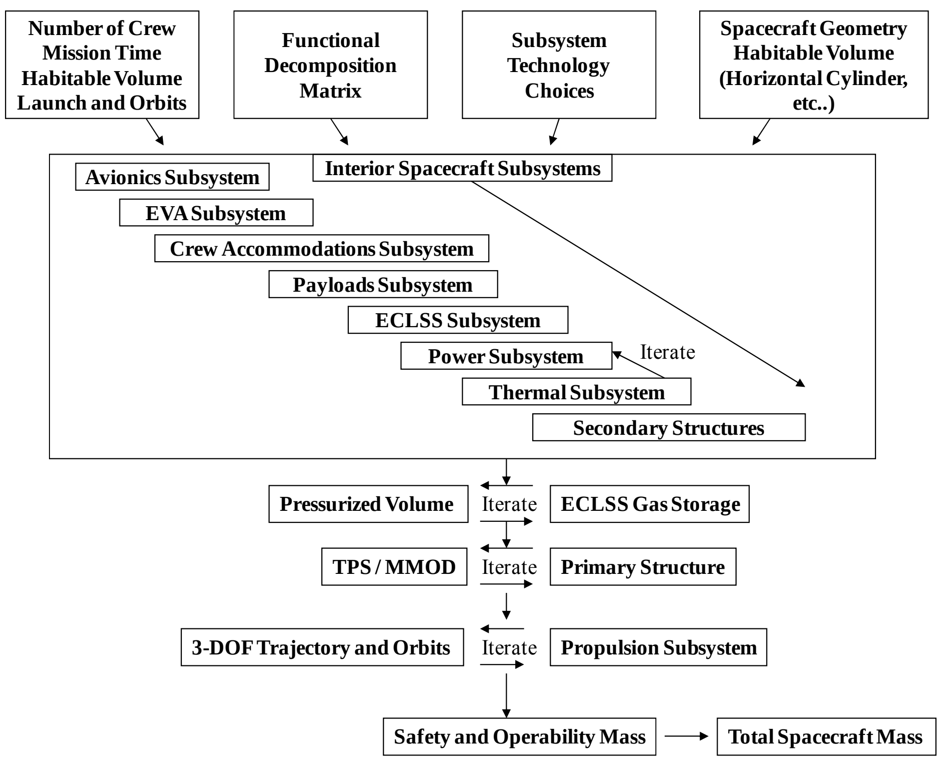


Figure D.1: Flowchart of CLAMP.

The program was written in MATLAB and utilizes a Microsoft Excel spreadsheet as an interface for the input variables. No additional computational programs are required to run the program and the typical time for a single run is less than 8 seconds depending upon the processing speed of the computer. The information presented in this appendix describes the logic, equations, and assumptions for key subroutines in the program

D.2 INPUT FILE

A Microsoft Excel spreadsheet format was used to provide the input variables and parameters for a given spacecraft configuration to the MATLAB code. The spreadsheet format allows the user to quickly identify specific variables in a large list instead of a text file based approach. Shown in Figure D.2 is an example of the main input file list.

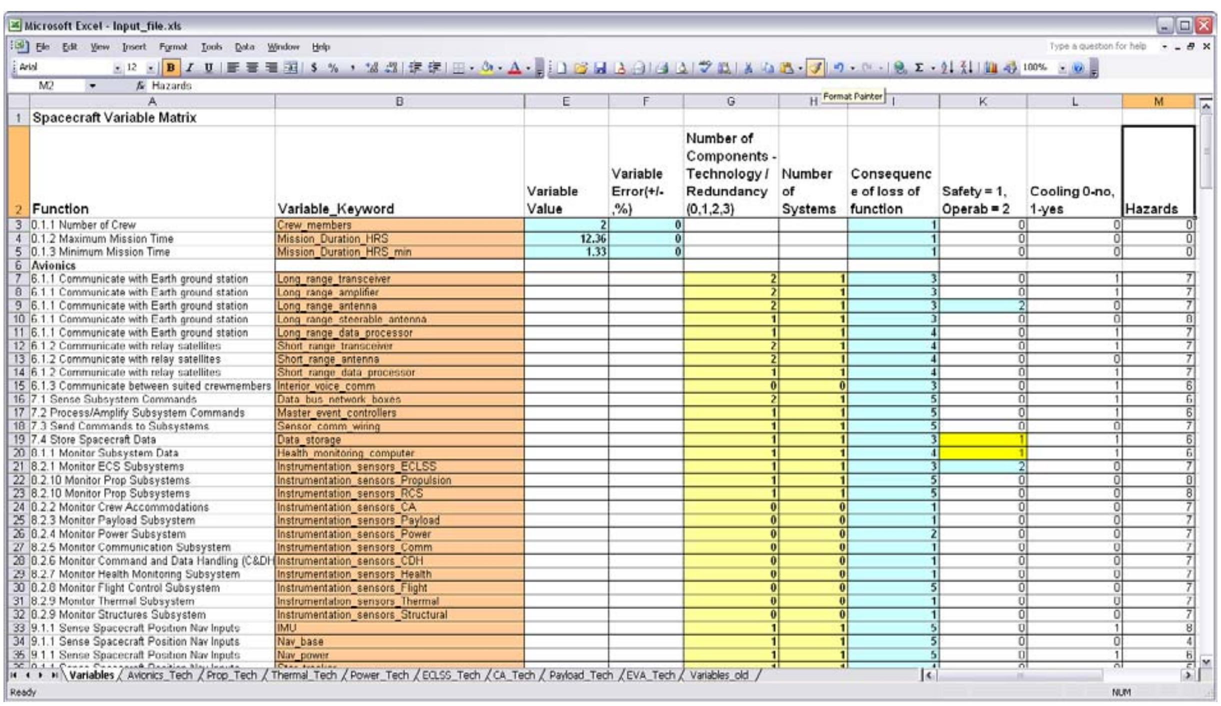
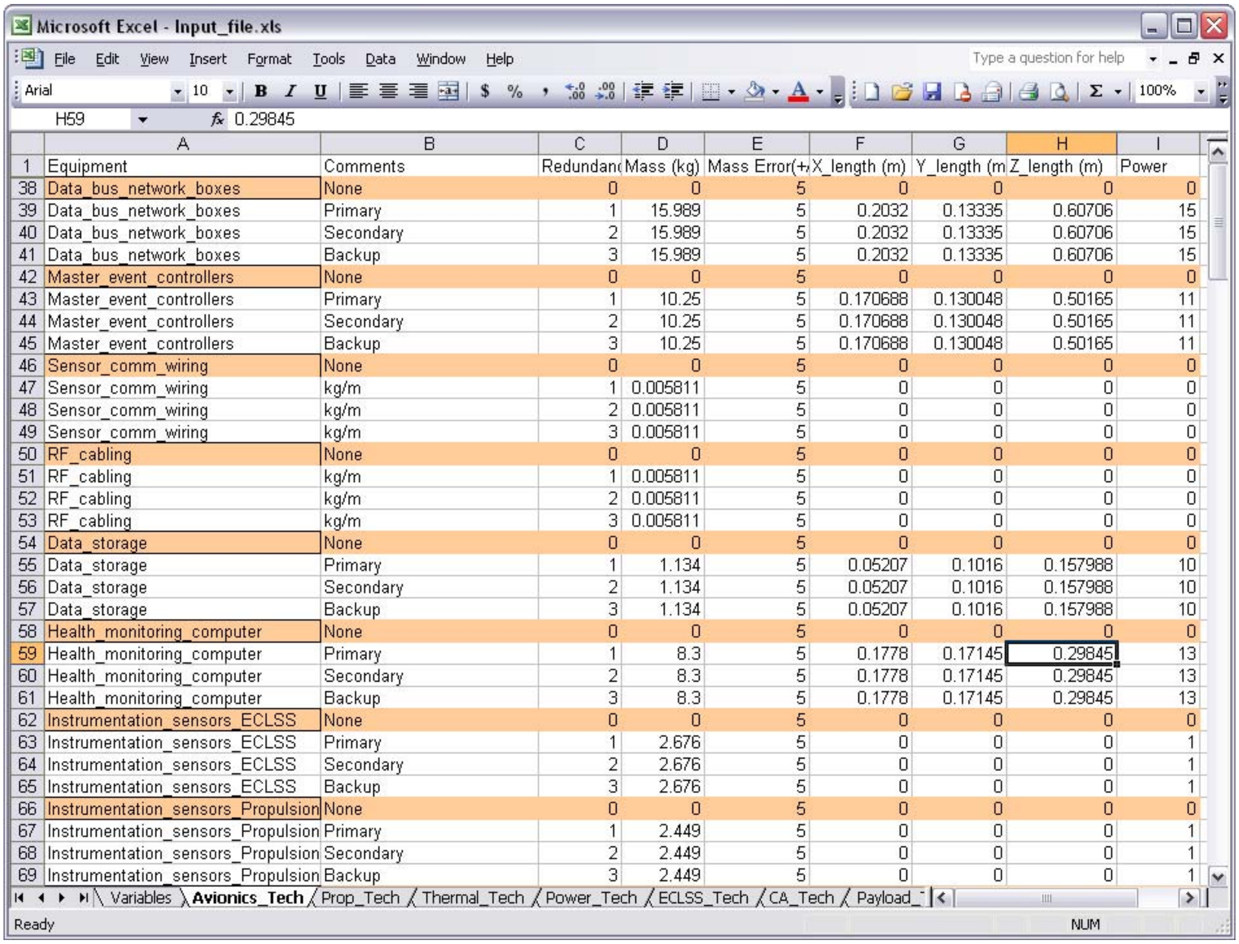


Figure D.2: Microsoft Excel input file.

The main sheet within the Microsoft Excel file is named "Variables". This spreadsheet contains all of the key variables and parameters used in the program. The sheet is loaded into a MATLAB array and referred to in the source code as "input_vars". The columns of the input file include the function, variable keyword, units, variable value, error, redundancy, and number of systems, consequence of the function, a flag for indicating Safety or Operability components, a flag for cooling, and the number of hazards associated with the component. In order for MATLAB to recognize a specific variable, the Variable_Keyword identifier is used to identify the input information that will be passed to a MATLAB variable. If the input value is a variable, the program will search for information in the variable value and error column. Otherwise, the program will utilize the variable keyword to identify the number of components in the redundancy column. The program will then look up the mass and volume information in the specific subsystem database tables.

Because the program is based upon a bottom up design philosophy, a list of components matched to variable keywords can be found in separate subsystem technology sheets within the Microsoft Excel input file. These sheets are named: Avionics_Tech, Prop_Tech, ECLSS_Tech, CA_Tech, Payload_Tech, EVA_Tech, Power_Tech, and Thermal_Tech. Shown in Figure D.3 is an example of a subsystem technology sheet. The subsystem technology sheet is a database that contains component mass, volume, power, and other information utilized in the spacecraft design configuration analysis.

For each variable keyword row, a set of units is specified. Some variables keywords call out metric units and others call out English units. The program always converts English units to metric during input to the subroutines and the metric system is utilized in all calculations within the program. The output of the program provides both metric and English units.



	Equipment	Comments	Redundanc	Mass (kg)	Mass Error(+/	X_length (m)	Y_length (m	Z_length (m)	Power
38	Data_bus_network_boxes	None	0	0	5	0	0	0	0
39	Data_bus_network_boxes	Primary	1	15.989	5	0.2032	0.13335	0.60706	15
40	Data_bus_network_boxes	Secondary	2	15.989	5	0.2032	0.13335	0.60706	15
41	Data_bus_network_boxes	Backup	3	15.989	5	0.2032	0.13335	0.60706	15
42	Master_event_controllers	None	0	0	5	0	0	0	0
43	Master_event_controllers	Primary	1	10.25	5	0.170688	0.130048	0.50165	11
44	Master_event_controllers	Secondary	2	10.25	5	0.170688	0.130048	0.50165	11
45	Master_event_controllers	Backup	3	10.25	5	0.170688	0.130048	0.50165	11
46	Sensor_comm_wiring	None	0	0	5	0	0	0	0
47	Sensor_comm_wiring	kg/m	1	0.005811	5	0	0	0	0
48	Sensor_comm_wiring	kg/m	2	0.005811	5	0	0	0	0
49	Sensor_comm_wiring	kg/m	3	0.005811	5	0	0	0	0
50	RF_cabling	None	0	0	5	0	0	0	0
51	RF_cabling	kg/m	1	0.005811	5	0	0	0	0
52	RF_cabling	kg/m	2	0.005811	5	0	0	0	0
53	RF_cabling	kg/m	3	0.005811	5	0	0	0	0
54	Data_storage	None	0	0	5	0	0	0	0
55	Data_storage	Primary	1	1.134	5	0.05207	0.1016	0.157988	10
56	Data_storage	Secondary	2	1.134	5	0.05207	0.1016	0.157988	10
57	Data_storage	Backup	3	1.134	5	0.05207	0.1016	0.157988	10
58	Health_monitoring_computer	None	0	0	5	0	0	0	0
59	Health_monitoring_computer	Primary	1	8.3	5	0.1778	0.17145	0.29845	13
60	Health_monitoring_computer	Secondary	2	8.3	5	0.1778	0.17145	0.29845	13
61	Health_monitoring_computer	Backup	3	8.3	5	0.1778	0.17145	0.29845	13
62	Instrumentation_sensors_ECLSS	None	0	0	5	0	0	0	0
63	Instrumentation_sensors_ECLSS	Primary	1	2.676	5	0	0	0	1
64	Instrumentation_sensors_ECLSS	Secondary	2	2.676	5	0	0	0	1
65	Instrumentation_sensors_ECLSS	Backup	3	2.676	5	0	0	0	1
66	Instrumentation_sensors_Propulsion	None	0	0	5	0	0	0	0
67	Instrumentation_sensors_Propulsion	Primary	1	2.449	5	0	0	0	1
68	Instrumentation_sensors_Propulsion	Secondary	2	2.449	5	0	0	0	1
69	Instrumentation_sensors_Propulsion	Backup	3	2.449	5	0	0	0	1

Figure D.3: Example of the subsystem technology sheet for Avionics.

In the subsystem technology sheet, the equipment name is the same as the variable keyword. For each piece of equipment, the redundancy of each component is specified for none, primary, secondary, and backup. This allows the user to specify different components that could be utilized to capture different or dissimilar redundancy. The code will use these parameters to calculate the mass of the individual components. In some instances, a parameter is used by the program to determine component mass such as wiring given in kg/m. Where volume information was available, the X length, Y length, and Z length values were updated.

D.3 MAIN PROGRAM FILE

A main program file is utilized to control the flow of the subroutines. The main program begins with loading the Microsoft Excel input file information and copying to MATLAB arrays. The first subroutine is named “Input_Excel” and the name of the input file is passed as a parameter to the subprogram. After the input file has been loaded into the MATLAB arrays, the original input file is saved as a copy and the program creates an input file for a minimum functionality baseline configuration. Once the lower mass boundary has been determined, a table of propulsion mass based on increased spacecraft payload mass is developed. The use of a propulsion flyout table increases the speed of the analysis because the code does not have to repeatedly perform the 3 degree of freedom trajectory analysis. After the propulsion flyout table has been constructed, the program creates two output files named CG_string_matrix, and SC_mass that are updated by the subsystem routines to capture the list of components.

The main output file, CG_string_matrix is a MATLAB cell string matrix that records information as strings instead of numbers. The maximum precision of this matrix is 4 decimal places. Throughout the development process, the precision did not pose a problem for generating accurate results. Although the use of a cell string matrix is not typical for a high fidelity analysis code, the method worked adequately for storing the information. At the end of the analysis, the CG_string_matrix is saved as a “.mat” file for post processing. Shown in Figure D.4 is an example of the CG_string_matrix.

Function	Keyword	Redundancy	System_ Num	Mass (kg)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)	Volume (m^3)	Consequence	Cooling	Safety / Operability	Thermal
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_transceiver	1	1	4.52	0	0	0	18	1	5	2	0	1
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_transceiver	2	1	4.5143	0	0	0	18	1	5	2	0	1
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_amplifier	1	1	4.2678	0	0	0	36	1	5	2	0	1
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_amplifier	2	1	4.2373	0	0	0	36	1	5	2	0	1
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_antenna	1	1	0.18103	0	0	0	0	1	5	2	0	0
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_antenna	2	1	0.18227	0	0	0	0	1	5	2	0	0
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_steerable_antenna	1	1	12.4363	0	0	0	60	1	5	2	1	0
6.1.1 Communicate with Earth ground station (External to Ascent Module)	Long_range_data_processor	1	1	4.6872	0	0	0	14.1	1	5	2	0	1
6.1.2 Communicate with relay satellites (External to Ascent Module)	Short_range_transceiver	1	1	2.9574	0	0	0	50	1	5	2	0	1
6.1.2 Communicate with relay satellites (External to Ascent Module)	Short_range_transceiver	2	1	2.9846	0	0	0	50	1	5	2	0	1
6.1.2 Communicate with relay satellites (External to Ascent Module)	Short_range_antenna	1	1	1.0978	0	0	0	0	1	5	2	0	0
6.1.2 Communicate with relay satellites (External to Ascent Module)	Short_range_antenna	2	1	1.0956	0	0	0	0	1	5	2	0	0
6.1.2 Communicate with relay satellites (External to Ascent Module)	Short_range_data_processor	1	1	1.1876	0	0	0	25	1	5	2	0	1
7.1 Sense Subsystem Commands	Data_bus_network_boxes	1	1	16.139	0	0	0	25	1	5	2	0	1
7.1 Sense Subsystem Commands	Data_bus_network_boxes	2	1	15.9286	0	0	0	25	1	5	2	0	1
7.2 Process/Amplify Subsystem Commands	Master_event_controllers	1	1	10.2651	0	0	0	25	1	5	2	0	1
7.3 Store Spacecraft Data	Data_storage	1	1	1.124	9.81E-05	9.82E-05	9.86E-05	25	0.00017	5	1	1	1
8.1.1 Monitor Vehicle Subsystem Data	Health_monitoring_computer	1	1	8.3223	0	0	0	25	1	5	2	1	1
8.1.2 Monitor ECS Subsystems	Instrumentation_sensors_ECLSS	1	1	2.6962	0	0	0	0	1	5	2	0	0
8.1.3 Monitor Prop Subsystems	Instrumentation_sensors_Propulsion	1	1	1.6233	0	0	0	0	1	5	2	0	0
8.1.4 Monitor RCS Subsystems	Instrumentation_sensors_RCS	1	1	2.22	0	0	0	0	1	5	2	0	0
9.1.1 Sense Spacecraft Position Nav Inputs	IMU	1	1	18.9655	0	0	0	0	1	5	1	0	1
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_base	1	1	1.8203	0.000164	0.000164	0.000166	0	0.000285	5	1	0	0
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_power	1	1	9.2509	0.000238	0.000238	0.000239	25	0.000413	5	1	0	1
9.1.1 Sense Spacecraft Position Nav Inputs	Star_tracker	1	1	10.5218	0.000342	0.000342	0.000342	0	0.000592	5	1	1	0
9.1.2 Convert Analog to Digital Nav Inputs	Nav_analog_digital	1	1	6.6465	0.00011	0.000111	0.00011	25	0.000191	5	1	0	1
9.1.3 Output Navigation to Guidance	Multiplexer_demultiplexers	1	1	16.842	9.85E-05	9.73E-05	9.74E-05	25	1	5	2	0	1
9.2.1 Calculate Spacecraft Guidance	Flight_control_computer	1	1	29.4103	0.000263	0.000263	0.000263	90	0.000456	5	1	0	1
9.2.2 Communicate with Instrumentation	Instrumentation_comm	1	1	1.7393	0	0	0	25	1	5	2	0	1
9.2.2 Communicate with Instrumentation	Instrumentation_comm	2	1	1.7416	0	0	0	25	1	5	2	0	1
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw	1	1	2.1701	9.82E-05	9.85E-05	9.89E-05	0	0.000171	5	1	0	0
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw	2	1	2.1567	9.93E-05	9.75E-05	9.79E-05	0	0.00017	5	1	0	0
9.3.1 Input Human Flight Controls	Flight_translation	1	1	2.3753	9.83E-05	9.78E-05	9.80E-05	0	0.00017	5	1	0	0
9.3.1 Input Human Flight Controls	Flight_translation	2	1	2.368	9.74E-05	9.74E-05	9.89E-05	0	0.00017	5	1	0	0
9.3.2 Input Human Navigation to Computer	Computer_keyboard	1	1	7.8537	0.000127	0.000125	0.000126	25	0.000218	5	1	0	0
9.3.3 Output Spacecraft Control to Propulsion	Control_propulsion	1	1	10.7549	0	0	0	25	1	5	2	0	0
9.3.5 Display Spacecraft Control	Crew_displays_control	1	1	16.4218	0	0	0	0	1	5	0	0	0
9.3.6 Display Spacecraft Navigation	Crew_displays_navigation	1	1	3.3769	0	0	0	0	1	5	0	0	0
9.3.7 Manual Control Spacecraft Subsystems	Crew_displays_subsystems	1	1	1.9943	0	0	0	0	1	5	0	0	0

Figure D.4: Example of the CG_string_matrix output file.

D.3 SUBROUTINES

The following sections provide a top level overview of how the CLAMP code calculates the mass, power, and volume for each of the subsystems. The intent of this section is to provide information about how the subsystem components are determined and integrated in the spacecraft. Key assumptions are made with respect to the sizing of various components in each subsystem and are described in the following sections.

D.3.1 AVIONICS SUBSYSTEM

The Avionics subroutine records the spacecraft components matched to functions related to Communications, Navigation, and Command and Data Handling. This subroutine uses parameters specified in the input file and records the mass, redundancy, volume, and other parameters in the CG_string_matrix output file. Because many of the avionics components are considered “black boxes”; the mass, volume, and power of the electronics components are input directly to the program. The wiring components, “Sensor_comm_wiring” and “RF_cabling” wire lengths are estimated based upon the total number of electronic boxes. A key assumption for sizing the communication and sensor wiring is that a single wire length to each box is estimated by adding the circumference to the length of the overall spacecraft. The number of cables is determined by counting the number of electronic boxes and multiplying by two for input and output wires. This subroutine does not derive the mass, volume, and power requirements for the electronics boxes. The reason for this assumption is that in many cases, the electronic components are procured through various suppliers. Listed in Table D.1 are the functions and variable keywords for the Avionics subsystem.

Table D.1: Avionics functions and Variable Keyword identifiers.

Avionics	
Function	**Variable_Keyword**
6.1.1 Communicate with Earth ground station	Long_range_transceiver
6.1.1 Communicate with Earth ground station	Long_range_amplifier
6.1.1 Communicate with Earth ground station	Long_range_antenna
6.1.1 Communicate with Earth ground station	Long_range_steerable_antenna
6.1.1 Communicate with Earth ground station	Long_range_data_processor
6.1.2 Communicate with relay satellites	Short_range_transceiver
6.1.2 Communicate with relay satellites	Short_range_antenna
6.1.2 Communicate with relay satellites	Short_range_data_processor
6.1.3 Communicate between suited crewmembers	Interior_voice_comm
7.1 Sense Subsystem Commands	Data_bus_network_boxes
7.2 Process/Amplify Subsystem Commands	Master_event_controllers
7.3 Send Commands to Subsystems	Sensor_comm_wiring
7.4 Store Spacecraft Data	Data_storage
8.1.1 Monitor Subsystem Data	Health_monitoring_computer
8.2.1 Monitor ECS Subsystems	Instrumentation_sensors_ECLSS
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_Propulsion
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_RCS
8.2.2 Monitor Crew Accommodations	Instrumentation_sensors_CA
8.2.3 Monitor Payload Subsystem	Instrumentation_sensors_Payload
8.2.4 Monitor Power Subsystem	Instrumentation_sensors_Power
8.2.5 Monitor Communication Subsystem	Instrumentation_sensors_Comm
8.2.6 Monitor Command and Data Handling (C&DH)	Instrumentation_sensors_CDH
8.2.7 Monitor Health Monitoring Subsystem	Instrumentation_sensors_Health
8.2.8 Monitor Flight Control Subsystem	Instrumentation_sensors_Flight
8.2.9 Monitor Thermal Subsystem	Instrumentation_sensors_Thermal
8.2.9 Monitor Structures Subsystem	Instrumentation_sensors_Structural
9.1.1 Sense Spacecraft Position Nav Inputs	IMU
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_base
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_power
9.1.1 Sense Spacecraft Position Nav Inputs	Star_tracker
9.1.2 Convert Analog to Digital Nav Inputs	Nav_analog_digital
9.1.3 Output Navigation to Guidance	Multiplexer_demultiplexers
9.2.1 Calculate Spacecraft Guidance	Flight_control_computer
9.2.2 Communicate with Instrumentation	Instrumentation_comm
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw
9.3.1 Input Human Flight Controls	Flight_translation
9.3.2 Input Human Navigation to Computer	Computer_keyboard
9.3.3 Output Spacecraft Control to Propulsion	Control_propulsion
9.3.4 Display Spacecraft Control	Crew_displays_control
9.3.5 Display Spacecraft Navigation	Crew_displays_navigation
9.3.6 Manual Control Spacecraft Subsystems	Crew_displays_subsystems
9.4.1 Sense Spacecraft Abort Position	Abort_navigation
9.4.2 Sense Spacecraft Abort Velocity Inputs	Abort_control
9.4.3 Calculate Spacecraft Abort Trajectory	Abort_guidance
9.4.4 Input Abort Commands	Abort_input
9.5 Provide Rendezvous Guidance	Rendezvous_radar
9.6 Provide Inertial Reference	Ordeal
9.7 Provide Additional Avionics	Misc_avionics

D.3.2 EXTRA VEHICULAR ACTIVITY SUBSYSTEM

The EVA subroutine is dedicated to the functions necessary to conduct EVA. Spacesuits, mobility aids (tools), and umbilical lines are components dedicated to EVA. Similar to the avionics system, the EVA subroutine records the components in the CG_string_matrix output file. The number of spacesuits and umbilical hoses are determined based upon the number of crew. The program has the capability to add redundant EVA components if necessary. For example, if two types of suits such as ascent pressure suits and EVA suits will be carried in the spacecraft, the EVA suits could be specified as redundant components and the mass identified as a "backup" mass. Listed in Table D.2 are the functions and variable keywords for the EVA subsystem.

Table D.2: EVA functions and Variable Keyword identifiers.

EVA	
Function	**Variable_Keyword**
12.1.1 Provide Space Suit	Spacesuits
12.2.1 Provide Mobility Aids	EVA_mobility
12.2.2 Provide Umbilical	Suit_umbilical

D.3.3 CREW ACCOMMODATIONS SUBSYSTEM

The Crew Accommodations subroutine calculates the components related to the functions of Crew Accommodations. The first step in the Crew Accommodations subroutine is to record the mass of the crew. Individual crew mass is an average mass of all the crew members. The nominal value for crew mass is specified in the input file and the crew members are recorded individually in the CG_string_matrix. Other components that are a function of the number of crew such as restraints, handholds, clothing, personal storage, sleep accommodations, and hygiene consumables are multiplied by the number of crew and recorded in the

CG_string_matrix. Lighting for the cabin, panel, and exterior is specified in the input file and recorded in the CG_string_matrix. Other crew accommodation components such as Entertainment, Exercise, and Medical Kit are recorded as specified by the input file. Listed in Table D.3 are the functions and variable keywords for the Crew Accommodations subsystem.

Table D.3: Crew Accommodations functions and Variable Keyword identifiers.

Crew Accommodations	
Function	**Variable_Keyword**
3.1.1 Provide Cabin Lighting	Cabin_lighting
3.1.2 Provide Panel Lighting	Panel_lighting
3.1.3 Provide Light for Exterior Viewing	Exterior_lighting
3.1.4 Provide Restraints	CA_restraints
3.1.5 Provide Handholds	CA_handholds
3.2.1 Provide Personal Storage	Personal_storage
3.2.2 Provide Additional Clothing Storage	Clothing
3.2.3 Provide Entertainment	Entertainment
3.3.2 Provide Medical Kit Storage	Medical_kit
3.3.3 Provide Exercise Capability	Exercise
3.3.4 Provide Sleep Accommodations	Sleep_accommodations
3.3.5 Provide Operational Supplies	Operational_supplies
3.3.1 Provide Hygiene Supplies	Hygiene_consumables
3.3.1 Provide Hygiene Supplies	Hygiene_kit

D.3.4 PAYLOAD SUBSYSTEM

The Payloads subroutine is dedicated to the functions for carrying payloads and performing mission operations. The parameters of payloads are specified in the input file and similar to the previous subroutines, the Payload subroutine records the science payload, general payload, tools and equipment, and consumable equipment storage mass, volume and power in the CG_string_matrix output file.

The two main types of payload specified in this subroutine are the Science Return and the General Return. Although both types can be used, the intent of dividing the two functions is to designate the amount of sample return from the amount of other mass that is returned from the lunar surface. However, the user has the flexibility to use either one or both; because the program carries the information from the input file and records in the CG_string_matrix output file. Listed in Table D.4 are the functions and variable keywords for the Payloads subsystem.

Table D.4: Payloads functions and Variable Keyword identifiers.

Payloads	
Function	**Variable_Keyword**
4.1.1 Provide Science Payload Storage	Science_return
4.1.2 Provide General Payload Storage	General_return
4.2.1 Provide Tools Storage	Tools_equipment
4.2.2 Provide Consumable Equipment Storage	Consumable_equipment
4.2.3 Provide Space Suit Storage	Spacesuit_storage
4.2.4 Provide Photography Equipment	Photography

D.3.5 ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

The ECLSS subsystem mass, volume, and power are calculated using two subroutines. The first subroutine records the information for all of the interior components. The second subroutine determines the amount of gas storage and is calculated after all of the interior volumes of the components are determined. The second subroutine is named the ECLSS Gas Storage System and is called after the primary structure has been sized. The first subroutine of the Environmental Control and Life Support System (ECLSS) is dedicated to the functions of cabin atmosphere management, food management, and waste management. A key assumption in this subroutine is that the Carbon Dioxide removal functions utilize consumable cartridges. These cartridges are determined based upon the given mission time and number of crew. Other time and crew dependent components such as food management, potable water, and waste management are calculated from heuristic values of (kg/crewmember) × time as specified in the input file. Listed in Table D.5 are the functions and variable keywords for the ECLSS subsystem.

Table D.5: ECLSS functions and Variable Keyword identifiers.

ECLSS	
Function	**Variable_Keyword**
2.1.1 Remove Carbon Dioxide	CO2_crew_day_rate
2.1.1 Remove Carbon Dioxide	CO2_removal_rate
2.1.1 Remove Carbon Dioxide	CO2_removal_mass
2.1.10 Provide Fire Detection and Suppression	Fire_extinguisher
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_density
2.1.2 Provide Metabolic Oxygen	O2_crew_day_rate
2.1.2.1 Store High Pressure Oxygen	Storage_tank_FoS
2.1.2.1 Store High Pressure Oxygen	PaO2
2.1.2.1 Store High Pressure Oxygen	Storage_tank_radius_length_ratio
2.1.2.1 Store High Pressure Oxygen	O2_storage_temperature
2.1.2.1 Store High Pressure Oxygen	O2_storage_pressure
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_yield_strength
2.1.2.1 Store High Pressure Oxygen	Storage_tank_redundancy
2.1.2.1 Store High Pressure Oxygen	Oxygen_tank_insulation
2.1.2.2 Transport Oxygen	Oxygen_gas_transport_lines
2.1.2.3 Control Oxygen Flow	Oxygen_tank_relief_valve
2.1.2.3 Control Oxygen Flow	Oxygen_tank_regulator_valve
2.1.2.4 Measure pp Oxygen Level in Cabin Air	Oxygen_pp_gauge
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_temperature
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_pressure
2.1.3.1 Store High Pressure Makeup Gas	Nitrogen_tank_insulation
2.1.3.2 Transport Makeup Gas	Nitrogen_gas_transport_lines
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_relief_valve
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_regulator_valve
2.1.4 Remove Trace Contaminants	Trace_Contaminants_filter
2.1.5 Remove Particulates	Particulates_filter
2.1.6 Remove Humidity	Humidity_percent
2.1.6.2 Condense Cabin Air	Condense_air_heat_exchanger
2.1.6.1 Provide Humidity Separator	Humidity_capture
2.1.7.1 Provide Air Circ Bypass	Mix_air_valve
2.1.7.2 Measure Cabin Air Temp	Cabin_temp
2.1.7.2 Measure Cabin Air Temp	Temperature_sensor
2.1.8.1 Circulate Air	Cabin_fan
2.1.8.2 Return Air	Return_air_ducts
2.1.8.3 Output Air	Direct_air_ducts
2.1.9.1 Provide Air Pressure Sensor	Air_Pressure_gauge
2.1.9.2 Provide Cabin Air Control Logic	Cabin_leak_percent
2.1.9.2 Provide Cabin Air Control Logic	Cabin_atm_pressure
2.1.9.2 Provide Cabin Air Control Logic	Cabin_air_control
2.2.1 Store Food	Food_storage
2.2.1 Store Food	Food_rate
2.2.2 Prepare Food	Galley_supplies
2.2.2 Prepare Food	Heat_food
2.3.1 Provide Potable Water	Potable_water
2.3.2 Provide Hygiene Water	Hygiene_water
2.4.1 Collect Urine	Urine_bags
2.4.2 Collect Fecal Solids	Diapers
2.4.2.2 Control Odor	Control_odor
2.4.2.3 WCS supplies	WCS_supplies
2.4.3 Collect Liquid Waste	Vacuum
2.4.4 Collect Solid Waste	Trash_bags
2.4.4 Collect Solid Waste	Housecleaning_supplies
2.4.5 Jettison Liquid Waste	Dump_Valve_Piping

D.3.6 POWER SUBSYSTEM

The Power subroutine calculates the power needs for the spacecraft. The total power required for the previous subsystems is determined and used to size batteries and power subsystem components. This subroutine is the first to utilize the information from the previous subsystems in order to determine mass and volume. The power subsystem is not entirely enclosed in the crew cabin and components that are located outside the pressurized volume can are specified in the input file. The power subsystem is a very challenging subsystem to calculate because of the dependencies to other subsystems. For this analysis code, the amount of battery mass is determined from the total amount of power required for the mission time. Redundancy in the batteries is determined by dividing the total amount of power needed for the mission by the amount of redundancy. The key assumption is that in the case of a failure in the batteries, a redundant set of batteries would be utilized and operational procedures would be set in place to mitigate the contingency. Battery mass is a huge driver in spacecraft mass and redundancy in batteries is a safety concern that should be addressed by the operational requirements at a later point in the design. Battery mass is calculated with heuristic values of kg/kW. Similar to battery mass, the wiring mass is estimated using a kg/kW relationship. As the spacecraft design matures, wiring mass can be more efficiently estimated through mockups or CAD. For the purposes of this estimating code, the heuristic approach was appropriate given the configuration of the spacecraft was yet to be determined. Other components such as power controllers, circuit breakers, inverters, and relays are recorded in the CG_string_matrix output file. Listed in Table D.6 are the functions and variable keywords for the Power subsystem.

Table D.6: Power functions and Variable Keyword identifiers.

Power	
Function	**Variable_Keyword**
5.1 Provide Power	Batteries
5.1 Provide Power	Depth_of_Discharge
5.1 Provide Power	Duty_cycle_coast
5.2 Distribute Power	Power_Distr_Wiring
5.3 Regulate Power	Power_Distr_Controller
5.3 Regulate Power	Power_inverters
5.3 Regulate Power	Power_relays
5.6 Provide Overload Protection	Power_Distr_CircBreak

D.3.7 ACTIVE THERMAL SYSTEM

The thermal subroutine calculates the functions associated with the thermal subsystem. This subroutine begins by summing the total power needs of the spacecraft in watts (W). The program does not include the amount of heat generated by the crew. The total amount of heat that needs to be rejected includes all of the components that utilize power and the batteries. The total heat is adjusted for mission time in heat rejection. For components that require cooling, a key assumption is that cold plates are used for components above a specified wattage. The number of cold plates is determined by summing all of the components that are greater than the minimum wattage level as specified in the input file. Another key assumption is that all batteries utilize cold plates. The cold plates are recorded in the CG_string_matrix according to the component that requires cooling in the format of component_coldplate. The number of coolant loops is determined by dividing the total heat by a “loop capacity”. The key assumption is that a single loop will remove a specified number of watts of heat. As the total heat number increases through redundant components, the total number of loops increases. For each loop, a set of valves and pumps is determined based on the input parameters and redundancy specified in the input file.

Radiators or sublimators are used to remove heat from the spacecraft in a vacuum environment. A key assumption in the program is that the spacecraft uses sublimators. The mass of the sublimator is determined from a kg/kW heat rejected relationship. The amount of water needed for the sublimator is determined by multiplying the total heat (W) by the mission duration and dividing by the latent heat of vaporization at vacuum conditions.

The last part of the thermal subroutine calculates the mass of the cooling loops and fluid. The cooling fluid density is given in the input file and the mass of coolant line is determined from the material density and the outer diameter of the line. The inner diameter of the coolant line is assumed to be 85% of the outer diameter. This assumption was made loosely on the wall thickness of a 1/2" inch outer diameter coolant line with approximately 7/16" inside diameter. The mass of the fluid is calculated based upon the volume of the inner diameter. A key assumption for a single coolant line length is the spacecraft circumference plus the length. Two single coolant line lengths are added to determine the overall loop mass (input and output). Although it is an over estimate of an actual thermal system, it is a conservative estimate for the thermal needs of the spacecraft. Listed in Table D.7 are the functions and variable keywords for the Thermal subsystem.

Table D.7: Thermal functions and Variable Keyword identifiers.

Thermal	
Function	**Variable_Keyword**
10.1 Collect Heat	Coldplate_min_power
10.1 Collect Heat	Loop_capacity
10.1 Collect Heat	Loop_diameter
10.1 Collect Heat	Thermal_interior_fluid_density
10.1 Collect Heat	Coolant_piping
10.1 Collect Heat	Coldplates
10.2 Transport Heat	Interior_coolant_pumps
10.2 Transport Heat	Interior_coolant_lines_valves
10.3 Remove Heat	Interior_exterior_heat_exchanger
10.3 Remove Heat	Water_system_sublimators
10.3 Remove Heat	Exterior_radiators

D.3.8 ECLSS GAS STORAGE SYSTEM

The ECLSS gas storage subroutine determines the mass of the high pressure gas storage system. In order to determine the amount of oxygen and makeup gas, the total spacecraft habitable volume is adjusted by the volume of the interior components. Because the interior components will not always be neatly packaged, a "packaging efficiency" is used to adjust the volume of the interior components. The two volumes are added to determine the total pressurized volume. Shown in Figure D.5 is an illustration of the derivation of total pressurized volume.

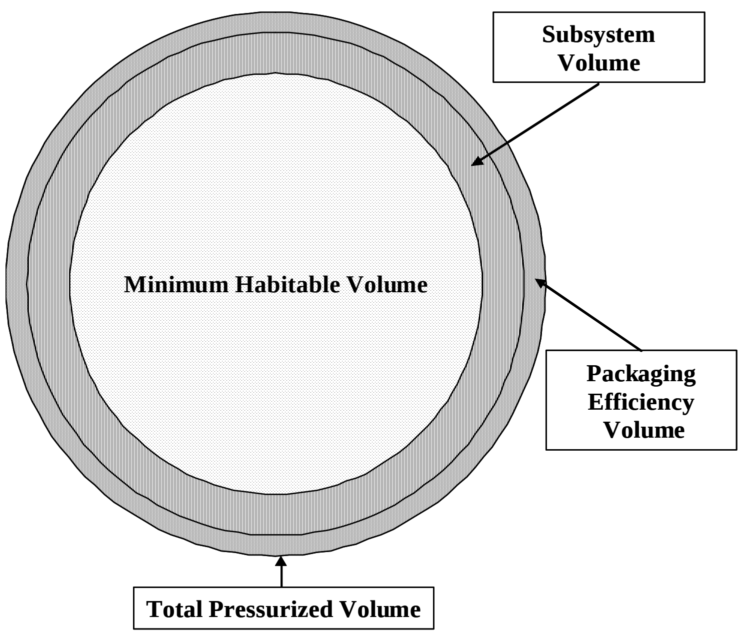


Figure D.5: Derivation of total pressurized volume.

The derived pressurized volume used to determine the inner diameter of the pressure vessel. This diameter is used in the primary structures subroutine to calculate the mass of the pressure vessel and supporting structure. The amount of oxygen needed is a function of the pressurized volume, crew members, and mission duration. The amount of makeup gas is a function of the atmospheric composition and cabin leak rate. The total mass of Oxygen needed by the crew is determined by:

$$\textit{Oxygen Mass} = \textit{Number Crew} \times (\textit{Oxygen Daily Rate}) \times (\textit{Mission Duration (hrs)} / 24) \quad \text{(D.1)}$$

The total mass of makeup gas needed for the spacecraft is determined by:

$$\begin{aligned}\textit{Oxygen Makeup} = {} & [(\textit{Cabin Daily Leak Percent} \times \textit{Cabin Oxygen Mass})/24] \\ & \times \textit{Mission Duration (hrs)} \qquad \text{(D.2)}\end{aligned}$$

$$\begin{aligned}\textit{Makeup Gas} = {} & [(\textit{Cabin Daily Leak Percent} \times \textit{Cabin Makeup Gas Mass})/24] \\ & \times \textit{Mission Duration (hrs)} \qquad \text{(D.3)}\end{aligned}$$

Where the Oxygen and Makeup Gas in the cabin are calculated using the ideal gas relationship:

$$\begin{aligned}\textit{Cabin Oxygen Mass} = {} & (\textit{Oxygen Partial Pressure} \times \textit{Cabin volume}) \\ & \div ((\textit{Ru} / \textit{Molecular weight of Oxygen}) \times \textit{Cabin Temperature}) \qquad \text{(D.4)}\end{aligned}$$

$$\begin{aligned}\textit{Cabin Makeup Gas Mass} = {} & (\textit{Makeup Gas Partial Pressure} \times \textit{Cabin volume}) \\ & \div ((\textit{Ru} / \textit{Molecular weight of Oxygen}) \times \textit{Cabin Temperature}) \qquad \text{(D.5)}\end{aligned}$$

The total amount of Oxygen and Makeup Gas is summed using the following expressions:

$$\textit{Total Oxygen Mass} = \textit{Crew Oxygen} + \textit{Oxygen Makeup} + \textit{Cabin Oxygen Mass} \quad \text{(D.6)}$$

$$\textit{Total Makeup Gas Mass} = \textit{Makeup Gas} + \textit{Cabin Makeup Gas} \quad \text{(D.7)}$$

The program has the capability to calculate a one or two gas system. Given the cabin atmospheric pressure and the partial pressure needed for Oxygen, the amount of makeup gas is based upon the partial pressure of the makeup gas. If the partial pressure for Oxygen is equal to the cabin atmospheric pressure, then makeup gas will not be included. The storage of Oxygen and Makeup gas is determined using ideal gas equation for the volume at high pressure. The storage tanks are sized according to the volume of the gas, the material density of the tank, and a factor of safety and are assumed to be spherical. The following expression is used to size the high pressure gas tanks:

$$\textit{Tank Volume} = (\textit{Mass} \times (\textit{Ru} / \textit{Molecular weight of gas})) \times \textit{Storage Temperature} \div \textit{Storage Pressure} \tag{D.8}$$

$$\textit{Spherical Radius} = \textit{Tank Volume} \div ((4/3\pi)^{1/3} \tag{D.9}$$

$$\textit{Tank Wall Thickness} = (\textit{Storage Pressure} \times \textit{Spherical Radius}) \div 2 \times (\textit{Storage Tank Yield Strength} / \textit{Factor of Safety}) \tag{D.10}$$

Once the tank mass and volume has been determined, the amount of insulation is calculated based upon the outer surface area of the tank. After the tanks have been sized, the transport lines are calculated. Using the same material, pressure and factor of safety for the storage tanks, the outer diameter of the lines are sized based on the given input diameter and the material. The storage tanks, insulation, and transport lines are recorded in the CG_string_matrix. Other components such as relief valves and regulator valves specified in the input file are recorded in the CG_string_matrix according to redundancy. The ECLSS Gas Storage components are assumed to be mounted external to the crew cabin.

B.3.9 SECONDARY STRUCTURES SUBSYSTEM

The Secondary Structures subroutine calculates the mass of secondary structural components. Because the exact configuration of the spacecraft has yet to be determined, this mass is difficult to determine in the early conceptual design phase. However, a factor can be utilized to estimate the mass of secondary structures for mounted components.

$$\textit{Secondary Structure Mass} = (\textit{Sum of Component Mass} - \textit{Crew Mass}) \times \textit{Factor} \quad \text{(D.11)}$$

The mass of Secondary Structures is on the order of 21% of the total component mass. This is a conservative estimate that can be adjusted according to the loading on the spacecraft and the configuration of the components. It is difficult to accurately predict secondary structure mass until detailed information about the dynamics of the spacecraft is quantified. The purpose of including Secondary Structures was to include mass that might otherwise be overlooked in the estimating process.

B.3.10 PRIMARY STRUCTURES SUBSYSTEM

The Primary Structures subroutine is one of the more important parts of the spacecraft estimating code. Because the exact configuration of the structure has yet to be determined, a skin-stringer approach was chosen to estimate the loading on the pressure vessel and the structure. The equations used for predicting loading on structural components were textbook equations with various conservative assumptions on the loading. To calculate the Primary Structure mass, the overall subroutine was divided into three parts. The first subroutine, named Struct_1, develops a node matrix of mounting locations to the primary structure. Assuming the mounting locations for components is located at the intersection of beams and hoops; the matrix for loading on the primary structure at the node locations is developed. Shown in Figure D.6 is an example of the node locations at the intersection of beams and hoops around the primary structure.

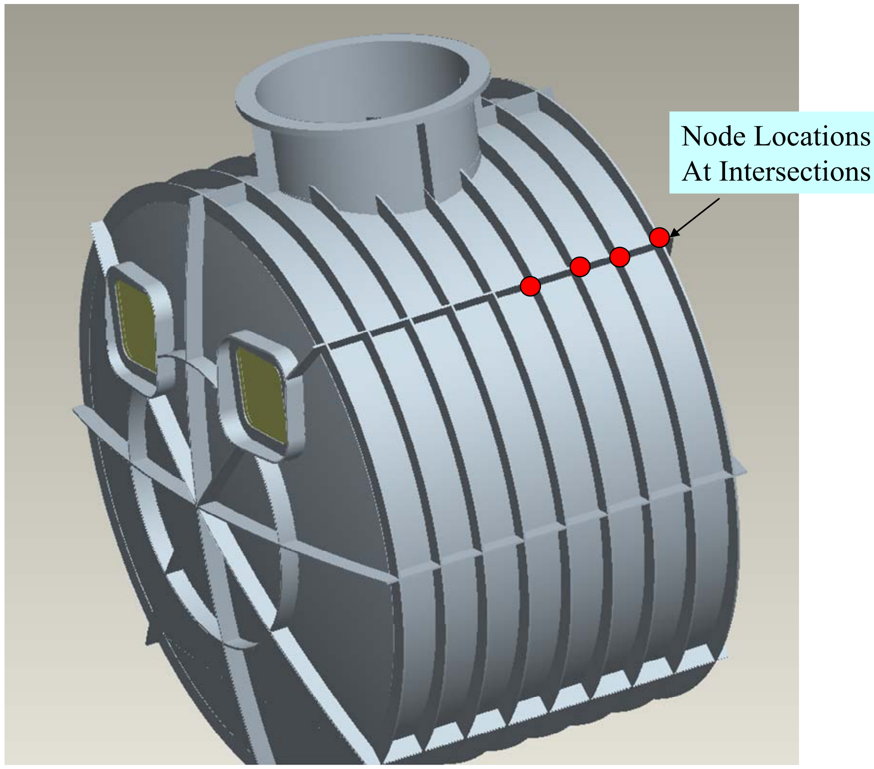


Figure D.6: Node locations at intersection of beams and hoops used for loading matrix.

In order to size the loading on the beams and hoops of the primary structure, the total amount of mass in the spacecraft is summed. This “payload” mass is used to create a distribution profile on the primary structure. To determine the amount of mass at the individual nodes, an assumption that the majority of the heavier components would be mounted toward the bottom of the primary structure and a slope of mass vs. angle was developed to simulate the mounting conditions of the components on the structure.

The next subroutine, Struct_2, calculates components in the primary structure such as the pressure vessel wall thickness, docking tunnel, floor beams, floor plate, docking ring flange, tunnel supports, docking flange support, hatch flanges, windows, and view ports. These components are updated in the node loading matrix.

Given the material yield strength, the cabin atmospheric pressure, factors of safety and loading, the pressure vessel cylinder wall thickness is calculated by:

$$\textit{Pressure Vessel Wall Thickness} = (\textit{Cabin Pressure} \times \textit{Inner Diameter}) \div 2 \times (\textit{Material Yield Strength}/(\textit{FoS} \times \textit{Loading})) \quad \text{(D.12)}$$

Because the ends of the cylinder are dome shaped, the equivalent spherical diameter is determined using the protrusion ratio given in the input file. This protrusion ratio is defined as:

$$\textit{Protrusion Distance} = \textit{Protrusion Ratio} \times \textit{Inner Diameter} \quad \text{(D.13)}$$

And the radius of curvature is determined as:

$$\textit{Radius of Curvature} = (\textit{Inner Diameter}/2)^2 + (\textit{Protrusion}^2) \div 2 \times \textit{Protrusion} \quad \text{(D.14)}$$

The thickness of the spherical section is calculated by:

$$\textit{Spherical Wall Thickness} = (\textit{Cabin Pressure} \times \textit{Radius of Curvature}) \div 2 \times (\textit{Material Yield Strength}/(\textit{FoS} \times \textit{Loading})) \quad \text{(D.15)}$$

The volume of the pressure vessel skin including the spherical end caps is calculated and the mass of the pressure vessel skin is determined by multiplying the volume by the material density.

Another important part of the primary structure is the floor beams and decking. The maximum loading on the floor beams is calculated as a concentrated load of the total crew mass times the acceleration loading in the center of the beam. Two floor beams provide support in the crew cabin. One beam is at the mid section of the cylinder and the main beam is in the middle of the floor and runs front to aft. Shown in Figure D.7 are the floor beams and decking in the primary structure.

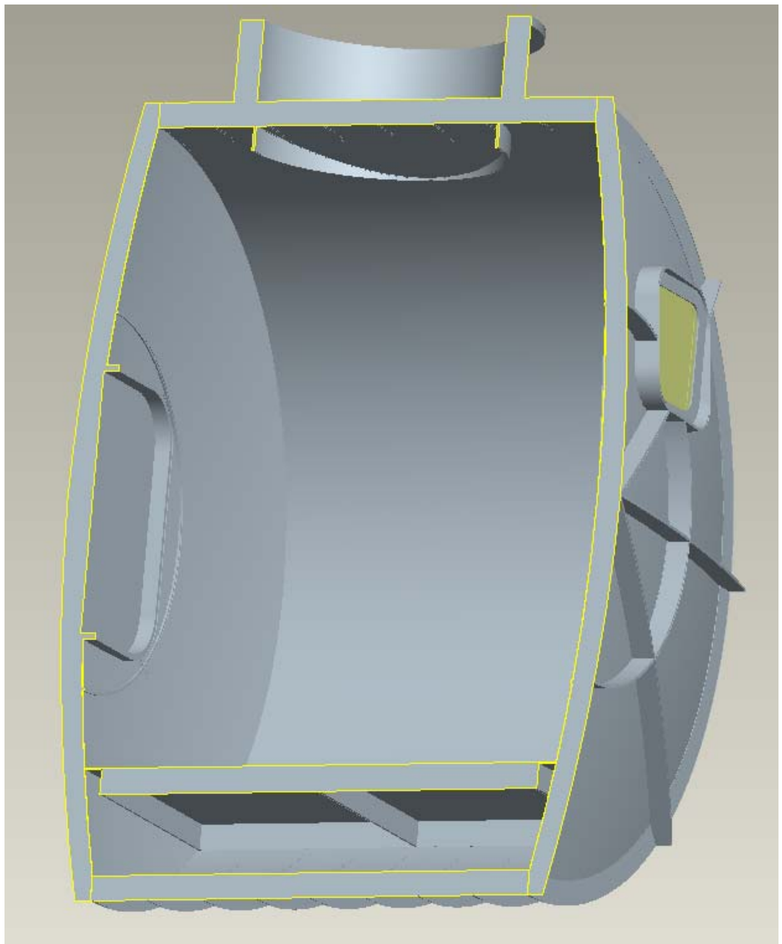


Figure D.7: Cross sectional view showing floor beams and decking.

The maximum moment due to bending in the floor beams was used to size the floor beams and was calculated using the following expression:

$$\textit{Floor Beam Bending Moment} = ((\textit{Crew Mass} \times \textit{Loading})/2) \times (\textit{Length of beam}/2) \qquad \text{(D.16)}$$

The base of the floor beam was assumed as a ratio of height to base as given in the input file

$$\textit{Beam base} = \textit{Beam height} / \textit{Height to Base Ratio} \tag{D.17}$$

The height of the floor beam is calculated as:

$$\begin{aligned}\textit{Beam height} = \textit{Height to Base Ratio} \times \textit{Bending Moment} \times 6 \\ \div \textit{Material Yield Strength} / (\textit{FoS} \times \textit{Loading})\end{aligned} \tag{D.18}$$

Given the height and base dimensions of the floor beam, the mass of all the beams was determined by multiplying the volume times the material density.

For this example, the simplest method of determining floor plate thickness is to assume the loading in the center of the floor plate. One floor plate is a "quadrant" of the floor beams. The floor plates were assumed constrained at the edges and the moments were distributed along the edges. The largest thickness was used in the calculation of the floor plate mass. The following equations were used to size the floor decking.

$$\begin{aligned}&Y = \textit{Width of Floor} / 2 \\ &Z = \textit{Length of Floor} / 2 \\ &\textit{Crew Weight} = \textit{Crew Mass} \times 9.81\end{aligned}$$

$$\begin{aligned}\textit{Thickness} = [(3/2) \times (Y / Z) \times \textit{Crew Weight} \\ \div (\textit{Beam Yield Strength} / (\textit{FoS} \times \textit{Loading}))]^{1/2}\end{aligned} \tag{D.19}$$

The docking ring flange was calculated using the maximum expected loading force on the ring. The maximum loading on the ring was assumed to be distributed, and the docking ring was sized according to equations of rings on multiple supports. Shown in Figure D.8 is the docking ring and supporting structure.

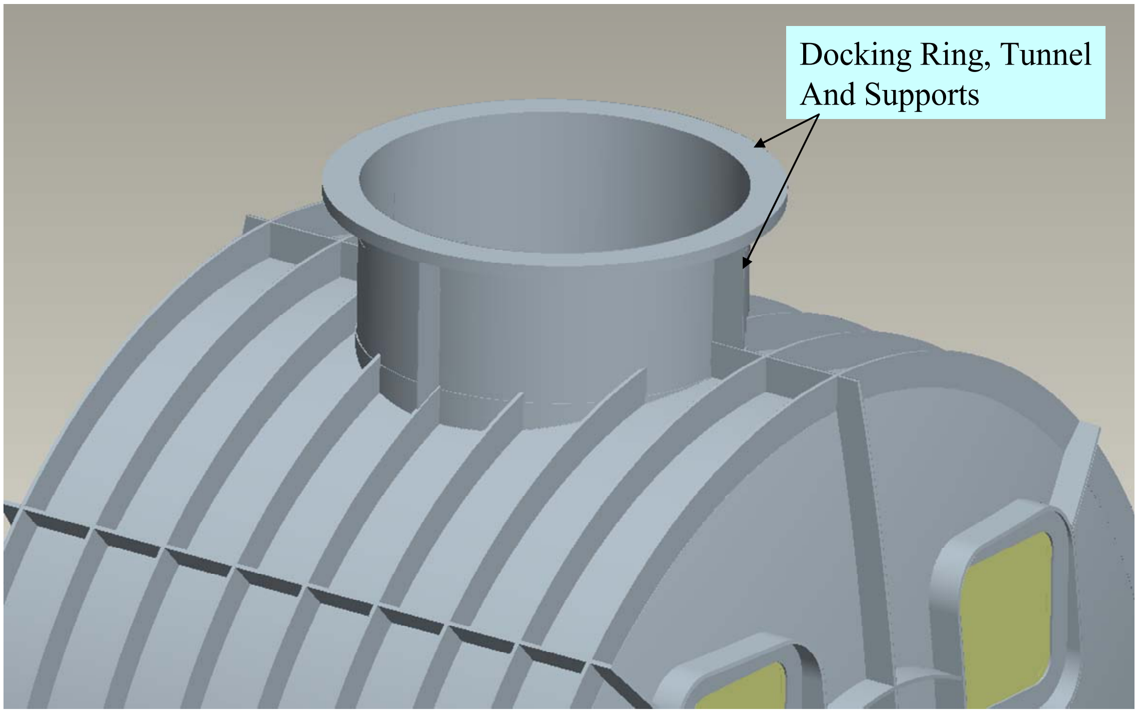


Figure D.8: Docking ring, tunnel and support structure.

The tunnel and docking flange support cross sections were sized according to the base and height dimensions of the docking ring. The docking hatch outer and inner structural rings that hold the hatch door was sized according to the same base and height dimensions.

The aft bulkhead hatch is sized according to dimensions given in the input file. The flanges of the hatch are calculated using a flange ratio of the major hatch dimension. Shown in Figure D.9 is the aft bulkhead hatch.

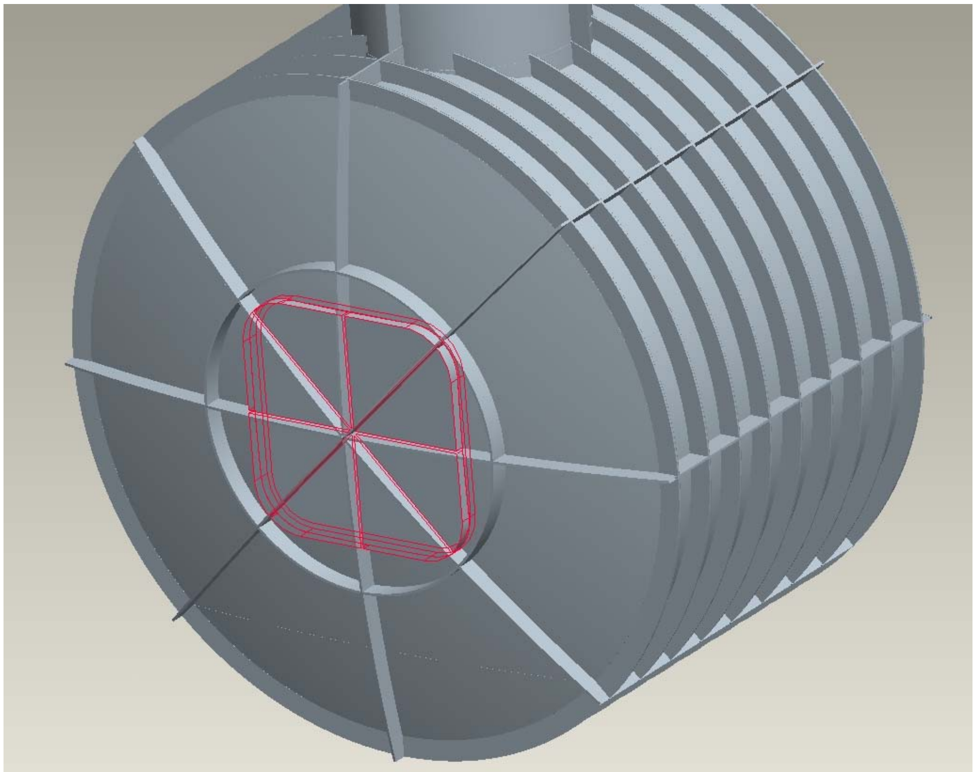


Figure D.9: Aft bulkhead hatch.

The primary structure assumes two windows with two window panes. One pane is for thermal management and the other for maintaining pressure. The overall window assembly frame thickness is 5 window pane thicknesses. Three window frames, outer, mid, and inner hold the window pane assembly and mount to the primary structure spherical hoops on the forward bulkhead. The window frame thickness is specified in the input file. The position of the windows on the forward bulkhead is a function of height above the crew floor. The center of the window is positioned at a specified height. The lateral distance of the window from the centerline axis of the vehicle is specified at angles from the mid planes of the cylinder. Shown in Figure D.10 are the forward windows.

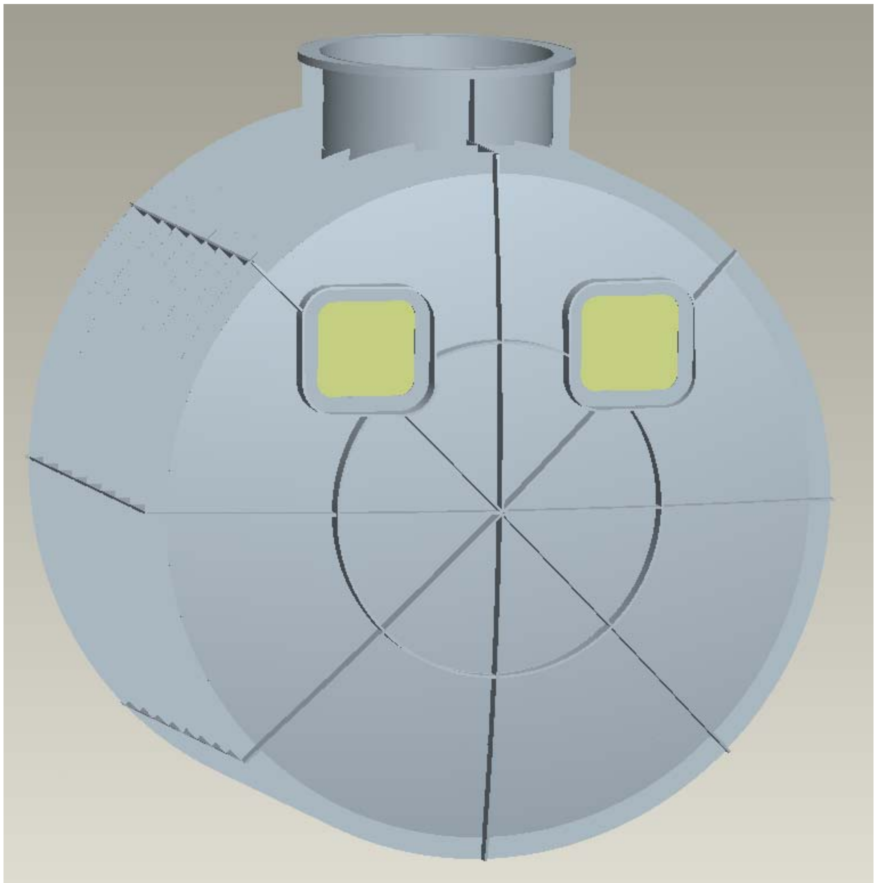


Figure D.10: Forward windows and front of primary structure.

A docking view port window was included in the design and is assumed to be a circular window above the commander's window. The total window frame assembly thickness is 5 window pane thicknesses, with two window panes. The mass for the docking frames and the window panes are added together for the docking view port assembly mass. An aft viewing port was included in the code and was calculated in the same manner as the docking port.

The last part of the Struct_2 subroutine updates the node loading matrix with the newly calculated components of the primary structure including pressure vessel skin, floor beams, floor plates, docking tunnel, docking ring, docking hatch, front windows, and aft hatch.

The final Primary Structure Subroutine is Struct_3. This subroutine sizes the outer hoops and stringers based on the mass information compiled in the node loading matrix. The subroutine first identifies the stringer with the worst loads. A single stringer is calculated and the mass of each stringer is added to the nodes in the node loading matrix. The Struct_3 subroutine also calculates the necessary height of the hoops based upon the loading at each of the nodes. Using the height to base ratio, the base of the hoop is calculated based on the hoop height. The mass at the nodes is adjusted for total expected acceleration loading and the moment at the node locations is calculated using the following equation:

$$\textit{Moment at Nodes} = PR\,(\cos\theta - 1) - HR\sin\theta \qquad \text{(D.20)}$$

Where P is the tangential force acting at the node, R is the radius of curvature, and H is the normal force acting at the node. Because the nodes are located circumferentially around the cylinder, the angle from the bottom of the cylinder is used as the 0 degree reference. The moments at the nodes are summed at the lower portion of the cylinder and the maximum bending stress in the hoop is calculated with the following equation:

$$\textit{Maximum Bending Stress} = \Sigma\,\textit{Node Moments} / ((\textit{base} \times \textit{height}^2)/6) \qquad \text{(D.21)}$$

The maximum bending stress is used to size the hoops according to the maximum allowable yield strength and the factor of safety. A data matrix named the Struct_mass matrix is used as a placeholder matrix to record the mass of all the primary structure components. The total Primary Structure mass is summed using the values in the Struct_mass matrix. The Primary Structure mass is not yet recorded in the CG_string_matrix because of the need to iterate with the TPS /

MMOD subroutine. Listed in Table D.8 are the functions and variable keywords for the Primary Structures subsystem.

Table D.8: Primary Structures functions and Variable Keyword identifiers.

Primary Structures	
Function	**Variable_Keyword**
1.1.1 Provide protection from vacuum	Factor_of_Safety
1.1.1 Provide protection from vacuum	Factor_of_Safety_PV_skin
1.1.1 Provide protection from vacuum	Interior_diameter_hab_volume
1.1.1 Provide protection from vacuum	Cylinder_length_hab_volume
1.1.1 Provide protection from vacuum	Packaging_efficiency
1.1.1 Provide protection from vacuum	Protrusion_ratio
1.1.1 Provide protection from vacuum	Pressure_vessel_material_yield_strength
1.1.1 Provide protection from vacuum	Pressure_vessel_material_density
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Elasticity
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Rigidity
1.2 Provide Load Bearing Capability	Max_acceleration_gs
1.2 Provide Load Bearing Capability	Number_hoops
1.2 Provide Load Bearing Capability	Number_stringers
1.2 Provide Load Bearing Capability	Height_to_base_ratio
1.2 Provide Load Bearing Capability	Fillet_ratio
1.5.3 Provide Docking Mechanism for Lunar Habitat to Lu	LIDS_published_weight
1.5.1 Support Docking Mechanism for Lunar Ascent Modu	Docking_ring_load_Force
1.5.2 Provide Ingress/Egress for Lunar Ascent Module to	LIDS_tunnel_diameter
1.5.2 Provide Ingress/Egress for Lunar Ascent Module to	LIDS_tunnel_height
1.5.4 Provide Ingress/Egress for Lunar Habitat to Lunar A	Hatch_width
1.5.4 Provide Ingress/Egress for Lunar Habitat to Lunar A	Hatch_height
1.5.4 Provide Ingress/Egress for Lunar Habitat to Lunar A	Hatch_corner_radius_ratio
1.5.4 Provide Ingress/Egress for Lunar Habitat to Lunar A	Hatch_flange_ratio
1.6.1 Provide Forward Windows	Window_material_density
1.6.1 Provide Forward Windows	Window_pane_thickness
1.6.1 Provide Forward Windows	Window_width_horizontal
1.6.1 Provide Forward Windows	Window_length
1.6.1 Provide Forward Windows	Window_corner_radius_ratio
1.6.1 Provide Forward Windows	Window_frame_ratio
1.6.1 Provide Forward Windows	Window_position_angle
1.6.1 Provide Forward Windows	Window_center_height
1.6.2 Provide Docking Viewing Ports	Docking_window
1.6.3 Provide Aft Viewing Ports	Aft_window
1.1.2 Provide MMOD/TPS protection	MMOD_standoff
1.1.2 Provide MMOD/TPS protection	LEO_time_days
1.1.2 Provide MMOD/TPS protection	PNP_target
1.1.2 Provide MMOD/TPS protection	TPS_blanket
1.1.2 Provide MMOD/TPS protection	LEO_altitude

D.3.11 THERMAL PROTECTION SYSTEM

The TPS_MMOD_iterate subroutine calculates the required TPS and MMOD thickness given a probability of no penetration. To determine the required shielding, the TPS_MMOD subroutine is iterated with Struct_3 to determine the optimum standoff height for a whipple shield of MMOD protection. The MMOD shield is a two layer whipple shield of the same material as the primary structure. The thicknesses of the outer and inner shield walls are calculated given the standoff distance of the hoop. The ballistic limit equations follow the instructions for sizing whipple shields as given in NASA TP – 2003 - 210788 Meteroid / Debris Shielding. Once the MMOD protection has been calculated, the surface area of the TPS is calculated and using the input surface area parameters specified in the input file for TPS blankets, the mass for TPS is calculated. The TPS protection is assumed to be mounted inside the MMOD protection. The TPS for the Lunar Ascent Module assumed Nextel and Saffil blanket insulation.

D.3.12 PROPULSION INERT COMPONENTS

The Propulsion inert components subroutine records the inert components of the propulsion subsystem. Propellant storage tanks and related components are calculated after the propellant mass has been determined. Many of the inert components include valves, filters, pressure regulators and relief valves. The two parts of the propulsion system are divided between the main engine and the RCS system. Each component is recorded in the CG_string_matrix according to redundancy or safety. The main engine pressurant, fuel, and oxidizer lines are sized according to specified pressure in the lines and the material yield strength. Trapped propellant in the Fuel and Oxidizer lines is also accounted for in the mass estimate. Components such as trim orifices, filters, isolation valve, pressure regulators, explosive valves, bipropellant valves, and solenoid valves are input in the CG_string matrix based on the desired redundancy specified in the input file. The RCS system is sized using the same methods as the Main Engine. For the RCS system, it is assumed that multiple systems of thrusters will be mounted on the spacecraft. The user has the option to choose the number of systems and the number of thrusters in each system. Listed in Table D.9 are the functions and variable keywords for the Propulsion subsystem.

Table D.9: Propulsion functions and Variable Keyword identifiers.

Propulsion	
Function	**Variable_Keyword**
11.1.6 Provide Main Engine	Specific_impulse
11.1.6 Provide Main Engine	Vacuum_thrust
11.1.6 Provide Main Engine	Main_engine_weight_thrust_ratio
0.1.4 Launch Trajectory	Gravity_constant
0.1.4 Launch Trajectory	Moon_gravity_parameter
0.1.4 Launch Trajectory	Moon_radius
0.1.4 Launch Trajectory	Orbital_apogee_altitude
0.1.4 Launch Trajectory	Orbital_perigee_altitude
0.1.4 Launch Trajectory	Launch_azimuth
0.1.4 Launch Trajectory	Launch_latitude
0.1.4 Launch Trajectory	Launch_longitude
11.1.6 Provide Main Engine	Oxidizer_fuel_ratio
11.1.2 Provide Main Engine Oxidizer	Oxidizer_density
11.1.1 Provide Main Engine Fuel	Fuel_density
11.1.7 Store Main Engine Propellant	Fuel_ox_tank_press
11.1.11 Provide Contingency Propellant	Contingency_propellant
11.1.3 Provide Main Engine Pressurant	Pressurant_pressure
11.2.3 Provide RCS Pressurant	Pressurant_pressure_RCS
11.1.3 Provide Main Engine Pressurant	Pressurant_tank_temperature
11.1.3 Provide Main Engine Pressurant	Pressurant_ratio_spec_heats
11.1.3 Provide Main Engine Pressurant	Pressurant_gas_constant

Table D.9: Propulsion functions and Variable Keyword identifiers (continued).

Propulsion	
Function	**Variable_Keyword**
11.1.7 Store Main Engine Propellant	Propulsion_tank_material_density
11.1.7 Store Main Engine Propellant	Propulsion_tank_yield_strength
11.1.7 Store Main Engine Propellant	Propulsion_tank_factor_of_safety
11.1.7 Store Main Engine Propellant	Propulsion_tank_unused_fraction
11.2.7 Store RCS Propellant	Fuel_ox_tank_press_RCS
11.1.9 Transport Main Engine Propellant	Propulsion_line_id
11.1.5 Transport Main Engine Pressurant	Main_engine_pressurant_line_id
11.2.9 Transport RCS Propellant	Propulsion_line_id_RCS
11.2.5 Transport RCS Pressurant	Pressurant_line_id_RCS
11.1.6 Provide Main Engine	Main_engine_linear_slope
11.1.6 Provide Main Engine	Main_engine_y_intercept
11.1.12 Support Main Engine Components	Propulsion_secondary_struct_ratio
11.2.6 Provide RCS thrusters	RCS_side_A_B_choice
11.2.6 Provide RCS thrusters	Specific_impulse_RCS
11.2.6 Provide RCS thrusters	Vacuum_thrust_RCS
11.2.6 Provide RCS thrusters	RCS_engine_linear_slope
11.2.6 Provide RCS thrusters	RCS_engine_y_intercept
11.2.6 Provide RCS thrusters	Num_RCS_engines_system
11.2.6 Provide RCS thrusters	Num_fire_RCS_thrust_ascent
11.2.11 Provide RCS contingency propellant	RCS_additional_propellant_mass
11.1.13 Insulate Main Engine Propellant	Propulsion_tanks_insulation
11.2.13 Insulate RCS Propellant	Propulsion_tanks_insulation_RCS
11.2.1 Provide RCS Propellant	RCS_propellant
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_valve
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_filter
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_solenoid_valve
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_pressure_regulator
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_check_valve
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_explosive_valve
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_relief_valve
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_trim_orifice
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_filter
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_isolation_valve
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_bipropellant_valve

Table D.9: Propulsion functions and Variable Keyword identifiers (continued).

Propulsion	
Function	**Variable_Keyword**
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_valve
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_filter
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_solenoid_valve
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_pressure_regulator
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_check_valve
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_explosive_valve
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_relief_valve
11.1.10.1 Control Main Engine Fuel	Fuel_trim_orifice
11.1.10.1 Control Main Engine Fuel	Fuel_filter
11.1.10.1 Control Main Engine Fuel	Fuel_isolation_valve
11.1.10.1 Control Main Engine Fuel	Fuel_bipropellant_valve
11.2.4 Control RCS Pressurant	System_pressurant_valve_RCS
11.2.4 Control RCS Pressurant	System_pressurant_filter_RCS
11.2.4 Control RCS Pressurant	System_pressurant_orifice_RCS
11.2.4 Control RCS Pressurant	System_pressurant_pressure_regulator_RCS
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_check_valve_RCS
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_relief_valve_RCS
11.2.4 Control RCS Pressurant	Fuel_pressurant_check_valve_RCS
11.2.4 Control RCS Pressurant	Fuel_pressurant_relief_valve_RCS
11.2.10.2 Control RCS Oxidizer	Oxidizer_shutoff_valve_RCS
11.2.10.1 Control RCS Fuel	Fuel_shutoff_valve_RCS
11.2.9.1 Filter RCS Propellant	Fuel_Ox_filter_RCS
11.2.10 Control RCS Propellant	Fuel_Ox_isolation_valve_RCS
11.2.14.2 Heat RCS Oxidizer lines	Oxidizer_heater_RCS
11.2.14.1 Heat RCS Fuel lines	Fuel_heater_RCS
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_valve
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_filter
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_valve
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_filter

D.3.13 PROPULSION FLYOUT TABLE

To reduce the computational burden of analyzing many trajectories during a Monte Carlo analysis, a propulsion look up table is developed before the analysis begins. Starting from a minimum functional configuration, the amount of main engine propellant and RCS propellant is derived for varying levels of spacecraft mass. The program calculates a range of potential spacecraft mass from 0.99 of the minimum functional configuration to 6 times the minimum functional mass. This range covers the anticipated spacecraft mass expected during a tradespace run. Using the orbital parameters given in the input file, an ideal delta V is calculated as a starting point condition and used to provide an initial value for the main engine propellant and RCS propellant mass. The program steps through an iteration routine where tanks are sized according to the predicted propellant mass ant the spacecraft is flown in a 3 degree of freedom trajectory from the lunar surface. Because the purpose of the subroutine is to determine an estimate for propellant mass, the trajectories are not optimized. The lunar ascent trajectory is a three part ascent with vertical rise, pitch over, and gravity turn to orbital insertion. The program flies the trajectory, determines the amount of propellant used to reach the orbital insertion and iterates again with new tank sizes based on the calculated propellant. This process repeats until the propellant mass converges to within 10 kg and the data is saved to a propellant mass look up table. Once the data has been compiled in the propellant mass look up table, a linear curve fit is used to smooth the data to reduce potential uncertainties calculated during the trajectory analysis. Shown in Figure D.11 is an illustration of the lunar ascent trajectory calculated using the CLAMP routine

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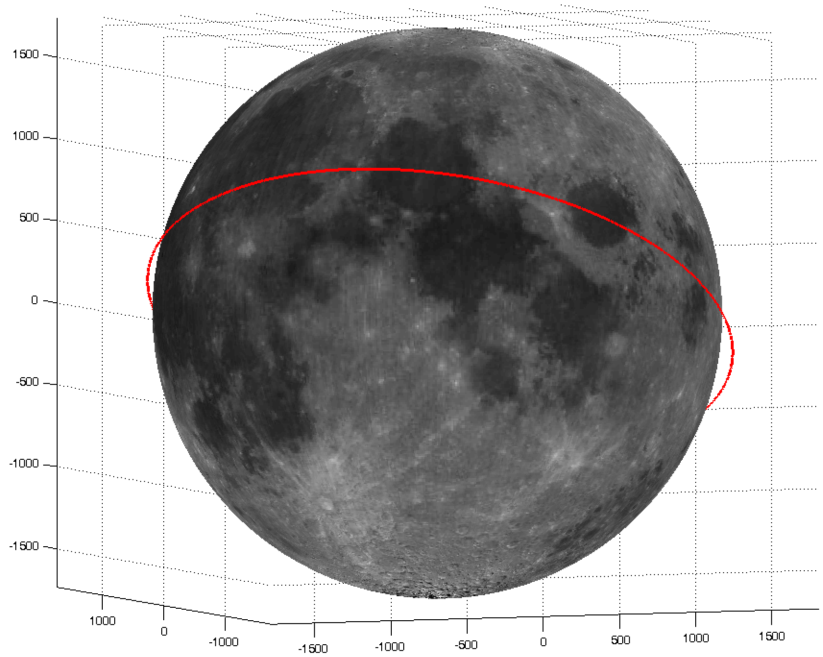

Figure D.11: Lunar ascent trajectory and orbit.

D.3.14 PROPULSION STORAGE SYSTEM

Using the main engine propellant and RCS propellant mass determined from the flyout table, the main engine and RCS pressurant, fuel, and oxidizer tanks are sized according to the propellant mass and the input parameters. The volume of the oxidizer and fuel needed for storage is calculated and the amount of pressurant needed to drive the propellant is determined based on the volume and differences in pressures. The pressurant, fuel, and oxidizer tanks are recorded in the CG_string_matrix. The RCS system uses the same equations as the main engine storage system and the tanks are sized according to the required propellant. A key assumption in the RCS system is that the thrusters utilize the same fuel and oxidizer propellant type as the main engine.

D.4 CLAMP ANALYSIS RUNS TO DETERMINE SAFETY AND OPERABILITY

The previous sections described the main subroutines in a single CLAMP analysis run. To calculate the amount of *Safety* and *Operability* in the spacecraft configuration, three runs are needed. Shown in Figure D.12 is a flowchart of the three CLAMP runs.

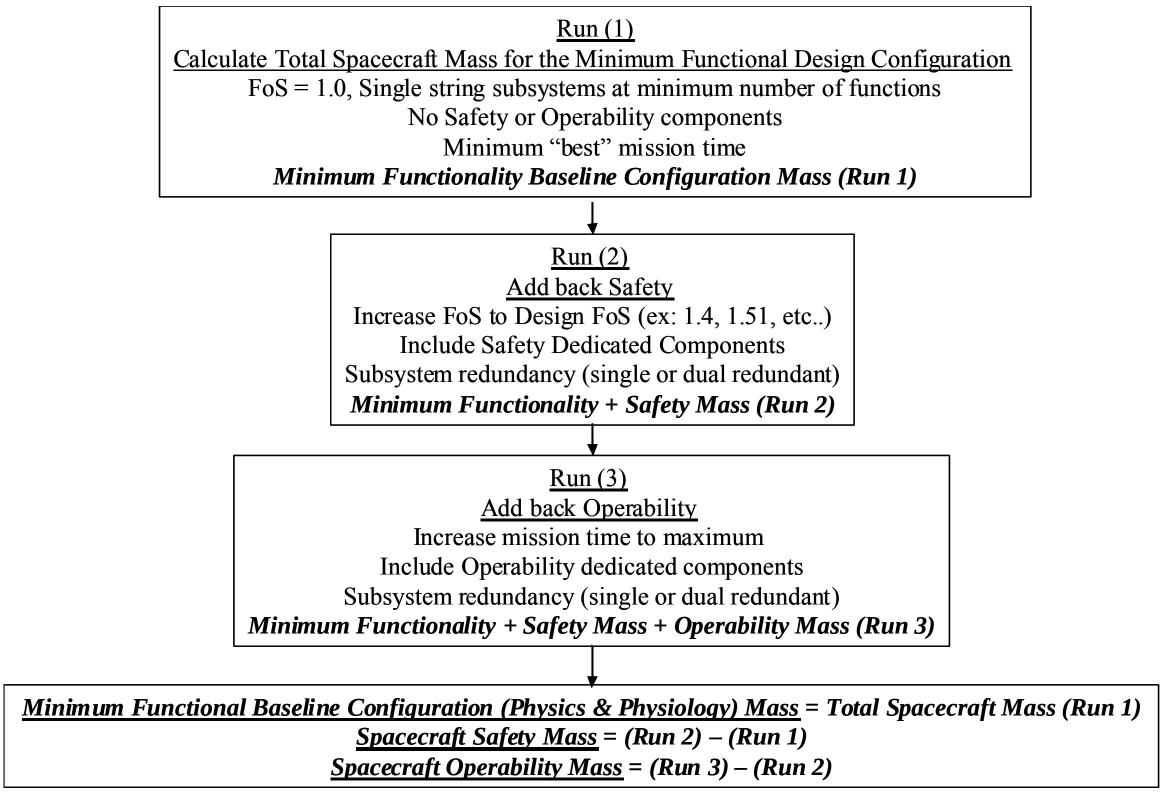

Figure D.12: Flowchart for calculating Minimum Functional, *Safety*, and *Operability* Mass.

D.4.1 MINIMUM FUNCTIONALITY CONFIGURATION – RUN 1

The program begins the analysis by first evaluating the minimum functionality configuration. Given an input file of a preferred spacecraft configuration, the Determine_Min_Functional subprogram strips away unnecessary components and modifies the original input file to be used by the program. This subroutine removes all redundant components, removes *Safety* and *Operability* identified components, specifies all Factors of Safety at 1.0, sets the mission time to the minimum mission time, and removes contingency or reserve propellants. In addition, error input information is set to zero for a deterministic analysis. After the minimum functionality input matrix has been created by the Determine_Min_Functional subprogram, the program runs the Min_Func_Config subprogram as a normal analysis with the previously mentioned subprograms. The information from the minimum functionality configuration run is written to output matrices named CG_string_matrix_minimum and SC_mass_minimum for later analysis to determine *Safety* and *Operability* mass.

D.4.2 SAFETY CONFIGURATION – RUN 2

Similar to the minimum functionality configuration, the Determine_Safety subprogram removes all *Operability* components from the original input file and creates a new input file to be used in the *Safety* configuration run. Minimum mission time is a part of this analysis. Error input is not changed in this subprogram to allow for Monte Carlo analysis. The program runs a subroutine called Safety_Config to determine the spacecraft configuration with additional safety. The output of the analysis run is stored in the matrices CG_string_matrix_safety and SC_mass_safety for later analysis.

D.4.3 OPERABILITY CONFIGURATION – RUN 3

The original input file is not modified for the *Operability* configuration run. The analysis run contains all elements of *Safety* and *Operability* for the desired configuration. Error input is not changed in this subprogram to allow for Monte Carlo analysis. The program runs a subroutine called Operability_Config to determine the spacecraft configuration all components as specified in the input file. The output of the analysis run is stored in the matrices CG_string_matrix_operability and SC_mass_operability for later analysis.

D.4.4 SAFETY AND OPERABILITY MASS

The reason for three analysis runs is to determine the mass of the spacecraft configuration at various configurations of *Safety* and *Operability*. The first run is a spacecraft configuration that does not include any additional *Safety* or *Operability* components beyond the minimum "best" mission design timeline. The second run adds mass associated with improving safety such as FoS, safety components, contingency propellant, etc.., for any item in the spacecraft that is necessary for *Safety*. The third run determines the amount of *Operability* mass in the spacecraft where mass is added for components that are related to increasing *Operability* factors of the mission. In addition, the mission time is increased to the maximum expected length. To calculate the amount of mass that contributes to an increase in the *Safety*, the first run is subtracted from the second run, the minimum functional baseline configuration. The same method is used to determine the amount of *Operability* mass where the second run is subtracted from the third run. This method calculates the difference in how much the total spacecraft mass increases with additions of mass due to *Safety* or *Operability*.

D.4.5 SAFETY AND OPERABILITY INDEX

The Safety Index and Operability Index are determined using the input and output files from each of the three analysis runs. The scores for the Safety Index and Operability Index are based upon the equations in Chapter 6. The scores are combined into a Total Spacecraft score and stored in a placeholder matrix named Safety_Oper_scores for later output.

D.4.6 OUTPUT FILE FORMATS

The last step in the CLAMP program is to write output files for the previous analysis. Three output files are written in MATLAB formats. The three output files are used in post processing program to generate figures and investigate subsystem information. These files are the following:

- Input file of the *Operability* configuration
- Output CG_string_matrix of the *Operability* configuration
- SC_mass file that includes all subsystem data, Safety Index, Operability Index, and Total Spacecraft scores for all three runs.

D.4.7 CONCLUSIONS

The previous sections provided a top level overview of the subroutines in the CLAMP program. Additional subroutines were used in the Monte Carlo sensitivity analyses and were not described here; but were utilized in generating data for the tradespaces in Chapter 3, 5, and 6. The CLAMP program contains over 10,000 lines of code and was a significant part of this dissertation.

APPENDIX E

APOLLO CONFIGURATION INPUT VARIABLE TABLES

The following tables were copied from the Microsoft Excel input file for the CLAMP analysis. The tables are grouped according to subsystems

Table E.1: Crew, Mission Time, and Avionics input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1-include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
0.1.1 Number of Crew	Crew_members	Num of Crew	2	0			1	0	0	0	0
0.1.2 Maximum Mission Time	Mission_Duration_HRS	Mission hours	12.36	0			1	0	0	0	0
0.1.3 Minimum Mission Time	Mission_Duration_HRS_min	Absolute Minimum Mission hours	1.33	0			1	0	0	0	0
Avionics											
6.1.1 Communicate with Earth ground station	Long_range_transceiver	S-Band Transceiver -2			2	1	3	2	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_amplifier	Power_amp & diplex - 1			2	1	3	2	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_antenna	S_Band Inflight Antennas - 2			2	1	3	2	2	0	7
6.1.1 Communicate with Earth ground station	Long_range_steerable_antenna	Steerable Antenna - 1			1	1	3	2	0	0	8
6.1.1 Communicate with Earth ground station	Long_range_data_processor	Signal Processor Assy			1	1	4	2	0	1	7
6.1.2 Communicate with relay satellites	Short_range_transceiver	VHF Xceiver and diplex			2	1	4	2	0	1	7
6.1.2 Communicate with relay satellites	Short_range_antenna	VHF Antenna			2	1	4	2	0	0	7
6.1.2 Communicate with relay satellites	Short_range_data_processor	VHF Ranging			1	1	4	2	0	1	7
6.1.3 Communicate between suited crewmembe	Interior_voice_comm	In VHF Xceiver			0	0	3	0	0	1	6
7.1 Sense Subsystem Commands	Data_bus_network_boxes	Signal conditioner Assy -2			2	1	5	1	0	1	6
7.2 Process/Amplify Subsystem Commands	Master_event_controllers	PCMETA -1			1	1	5	1	0	1	6
7.3 Send Commands to Subsystems	Sensor_comm_wiring	Wire harness A&B			1	1	5	2	0	0	7
7.4 Store Spacecraft Data	Data_storage	Data Storage			1	1	3	1	1	1	6
8.1.1 Monitor Subsystem Data	Health_monitoring_computer	Caution and Warning			1	1	4	1	1	1	6
8.2.1 Monitor ECS Subsystems	Instrumentation_sensors_ECLSS	ECS Sensors - total mass			1	1	3	2	2	0	7
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_Propulsion	Prop Sensors - total mass			1	1	5	2	0	0	8
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_RCS	RCS Sensors - total mass			1	1	5	2	0	0	8
8.2.2 Monitor Crew Accommodations	Instrumentation_sensors_CA	CA Sensors			0	0	1	1	0	0	7
8.2.3 Monitor Payload Subsystem	Instrumentation_sensors_Payload	Payload Sensors			0	0	1	2	0	0	7
8.2.4 Monitor Power Subsystem	Instrumentation_sensors_Power	Power Sensors			0	0	2	2	0	0	7
8.2.5 Monitor Communication Subsystem	Instrumentation_sensors_Comm	Comm Sensors			0	0	1	1	0	0	7
8.2.6 Monitor Command and Data Handling (C&	Instrumentation_sensors_CDH	CDH Sensors			0	0	1	1	0	0	7
8.2.7 Monitor Health Monitoring Subsystem	Instrumentation_sensors_Health	Health Sensors			0	0	1	1	0	0	7
8.2.8 Monitor Flight Control Subsystem	Instrumentation_sensors_Flight	Flight Sensors			0	0	5	1	0	0	7
8.2.9 Monitor Thermal Subsystem	Instrumentation_sensors_Thermal	Thermal Sensors			0	0	1	1	0	0	7
8.2.9 Monitor Structures Subsystem	Instrumentation_sensors_Structural	Structural Sensors			0	0	1	2	0	0	7
9.1.1 Sense Spacecraft Position Nav Inputs	IMU	IMU			1	1	5	1	0	1	8
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_base	IMU mounting			1	1	5	1	0	0	4
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_power	Power and Servo Assy			1	1	5	1	0	1	6
9.1.1 Sense Spacecraft Position Nav Inputs	Star_tracker	AOT Telescope / Computer Con			1	1	4	1	0	0	5
9.1.2 Convert Analog to Digital Nav Inputs	Nav_analog_digital	Pulse Torque Assembly			1	1	5	1	0	1	6
9.1.3 Output Navigation to Guidance	Multiplexer_demultiplexers	Coupling Data Unit			1	1	5	1	0	1	7
9.2.1 Calculate Spacecraft Guidance	Flight_control_computer	LM Guidance Computer			1	1	5	1	0	1	8
9.2.2 Communicate with Instrumentation	Instrumentation_comm	Signal Conditioner Assy			2	1	5	1	0	1	7
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw	ACA - Attitude Controller Assem			2	1	5	1	0	0	8
9.3.1 Input Human Flight Controls	Flight_translation	TTCA - Thrust and translation co			2	1	5	1	0	0	8
9.3.2 Input Human Navigation to Computer	Computer_keyboard	DSKD			1	1	5	1	0	0	8
9.3.3 Output Spacecraft Control to Propulsion	Control_propulsion	ATCA - Attitude and Translation			1	1	5	2	0	0	8
9.3.4 Display Spacecraft Control	Crew_displays_control	Panel 3 4 5			1	1	5	1	0	0	8
9.3.5 Display Spacecraft Navigation	Crew_displays_navigation	Panel 1 2			1	1	5	1	0	0	8
9.3.6 Manual Control Spacecraft Subsystems	Crew_displays_subsystems	Panel 6 8 11 12 14 16			1	1	4	1	0	0	8
9.4.1 Sense Spacecraft Abort Position	Abort_navigation	Abort Sensor Assy			1	1	5	1	1	1	8
9.4.2 Sense Spacecraft Abort Velocity Inputs	Abort_control	Rate Gyro Assy			1	1	5	1	1	1	7
9.4.3 Calculate Spacecraft Abort Trajectory	Abort_guidance	Abort Electronics Assy			1	1	5	1	1	1	8
9.4.4 Input Abort Commands	Abort_input	DEDA -Data Entry and Display A			1	1	5	1	1	0	8
9.5 Provide Rendezvous Guidance	Rendezvous_radar	Rendezvous Radar			1	1	4	2	0	0	10
9.6 Provide Inertial Reference	Ordeal	Ordeal			1	1	2	1	2	0	6
9.7 Provide Additional Avionics	Misc_avionics	Misc Avionics			0	0	1	0	2	0	4

Table E.2: EVA, Crew Accommodations, and Payloads input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1- include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
EVAs											
12.1.1 Provide Space Suit	Spacesuits	Pressure Suit, no PLSS			**1**	**1**	**3**	0	1	0	8
12.2.1 Provide Mobility Aids	EVA_mobility	Mobility devices			**0**	**0**	**1**	0	0	0	6
12.2.2 Provide Umbilical	Suit_umbilical	Umbilical hoses			**0**	**0**	**3**	0	0	0	7
Crew Accommodations											
3.1.1 Provide Cabin Lighting	Cabin_lighting	Lighting - normalized to one day			**1**	**1**	**2**	1	1	0	5
3.1.2 Provide Panel Lighting	Panel_lighting	Assumed same as cabin light			**1**	**1**	**3**	1	0	0	6
3.1.3 Provide Light for Exterior Viewing	Exterior_lighting	Lighting			**2**	**1**	**3**	2	1	0	7
3.1.4 Provide Restraints	CA_restraints	kg			**1**	**1**	**2**	1	1	0	4
3.1.5 Provide Handholds	CA_handholds	handholds			**1**	**1**	**1**	1	2	0	4
3.2.1 Provide Personal Storage	Personal_storage	kg/person/day -normalized from HSMAD			**0**	**0**	**1**	1	0	0	2
3.2.2 Provide Additional Clothing Storage	Clothing	kg/person/day - normalized from HSMAD			**0**	**0**	**1**	1	0	0	3
3.2.3 Provide Entertainment	Entertainment	Not included			**0**	**0**	**1**	1	0	0	5
3.3.2 Provide Medical Kit Storage	Medical_kit	kg/day-normalized from HSMAS			**1**	**1**	**3**	1	1	0	5
3.3.3 Provide Exercise Capability	Exercise	Not included			**0**	**0**	**1**	1	0	0	6
3.3.4 Provide Sleep Accommodations	Sleep_accommodations	Not included			**0**	**0**	**1**	1	0	0	4
3.3.5 Provide Operational Supplies	Operational_supplies	kg/person			**1**	**1**	**3**	1	1	0	6
3.3.1 Provide Hygiene Supplies	Hygiene_consumables	kg/person/day			**1**	**1**	**1**	1	2	0	4
3.3.1 Provide Hygiene Supplies	Hygiene_kit	kg/person			**0**	**0**	**1**	1	2	0	4
Payloads											
4.1.1 Provide Science Payload Storage	Science_return	http://history.nasa.gov/SP-4029/			**1**	**1**	**1**	1	2	0	6
4.1.2 Provide General Payload Storage	General_return	Not included			**0**	**0**	**1**	1	2	0	6
4.2.1 Provide Tools Storage	Tools_equipment	1.67 kg estimated			**1**	**1**	**2**	1	1	0	5
4.2.2 Provide Consumable Equipment Storage	Consumable_equipment	Not included			**0**	**0**	**2**	1	0	0	6
4.2.3 Provide Space Suit Storage	Spacesuit_storage	Not included					**1**	1	0	0	2
4.2.4 Provide Photography Equipment	Photography	19kg estimated Lunar Mission			**1**	**1**	**1**	1	2	0	9

Table E.3: ECLSS input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1- include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS											
2.1.1 Remove Carbon Dioxide	CO2_crew_day_rate	kg/person/day	**0.998**	**0**			**5**	0	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_rate	CO2 Canister Perf			**1**	**1**	**5**	1	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_mass	CO2 Canister Mass			**2**	**1**	**5**	1	0	0	6
2.1.10 Provide Fire Detection and Suppression	Fire_extinguisher	Not included			**0**	**0**	**4**	1	0	0	6
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_density	lbm/in^3	**0.296**	**0**			**5**	0	0	0	7
2.1.2 Provide Metabolic Oxygen	O2_crew_day_rate	kg/person/day	**0.835**	**0**			**5**	0	0	0	2
2.1.2.1 Store High Pressure Oxygen	Storage_tank_FoS	Storage Tank FOS	**1.5**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	PaO2	psi - Apollo all Oxygen	**5.2**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_radius_length_ratio	ratio	**1**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_temperature	Celsius	**26.67**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_pressure	psi	**840**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_yield_strength	psi	**142000**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_redundancy	Redundancy in Oxygen	**2**	**0**			**5**	0	0	0	7
2.1.2.1 Store High Pressure Oxygen	Oxygen_tank_insulation	Use same as MMOD			**1**	**1**	**4**	2	0	0	7
2.1.2.2 Transport Oxygen	Oxygen_gas_transport_lines	Assume 1/2 inch line			**1**	**1**	**5**	2	0	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_relief_valve	Used 1/2 ISS regulator valve			**1**	**1**	**4**	2	1	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_regulator_valve	Used 1/2 ISS relief valve			**1**	**1**	**5**	2	0	0	9
2.1.2.4 Measure pp Oxygen Level in Cabin Air	Oxygen_pp_gauge	Not included			**0**	**0**	**4**	2	0	0	5
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_temperature	Celsius	**26.67**	**0**			**4**	0	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_pressure	psi	**840**	**0**			**4**	0	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	Nitrogen_tank_insulation	Use same as MMOD			**0**	**0**	**4**	2	0	0	7
2.1.3.2 Transport Makeup Gas	Nitrogen_gas_transport_lines	Assume1/2 inch line			**0**	**0**	**4**	2	0	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_relief_valve	Used 1/2 ISS relief valve			**0**	**0**	**4**	2	1	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_regulator_valve	Used 1/2 ISS regulator valve			**0**	**0**	**4**	2	0	0	9

Table E.3: ECLSS input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1-include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS											
2.1.5 Remove Particulates	Particulates_filter	Assumed 1/2 Shuttle Filter			1	1	2	1	1	0	5
2.1.6 Remove Humidity	Humidity_percent	Percent (40% = 0.4)	0.6	0			3	0	0	0	7
2.1.6.2 Condense Cabin Air	Condense_air_heat_exchanger	Assumed 1/2 Spacelab Conds Heat Exch			1	1	3	1	0	0	7
2.1.6.1 Provide Humidity Separator	Humidity_capture	Assumed 1/2 Spacelab Separator			1	1	3	1	0	0	5
2.1.7.1 Provide Air Circ Bypass	Mix_air_valve	Assumed Spacelab 1/2 TCV			1	1	2	1	2	0	7
2.1.7.2 Measure Cabin Air Temp	Cabin_temp	Celsius	22.05	0			2	0	0	0	6
2.1.7.2 Measure Cabin Air Temp	Temperature_sensor	Assumed Shuttle			1	1	1	1	1	0	6
2.1.8.1 Circulate Air	Cabin_fan	Assumed 1/2 Spacelab Fan			1	1	4	1	0	1	7
2.1.8.2 Return Air	Return_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	1	1	0	4
2.1.8.3 Output Air	Direct_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	1	1	0	4
2.1.9.1 Provide Air Pressure Sensor	Air_Pressure_gauge	Assumed Shuttle Sensor			1	1	4	1	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_leak_percent	%/day	0.05	0			4	0	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_atm_pressure	psi	5.2	0			5	0	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_air_control	Assumed 1/2 Shuttle Cabin Temp Control			1	1	3	1	0	0	7
2.2.1 Store Food	Food_storage	Assumed in Food mass			0	0	3	1	0	0	4
2.2.1 Store Food	Food_rate	kg/person/day			1	1	3	1	0	0	4
2.2.2 Prepare Food	Galley_supplies	kg/person/day			0	0	1	1	2	0	4
2.2.2 Prepare Food	Heat_food	Assumed HSMAD			0	0	1	1	2	0	4
2.3.1 Provide Potable Water	Potable_water	kg/person/day			1	1	4	2	0	0	5
2.3.2 Provide Hygiene Water	Hygiene_water	kg/person/day			1	1	1	2	2	0	5
2.4.1 Collect Urine	Urine_bags	kg/person/day			1	1	1	1	0	0	4
2.4.2 Collect Fecal Solids	Diapers	kg/person/day			0	0	1	1	0	0	4
2.4.2.2 Control Odor	Control_odor	kg/person/day			0	0	1	1	2	0	3
2.4.2.3 WCS supplies	WCS_supplies	kg/person/day			1	1	1	1	2	0	3
2.4.3 Collect Liquid Waste	Vacuum	Assumed HSMAD			0	0	1	1	2	0	6
2.4.4 Collect Solid Waste	Trash_bags	kg/person/day			0	0	1	1	0	0	3
2.4.4 Collect Solid Waste	Housecleaning_supplies	kg/person/day			0	0	1	1	0	0	3
2.4.5 Jettison Liquid Waste	Dump_Valve_Piping	Assumed ISS Overboard Water			1	1	2	1	2	0	7

Table E.4: Power and Thermal input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1-include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Power											
5.1 Provide Power	Batteries	Adapted from Apollo			2	1	5	2	0	1	7
5.1 Provide Power	Depth_of_Discharge		1	0			5	0	0	0	7
5.1 Provide Power	Duty_cycle_coast		1	0			5	0	0	0	7
5.2 Distribute Power	Power_Distr_Wiring	Adapted from Apollo			2	1	5	2	0	0	7
5.3 Regulate Power	Power_Distr_Controller	Power Controller ECA			2	1	5	2	0	1	8
5.3 Regulate Power	Power_inverters	Power Inverter			2	1	5	2	0	1	8
5.3 Regulate Power	Power_relays	Power relay - Electronic			2	1	5	2	0	1	7
5.6 Provide Overload Protection	Power_Distr_CircBreak	Panels/Circuit Breakers			2	1	4	1	1	0	7
Thermal											
10.1 Collect Heat	Coldplate_min_power	Minimum watts for a coldplate	20	0			4	0	0	0	6
10.1 Collect Heat	Loop_capacity	Watts of heat removed single loo	500	0			4	0	0	0	6
10.1 Collect Heat	Loop_diameter	inches	0.518	0			4	0	0	0	6
10.1 Collect Heat	Thermal_interior_fluid_density	kg/m3	1040	0			4	0	0	0	6
10.1 Collect Heat	Coolant_piping	kg/m3			1	1	4	1	0	0	6
10.1 Collect Heat	Coldplates	5.25 kg/kW derived from Apollo			1	1	4	1	0	0	6
10.2 Transport Heat	Interior_coolant_pumps	Assumed Shuttle Water Pump			1	1	4	1	0	0	7
10.2 Transport Heat	Interior_coolant_lines_valves	Assumed Shuttle Check Valve			1	1	4	1	0	0	8
10.3 Remove Heat	Interior_exterior_heat_exchanger	3.2 kg/kW derived from Apollo			2	1	4	1	0	0	7
10.3 Remove Heat	Water_system_sublimators	Latent Heat of Vaporization	2501.3	0			4	2	0	0	7
10.3 Remove Heat	Exterior_radiators	Radiator Panels			0	0	4	2	0	0	8

Table E.5: Secondary Structures and Structures input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1- include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Secondary Structures											
1.4 Support Internal Subsystems and Componen	Secondary_struct_ratio	ratio	**0.21**	**0**			**4**	0	0	0	6
Structures											
1.1.1 Provide protection from vacuum	Factor_of_Safety	Structural FoS	**1.41**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Factor_of_Safety_PV_skin	Structural FoS	**2.00**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Interior_diameter_hab_volume	meters	**2.159**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Cylinder_length_hab_volume	meters	**1.455**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Packaging_efficiency	ratio	**0.35**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Protrusion_ratio	(0.001 -0.25)	**0.065**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_yield_strength	psi	**73000**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_density	lbm/in^3	**0.102**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Elasticity	10^6 psi	**10.4**	**0**			**5**	0	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Rigidity	10^6 psi	**3.9**	**0**			**5**	0	0	0	6
1.2 Provide Load Bearing Capability	Max_acceleration_gs	gs of acceleration	**4.7**	**0**			**5**	0	0	0	6
1.2 Provide Load Bearing Capability	Number_hoops	Odd Number	**19**	**0**			**5**	0	0	0	6
1.2 Provide Load Bearing Capability	Number_stringers	Multiple of 4	**20**	**0**			**5**	0	0	0	6
1.2 Provide Load Bearing Capability	Height_to_base_ratio	ratio	**8.2**	**0**			**5**	0	0	0	6
1.2 Provide Load Bearing Capability	Fillet_ratio	ratio	**0.1**	**0**			**5**	0	0	0	6
1.5.3 Provide Docking Mechanism for Lunar Hab	LIDS_published_weight	lbs	**870**	**0**			**4**	0	0	0	8
1.5.1 Support Docking Mechanism for Lunar Asc	Docking_ring_load_Force	lbf	**23300**	**0**			**4**	0	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_diameter	inches	**32**	**0**			**4**	0	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_height	inches	**16**	**0**			**4**	0	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_width	inches	**32**	**0**			**4**	0	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_height	inches	**32**	**0**			**4**	0	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_corner_radius_ratio	ratio	**5.33**	**0**			**4**	0	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_flange_ratio	ratio	**20**	**0**			**4**	0	0	0	6
1.6.1 Provide Forward Windows	Window_material_density	lbm/in^3	**0.091402**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_pane_thickness	inches	**0.2**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_width_horizontal	inches	**18**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_length	inches	**17**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_corner_radius_ratio	ratio	**4**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_frame_ratio	ratio	**4**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_position_angle	degrees	**45**	**0**			**5**	0	0	0	7
1.6.1 Provide Forward Windows	Window_center_height	inches	**60**	**0**			**5**	0	0	0	7
1.6.2 Provide Docking Viewing Ports	Docking_window	area in^2	**60**	**0**			**4**	0	0	0	7
1.6.3 Provide Aft Viewing Ports	Aft_window	area in^2	**40**	**0**			**3**	0	0	0	7
1.1.2 Provide MMOD/TPS protection	MMOD_standoff	Minimum standoff inches	**3**	**0**			**5**	0	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_time_days	days	**8.6**	**0**			**5**	0	0	0	6
1.1.2 Provide MMOD/TPS protection	PNP_target	Probability	**0.9995**	**0**			**5**	0	0	0	6
1.1.2 Provide MMOD/TPS protection	TPS_blanket	kg/m^2 - 1" Thermal Blanket 6pc	**2.15**	**3**			**4**	0	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_altitude	km	**350**	**0**			**5**	0	0	0	6

Table E.6: Propulsion input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1- include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion											
11.1.6 Provide Main Engine	Specific_impulse	Isp (seconds)	**311.7**	**0**			**5**	0	0	0	9
11.1.6 Provide Main Engine	Vacuum_thrust	N (newtons)	**15746**	**0**			**5**	0	0	0	9
11.1.6 Provide Main Engine	Main_engine_weight_thrust_ratio	ratio	**3.17**	**0**			**5**	0	0	0	9
0.1.4 Launch Trajectory	Gravity_constant	m/s2	**9.81**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Moon_gravity_parameter	km3/sec2	**4902.87**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Moon_radius	km	**1738**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Orbital_apogee_altitude	km	**78.71**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Orbital_perigee_altitude	km	**16.67**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Launch_azimuth	degrees North	**270**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Launch_latitude	degrees	**26.1322**	**0**			**1**	0	0	0	0
0.1.4 Launch Trajectory	Launch_longitude	degrees	**3.63386**	**0**			**1**	0	0	0	0

Table E.6: Propulsion input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Pressurized Vol In_Out (0 not included, 1- include in volume, 2 outside)	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion											
11.1.6 Provide Main Engine	Oxidizer_fuel_ratio	ratio	**1.61**	**0**			**5**	0	0	0	9
11.1.2 Provide Main Engine Oxidizer	Oxidizer_density	kg/m3	**1450**	**0**			**5**	0	0	0	3
11.1.1 Provide Main Engine Fuel	Fuel_density	kg/m3	**903**	**0**			**5**	0	0	0	3
11.1.7 Store Main Engine Propellant	Fuel_ox_tank_press	psi	**179**	**0**			**5**	0	0	0	7
11.1.11 Provide Contingency Propellant	Contingency_propellant	ratio	**0.031**	**0**			**4**	0	1	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_pressure	psi	**3050**	**0**			**5**	0	0	0	3
11.2.3 Provide RCS Pressurant	Pressurant_pressure_RCS	psi	**3050**	**0**			**5**	0	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_tank_temperature	degrees K	**294.26**	**0**			**5**	0	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_ratio_spec_heats	ratio	**1.667**	**0**			**5**	0	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_gas_constant	R J/kg*K	**2076.9**	**0**			**5**	0	0	0	3
11.1.7 Store Main Engine Propellant	Propulsion_tank_material_density	lbm/in^3	**0.16**	**0**			**5**	0	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_yield_strength	psi	**140000**	**0**			**5**	0	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_factor_of_safety	FoS	**1.5**	**0**			**5**	0	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_unused_fraction	ratio	**0.01**	**0**			**5**	0	0	0	7
11.2.7 Store RCS Propellant	Fuel_ox_tank_press_RCS	psi	**179**	**0**			**5**	0	0	0	7
11.1.9 Transport Main Engine Propellant	Propulsion_line_id	m	**0.0254**	**0**			**5**	0	0	0	7
11.1.5 Transport Main Engine Pressurant	Main_engine_pressurant_line_id	m	**0.015875**	**0**			**5**	0	0	0	7
11.2.9 Transport RCS Propellant	Propulsion_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	0	7
11.2.5 Transport RCS Pressurant	Pressurant_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	0	7
11.1.6 Provide Main Engine	Main_engine_linear_slope	linear_slope kg/kN	**0.5786**	**0**			**5**	2	0	0	9
11.1.6 Provide Main Engine	Main_engine_y_intercept	linear_curve_intercept - kg	**65**	**0**			**5**	2	0	0	9
11.1.12 Support Main Engine Components	Propulsion_secondary_struct_ratio	ratio	**0.2**	**0**			**5**	0	0	0	6
11.2.6 Provide RCS thrusters	RCS_side_A_B_choice	Integer, 1,2,3	**2**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	Specific_impulse_RCS	Isp (seconds)	**240**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	Vacuum_thrust_RCS	N (newtons)	**444.822**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_linear_slope	linear_slope kg/kN	**6.48**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_y_intercept	linear_curve_intercept - kg	**0**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	Num_RCS_engines_system	Integer, 4,8,16	**8**	**0**			**5**	0	0	0	9
11.2.6 Provide RCS thrusters	Num_fire_RCS_thrust_ascent	Num firing during ascent	**0.57**	**0**			**5**	0	0	0	9
11.2.11 Provide RCS contingency propellant	RCS_additional_propellant_mass	Rendezvous -kg	**96.2**	**3**			**4**	0	1	0	3
11.1.13 Insulate Main Engine Propellant	Propulsion_tanks_insulation	Use same as TPS			**1**	**1**	**4**	2	0	0	4
11.2.13 Insulate RCS Propellant	Propulsion_tanks_insulation_RCS	Placeholder			**1**	**1**	**4**	2	0	0	4
11.2.1 Provide RCS Propellant	RCS_propellant	Placeholder			**1**	**1**	**5**	0	0	0	3
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_valve	kg			**1**	**1**	**5**	2	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_filter	kg			**1**	**1**	**4**	2	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_solenoid_valve	kg			**1**	**1**	**5**	2	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_pressure_regulator	kg			**2**	**1**	**5**	2	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_check_valve	kg			**2**	**1**	**4**	2	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_explosive_valve	kg			**2**	**1**	**5**	2	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_relief_valve	kg			**1**	**1**	**4**	2	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_trim_orifice	kg			**1**	**1**	**5**	2	0	0	5
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_filter	kg			**1**	**1**	**4**	2	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_isolation_valve	kg			**2**	**1**	**5**	2	0	0	9
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_bipropellant_valve	kg			**2**	**1**	**5**	2	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_valve	kg			**1**	**1**	**5**	2	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_filter	kg			**1**	**1**	**4**	2	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_solenoid_valve	kg			**1**	**1**	**5**	2	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_pressure_regulator	kg			**2**	**1**	**5**	2	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_check_valve	kg			**2**	**1**	**4**	2	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_explosive_valve	kg			**2**	**1**	**5**	2	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_relief_valve	kg			**1**	**1**	**4**	2	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_trim_orifice	kg			**1**	**1**	**5**	2	0	0	5
11.1.10.1 Control Main Engine Fuel	Fuel_filter	kg			**1**	**1**	**4**	2	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_isolation_valve	kg			**2**	**1**	**5**	2	0	0	9
11.1.10.1 Control Main Engine Fuel	Fuel_bipropellant_valve	kg			**2**	**1**	**5**	2	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_valve_RCS	kg			**2**	**1**	**5**	2	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_filter_RCS	kg			**1**	**1**	**4**	2	1	0	7
11.2.4 Control RCS Pressurant	System_pressurant_orifice_RCS	kg			**1**	**1**	**5**	2	0	0	5
11.2.4 Control RCS Pressurant	System_pressurant_pressure_regulator_R	kg			**2**	**1**	**5**	2	0	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_check_valve_RCS	kg			**1**	**1**	**4**	2	1	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_relief_valve_RCS	kg			**1**	**1**	**4**	2	1	0	6
11.2.4 Control RCS Pressurant	Fuel_pressurant_check_valve_RCS	kg			**1**	**1**	**4**	2	1	0	7
11.2.4 Control RCS Pressurant	Fuel_pressurant_relief_valve_RCS	kg			**1**	**1**	**4**	2	1	0	7
11.2.10.2 Control RCS Oxidizer	Oxidizer_shutoff_valve_RCS	kg			**1**	**1**	**5**	2	0	0	9
11.2.10.1 Control RCS Fuel	Fuel_shutoff_valve_RCS	kg			**1**	**1**	**5**	2	0	0	9
11.2.9.1 Filter RCS Propellant	Fuel_Ox_filter_RCS	kg			**1**	**1**	**4**	2	1	0	7
11.2.10 Control RCS Propellant	Fuel_Ox_isolation_valve_RCS	kg			**1**	**1**	**5**	2	0	0	9
11.2.14.2 Heat RCS Oxidizer lines	Oxidizer_heater_RCS	kg			**1**	**1**	**4**	2	1	0	6
11.2.14.1 Heat RCS Fuel lines	Fuel_heater_RCS	kg			**1**	**1**	**4**	2	1	0	6
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_valve	kg			**2**	**1**	**4**	2	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_filter	kg			**2**	**1**	**4**	2	1	0	7
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_valve	kg			**2**	**1**	**4**	2	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_filter	kg			**2**	**1**	**4**	2	1	0	7

Table E.7: Avionics technology database.

Equipment	Mass (kg)	Mass Error (+/-,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Abort_control	0.91	5	0.000	0.000	0.000	2
Abort_guidance	14.74	5	0.603	0.203	0.133	96
Abort_input	3.40	5	0.140	0.152	0.132	10
Abort_navigation	9.38	5	0.130	0.229	0.292	74
Computer_keyboard	7.71	5	0.203	0.203	0.178	25
Control_propulsion	10.75	5	0.000	0.000	0.000	50
Control_propulsion_thrusters	0.00	5	0.000	0.000	0.000	0
Crew_displays_control	7.44	5	0.000	0.000	0.000	0
Crew_displays_navigation	15.79	5	0.000	0.000	0.000	0
Crew_displays_subsystems	5.35	5	0.000	0.000	0.000	0
Data_bus_network_boxes	15.99	5	0.203	0.133	0.607	15
Data_storage	1.13	5	0.052	0.102	0.158	10
Flight_control_computer	31.75	5	0.152	0.318	0.610	70
Flight_roll_pitch_yaw	2.16	5	0.000	0.000	0.000	0
Flight_translation	2.38	5	0.000	0.000	0.000	0
Health_monitoring_computer	8.30	5	0.178	0.171	0.298	13
IMU	19.10	5	0.000	0.000	0.000	100
Instrumentation_comm	1.27	5	0.084	0.224	0.284	25
Instrumentation_sensors_CA	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_CDH	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Comm	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_ECLSS	2.68	5	0.000	0.000	0.000	1
Instrumentation_sensors_Flight	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Health	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Payload	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Power	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Propulsion	2.45	5	0.000	0.000	0.000	1
Instrumentation_sensors_RCS	2.22	5	0.000	0.000	0.000	1
Instrumentation_sensors_Thermal	0.00	5	0.000	0.000	0.000	0
Interior_voice_comm	0.00	5	0.000	0.000	0.000	0
Long_range_amplifier	4.24	5	0.000	0.000	0.000	36
Long_range_antenna	0.18	5	0.000	0.000	0.000	1
Long_range_data_processor	4.72	5	0.000	0.000	0.000	27
Long_range_steerable_antenna	12.47	5	0.000	0.000	0.000	60
Long_range_transceiver	4.54	5	0.000	0.000	0.000	18
Master_event_controllers	10.25	5	0.171	0.130	0.502	11
Misc_avionics	0.00	5	0.000	0.000	0.000	0
Multiplexer_demultiplexers	15.88	5	0.140	0.288	0.508	25
Nav_analog_digital	6.80	5	0.064	0.279	0.330	25
Nav_base	1.36	5	0.287	0.287	0.287	0
Nav_power	9.07	5	0.067	0.225	0.597	100
Ordeal	3.13	5	0.000	0.000	0.000	4
Rendezvous_radar	35.43	5	0.000	0.000	0.000	95
Short_range_antenna	1.09	5	0.000	0.000	0.000	1
Short_range_data_processor	1.18	5	0.000	0.000	0.000	5
Short_range_transceiver	2.97	5	0.000	0.000	0.000	32
Star_tracker	10.48	5	0.000	0.000	0.000	0

Table E.8: EVA technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
EVA_mobility	0.00	5	0.000	0.000	0.000	0
Spacesuits	33.00	5	0.000	0.000	0.000	0
Suit_umbilical	0.00	5	0.000	0.000	0.000	0

Table E.9: Payloads technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Science_return	115.62	5	0.000	0.000	0.000	0
Consumable_equipment	0.00	5	0.000	0.000	0.000	0
General_return	0.00	5	0.000	0.000	0.000	0
Photography	19.00	5	0.437	0.437	0.437	0
Tools_equipment	1.67	5	0.149	0.149	0.149	0

Table E.10: Power technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Batteries	154.27	5	0.00	0.00	0.00	1000
Power_Distr_CircBreak	17.89	5	0.000	0.000	0.000	0
Power_Distr_Controller	4.76	5	0.000	0.000	0.000	0
Power_Distr_Wiring	47.14	5	0.000	0.000	0.000	0
Power_inverters	6.98	5	0.000	0.000	0.000	0
Power_relays	4.56	5	0.000	0.000	0.000	0

Table E.11: Thermal technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Coldplates	5.50	5	0.000	0.000	0.000	0
Exterior_radiators	0.00	5	0.000	0.000	0.000	0
Interior_coolant_lines_valves	0.90	5	0.000	0.000	0.000	0
Interior_coolant_pumps	1.88	5	0.000	0.000	0.000	0

Table E.12: Crew Accommodations technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Cabin_lighting	1.00	5	0.100	0.100	0.100	25
CA_handholds	0.60	5	0.100	0.100	0.100	0
CA_restraints	4.16	5	0.100	0.100	0.100	0
Clothing	1.53	5	0.100	0.100	0.100	0
Entertainment	0.00	5	0.000	0.000	0.000	0
Exercise	0.00	5	0.000	0.000	0.000	0
Exterior_lighting	6.00	5	0.000	0.000	0.000	100
Medical_kit	0.32	5	0.100	0.100	0.100	0
Operational_supplies	6.50	5	0.100	0.100	0.100	0
Panel_lighting	1.00	5	0.100	0.100	0.100	25
Personal_storage	1.66	5	0.100	0.100	0.100	0
Sleep_accommodations	0.00	5	0.000	0.000	0.000	0

Table E.13: ECLSS technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Oxygen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Air_Pressure_gauge	0.25	5	0.000	0.000	0.000	1
Cabin_air_control	2.22	5	0.006	0.006	0.006	16
Cabin_fan	4.65	5	0.002	0.002	0.002	70
Condense_air_heat_exchanger	8.89	5	0.045	0.045	0.045	0
Control_odor	0.00	5	0.000	0.000	0.000	0
Diapers	0.00	5	0.000	0.000	0.000	0
Dump_Valve_Piping	1.46	5	0.005	0.005	0.005	0
Fecal_bags	0.00	5	0.000	0.000	0.000	0
Fire_extinguisher	0.00	5	0.000	0.000	0.000	0
Food_rate	2.30	5	0.005	0.005	0.005	0
Food_storage	0.00	5	0.000	0.000	0.000	0
Galley_supplies	0.50	5	0.001	0.001	0.001	0
Heat_food	5.00	5	0.000	0.000	0.000	0
Housecleaning_supplies	0.00	5	0.000	0.000	0.000	0
Humidity_capture	1.28	5	0.004	0.004	0.004	0
Hygiene_water	4.10	5	0.000	0.000	0.000	0
Mix_air_valve	1.15	5	0.008	0.008	0.008	0
Nitrogen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Nitrogen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Oxygen_pp_gauge	0.00	5	0.000	0.000	0.000	0
Oxygen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Particulates_filter	1.25	5	0.000	0.000	0.000	0
Potable_water	1.60	5	0.000	0.000	0.000	0
Temperature_sensor	0.08	5	0.000	0.000	0.000	0
Trace_Contaminants_filter	1.25	5	0.000	0.000	0.000	0
Trash_bags	0.00	5	0.000	0.000	0.000	0
Urine_bags	0.23	5	0.000	0.000	0.000	0
Vacuum	0.00	5	0.000	0.000	0.000	0
WCS_supplies	0.05	5	0.001	0.001	0.001	0

Table E.14: Propulsion technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
APS_interconnect_Fuel_filter	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Fuel_valve	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_filter	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_valve	0.76	5	0.050	0.050	0.050	0
Fuel_bipropellant_valve	0.70	5	0.050	0.050	0.050	0
Fuel_filter	0.70	5	0.050	0.050	0.050	0
Fuel_heater_RCS	0.60	5	0.050	0.050	0.050	0
Fuel_isolation_valve	0.70	5	0.050	0.050	0.050	0
Fuel_Ox_filter_RCS	0.11	5	0.050	0.050	0.050	0
Fuel_Ox_isolation_valve_RCS	0.11	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve_RCS	0.51	5	0.050	0.050	0.050	0
Fuel_pressurant_explosive_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_filter	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_pressure_regulator	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve_RCS	0.51	5	0.050	0.050	0.050	0
Fuel_pressurant_solenoid_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_valve	0.83	5	0.050	0.050	0.050	0
Fuel_shutoff_valve_RCS	0.60	5	0.050	0.050	0.050	0
Fuel_trim_orifice	0.70	5	0.050	0.050	0.050	0
Oxidizer_bipropellant_valve	0.70	5	0.050	0.050	0.050	0
Oxidizer_filter	0.70	5	0.050	0.050	0.050	0
Oxidizer_heater_RCS	0.60	5	0.050	0.050	0.050	0
Oxidizer_isolation_valve	0.70	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve_RCS	0.51	5	0.050	0.050	0.050	0
Oxidizer_pressurant_explosive_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_filter	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_pressure_regulator	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve_RCS	0.51	5	0.050	0.050	0.050	0
Oxidizer_pressurant_solenoid_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_shutoff_valve_RCS	0.60	5	0.050	0.050	0.050	0
Oxidizer_trim_orifice	0.70	5	0.050	0.050	0.050	0
System_pressurant_filter_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_orifice_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_pressure_regulator_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_valve_RCS	0.51	5	0.050	0.050	0.050	0

APPENDIX F

APOLLO ONE MAN INPUT VARIABLE TABLES

The following tables were copied from the Microsoft Excel input file for the CLAMP analysis. The tables are grouped according to subsystems.

Table F.1: Crew, Mission Time, and Avionics input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
0.1.1 Number of Crew	Crew_members	Num of Crew	1	0			1	0	0	0
0.1.2 Maximum Mission Time	Mission_Duration_HRS	Mission hours	6.18	0			1	0	0	0
0.1.3 Minimum Mission Time	Mission_Duration_HRS_min	Absolute Minimum Mission ho	1.33	0			1	0	0	0
Avionics										
6.1.1 Communicate with Earth ground station	Long_range_transceiver	S-Band Transceiver -2			0	0	3	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_amplifier	Power_amp & diplex - 1			0	0	3	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_antenna	S_Band Inflight Antennas - 2			0	0	3	2	0	7
6.1.1 Communicate with Earth ground station	Long_range_steerable_antenna	Steerable Antenna - 1			0	0	3	0	0	8
6.1.1 Communicate with Earth ground station	Long_range_data_processor	Signal Processor Assy			1	1	4	0	1	7
6.1.2 Communicate with relay satellites	Short_range_transceiver	VHF Xceiver and diplex			2	1	4	0	1	7
6.1.2 Communicate with relay satellites	Short_range_antenna	VHF Antenna			2	1	4	0	0	7
6.1.2 Communicate with relay satellites	Short_range_data_processor	VHF Ranging			1	1	4	0	1	7
6.1.3 Communicate between suited crewmembe	Interior_voice_comm	In VHF Xceiver			0	0	3	0	1	6
7.1 Sense Subsystem Commands	Data_bus_network_boxes	Signal conditioner Assy -2			2	1	5	0	1	6
7.2 Process/Amplify Subsystem Commands	Master_event_controllers	PCMETA -1			1	1	5	0	1	6
7.3 Send Commands to Subsystems	Sensor_comm_wiring	Wire harness A&B			1	1	5	0	0	7
7.4 Store Spacecraft Data	Data_storage	Data Storage			1	1	3	1	1	6
8.1.1 Monitor Subsystem Data	Health_monitoring_computer	Caution and Warning			1	1	4	1	1	6
8.2.1 Monitor ECS Subsystems	Instrumentation_sensors_ECLSS	ECS Sensors - total mass			0	0	3	2	0	7
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_Propulsion	Prop Sensors - total mass			1	1	5	0	0	8
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_RCS	RCS Sensors - total mass			1	1	5	0	0	8
8.2.2 Monitor Crew Accommodations	Instrumentation_sensors_CA	CA Sensors			0	0	1	0	0	7
8.2.3 Monitor Payload Subsystem	Instrumentation_sensors_Payload	Payload Sensors			0	0	1	0	0	7
8.2.4 Monitor Power Subsystem	Instrumentation_sensors_Power	Power Sensors			0	0	2	0	0	7
8.2.5 Monitor Communication Subsystem	Instrumentation_sensors_Comm	Comm Sensors			0	0	1	0	0	7
8.2.6 Monitor Command and Data Handling (C&	Instrumentation_sensors_CDH	CDH Sensors			0	0	1	0	0	7
8.2.7 Monitor Health Monitoring Subsystem	Instrumentation_sensors_Health	Health Sensors			0	0	1	0	0	7
8.2.8 Monitor Flight Control Subsystem	Instrumentation_sensors_Flight	Flight Sensors			0	0	5	0	0	7
8.2.9 Monitor Thermal Subsystem	Instrumentation_sensors_Thermal	Thermal Sensors			0	0	1	0	0	7
8.2.9 Monitor Structures Subsystem	Instrumentation_sensors_Structural	Structural Sensors			0	0	1	0	0	7
9.1.1 Sense Spacecraft Position Nav Inputs	IMU	IMU			1	1	5	0	1	8
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_base	IMU mounting			1	1	5	0	0	4
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_power	Power and Servo Assy			1	1	5	0	1	6
9.1.1 Sense Spacecraft Position Nav Inputs	Star_tracker	AOT Telescope / Computer Control and Reticle dimm			1	1	4	0	0	5
9.1.2 Convert Analog to Digital Nav Inputs	Nav_analog_digital	Pulse Torque Assembly			1	1	5	0	1	6
9.1.3 Output Navigation to Guidance	Multiplexer_demultiplexers	Coupling Data Unit			1	1	5	0	1	7
9.2.1 Calculate Spacecraft Guidance	Flight_control_computer	LM Guidance Computer			1	1	5	0	1	8
9.2.2 Communicate with Instrumentation	Instrumentation_comm	Signal Conditioner Assy			2	1	5	0	1	7
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw	ACA - Attitude Controller Assembly			1	1	5	0	0	8
9.3.1 Input Human Flight Controls	Flight_translation	TTCA - Thrust and translation controller assembly			1	1	5	0	0	8
9.3.2 Input Human Navigation to Computer	Computer_keyboard	DSKD			1	1	5	0	0	8
9.3.3 Output Spacecraft Control to Propulsion	Control_propulsion	ATCA - Attitude and Translation Control Assy			1	1	5	0	0	8
9.3.4 Display Spacecraft Control	Crew_displays_control	Panel 3 4 5			1	1	5	0	0	8
9.3.5 Display Spacecraft Navigation	Crew_displays_navigation	Panel 1 2			1	1	5	0	0	8
9.3.6 Manual Control Spacecraft Subsystems	Crew_displays_subsystems	Panel 6 8 11 12 14 16			1	1	4	0	0	8
9.4.1 Sense Spacecraft Abort Position	Abort_navigation	Abort Sensor Assy			1	1	5	1	1	8
9.4.2 Sense Spacecraft Abort Velocity Inputs	Abort_control	Rate Gyro Assy			1	1	5	1	1	7
9.4.3 Calculate Spacecraft Abort Trajectory	Abort_guidance	Abort Electronics Assy			1	1	5	1	1	8
9.4.4 Input Abort Commands	Abort_input	DEDA -Data Entry and Display Assembly			1	1	5	1	0	8
9.5 Provide Rendezvous Guidance	Rendezvous_radar	Rendezvous Radar			1	1	4	0	0	10
9.6 Provide Inertial Reference	Ordeal	Ordeal			0	0	2	2	0	6
9.7 Provide Additional Avionics	Misc_avionics	Misc Avionics			0	0	1	2	0	4

Table F.2: EVA, Crew Accommodations, and Payloads input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
EVAs										
12.1.1 Provide Space Suit	Spacesuits	Pressure Suit, no PLSS			**1**	**1**	**3**	1	0	8
12.2.1 Provide Mobility Aids	EVA_mobility	Mobility devices			**0**	**0**	**1**	0	0	6
12.2.2 Provide Umbilical	Suit_umbilical	Umbilical hoses			**0**	**0**	**3**	0	0	7
Crew Accommodations										
3.1.1 Provide Cabin Lighting	Cabin_lighting	Lighting - normalized to one day			**1**	**1**	**2**	1	0	5
3.1.2 Provide Panel Lighting	Panel_lighting	Assumed same as cabin light			**1**	**1**	**3**	0	0	6
3.1.3 Provide Light for Exterior Viewing	Exterior_lighting	Lighting			**2**	**1**	**3**	1	0	7
3.1.4 Provide Restraints	CA_restraints	kg			**1**	**1**	**2**	1	0	4
3.1.5 Provide Handholds	CA_handholds	handholds			**1**	**1**	**1**	2	0	4
3.2.1 Provide Personal Storage	Personal_storage	kg/person/day -normalized from HSMAD			**0**	**0**	**1**	0	0	2
3.2.2 Provide Additional Clothing Storage	Clothing	kg/person/day - normalized from HSMAD			**0**	**0**	**1**	0	0	3
3.2.3 Provide Entertainment	Entertainment	Not included			**0**	**0**	**1**	0	0	5
3.3.2 Provide Medical Kit Storage	Medical_kit	kg/day-normalized from HSMAS			**1**	**1**	**3**	1	0	5
3.3.3 Provide Exercise Capability	Exercise	Not included			**0**	**0**	**1**	0	0	6
3.3.4 Provide Sleep Accommodations	Sleep_accommodations	Not included			**0**	**0**	**1**	0	0	4
3.3.5 Provide Operational Supplies	Operational_supplies	kg/person			**1**	**1**	**3**	1	0	6
3.3.1 Provide Hygiene Supplies	Hygiene_consumables	kg/person/day			**1**	**1**	**1**	2	0	4
3.3.1 Provide Hygiene Supplies	Hygiene_kit	kg/person			**0**	**0**	**1**	2	0	4
Payloads										
4.1.1 Provide Science Payload Storage	Science_return	http://history.nasa.gov/SP-4029			**1**	**1**	**1**	2	0	6
4.1.2 Provide General Payload Storage	General_return	Not included			**0**	**0**	**1**	2	0	6
4.2.1 Provide Tools Storage	Tools_equipment	1.67 kg estimated			**1**	**1**	**2**	1	0	5
4.2.2 Provide Consumable Equipment Storage	Consumable_equipment	Not included			**0**	**0**	**2**	0	0	6
4.2.3 Provide Space Suit Storage	Spacesuit_storage	Not included					**1**	0	0	2
4.2.4 Provide Photography Equipment	Photography	19kg estimated Lunar Mission			**1**	**1**	**1**	2	0	9

Table F.3: ECLSS input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS										
2.1.1 Remove Carbon Dioxide	CO2_crew_day_rate	kg/person/day	**0.998**	**0**			**5**	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_rate	CO2 Canister Perf			**1**	**1**	**5**	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_mass	CO2 Canister Mass			**2**	**1**	**5**	0	0	6
2.1.10 Provide Fire Detection and Suppression	Fire_extinguisher	Not included			**0**	**0**	**4**	0	0	6
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_density	lbm/in^3	**0.296**	**0**			**5**	0	0	7
2.1.2 Provide Metabolic Oxygen	O2_crew_day_rate	kg/person/day	**0.835**	**0**			**5**	0	0	2
2.1.2.1 Store High Pressure Oxygen	Storage_tank_FoS	Storage Tank FOS	**1.5**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	PaO2	psi - Apollo all Oxygen	**5.2**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_radius_length_ratio	ratio	**1**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_temperature	Celsius	**26.67**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_pressure	psi	**840**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_yield_strength	psi	**142000**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_redundancy	Redundancy in Oxygen	**2**	**0**			**5**	0	0	7
2.1.2.1 Store High Pressure Oxygen	Oxygen_tank_insulation	Use same as MMOD			**1**	**1**	**4**	0	0	7
2.1.2.2 Transport Oxygen	Oxygen_gas_transport_lines	Assume 1/2 inch line			**1**	**1**	**5**	0	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_relief_valve	Used 1/2 ISS regulator valve			**1**	**1**	**4**	1	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_regulator_valve	Used 1/2 ISS relief valve			**1**	**1**	**5**	0	0	9
2.1.2.4 Measure pp Oxygen Level in Cabin Air	Oxygen_pp_gauge	Not included			**0**	**0**	**4**	0	0	5
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_temperature	Celsius	**26.67**	**0**			**4**	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_pressure	psi	**840**	**0**			**4**	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	Nitrogen_tank_insulation	Use same as MMOD			**0**	**0**	**4**	0	0	7
2.1.3.2 Transport Makeup Gas	Nitrogen_gas_transport_lines	Assume1/2 inch line			**0**	**0**	**4**	0	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_relief_valve	Used 1/2 ISS relief valve			**0**	**0**	**4**	1	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_regulator_valve	Used 1/2 ISS regulator valve			**0**	**0**	**4**	0	0	9

Table F.3: ECLSS input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS										
2.1.4 Remove Trace Contaminants	Trace_Contaminants_filter	Assumed 1/2 Shuttle Filter			1	1	2	1	0	5
2.1.5 Remove Particulates	Particulates_filter	Assumed 1/2 Shuttle Filter			1	1	2	1	0	5
2.1.6 Remove Humidity	Humidity_percent	Percent (40% = 0.4)	0.6	0			3	0	0	7
2.1.6.2 Condense Cabin Air	Condense_air_heat_exchanger	Assumed 1/2 Spacelab Conds Heat Exch			1	1	3	0	0	7
2.1.6.1 Provide Humidity Separator	Humidity_capture	Assumed 1/2 Spacelab Separator			1	1	3	0	0	5
2.1.7.1 Provide Air Circ Bypass	Mix_air_valve	Assumed Spacelab 1/2 TCV			1	1	2	2	0	7
2.1.7.2 Measure Cabin Air Temp	Cabin_temp	Celsius	22.05	0			2	0	0	6
2.1.7.2 Measure Cabin Air Temp	Temperature_sensor	Assumed Shuttle			1	1	1	1	0	6
2.1.8.1 Circulate Air	Cabin_fan	Assumed 1/2 Spacelab Fan			1	1	4	0	1	7
2.1.8.2 Return Air	Return_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	1	0	4
2.1.8.3 Output Air	Direct_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	1	0	4
2.1.9.1 Provide Air Pressure Sensor	Air_Pressure_gauge	Assumed Shuttle Sensor			1	1	4	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_leak_percent	%/day	0.05	0			4	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_atm_pressure	psi	5.2	0			5	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_air_control	Assumed 1/2 Shuttle Cabin Temp Control			1	1	3	0	0	7
2.2.1 Store Food	Food_storage	Assumed in Food mass			0	0	3	0	0	4
2.2.1 Store Food	Food_rate	kg/person/day			1	1	3	0	0	4
2.2.2 Prepare Food	Galley_supplies	kg/person/day			0	0	1	2	0	4
2.2.2 Prepare Food	Heat_food	Assumed HSMAD			0	0	1	2	0	4
2.3.1 Provide Potable Water	Potable_water	kg/person/day			1	1	4	0	0	5
2.3.2 Provide Hygiene Water	Hygiene_water	kg/person/day			1	1	1	2	0	5
2.4.1 Collect Urine	Urine_bags	kg/person/day			1	1	1	0	0	4
2.4.2 Collect Fecal Solids	Diapers	kg/person/day			0	0	1	0	0	4
2.4.2.2 Control Odor	Control_odor	kg/person/day			0	0	1	2	0	3
2.4.2.3 WCS supplies	WCS_supplies	kg/person/day			1	1	1	2	0	3
2.4.3 Collect Liquid Waste	Vacuum	Assumed HSMAD			0	0	1	2	0	6
2.4.4 Collect Solid Waste	Trash_bags	kg/person/day			0	0	1	0	0	3
2.4.4 Collect Solid Waste	Housecleaning_supplies	kg/person/day			0	0	1	0	0	3
2.4.5 Jettison Liquid Waste	Dump_Valve_Piping	Assumed ISS Overboard Water			1	1	2	2	0	7

Table F.4: Power and Thermal input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Power										
5.1 Provide Power	Batteries	Adapted from Apollo			2	1	5	0	1	7
5.1 Provide Power	Depth_of_Discharge		1	0			5	0	0	7
5.1 Provide Power	Duty_cycle_coast		1	0			5	0	0	7
5.2 Distribute Power	Power_Distr_Wiring	Adapted from Apollo			2	1	5	0	0	7
5.3 Regulate Power	Power_Distr_Controller	Power Controller ECA			2	1	5	0	1	8
5.3 Regulate Power	Power_inverters	Power Inverter			2	1	5	0	1	8
5.3 Regulate Power	Power_relays	Power relay - Electronic			2	1	5	0	1	7
5.6 Provide Overload Protection	Power_Distr_CircBreak	Panels/Circuit Breakers			2	1	4	1	0	7
Thermal										
10.1 Collect Heat	Coldplate_min_power	Minimum watts for a coldplate	20	0			4	0	0	6
10.1 Collect Heat	Loop_capacity	Watts of heat removed single	500	0			4	0	0	6
10.1 Collect Heat	Loop_diameter	inches	0.518	0			4	0	0	6
10.1 Collect Heat	Thermal_interior_fluid_density	kg/m3	1040	0			4	0	0	6
10.1 Collect Heat	Coolant_piping	kg/m3			1	1	4	0	0	6
10.1 Collect Heat	Coldplates	5.25 kg/kW derived from Apollo			1	1	4	0	0	6
10.2 Transport Heat	Interior_coolant_pumps	Assumed Shuttle Water Pump			1	1	4	0	0	7
10.2 Transport Heat	Interior_coolant_lines_valves	Assumed Shuttle Check Valve			1	1	4	0	0	8
10.3 Remove Heat	Interior_exterior_heat_exchanger	3.2 kg/kW derived from Apollo			2	1	4	0	0	7
10.3 Remove Heat	Water_system_sublimators	Latent Heat of Vaporization	2501.3	0			4	0	0	7
10.3 Remove Heat	Exterior_radiators	Radiator Panels			0	0	4	0	0	8

Table F.5: Secondary Structures and Structures input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Secondary Structures										
1.4 Support Internal Subsystems and Componen	Secondary_struct_ratio	ratio	**0.21**	**0**			**4**	0	0	6
Structures										
1.1.1 Provide protection from vacuum	Factor_of_Safety	Structural FoS	**1.41**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Factor_of_Safety_PV_skin	Structural FoS	**2.00**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Interior_diameter_hab_volume	meters	**2**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Cylinder_length_hab_volume	meters	**0.762**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Packaging_efficiency	ratio	**0.55**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Protrusion_ratio	(0.001 -0.25)	**0.065**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_yield_strength	psi	**73000**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_density	lbm/in^3	**0.102**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Elasticity	10^6 psi	**10.4**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Rigidity	10^6 psi	**3.9**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Max_acceleration_gs	gs of acceleration	**4.7**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Number_hoops	Odd Number	**15**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Number_stringers	Multiple of 4	**20**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Height_to_base_ratio	ratio	**8.2**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Fillet_ratio	ratio	**0.1**	**0**			**5**	0	0	6
1.5.3 Provide Docking Mechanism for Lunar Hal	LIDS_published_weight	lbs	**870**	**0**			**4**	0	0	8
1.5.1 Support Docking Mechanism for Lunar Asc	Docking_ring_load_Force	lbf	**23300**	**0**			**4**	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_diameter	inches	**32**	**0**			**4**	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_height	inches	**16**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat tc	Hatch_width	inches	**32**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat tc	Hatch_height	inches	**32**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat tc	Hatch_corner_radius_ratio	ratio	**5.33**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat tc	Hatch_flange_ratio	ratio	**20**	**0**			**4**	0	0	6
1.6.1 Provide Forward Windows	Window_material_density	lbm/in^3	**0.091402**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_pane_thickness	inches	**0.2**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_width_horizontal	inches	**18**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_length	inches	**17**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_corner_radius_ratio	ratio	**4**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_frame_ratio	ratio	**4**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_position_angle	degrees	**45**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_center_height	inches	**60**	**0**			**5**	0	0	7
1.6.2 Provide Docking Viewing Ports	Docking_window	area in^2	**60**	**0**			**4**	0	0	7
1.6.3 Provide Aft Viewing Ports	Aft_window	area in^2	**40**	**0**			**3**	0	0	7
1.1.2 Provide MMOD/TPS protection	MMOD_standoff	Minimum standoff inches	**3**	**0**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_time_days	days	**8.6**	**0**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	PNP_target	Probability	**0.9995**	**0**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	TPS_blanket	kg/m^2 - 1" Thermal Blanket 6	**2.15**	**3**			**4**	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_altitude	km	**350**	**0**			**5**	0	0	6

Table F.6: Propulsion input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion										
11.1.6 Provide Main Engine	Specific_impulse	Isp (seconds)	**311.7**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Vacuum_thrust	N (newtons)	**15746**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Main_engine_weight_thrust_ratio	ratio	**3.17**	**0**			**5**	0	0	9
0.1.4 Launch Trajectory	Gravity_constant	m/s2	**9.81**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Moon_gravity_parameter	km3/sec2	**4902.87**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Moon_radius	km	**1738**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Orbital_apogee_altitude	km	**78.71**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Orbital_perigee_altitude	km	**16.67**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_azimuth	degrees North	**270**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_latitude	degrees	**26.1322**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_longitude	degrees	**3.63386**	**0**			**1**	0	0	0

Table F.6: Propulsion input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion										
11.1.6 Provide Main Engine	Oxidizer_fuel_ratio	ratio	**1.61**	**0**			**5**	0	0	9
11.1.2 Provide Main Engine Oxidizer	Oxidizer_density	kg/m3	**1450**	**0**			**5**	0	0	3
11.1.1 Provide Main Engine Fuel	Fuel_density	kg/m3	**903**	**0**			**5**	0	0	3
11.1.7 Store Main Engine Propellant	Fuel_ox_tank_press	psi	**179**	**0**			**5**	0	0	7
11.1.11 Provide Contingency Propellant	Contingency_propellant	ratio	**0.031**	**0**			**4**	1	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_pressure	psi	**3050**	**0**			**5**	0	0	3
11.2.3 Provide RCS Pressurant	Pressurant_pressure_RCS	psi	**3050**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_tank_temperature	degrees K	**294.26**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_ratio_spec_heats	ratio	**1.667**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_gas_constant	R J/kg*K	**2076.9**	**0**			**5**	0	0	3
11.1.7 Store Main Engine Propellant	Propulsion_tank_material_density	lbm/in^3	**0.16**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_yield_strength	psi	**140000**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_factor_of_safety	FoS	**1.5**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_unused_fraction	ratio	**0.01**	**0**			**5**	0	0	7
11.2.7 Store RCS Propellant	Fuel_ox_tank_press_RCS	psi	**179**	**0**			**5**	0	0	7
11.1.9 Transport Main Engine Propellant	Propulsion_line_id	m	**0.0254**	**0**			**5**	0	0	7
11.1.5 Transport Main Engine Pressurant	Main_engine_pressurant_line_id	m	**0.015875**	**0**			**5**	0	0	7
11.2.9 Transport RCS Propellant	Propulsion_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	7
11.2.5 Transport RCS Pressurant	Pressurant_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	7
11.1.6 Provide Main Engine	Main_engine_linear_slope	linear_slope kg/kN	**0.5786**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Main_engine_y_intercept	linear_curve_intercept - kg	**65**	**0**			**5**	0	0	9
11.1.12 Support Main Engine Components	Propulsion_secondary_struct_ratio	ratio	**0.2**	**0**			**5**	0	0	6
11.2.6 Provide RCS thrusters	RCS_side_A_B_choice	Integer, 1,2,3	**2**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Specific_impulse_RCS	Isp (seconds)	**240**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Vacuum_thrust_RCS	N (newtons)	**444.822**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_linear_slope	linear_slope kg/kN	**6.48**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_y_intercept	linear_curve_intercept - kg	**0**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Num_RCS_engines_system	Integer, 4,8,16	**8**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Num_fire_RCS_thrust_ascent	Num firing during ascent	**0.57**	**0**			**5**	0	0	9
11.2.11 Provide RCS contingency propellant	RCS_additional_propellant_mass	Rendezvous -kg	**96.2**	**3**			**4**	1	0	3
11.1.13 Insulate Main Engine Propellant	Propulsion_tanks_insulation	Use same as TPS			**1**	**1**	**4**	0	0	4
11.2.13 Insulate RCS Propellant	Propulsion_tanks_insulation_RCS	Placeholder			**1**	**1**	**4**	0	0	4
11.2.1 Provide RCS Propellant	RCS_propellant	Placeholder			**1**	**1**	**5**	0	0	3
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_valve	kg			**1**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_filter	kg			**1**	**1**	**4**	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_solenoid_valve	kg			**1**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_pressure_regulator	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_check_valve	kg			**2**	**1**	**4**	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_explosive_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_relief_valve	kg			**1**	**1**	**4**	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_trim_orifice	kg			**1**	**1**	**5**	0	0	5
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_filter	kg			**1**	**1**	**4**	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_isolation_valve	kg			**2**	**1**	**5**	0	0	9
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_bipropellant_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_valve	kg			**1**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_filter	kg			**1**	**1**	**4**	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_solenoid_valve	kg			**1**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_pressure_regulator	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_check_valve	kg			**2**	**1**	**4**	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_explosive_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_relief_valve	kg			**1**	**1**	**4**	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_trim_orifice	kg			**1**	**1**	**5**	0	0	5
11.1.10.1 Control Main Engine Fuel	Fuel_filter	kg			**1**	**1**	**4**	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_isolation_valve	kg			**2**	**1**	**5**	0	0	9
11.1.10.1 Control Main Engine Fuel	Fuel_bipropellant_valve	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_valve_RCS	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_filter_RCS	kg			**1**	**1**	**4**	1	0	7
11.2.4 Control RCS Pressurant	System_pressurant_orifice_RCS	kg			**1**	**1**	**5**	0	0	5
11.2.4 Control RCS Pressurant	System_pressurant_pressure_regulator_R	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_check_valve_RCS	kg			**1**	**1**	**4**	1	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_relief_valve_RCS	kg			**1**	**1**	**4**	1	0	6
11.2.4 Control RCS Pressurant	Fuel_pressurant_check_valve_RCS	kg			**1**	**1**	**4**	1	0	7
11.2.4 Control RCS Pressurant	Fuel_pressurant_relief_valve_RCS	kg			**1**	**1**	**4**	1	0	7
11.2.10.2 Control RCS Oxidizer	Oxidizer_shutoff_valve_RCS	kg			**1**	**1**	**5**	0	0	9
11.2.10.1 Control RCS Fuel	Fuel_shutoff_valve_RCS	kg			**1**	**1**	**5**	0	0	9
11.2.9.1 Filter RCS Propellant	Fuel_Ox_filter_RCS	kg			**1**	**1**	**4**	1	0	7
11.2.10 Control RCS Propellant	Fuel_Ox_isolation_valve_RCS	kg			**1**	**1**	**5**	0	0	9
11.2.14.2 Heat RCS Oxidizer lines	Oxidizer_heater_RCS	kg			**1**	**1**	**4**	1	0	6
11.2.14.1 Heat RCS Fuel lines	Fuel_heater_RCS	kg			**1**	**1**	**4**	1	0	6
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_valve	kg			**2**	**1**	**4**	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_filter	kg			**2**	**1**	**4**	1	0	7
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_valve	kg			**2**	**1**	**4**	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_filter	kg			**2**	**1**	**4**	1	0	7

Table F.7: Avionics technology database.

Equipment	Mass (kg)	Mass Error (+/-,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Abort_control	0.91	5	0.000	0.000	0.000	2
Abort_guidance	14.74	5	0.603	0.203	0.133	96
Abort_input	3.40	5	0.140	0.152	0.132	10
Abort_navigation	9.38	5	0.130	0.229	0.292	74
Computer_keyboard	7.71	5	0.203	0.203	0.178	25
Control_propulsion	10.75	5	0.000	0.000	0.000	50
Control_propulsion_thrusters	0.00	5	0.000	0.000	0.000	0
Crew_displays_control	7.44	5	0.000	0.000	0.000	0
Crew_displays_navigation	15.79	5	0.000	0.000	0.000	0
Crew_displays_subsystems	5.35	5	0.000	0.000	0.000	0
Data_bus_network_boxes	15.99	5	0.203	0.133	0.607	15
Data_storage	1.13	5	0.052	0.102	0.158	10
Flight_control_computer	31.75	5	0.152	0.318	0.610	70
Flight_roll_pitch_yaw	2.16	5	0.000	0.000	0.000	0
Flight_translation	2.38	5	0.000	0.000	0.000	0
Health_monitoring_computer	8.30	5	0.178	0.171	0.298	13
IMU	19.10	5	0.000	0.000	0.000	100
Instrumentation_comm	1.27	5	0.084	0.224	0.284	25
Instrumentation_sensors_CA	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_CDH	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Comm	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_ECLSS	2.68	5	0.000	0.000	0.000	1
Instrumentation_sensors_Flight	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Health	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Payload	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Power	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Propulsion	2.45	5	0.000	0.000	0.000	1
Instrumentation_sensors_RCS	2.22	5	0.000	0.000	0.000	1
Instrumentation_sensors_Thermal	0.00	5	0.000	0.000	0.000	0
Interior_voice_comm	0.00	5	0.000	0.000	0.000	0
Long_range_amplifier	4.24	5	0.000	0.000	0.000	36
Long_range_antenna	0.18	5	0.000	0.000	0.000	1
Long_range_data_processor	4.72	5	0.000	0.000	0.000	27
Long_range_steerable_antenna	12.47	5	0.000	0.000	0.000	60
Long_range_transceiver	4.54	5	0.000	0.000	0.000	18
Master_event_controllers	10.25	5	0.171	0.130	0.502	11
Misc_avionics	0.00	5	0.000	0.000	0.000	0
Multiplexer_demultiplexers	15.88	5	0.140	0.288	0.508	25
Nav_analog_digital	6.80	5	0.064	0.279	0.330	25
Nav_base	1.36	5	0.287	0.287	0.287	0
Nav_power	9.07	5	0.067	0.225	0.597	100
Ordeal	3.13	5	0.000	0.000	0.000	4
Rendezvous_radar	35.43	5	0.000	0.000	0.000	95
Short_range_antenna	1.09	5	0.000	0.000	0.000	1
Short_range_data_processor	1.18	5	0.000	0.000	0.000	5
Short_range_transceiver	2.97	5	0.000	0.000	0.000	32
Star_tracker	10.48	5	0.000	0.000	0.000	0

Table F.8: EVA technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
EVA_mobility	0.00	5	0.000	0.000	0.000	0
Spacesuits	33.00	5	0.000	0.000	0.000	0
Suit_umbilical	0.00	5	0.000	0.000	0.000	0

Table F.9: Payloads technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Science_return	115.62	5	0.000	0.000	0.000	0
Consumable_equipment	0.00	5	0.000	0.000	0.000	0
General_return	0.00	5	0.000	0.000	0.000	0
Photography	19.00	5	0.437	0.437	0.437	0
Tools_equipment	1.67	5	0.149	0.149	0.149	0

Table F.10: Power technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Batteries	154.27	5	0.00	0.00	0.00	1000
Power_Distr_CircBreak	17.89	5	0.000	0.000	0.000	0
Power_Distr_Controller	4.76	5	0.000	0.000	0.000	0
Power_Distr_Wiring	47.14	5	0.000	0.000	0.000	0
Power_inverters	6.98	5	0.000	0.000	0.000	0
Power_relays	4.56	5	0.000	0.000	0.000	0

Table F.11: Thermal technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Coldplates	5.50	5	0.000	0.000	0.000	0
Exterior_radiators	0.00	5	0.000	0.000	0.000	0
Interior_coolant_lines_valves	0.90	5	0.000	0.000	0.000	0
Interior_coolant_pumps	1.88	5	0.000	0.000	0.000	0

Table F.12: Crew Accommodations technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Cabin_lighting	1.00	5	0.100	0.100	0.100	25
CA_handholds	0.60	5	0.100	0.100	0.100	0
CA_restraints	4.16	5	0.100	0.100	0.100	0
Clothing	1.53	5	0.100	0.100	0.100	0
Entertainment	0.00	5	0.000	0.000	0.000	0
Exercise	0.00	5	0.000	0.000	0.000	0
Exterior_lighting	6.00	5	0.000	0.000	0.000	100
Medical_kit	0.32	5	0.100	0.100	0.100	0
Operational_supplies	6.50	5	0.100	0.100	0.100	0
Panel_lighting	1.00	5	0.100	0.100	0.100	25
Personal_storage	1.66	5	0.100	0.100	0.100	0
Sleep_accommodations	0.00	5	0.000	0.000	0.000	0

Table F.13: ECLSS technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Oxygen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Air_Pressure_gauge	0.25	5	0.000	0.000	0.000	1
Cabin_air_control	2.22	5	0.006	0.006	0.006	16
Cabin_fan	4.65	5	0.002	0.002	0.002	70
Condense_air_heat_exchanger	8.89	5	0.045	0.045	0.045	0
Control_odor	0.00	5	0.000	0.000	0.000	0
Diapers	0.00	5	0.000	0.000	0.000	0
Dump_Valve_Piping	1.46	5	0.005	0.005	0.005	0
Fecal_bags	0.00	5	0.000	0.000	0.000	0
Fire_extinguisher	0.00	5	0.000	0.000	0.000	0
Food_rate	2.30	5	0.005	0.005	0.005	0
Food_storage	0.00	5	0.000	0.000	0.000	0
Galley_supplies	0.50	5	0.001	0.001	0.001	0
Heat_food	5.00	5	0.000	0.000	0.000	0
Housecleaning_supplies	0.00	5	0.000	0.000	0.000	0
Humidity_capture	1.28	5	0.004	0.004	0.004	0
Hygiene_water	4.10	5	0.000	0.000	0.000	0
Mix_air_valve	1.15	5	0.008	0.008	0.008	0
Nitrogen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Nitrogen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Oxygen_pp_gauge	0.00	5	0.000	0.000	0.000	0
Oxygen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Particulates_filter	1.25	5	0.000	0.000	0.000	0
Potable_water	1.60	5	0.000	0.000	0.000	0
Temperature_sensor	0.08	5	0.000	0.000	0.000	0
Trace_Contaminants_filter	1.25	5	0.000	0.000	0.000	0
Trash_bags	0.00	5	0.000	0.000	0.000	0
Urine_bags	0.23	5	0.000	0.000	0.000	0
Vacuum	0.00	5	0.000	0.000	0.000	0
WCS_supplies	0.05	5	0.001	0.001	0.001	0

Table F.14: Propulsion technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
APS_interconnect_Fuel_filter	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Fuel_valve	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_filter	0.76	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_valve	0.76	5	0.050	0.050	0.050	0
Fuel_bipropellant_valve	0.70	5	0.050	0.050	0.050	0
Fuel_filter	0.70	5	0.050	0.050	0.050	0
Fuel_heater_RCS	0.60	5	0.050	0.050	0.050	0
Fuel_isolation_valve	0.70	5	0.050	0.050	0.050	0
Fuel_Ox_filter_RCS	0.11	5	0.050	0.050	0.050	0
Fuel_Ox_isolation_valve_RCS	0.11	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve_RCS	0.51	5	0.050	0.050	0.050	0
Fuel_pressurant_explosive_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_filter	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_pressure_regulator	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve_RCS	0.51	5	0.050	0.050	0.050	0
Fuel_pressurant_solenoid_valve	0.83	5	0.050	0.050	0.050	0
Fuel_pressurant_valve	0.83	5	0.050	0.050	0.050	0
Fuel_shutoff_valve_RCS	0.60	5	0.050	0.050	0.050	0
Fuel_trim_orifice	0.70	5	0.050	0.050	0.050	0
Oxidizer_bipropellant_valve	0.70	5	0.050	0.050	0.050	0
Oxidizer_filter	0.70	5	0.050	0.050	0.050	0
Oxidizer_heater_RCS	0.60	5	0.050	0.050	0.050	0
Oxidizer_isolation_valve	0.70	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve_RCS	0.51	5	0.050	0.050	0.050	0
Oxidizer_pressurant_explosive_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_filter	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_pressure_regulator	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve_RCS	0.51	5	0.050	0.050	0.050	0
Oxidizer_pressurant_solenoid_valve	0.83	5	0.050	0.050	0.050	0
Oxidizer_shutoff_valve_RCS	0.60	5	0.050	0.050	0.050	0
Oxidizer_trim_orifice	0.70	5	0.050	0.050	0.050	0
System_pressurant_filter_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_orifice_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_pressure_regulator_RCS	0.51	5	0.050	0.050	0.050	0
System_pressurant_valve_RCS	0.51	5	0.050	0.050	0.050	0

APPENDIX G

ESAS CONFIGURATION INPUT VARIABLE TABLES

The following tables were copied from the Microsoft Excel input file for the CLAMP analysis. The tables are grouped according to subsystems.

Table G.1: Crew, Mission Time, and Avionics input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
0.1.1 Number of Crew	Crew_members	Num of Crew	4	0			1	0	0	0
0.1.2 Maximum Mission Time	Mission_Duration_HRS	Mission hours	3	0			1	0	0	0
0.1.3 Minimum Mission Time	Mission_Duration_HRS_min	Absolute Minimum Mission hours	1.33	0			1	0	0	0
Avionics										
6.1.1 Communicate with Earth ground station	Long_range_transceiver	S-Band Transceiver -2			2	1	3	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_amplifier	Power_amp & diplex - 1			2	1	3	0	1	7
6.1.1 Communicate with Earth ground station	Long_range_antenna	S_Band Inflight Antennas - 2			2	1	3	0	0	7
6.1.1 Communicate with Earth ground station	Long_range_steerable_antenna	Steerable Antenna - 1			1	1	3	0	0	8
6.1.1 Communicate with Earth ground station	Long_range_data_processor	Signal Processor Assy			2	1	4	0	1	7
6.1.2 Communicate with relay satellites	Short_range_transceiver	VHF Xceiver and diplex			2	1	4	0	1	7
6.1.2 Communicate with relay satellites	Short_range_antenna	VHF Antenna			2	1	4	0	0	7
6.1.2 Communicate with relay satellites	Short_range_data_processor	VHF Ranging			2	1	4	0	1	7
6.1.3 Communicate between suited crewmembe	Interior_voice_comm	In VHF Xceiver			0	0	3	0	1	6
7.1 Sense Subsystem Commands	Data_bus_network_boxes	Signal conditioner Assy -2			2	1	5	0	1	6
7.2 Process/Amplify Subsystem Commands	Master_event_controllers	PCMETA -1			2	1	5	0	1	6
7.3 Send Commands to Subsystems	Sensor_comm_wiring	Wire harness A&B			2	1	5	0	0	7
7.4 Store Spacecraft Data	Data_storage	Data Storage			1	1	3	2	1	6
8.1.1 Monitor Subsystem Data	Health_monitoring_computer	Caution and Warning			1	1	4	1	1	6
8.2.1 Monitor ECS Subsystems	Instrumentation_sensors_ECLSS	ECS Sensors - total mass			1	1	3	2	0	7
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_Propulsion	Prop Sensors - total mass			1	1	5	0	0	8
8.2.10 Monitor Prop Subsystems	Instrumentation_sensors_RCS	RCS Sensors - total mass			1	1	5	0	0	8
8.2.2 Monitor Crew Accommodations	Instrumentation_sensors_CA	CA Sensors			0	0	1	0	0	7
8.2.3 Monitor Payload Subsystem	Instrumentation_sensors_Payload	Payload Sensors			0	0	1	0	0	7
8.2.4 Monitor Power Subsystem	Instrumentation_sensors_Power	Power Sensors			0	0	2	0	0	7
8.2.5 Monitor Communication Subsystem	Instrumentation_sensors_Comm	Comm Sensors			0	0	1	0	0	7
8.2.6 Monitor Command and Data Handling (C&	Instrumentation_sensors_CDH	CDH Sensors			0	0	1	0	0	7
8.2.7 Monitor Health Monitoring Subsystem	Instrumentation_sensors_Health	Health Sensors			0	0	1	0	0	7
8.2.8 Monitor Flight Control Subsystem	Instrumentation_sensors_Flight	Flight Sensors			0	0	5	0	0	7
8.2.9 Monitor Thermal Subsystem	Instrumentation_sensors_Thermal	Thermal Sensors			0	0	1	0	0	7
8.2.9 Monitor Structures Subsystem	Instrumentation_sensors_Structural	Structural Sensors			0	0	1	0	0	7
9.1.1 Sense Spacecraft Position Nav Inputs	IMU	IMU			2	1	5	0	1	8
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_base	IMU mounting			2	1	5	0	0	4
9.1.1 Sense Spacecraft Position Nav Inputs	Nav_power	Power and Servo Assy			2	1	5	0	1	6
9.1.1 Sense Spacecraft Position Nav Inputs	Star_tracker	AOT Telescope / Computer Control and Reticle dimmer a			1	1	4	0	0	5
9.1.2 Convert Analog to Digital Nav Inputs	Nav_analog_digital	Pulse Torque Assembly			2	1	5	0	1	6
9.1.3 Output Navigation to Guidance	Multiplexer_demultiplexers	Coupling Data Unit			2	1	5	0	1	7
9.2.1 Calculate Spacecraft Guidance	Flight_control_computer	LM Guidance Computer			2	1	5	0	1	8
9.2.2 Communicate with Instrumentation	Instrumentation_comm	Signal Conditioner Assy			2	1	5	0	1	7
9.3.1 Input Human Flight Controls	Flight_roll_pitch_yaw	ACA - Attitude Controller Assembly			2	1	5	0	0	8
9.3.1 Input Human Flight Controls	Flight_translation	TTCA - Thrust and translation controller assembly			2	1	5	0	0	8
9.3.2 Input Human Navigation to Computer	Computer_keyboard	DSKD			2	1	5	0	0	8
9.3.3 Output Spacecraft Control to Propulsion	Control_propulsion	ATCA - Attitude and Translation Control Assy			1	1	5	0	0	8
9.3.4 Display Spacecraft Control	Crew_displays_control	Panel 3 4 5			1	1	5	0	0	8
9.3.5 Display Spacecraft Navigation	Crew_displays_navigation	Panel 1 2			1	1	5	0	0	8
9.3.6 Manual Control Spacecraft Subsystems	Crew_displays_subsystems	Panel 6 8 11 12 14 16			1	1	4	0	0	8
9.4.1 Sense Spacecraft Abort Position	Abort_navigation	Abort Sensor Assy			1	1	5	1	1	8
9.4.2 Sense Spacecraft Abort Velocity Inputs	Abort_control	Rate Gyro Assy			1	1	5	1	1	7
9.4.3 Calculate Spacecraft Abort Trajectory	Abort_guidance	Abort Electronics Assy			1	1	5	1	1	8
9.4.4 Input Abort Commands	Abort_input	DEDA -Data Entry and Display Assembly			1	1	5	1	0	8
9.5 Provide Rendezvous Guidance	Rendezvous_radar	Rendezvous Radar			1	1	4	0	0	10
9.6 Provide Inertial Reference	Ordeal	Ordeal			1	1	2	2	0	6
9.7 Provide Additional Avionics	Misc_avionics	Misc Avionics			0	0	1	2	0	4

Table G.2: EVA, Crew Accommodations, and Payloads input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
EVAs										
12.1.1 Provide Space Suit	Spacesuits	Pressure Suit, no PLSS			1	1	3	2	0	8
12.2.1 Provide Mobility Aids	EVA_mobility	Mobility devices			0	0	1	0	0	6
12.2.2 Provide Umbilical	Suit_umbilical	Umbilical hoses			0	0	3	0	0	7
Crew Accommodations										
3.1.1 Provide Cabin Lighting	Cabin_lighting	Lighting - normalized to one day			2	1	2	0	0	5
3.1.2 Provide Panel Lighting	Panel_lighting	Assumed same as cabin light			2	1	3	0	0	6
3.1.3 Provide Light for Exterior Viewing	Exterior_lighting	Lighting			2	1	3	0	0	7
3.1.4 Provide Restraints	CA_restraints	kg			2	1	2	2	0	4
3.1.5 Provide Handholds	CA_handholds	handholds			2	1	1	2	0	4
3.2.1 Provide Personal Storage	Personal_storage	kg/person/day -normalized from HSMAD			0	0	1	0	0	2
3.2.2 Provide Additional Clothing Storage	Clothing	kg/person/day - normalized from HSMAD			0	0	1	0	0	3
3.2.3 Provide Entertainment	Entertainment	Not included			0	0	1	0	0	5
3.3.2 Provide Medical Kit Storage	Medical_kit	kg/day-normalized from HSMAS			1	1	3	2	0	5
3.3.3 Provide Exercise Capability	Exercise	Not included			0	0	1	0	0	6
3.3.4 Provide Sleep Accommodations	Sleep_accommodations	Not included			0	0	1	0	0	4
3.3.5 Provide Operational Supplies	Operational_supplies	kg/person			1	1	3	2	0	6
3.3.1 Provide Hygiene Supplies	Hygiene_consumables	kg/person/day			1	1	1	2	0	4
3.3.1 Provide Hygiene Supplies	Hygiene_kit	kg/person			0	0	1	2	0	4
Payloads										
4.1.1 Provide Science Payload Storage	Science_return	http://history.nasa.gov/SP-4029/Apollo_15a_Summary.ht			0	0	1	2	0	6
4.1.2 Provide General Payload Storage	General_return	Not included			0	0	1	2	0	6
4.2.1 Provide Tools Storage	Tools_equipment	1.67 kg estimated			0	0	2	2	0	5
4.2.2 Provide Consumable Equipment Storage	Consumable_equipment	Not included			0	0	2	0	0	6
4.2.3 Provide Space Suit Storage	Spacesuit_storage	Not included					1	0	0	2
4.2.4 Provide Photography Equipment	Photography	19kg estimated Lunar Mission			0	0	1	2	0	9

Table G.3: ECLSS input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS										
2.1.1 Remove Carbon Dioxide	CO2_crew_day_rate	kg/person/day	0.998	0			5	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_rate	CO2 Canister Perf			1	1	5	0	0	6
2.1.1 Remove Carbon Dioxide	CO2_removal_mass	CO2 Canister Mass			2	1	5	0	0	6
2.1.10 Provide Fire Detection and Suppression	Fire_extinguisher	Not included			0	0	4	0	0	6
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_density	lbm/in^3	0.296	0			5	0	0	7
2.1.2 Provide Metabolic Oxygen	O2_crew_day_rate	kg/person/day	0.835	0			5	0	0	2
2.1.2.1 Store High Pressure Oxygen	Storage_tank_FoS	Storage Tank FOS	1.5	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	PaO2	psi - Apollo all Oxygen	3	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_radius_length_ratio	ratio	1	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_temperature	Celsius	26.67	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	O2_storage_pressure	psi	840	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_material_yield_strength	psi	142000	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	Storage_tank_redundancy	Redundancy in Oxygen	2	0			5	0	0	7
2.1.2.1 Store High Pressure Oxygen	Oxygen_tank_insulation	Use same as MMOD			1	1	4	0	0	7
2.1.2.2 Transport Oxygen	Oxygen_gas_transport_lines	Assume 1/2 inch line			1	1	5	0	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_relief_valve	Used 1/2 ISS regulator valve			1	1	4	1	0	7
2.1.2.3 Control Oxygen Flow	Oxygen_tank_regulator_valve	Used 1/2 ISS relief valve			1	1	5	0	0	9
2.1.2.4 Measure pp Oxygen Level in Cabin Air	Oxygen_pp_gauge	Not included			0	0	4	0	0	5
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_temperature	Celsius	26.67	0			4	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	N2_storage_pressure	psi	840	0			4	0	0	7
2.1.3.1 Store High Pressure Makeup Gas	Nitrogen_tank_insulation	Use same as MMOD			1	1	4	0	0	7
2.1.3.2 Transport Makeup Gas	Nitrogen_gas_transport_lines	Assume1/2 inch line			1	1	4	0	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_relief_valve	Used 1/2 ISS relief valve			1	1	4	1	0	7
2.1.3.3 Control Makeup Gas Flow	Nitrogen_tank_regulator_valve	Used 1/2 ISS regulator valve			1	1	4	0	0	9

Table G.3: ECLSS input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
ECLSS										
2.1.4 Remove Trace Contaminants	Trace_Contaminants_filter	Assumed 1/2 Shuttle Filter			1	1	2	0	0	5
2.1.5 Remove Particulates	Particulates_filter	Assumed 1/2 Shuttle Filter			1	1	2	0	0	5
2.1.6 Remove Humidity	Humidity_percent	Percent (40% = 0.4)	0.6	0			3	0	0	7
2.1.6.2 Condense Cabin Air	Condense_air_heat_exchanger	Assumed 1/2 Spacelab Conds Heat Exch			1	1	3	0	0	7
2.1.6.1 Provide Humidity Separator	Humidity_capture	Assumed 1/2 Spacelab Separator			1	1	3	0	0	5
2.1.7.1 Provide Air Circ Bypass	Mix_air_valve	Assumed Spacelab 1/2 TCV			1	1	2	2	0	7
2.1.7.2 Measure Cabin Air Temp	Cabin_temp	Celsius	22.05	0			2	0	0	6
2.1.7.2 Measure Cabin Air Temp	Temperature_sensor	Assumed Shuttle			1	1	1	0	0	6
2.1.8.1 Circulate Air	Cabin_fan	Assumed 1/2 Spacelab Fan			1	1	4	0	1	7
2.1.8.2 Return Air	Return_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	0	0	4
2.1.8.3 Output Air	Direct_air_ducts	Unknown - Assumed 4 in diameterflex line			1	1	1	0	0	4
2.1.9.1 Provide Air Pressure Sensor	Air_Pressure_gauge	Assumed Shuttle Sensor			1	1	4	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_leak_percent	%/day	0.05	0			4	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_atm_pressure	psi	9.5	0			5	0	0	7
2.1.9.2 Provide Cabin Air Control Logic	Cabin_air_control	Assumed 1/2 Shuttle Cabin Temp Control			1	1	3	0	0	7
2.2.1 Store Food	Food_storage	Assumed in Food mass			1	1	3	0	0	4
2.2.1 Store Food	Food_rate	kg/person/day			1	1	3	0	0	4
2.2.2 Prepare Food	Galley_supplies	kg/person/day			1	1	1	2	0	4
2.2.2 Prepare Food	Heat_food	Assumed HSMAD			1	1	1	2	0	4
2.3.1 Provide Potable Water	Potable_water	kg/person/day			1	1	4	0	0	5
2.3.2 Provide Hygiene Water	Hygiene_water	kg/person/day			1	1	1	2	0	5
2.4.1 Collect Urine	Urine_bags	kg/person/day			1	1	1	0	0	4
2.4.2 Collect Fecal Solids	Diapers	kg/person/day			1	1	1	0	0	4
2.4.2.2 Control Odor	Control_odor	kg/person/day			0	0	1	2	0	3
2.4.2.3 WCS supplies	WCS_supplies	kg/person/day			1	1	1	2	0	3
2.4.3 Collect Liquid Waste	Vacuum	Assumed HSMAD			0	0	1	2	0	6
2.4.4 Collect Solid Waste	Trash_bags	kg/person/day			0	0	1	0	0	3
2.4.4 Collect Solid Waste	Housecleaning_supplies	kg/person/day			0	0	1	0	0	3
2.4.5 Jettison Liquid Waste	Dump_Valve_Piping	Assumed ISS Overboard Water			1	1	2	2	0	7

Table G.4: Power and Thermal input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Power										
5.1 Provide Power	Batteries	Adapted from Apollo			2	1	5	0	1	7
5.1 Provide Power	Depth_of_Discharge		1	0			5	0	0	7
5.1 Provide Power	Duty_cycle_coast		1	0			5	0	0	7
5.2 Distribute Power	Power_Distr_Wiring	Adapted from Apollo			2	1	5	0	0	7
5.3 Regulate Power	Power_Distr_Controller	Power Controller ECA			2	1	5	0	1	8
5.3 Regulate Power	Power_inverters	Power Inverter			2	1	5	0	1	8
5.3 Regulate Power	Power_relays	Power relay - Electronic			2	1	5	0	1	7
5.6 Provide Overload Protection	Power_Distr_CircBreak	Panels/Circuit Breakers			2	1	4	1	0	7
Thermal										
10.1 Collect Heat	Coldplate_min_power	Minimum watts for a coldplate	20	0			4	0	0	6
10.1 Collect Heat	Loop_capacity	Watts of heat removed single loo	500	0			4	0	0	6
10.1 Collect Heat	Loop_diameter	inches	0.518	0			4	0	0	6
10.1 Collect Heat	Thermal_interior_fluid_density	kg/m3	1040	0			4	0	0	6
10.1 Collect Heat	Coolant_piping	kg/m3			2	1	4	0	0	6
10.1 Collect Heat	Coldplates	5.25 kg/kW derived from Apollo			2	1	4	0	0	6
10.2 Transport Heat	Interior_coolant_pumps	Assumed Shuttle Water Pump			2	1	4	0	0	7
10.2 Transport Heat	Interior_coolant_lines_valves	Assumed Shuttle Check Valve			2	1	4	0	0	8
10.3 Remove Heat	Interior_exterior_heat_exchanger	3.2 kg/kW derived from Apollo			2	1	4	0	0	7
10.3 Remove Heat	Water_system_sublimators	Latent Heat of Vaporization	2501.3	0			4	0	0	7
10.3 Remove Heat	Exterior_radiators	Radiator Panels			0	0	4	0	0	8

Table G.5: Secondary Structures and Structures input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Secondary Structures										
1.4 Support Internal Subsystems and Componen	Secondary_struct_ratio	ratio	**0.2155**	**0**			**4**	0	0	6
Structures										
1.1.1 Provide protection from vacuum	Factor_of_Safety	Structural FoS	**1.41**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Factor_of_Safety_PV_skin	Structural FoS	**1.41**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Interior_diameter_hab_volume	meters	**3**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Cylinder_length_hab_volume	meters	**4.245**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Packaging_efficiency	ratio	**0.75**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Protrusion_ratio	(0.001 -0.25)	**0.08**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_yield_strength	psi	**73000**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_material_density	lbm/in^3	**0.102**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Elasticity	10^6 psi	**10.4**	**0**			**5**	0	0	6
1.1.1 Provide protection from vacuum	Pressure_vessel_mod_of_Rigidity	10^6 psi	**3.9**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Max_acceleration_gs	gs of acceleration	**3.5**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Number_hoops	Odd Number	**25**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Number_stringers	Multiple of 4	**20**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Height_to_base_ratio	ratio	**9.99**	**0**			**5**	0	0	6
1.2 Provide Load Bearing Capability	Fillet_ratio	ratio	**0.1**	**0**			**5**	0	0	6
1.5.3 Provide Docking Mechanism for Lunar Hat	LIDS_published_weight	lbs	**870**	**0**			**4**	0	0	8
1.5.1 Support Docking Mechanism for Lunar Asc	Docking_ring_load_Force	lbf	**23300**	**0**			**4**	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_diameter	inches	**32**	**0**			**4**	0	0	6
1.5.2 Provide Ingress/Egress for Lunar Ascent M	LIDS_tunnel_height	inches	**16**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_width	inches	**32**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_height	inches	**32**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_corner_radius_ratio	ratio	**5.33**	**0**			**4**	0	0	6
1.5.4 Provide Ingress/Egress for Lunar Habitat to	Hatch_flange_ratio	ratio	**20**	**0**			**4**	0	0	6
1.6.1 Provide Forward Windows	Window_material_density	lbm/in^3	**0.091402**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_pane_thickness	inches	**0.2**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_width_horizontal	inches	**18**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_length	inches	**17**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_corner_radius_ratio	ratio	**4**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_frame_ratio	ratio	**4**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_position_angle	degrees	**45**	**0**			**5**	0	0	7
1.6.1 Provide Forward Windows	Window_center_height	inches	**60**	**0**			**5**	0	0	7
1.6.2 Provide Docking Viewing Ports	Docking_window	area in^2	**60**	**0**			**4**	0	0	7
1.6.3 Provide Aft Viewing Ports	Aft_window	area in^2	**40**	**0**			**3**	0	0	7
1.1.2 Provide MMOD/TPS protection	MMOD_standoff	Minimum standoff inches	**3**	**0**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_time_days	days	**8.6**	**0**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	PNP_target	Probability	**0.9905**	**7**			**5**	0	0	6
1.1.2 Provide MMOD/TPS protection	TPS_blanket	kg/m^2 - 1" Thermal Blanket 6pc	**4.37**	**3**			**4**	0	0	6
1.1.2 Provide MMOD/TPS protection	LEO_altitude	km	**350**	**0**			**5**	0	0	6

Table G.6: Propulsion input variables list.

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion										
11.1.6 Provide Main Engine	Specific_impulse	Isp (seconds)	**320.7**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Vacuum_thrust	N (newtons)	**44500**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Main_engine_weight_thrust_ratio	ratio	**3.06**	**0**			**5**	0	0	9
0.1.4 Launch Trajectory	Gravity_constant	m/s2	**9.81**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Moon_gravity_parameter	km3/sec2	**4902.87**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Moon_radius	km	**1738**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Orbital_apogee_altitude	km	**78.71**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Orbital_perigee_altitude	km	**16.67**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_azimuth	degrees North	**270**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_latitude	degrees	**26.1322**	**0**			**1**	0	0	0
0.1.4 Launch Trajectory	Launch_longitude	degrees	**3.63386**	**0**			**1**	0	0	0

Table G.6: Propulsion input variables list (continued).

Function	Variable_Keyword	Units	Variable Value	Variable Error(+/-,%)	Number of Components -Technology / Redundancy (0,1,2,3)	# Systems	Criticality	Safety = 1, Operab = 2	Cooling 0-no,1-yes	Hazards
Propulsion										
11.1.6 Provide Main Engine	Oxidizer_fuel_ratio	ratio	**3.4**	**0**			**5**	0	0	9
11.1.2 Provide Main Engine Oxidizer	Oxidizer_density	kg/m3	**1140**	**0**			**5**	0	0	3
11.1.1 Provide Main Engine Fuel	Fuel_density	kg/m3	**445**	**0**			**5**	0	0	3
11.1.7 Store Main Engine Propellant	Fuel_ox_tank_press	psi	**179**	**0**			**5**	0	0	7
11.1.11 Provide Contingency Propellant	Contingency_propellant	ratio	**0.083**	**0**			**4**	1	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_pressure	psi	**3050**	**0**			**5**	0	0	3
11.2.3 Provide RCS Pressurant	Pressurant_pressure_RCS	psi	**3050**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_tank_temperature	degrees K	**294.26**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_ratio_spec_heats	ratio	**1.667**	**0**			**5**	0	0	3
11.1.3 Provide Main Engine Pressurant	Pressurant_gas_constant	R J/kg*K	**2076.9**	**0**			**5**	0	0	3
11.1.7 Store Main Engine Propellant	Propulsion_tank_material_density	lbm/in^3	**0.16**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_yield_strength	psi	**140000**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_factor_of_safety	FoS	**1.5**	**0**			**5**	0	0	7
11.1.7 Store Main Engine Propellant	Propulsion_tank_unused_fraction	ratio	**0.01**	**0**			**5**	0	0	7
11.2.7 Store RCS Propellant	Fuel_ox_tank_press_RCS	psi	**179**	**0**			**5**	0	0	7
11.1.9 Transport Main Engine Propellant	Propulsion_line_id	m	**0.0254**	**0**			**5**	0	0	7
11.1.5 Transport Main Engine Pressurant	Main_engine_pressurant_line_id	m	**0.015875**	**0**			**5**	0	0	7
11.2.9 Transport RCS Propellant	Propulsion_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	7
11.2.5 Transport RCS Pressurant	Pressurant_line_id_RCS	m	**0.015875**	**0**			**5**	0	0	7
11.1.6 Provide Main Engine	Main_engine_linear_slope	linear_slope kg/kN	**2.65**	**0**			**5**	0	0	9
11.1.6 Provide Main Engine	Main_engine_y_intercept	linear_curve_intercept - kg	**42.953**	**0**			**5**	0	0	9
11.1.12 Support Main Engine Components	Propulsion_secondary_struct_ratio	ratio	**0.2**	**0**			**5**	0	0	6
11.2.6 Provide RCS thrusters	RCS_side_A_B_choice	Integer, 1,2,3	**2**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Specific_impulse_RCS	Isp (seconds)	**240**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Vacuum_thrust_RCS	N (newtons)	**444.822**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_linear_slope	linear_slope kg/kN	**6.48**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	RCS_engine_y_intercept	linear_curve_intercept - kg	**0**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Num_RCS_engines_system	Integer, 4,8,16	**8**	**0**			**5**	0	0	9
11.2.6 Provide RCS thrusters	Num_fire_RCS_thrust_ascent	Num firing during ascent	**0.57**	**0**			**5**	0	0	9
11.2.11 Provide RCS contingency propellant	RCS_additional_propellant_mass	Rendezvous -kg	**200**	**3**			**4**	2	0	3
11.1.13 Insulate Main Engine Propellant	Propulsion_tanks_insulation	Use same as TPS			**1**	**1**	**4**	0	0	4
11.2.13 Insulate RCS Propellant	Propulsion_tanks_insulation_RCS	Placeholder			**1**	**1**	**4**	0	0	4
11.2.1 Provide RCS Propellant	RCS_propellant	Placeholder			**1**	**1**	**5**	0	0	3
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_filter	kg			**2**	**1**	**4**	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_solenoid_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_pressure_regulator	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_check_valve	kg			**2**	**1**	**4**	1	0	7
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_explosive_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.2 Control ME Pressurant - Oxidizer	Oxidizer_pressurant_relief_valve	kg			**2**	**1**	**4**	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_trim_orifice	kg			**2**	**1**	**5**	0	0	5
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_filter	kg			**2**	**1**	**4**	1	0	7
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_isolation_valve	kg			**2**	**1**	**5**	0	0	9
11.1.10.2 Control Main Engine Oxidizer	Oxidizer_bipropellant_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_filter	kg			**2**	**1**	**4**	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_solenoid_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_pressure_regulator	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_check_valve	kg			**2**	**1**	**4**	1	0	7
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_explosive_valve	kg			**2**	**1**	**5**	0	0	9
11.1.4.1 Control ME Pressurant - Fuel	Fuel_pressurant_relief_valve	kg			**2**	**1**	**4**	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_trim_orifice	kg			**2**	**1**	**5**	0	0	5
11.1.10.1 Control Main Engine Fuel	Fuel_filter	kg			**2**	**1**	**4**	1	0	7
11.1.10.1 Control Main Engine Fuel	Fuel_isolation_valve	kg			**2**	**1**	**5**	0	0	9
11.1.10.1 Control Main Engine Fuel	Fuel_bipropellant_valve	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_valve_RCS	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	System_pressurant_filter_RCS	kg			**2**	**1**	**4**	1	0	7
11.2.4 Control RCS Pressurant	System_pressurant_orifice_RCS	kg			**2**	**1**	**5**	0	0	5
11.2.4 Control RCS Pressurant	System_pressurant_pressure_regulator_R	kg			**2**	**1**	**5**	0	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_check_valve_RCS	kg			**2**	**1**	**4**	1	0	9
11.2.4 Control RCS Pressurant	Oxidizer_pressurant_relief_valve_RCS	kg			**2**	**1**	**4**	1	0	6
11.2.4 Control RCS Pressurant	Fuel_pressurant_check_valve_RCS	kg			**2**	**1**	**4**	1	0	7
11.2.4 Control RCS Pressurant	Fuel_pressurant_relief_valve_RCS	kg			**2**	**1**	**4**	1	0	7
11.2.10.2 Control RCS Oxidizer	Oxidizer_shutoff_valve_RCS	kg			**2**	**1**	**5**	0	0	9
11.2.10.1 Control RCS Fuel	Fuel_shutoff_valve_RCS	kg			**2**	**1**	**5**	0	0	9
11.2.9.1 Filter RCS Propellant	Fuel_Ox_filter_RCS	kg			**2**	**1**	**4**	1	0	7
11.2.10 Control RCS Propellant	Fuel_Ox_isolation_valve_RCS	kg			**2**	**1**	**5**	0	0	9
11.2.14.2 Heat RCS Oxidizer lines	Oxidizer_heater_RCS	kg			**2**	**1**	**4**	1	0	6
11.2.14.1 Heat RCS Fuel lines	Fuel_heater_RCS	kg			**2**	**1**	**4**	1	0	6
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_valve	kg			**2**	**1**	**4**	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Oxidizer_filter	kg			**2**	**1**	**4**	1	0	7
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_valve	kg			**2**	**1**	**4**	1	0	9
11.1.14 Control Main Engine Bypass	APS_interconnect_Fuel_filter	kg			**2**	**1**	**4**	1	0	7

Table G.7: Avionics technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Abort_control	0.91	5	0.000	0.000	0.000	4
Abort_guidance	14.74	5	0.603	0.203	0.133	200
Abort_input	3.40	5	0.140	0.152	0.132	20
Abort_navigation	9.38	5	0.130	0.229	0.292	145
Computer_keyboard	7.71	5	0.203	0.203	0.178	25
Control_propulsion	10.00	5	0.000	0.000	0.000	100
Control_propulsion_thrusters	0.00	5	0.000	0.000	0.000	0
Crew_displays_control	7.44	5	0.000	0.000	0.000	0
Crew_displays_navigation	15.79	5	0.000	0.000	0.000	0
Crew_displays_subsystems	5.35	5	0.000	0.000	0.000	0
Data_bus_network_boxes	15.99	5	0.203	0.133	0.607	15
Data_storage	1.13	5	0.052	0.102	0.158	10
Flight_control_computer	4.53	5	0.152	0.318	0.610	250
Flight_roll_pitch_yaw	2.16	5	0.000	0.000	0.000	0
Flight_translation	2.38	5	0.000	0.000	0.000	0
Health_monitoring_computer	8.30	5	0.178	0.171	0.298	100
IMU	9.55	5	0.000	0.000	0.000	150
Instrumentation_comm	1.27	5	0.084	0.224	0.284	50
Instrumentation_sensors_CA	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_CDH	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Comm	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_ECLSS	2.68	5	0.000	0.000	0.000	1
Instrumentation_sensors_Flight	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Health	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Payload	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Power	0.00	5	0.000	0.000	0.000	0
Instrumentation_sensors_Propulsion	2.45	5	0.000	0.000	0.000	1
Instrumentation_sensors_RCS	2.22	5	0.000	0.000	0.000	1
Instrumentation_sensors_Thermal	0.00	5	0.000	0.000	0.000	0
Interior_voice_comm	0.00	5	0.000	0.000	0.000	0
Long_range_amplifier	4.24	5	0.000	0.000	0.000	36
Long_range_antenna	0.18	5	0.000	0.000	0.000	1
Long_range_data_processor	4.72	5	0.000	0.000	0.000	27
Long_range_steerable_antenna	12.47	5	0.000	0.000	0.000	300
Long_range_transceiver	4.54	5	0.000	0.000	0.000	18
Master_event_controllers	10.25	5	0.171	0.130	0.502	11
Misc_avionics	0.00	5	0.000	0.000	0.000	0
Multiplexer_demultiplexers	7.94	5	0.140	0.288	0.508	50
Nav_analog_digital	6.80	5	0.064	0.279	0.330	50
Nav_base	1.36	5	0.287	0.287	0.287	0
Nav_power	9.07	5	0.067	0.225	0.597	100
Ordeal	3.13	5	0.000	0.000	0.000	8
Rendezvous_radar	35.43	5	0.000	0.000	0.000	200
Short_range_antenna	1.09	5	0.000	0.000	0.000	1
Short_range_data_processor	1.18	5	0.000	0.000	0.000	5
Short_range_transceiver	2.97	5	0.000	0.000	0.000	32
Star_tracker	10.48	5	0.000	0.000	0.000	0

Table G.8: EVA technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
EVA_mobility	0.00	5	0.000	0.000	0.000	0
Spacesuits	33.00	5	0.000	0.000	0.000	0
Suit_umbilical	0.00	5	0.000	0.000	0.000	0

Table G.9: Payloads technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Science_return	115.62	5	0.000	0.000	0.000	0
Consumable_equipment	0.00	5	0.000	0.000	0.000	0
General_return	0.00	5	0.000	0.000	0.000	0
Photography	19.00	5	0.437	0.437	0.437	0
Tools_equipment	1.67	5	0.149	0.149	0.149	0

Table G.10: Power technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Batteries	100.00	5	0.000	0.000	0.000	1000
Power_Distr_CircBreak	17.89	5	0.000	0.000	0.000	0
Power_Distr_Controller	9.52	5	0.000	0.000	0.000	0
Power_Distr_Wiring	47.14	5	0.000	0.000	0.000	0
Power_inverters	13.96	5	0.000	0.000	0.000	0
Power_relays	9.12	5	0.000	0.000	0.000	0

Table G.11: Thermal technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Coldplates	5.50	5	0.000	0.000	0.000	0
Exterior_radiators	0.00	5	0.000	0.000	0.000	0
Interior_coolant_lines_valves	0.90	5	0.000	0.000	0.000	0
Interior_coolant_pumps	1.88	5	0.000	0.000	0.000	0

Table G.12: Crew Accommodations technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
CA_handholds	5.00	5	0.100	0.100	0.100	0
CA_restraints	5.00	5	0.100	0.100	0.100	0
Cabin_lighting	2.00	5	0.100	0.100	0.100	100
Clothing	4.60	5	0.100	0.100	0.100	0
Entertainment	0.00	5	0.000	0.000	0.000	0
Exercise	0.00	5	0.000	0.000	0.000	0
Exterior_lighting	10.00	5	0.000	0.000	0.000	300
Medical_kit	1.00	5	0.100	0.100	0.100	0
Operational_supplies	5.00	5	0.100	0.100	0.100	0
Panel_lighting	1.00	5	0.100	0.100	0.100	100
Personal_storage	1.66	5	0.100	0.100	0.100	0
Sleep_accommodations	0.00	5	0.000	0.000	0.000	0

Table G.13: ECLSS technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
Air_Pressure_gauge	0.25	5	0.000	0.000	0.000	1
Cabin_air_control	2.22	5	0.006	0.006	0.006	16
Cabin_fan	9.30	5	0.002	0.002	0.002	100
Condense_air_heat_exchanger	13.52	5	0.045	0.045	0.045	0
Control_odor	0.00	5	0.000	0.000	0.000	0
Diapers	0.00	5	0.000	0.000	0.000	0
Dump_Valve_Piping	3.00	5	0.005	0.005	0.005	0
Fecal_bags	0.00	5	0.000	0.000	0.000	0
Fire_extinguisher	0.00	5	0.000	0.000	0.000	0
Food_rate	2.30	5	0.005	0.005	0.005	0
Food_storage	0.00	5	0.000	0.000	0.000	0
Galley_supplies	0.50	5	0.001	0.001	0.001	0
Heat_food	0.00	5	0.000	0.000	0.000	0
Housecleaning_supplies	0.00	5	0.000	0.000	0.000	0
Humidity_capture	2.55	5	0.004	0.004	0.004	0
Hygiene_water	4.10	5	0.000	0.000	0.000	0
Mix_air_valve	2.30	5	0.008	0.008	0.008	0
Nitrogen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Nitrogen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Oxygen_pp_gauge	0.00	5	0.000	0.000	0.000	0
Oxygen_tank_regulator_valve	0.49	5	0.000	0.000	0.000	0
Oxygen_tank_relief_valve	0.49	5	0.000	0.000	0.000	0
Particulates_filter	2.50	5	0.000	0.000	0.000	0
Potable_water	2.00	5	0.000	0.000	0.000	0
Temperature_sensor	0.08	5	0.000	0.000	0.000	0
Trace_Contaminants_filter	2.50	5	0.000	0.000	0.000	0
Trash_bags	0.00	5	0.000	0.000	0.000	0
Urine_bags	0.23	5	0.000	0.000	0.000	0
Vacuum	0.00	5	0.000	0.000	0.000	0
WCS_supplies	0.05	5	0.001	0.001	0.001	0

Table G.14: Propulsion technology database.

Equipment	Mass (kg)	Mass Error (+/- ,%)	X_length (m)	Y_length (m)	Z_length (m)	Power (W)
APS_interconnect_Fuel_filter	2.28	5	0.050	0.050	0.050	0
APS_interconnect_Fuel_valve	2.28	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_filter	2.28	5	0.050	0.050	0.050	0
APS_interconnect_Oxidizer_valve	2.28	5	0.050	0.050	0.050	0
Fuel_bipropellant_valve	2.10	5	0.050	0.050	0.050	0
Fuel_filter	2.10	5	0.050	0.050	0.050	0
Fuel_heater_RCS	1.79	5	0.050	0.050	0.050	0
Fuel_isolation_valve	2.10	5	0.050	0.050	0.050	0
Fuel_Ox_filter_RCS	0.33	5	0.050	0.050	0.050	0
Fuel_Ox_isolation_valve_RCS	0.33	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_check_valve_RCS	1.53	5	0.050	0.050	0.050	0
Fuel_pressurant_explosive_valve	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_filter	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_pressure_regulator	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_relief_valve_RCS	1.53	5	0.050	0.050	0.050	0
Fuel_pressurant_solenoid_valve	2.49	5	0.050	0.050	0.050	0
Fuel_pressurant_valve	2.49	5	0.050	0.050	0.050	0
Fuel_shutoff_valve_RCS	1.79	5	0.050	0.050	0.050	0
Fuel_trim_orifice	1.05	5	0.050	0.050	0.050	0
Oxidizer_bipropellant_valve	2.10	5	0.050	0.050	0.050	0
Oxidizer_filter	2.10	5	0.050	0.050	0.050	0
Oxidizer_heater_RCS	1.79	5	0.050	0.050	0.050	0
Oxidizer_isolation_valve	2.10	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_check_valve_RCS	1.53	5	0.050	0.050	0.050	0
Oxidizer_pressurant_explosive_valve	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_filter	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_pressure_regulator	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_relief_valve_RCS	1.53	5	0.050	0.050	0.050	0
Oxidizer_pressurant_solenoid_valve	2.49	5	0.050	0.050	0.050	0
Oxidizer_pressurant_valve	2.49	5	0.050	0.050	0.050	0
Oxidizer_shutoff_valve_RCS	1.79	5	0.050	0.050	0.050	0
Oxidizer_trim_orifice	1.05	5	0.050	0.050	0.050	0
System_pressurant_filter_RCS	1.53	5	0.050	0.050	0.050	0
System_pressurant_orifice_RCS	1.53	5	0.050	0.050	0.050	0
System_pressurant_pressure_regulator_RCS	1.53	5	0.050	0.050	0.050	0
System_pressurant_valve_RCS	1.53	5	0.050	0.050	0.050	0

APPENDIX H

SUBSYSTEM MASS TABLES

Table H.1: Apollo subsystem mass at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	554	0	330	2	0	138	99	64	180	463	324	356	2372	270	5152	0	0
2	FoS	Minimum Time	554	0	330	2	0	138	99	65	180	664	330	410	2584	269	5626	474	0
3	LOC 1FT	Minimum Time	944	0	330	2	0	363	162	83	326	788	356	577	3660	307	7899	2747	0
4	LOM 1FT	Minimum Time	1086	0	330	2	0	415	301	94	399	834	359	601	4120	307	8848	3696	0
5	Safety Comp	Minimum Time	1176	146	330	65	4	529	361	117	504	891	366	742	5350	536	11117	5965	0
6	All Comp 1FT	Minimum Time	1226	146	330	68	4	564	411	145	538	916	367	754	5576	536	11582	6430	0
7	LOC 2FT	Minimum Time	1643	146	330	68	4	956	481	164	727	1050	385	865	6850	539	14204	9052	0
8	LOM 2FT	Minimum Time	1789	146	330	68	4	1032	769	175	836	1107	387	902	7576	539	15660	10508	0
9	Safety Comp	Minimum Time	1882	291	330	131	7	1181	855	198	955	1178	403	1001	8398	540	17349	12197	0
10	All Comp 2FT	Minimum Time	1933	291	330	133	7	1232	930	226	998	1210	404	1015	8680	540	17930	12778	0
11	Min Funct	Nominal Time	554	0	330	2	0	270	146	81	221	480	325	365	2605	270	5648	0	522
12	FoS	Nominal Time	554	0	330	2	0	270	146	83	222	683	332	423	2826	269	6140	474	538
13	LOC 1FT	Nominal Time	944	0	330	2	0	565	237	101	388	811	357	596	4015	307	8654	2747	780
14	LOM 1FT	Nominal Time	1086	0	330	2	0	653	388	117	472	857	360	623	4537	307	9733	3696	910
15	Safety Comp	Nominal Time	1176	146	330	65	4	820	466	140	591	941	368	771	5905	536	12260	5965	1168
16	All Comp 1FT	Nominal Time	1226	146	330	68	4	879	523	173	634	968	370	785	6176	537	12817	6430	1260
17	LOC 2FT	Nominal Time	1643	146	330	68	4	1341	620	192	843	1107	387	902	7574	539	15695	9052	1515
18	LOM 2FT	Nominal Time	1789	146	330	68	4	1454	921	207	964	1167	390	943	8368	539	17289	10508	1654
19	Safety Comp	Nominal Time	1882	291	330	131	7	1655	1025	231	1096	1241	405	1046	9277	541	19156	12197	1832
20	All Comp 2FT	Nominal Time	1933	291	330	133	7	1730	1106	264	1148	1273	407	1062	9604	541	19830	12778	1924
21	Min Funct	Operability	569	0	330	13	297	272	146	106	295	524	344	382	3052	271	6600	0	1473
22	FoS	Operability	569	0	330	13	297	272	146	108	295	760	351	449	3313	270	7172	474	1571
23	LOC 1FT	Operability	962	0	330	13	297	567	237	126	462	875	368	620	4485	308	9651	2747	1777
24	LOM 1FT	Operability	1104	0	330	13	297	656	433	142	555	925	371	650	5060	309	10844	3696	2021
25	Safety Comp	Operability	1194	146	330	76	300	822	511	165	675	1013	379	800	6467	538	13417	5965	2325
26	All Comp 1FT	Operability	1244	146	330	78	300	882	523	198	708	1014	380	810	6657	538	13807	6430	2250
27	LOC 2FT	Operability	1662	146	330	78	300	1344	620	216	917	1158	405	928	8071	540	16717	9052	2538
28	LOM 2FT	Operability	1808	146	330	78	300	1457	922	232	1038	1220	408	969	8869	541	18317	10508	2683
29	Safety Comp	Operability	1901	291	330	142	304	1658	1025	255	1171	1296	416	1071	9772	542	20173	12197	2849
30	All Comp 2FT	Operability	1953	291	330	144	304	1733	1106	288	1222	1329	417	1088	10100	542	20849	12778	2943
31	Min Funct	2x Operability	585	0	330	25	594	423	200	170	419	617	362	411	3813	272	8219	0	3092
32	FoS	2x Operability	585	0	330	25	594	423	200	174	420	852	368	489	4089	272	8817	474	3216
33	LOC 1FT	2x Operability	979	0	330	25	594	798	322	192	611	1004	386	670	5423	310	11643	2747	3769
34	LOM 1FT	2x Operability	1121	0	330	25	594	927	532	212	716	1060	389	703	6070	311	12990	3696	4167
35	Safety Comp	2x Operability	1211	146	330	89	597	1152	629	236	852	1130	405	860	7628	540	15805	5965	4713
36	All Comp 1FT	2x Operability	1261	146	330	91	597	1239	649	274	894	1160	406	874	7898	540	16360	6430	4802
37	LOC 2FT	2x Operability	1682	146	330	91	597	1781	777	293	1127	1319	424	999	9454	543	19563	9052	5384
38	LOM 2FT	2x Operability	1828	146	330	91	597	1935	1093	313	1260	1387	427	1044	10333	543	21326	10508	5692
39	Safety Comp	2x Operability	1921	291	330	155	601	2194	1216	336	1410	1472	435	1152	11341	544	23397	12197	6073
40	All Comp 2FT	2x Operability	1972	291	330	157	601	2297	1305	375	1469	1509	437	1171	11722	545	24180	12778	6274

Table H.2: Apollo subsystem mass fractions at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	11%	0%	6%	0%	0%	3%	2%	1%	3%	9%	6%	7%	46%	5%	100%	0%	0%
2	FoS	Minimum Time	10%	0%	6%	0%	0%	2%	2%	1%	3%	12%	6%	7%	46%	5%	100%	8%	0%
3	LOC 1FT	Minimum Time	12%	0%	4%	0%	0%	5%	2%	1%	4%	10%	5%	7%	46%	4%	100%	35%	0%
4	LOM 1FT	Minimum Time	12%	0%	4%	0%	0%	5%	3%	1%	5%	9%	4%	7%	47%	3%	100%	42%	0%
5	Safety Comp	Minimum Time	11%	1%	3%	1%	0%	5%	3%	1%	5%	8%	3%	7%	48%	5%	100%	54%	0%
6	All Comp 1FT	Minimum Time	11%	1%	3%	1%	0%	5%	4%	1%	5%	8%	3%	7%	48%	5%	100%	56%	0%
7	LOC 2FT	Minimum Time	12%	1%	2%	0%	0%	7%	3%	1%	5%	7%	3%	6%	48%	4%	100%	64%	0%
8	LOM 2FT	Minimum Time	11%	1%	2%	0%	0%	7%	5%	1%	5%	7%	2%	6%	48%	3%	100%	67%	0%
9	Safety Comp	Minimum Time	11%	2%	2%	1%	0%	7%	5%	1%	6%	7%	2%	6%	48%	3%	100%	70%	0%
10	All Comp 2FT	Minimum Time	11%	2%	2%	1%	0%	7%	5%	1%	6%	7%	2%	6%	48%	3%	100%	71%	0%
11	Min Funct	Nominal Time	10%	0%	6%	0%	0%	5%	3%	1%	4%	8%	6%	6%	46%	5%	100%	0%	9%
12	FoS	Nominal Time	9%	0%	5%	0%	0%	4%	2%	1%	4%	11%	5%	7%	46%	4%	100%	8%	9%
13	LOC 1FT	Nominal Time	11%	0%	4%	0%	0%	7%	3%	1%	4%	9%	4%	7%	46%	4%	100%	32%	9%
14	LOM 1FT	Nominal Time	11%	0%	3%	0%	0%	7%	4%	1%	5%	9%	4%	6%	47%	3%	100%	38%	9%
15	Safety Comp	Nominal Time	10%	1%	3%	1%	0%	7%	4%	1%	5%	8%	3%	6%	48%	4%	100%	49%	10%
16	All Comp 1FT	Nominal Time	10%	1%	3%	1%	0%	7%	4%	1%	5%	8%	3%	6%	48%	4%	100%	50%	10%
17	LOC 2FT	Nominal Time	10%	1%	2%	0%	0%	9%	4%	1%	5%	7%	2%	6%	48%	3%	100%	58%	10%
18	LOM 2FT	Nominal Time	10%	1%	2%	0%	0%	8%	5%	1%	6%	7%	2%	5%	48%	3%	100%	61%	10%
19	Safety Comp	Nominal Time	10%	2%	2%	1%	0%	9%	5%	1%	6%	6%	2%	5%	48%	3%	100%	64%	10%
20	All Comp 2FT	Nominal Time	10%	1%	2%	1%	0%	9%	6%	1%	6%	6%	2%	5%	48%	3%	100%	64%	10%
21	Min Funct	Operability	9%	0%	5%	0%	4%	4%	2%	2%	4%	8%	5%	6%	46%	4%	100%	0%	22%
22	FoS	Operability	8%	0%	5%	0%	4%	4%	2%	2%	4%	11%	5%	6%	46%	4%	100%	7%	22%
23	LOC 1FT	Operability	10%	0%	3%	0%	3%	6%	2%	1%	5%	9%	4%	6%	46%	3%	100%	28%	18%
24	LOM 1FT	Operability	10%	0%	3%	0%	3%	6%	4%	1%	5%	9%	3%	6%	47%	3%	100%	34%	19%
25	Safety Comp	Operability	9%	1%	2%	1%	2%	6%	4%	1%	5%	8%	3%	6%	48%	4%	100%	44%	17%
26	All Comp 1FT	Operability	9%	1%	2%	1%	2%	6%	4%	1%	5%	7%	3%	6%	48%	4%	100%	47%	16%
27	LOC 2FT	Operability	10%	1%	2%	0%	2%	8%	4%	1%	5%	7%	2%	6%	48%	3%	100%	54%	15%
28	LOM 2FT	Operability	10%	1%	2%	0%	2%	8%	5%	1%	6%	7%	2%	5%	48%	3%	100%	57%	15%
29	Safety Comp	Operability	9%	1%	2%	1%	2%	8%	5%	1%	6%	6%	2%	5%	48%	3%	100%	60%	14%
30	All Comp 2FT	Operability	9%	1%	2%	1%	1%	8%	5%	1%	6%	6%	2%	5%	48%	3%	100%	61%	14%
31	Min Funct	2x Operability	7%	0%	4%	0%	7%	5%	2%	2%	5%	8%	4%	5%	46%	3%	100%	0%	38%
32	FoS	2x Operability	7%	0%	4%	0%	7%	5%	2%	2%	5%	10%	4%	6%	46%	3%	100%	5%	36%
33	LOC 1FT	2x Operability	8%	0%	3%	0%	5%	7%	3%	2%	5%	9%	3%	6%	47%	3%	100%	24%	32%
34	LOM 1FT	2x Operability	9%	0%	3%	0%	5%	7%	4%	2%	6%	8%	3%	5%	47%	2%	100%	28%	32%
35	Safety Comp	2x Operability	8%	1%	2%	1%	4%	7%	4%	1%	5%	7%	3%	5%	48%	3%	100%	38%	30%
36	All Comp 1FT	2x Operability	8%	1%	2%	1%	4%	8%	4%	2%	5%	7%	2%	5%	48%	3%	100%	39%	29%
37	LOC 2FT	2x Operability	9%	1%	2%	0%	3%	9%	4%	1%	6%	7%	2%	5%	48%	3%	100%	46%	28%
38	LOM 2FT	2x Operability	9%	1%	2%	0%	3%	9%	5%	1%	6%	7%	2%	5%	48%	3%	100%	49%	27%
39	Safety Comp	2x Operability	8%	1%	1%	1%	3%	9%	5%	1%	6%	6%	2%	5%	48%	2%	100%	52%	26%
40	All Comp 2FT	2x Operability	8%	1%	1%	1%	2%	9%	5%	2%	6%	6%	2%	5%	48%	2%	100%	53%	26%

Table H.3: Apollo One Man subsystem mass at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	472	0	165	2	0	124	89	60	157	326	185	332	1813	263	3989	0	0
2	FoS	Minimum Time	472	0	165	2	0	124	89	60	157	463	191	376	1971	263	4334	345	0
3	LOC 1FT	Minimum Time	831	0	165	2	0	337	128	78	289	556	205	535	2920	296	6341	2352	0
4	LOM 1FT	Minimum Time	966	0	165	2	0	389	275	89	362	606	208	559	3385	296	7303	3314	0
5	Safety Comp	Minimum Time	1055	73	165	42	4	503	331	110	445	649	219	692	4442	526	9256	5267	0
6	All Comp 1FT	Minimum Time	1055	73	165	44	4	513	333	137	453	650	219	694	4491	526	9356	5367	0
7	LOC 2FT	Minimum Time	1435	73	165	44	4	881	438	155	636	772	233	803	5727	529	11895	7906	0
8	LOM 2FT	Minimum Time	1573	73	165	44	4	957	702	166	739	816	235	838	6397	530	13238	9249	0
9	Safety Comp	Minimum Time	1665	146	165	84	7	1105	782	187	835	871	241	928	7067	531	14613	10624	0
10	All Comp 2FT	Minimum Time	1665	146	165	86	7	1119	783	215	844	894	242	932	7146	531	14775	10786	0
11	Min Funct	Nominal Time	472	0	165	2	0	174	107	64	172	338	186	336	1905	263	4185	0	198
12	FoS	Nominal Time	472	0	165	2	0	174	107	64	172	463	191	380	2053	263	4508	345	176
13	LOC 1FT	Nominal Time	831	0	165	2	0	418	159	82	313	572	206	543	3067	296	6653	2352	314
14	LOM 1FT	Nominal Time	966	0	165	2	0	486	311	94	391	606	208	568	3543	296	7636	3314	335
15	Safety Comp	Nominal Time	1055	73	165	42	4	623	375	115	480	668	220	704	4665	526	9715	5267	461
16	All Comp 1FT	Nominal Time	1055	73	165	44	4	635	377	144	490	668	220	706	4720	527	9827	5367	472
17	LOC 2FT	Nominal Time	1435	73	165	44	4	1034	495	162	682	794	234	818	6011	529	12478	7906	585
18	LOM 2FT	Nominal Time	1573	73	165	44	4	1127	764	174	789	838	236	854	6709	530	13880	9249	644
19	Safety Comp	Nominal Time	1665	146	165	84	7	1298	852	195	892	894	242	946	7418	531	15333	10624	722
20	All Comp 2FT	Nominal Time	1665	146	165	86	7	1314	854	223	902	918	243	951	7503	531	15508	10786	735
21	Min Funct	Operability	487	0	165	8	169	175	108	80	216	357	200	347	2169	264	4744	0	757
22	FoS	Operability	487	0	165	8	169	175	108	81	216	503	200	395	2335	264	5106	345	774
23	LOC 1FT	Operability	848	0	165	8	169	419	159	98	357	620	220	559	3366	297	7286	2352	947
24	LOM 1FT	Operability	983	0	165	8	169	488	312	111	435	657	222	584	3845	298	8275	3314	974
25	Safety Comp	Operability	1072	73	165	47	173	625	375	131	524	703	227	719	4963	528	10327	5267	1073
26	All Comp 1FT	Operability	1072	73	165	49	173	637	377	160	534	723	228	723	5039	528	10482	5367	1127
27	LOC 2FT	Operability	1453	73	165	49	173	1037	495	178	726	833	242	834	6316	531	13106	7906	1213
28	LOM 2FT	Operability	1592	73	165	49	173	1129	764	190	834	903	245	871	7042	531	14562	9249	1327
29	Safety Comp	Operability	1683	146	165	89	177	1300	852	211	936	962	250	964	7754	532	16022	10624	1411
30	All Comp 2FT	Operability	1683	146	165	91	177	1317	854	240	947	962	250	967	7814	532	16146	10786	1372
31	Min Funct	2x Operability	502	0	165	13	339	241	132	106	280	404	209	362	2566	265	5584	0	1597
32	FoS	2x Operability	502	0	165	13	339	241	132	107	280	563	209	417	2749	265	5981	345	1650
33	LOC 1FT	2x Operability	864	0	165	13	339	525	198	125	434	672	228	584	3837	299	8283	2352	1944
34	LOM 1FT	2x Operability	1000	0	165	13	339	614	358	139	517	710	230	611	4353	300	9349	3314	2048
35	Safety Comp	2x Operability	1089	73	165	53	342	780	432	160	615	781	237	751	5574	530	11580	5267	2327
36	All Comp 1FT	2x Operability	1089	73	165	55	342	796	434	190	626	781	237	754	5635	530	11707	5367	2353
37	LOC 2FT	2x Operability	1472	73	165	55	342	1236	568	208	830	922	251	870	7014	532	14538	7906	2645
38	LOM 2FT	2x Operability	1610	73	165	55	342	1349	844	221	944	971	253	908	7755	533	16024	9249	2789
39	Safety Comp	2x Operability	1702	146	165	95	346	1549	942	242	1054	1033	259	1003	8517	534	17587	10624	2976
40	All Comp 2FT	2x Operability	1702	146	165	97	346	1569	944	272	1066	1034	259	1007	8584	534	17725	10786	2952

Table H.4: Apollo One Man subsystem mass fractions at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	12%	0%	4%	0%	0%	3%	2%	2%	4%	8%	5%	8%	45%	7%	100%	0%	0%
2	FoS	Minimum Time	11%	0%	4%	0%	0%	3%	2%	1%	4%	11%	4%	9%	45%	6%	100%	8%	0%
3	LOC 1FT	Minimum Time	13%	0%	3%	0%	0%	5%	2%	1%	5%	9%	3%	8%	46%	5%	100%	37%	0%
4	LOM 1FT	Minimum Time	13%	0%	2%	0%	0%	5%	4%	1%	5%	8%	3%	8%	46%	4%	100%	45%	0%
5	Safety Comp	Minimum Time	11%	1%	2%	0%	0%	5%	4%	1%	5%	7%	2%	7%	48%	6%	100%	57%	0%
6	All Comp 1FT	Minimum Time	11%	1%	2%	0%	0%	5%	4%	1%	5%	7%	2%	7%	48%	6%	100%	57%	0%
7	LOC 2FT	Minimum Time	12%	1%	1%	0%	0%	7%	4%	1%	5%	6%	2%	7%	48%	4%	100%	66%	0%
8	LOM 2FT	Minimum Time	12%	1%	1%	0%	0%	7%	5%	1%	6%	6%	2%	6%	48%	4%	100%	70%	0%
9	Safety Comp	Minimum Time	11%	1%	1%	1%	0%	8%	5%	1%	6%	6%	2%	6%	48%	4%	100%	73%	0%
10	All Comp 2FT	Minimum Time	11%	1%	1%	1%	0%	8%	5%	1%	6%	6%	2%	6%	48%	4%	100%	73%	0%
11	Min Funct	Nominal Time	11%	0%	4%	0%	0%	4%	3%	2%	4%	8%	4%	8%	46%	6%	100%	0%	5%
12	FoS	Nominal Time	10%	0%	4%	0%	0%	4%	2%	1%	4%	10%	4%	8%	46%	6%	100%	8%	4%
13	LOC 1FT	Nominal Time	12%	0%	2%	0%	0%	6%	2%	1%	5%	9%	3%	8%	46%	4%	100%	35%	5%
14	LOM 1FT	Nominal Time	13%	0%	2%	0%	0%	6%	4%	1%	5%	8%	3%	7%	46%	4%	100%	43%	4%
15	Safety Comp	Nominal Time	11%	1%	2%	0%	0%	6%	4%	1%	5%	7%	2%	7%	48%	5%	100%	54%	5%
16	All Comp 1FT	Nominal Time	11%	1%	2%	0%	0%	6%	4%	1%	5%	7%	2%	7%	48%	5%	100%	55%	5%
17	LOC 2FT	Nominal Time	11%	1%	1%	0%	0%	8%	4%	1%	5%	6%	2%	7%	48%	4%	100%	63%	5%
18	LOM 2FT	Nominal Time	11%	1%	1%	0%	0%	8%	6%	1%	6%	6%	2%	6%	48%	4%	100%	67%	5%
19	Safety Comp	Nominal Time	11%	1%	1%	1%	0%	8%	6%	1%	6%	6%	2%	6%	48%	3%	100%	69%	5%
20	All Comp 2FT	Nominal Time	11%	1%	1%	1%	0%	8%	6%	1%	6%	6%	2%	6%	48%	3%	100%	70%	5%
21	Min Funct	Operability	10%	0%	3%	0%	4%	4%	2%	2%	5%	8%	4%	7%	46%	6%	100%	0%	16%
22	FoS	Operability	10%	0%	3%	0%	3%	3%	2%	2%	4%	10%	4%	8%	46%	5%	100%	7%	15%
23	LOC 1FT	Operability	12%	0%	2%	0%	2%	6%	2%	1%	5%	9%	3%	8%	46%	4%	100%	32%	13%
24	LOM 1FT	Operability	12%	0%	2%	0%	2%	6%	4%	1%	5%	8%	3%	7%	46%	4%	100%	40%	12%
25	Safety Comp	Operability	10%	1%	2%	0%	2%	6%	4%	1%	5%	7%	2%	7%	48%	5%	100%	51%	10%
26	All Comp 1FT	Operability	10%	1%	2%	0%	2%	6%	4%	2%	5%	7%	2%	7%	48%	5%	100%	51%	11%
27	LOC 2FT	Operability	11%	1%	1%	0%	1%	8%	4%	1%	6%	6%	2%	6%	48%	4%	100%	60%	9%
28	LOM 2FT	Operability	11%	0%	1%	0%	1%	8%	5%	1%	6%	6%	2%	6%	48%	4%	100%	64%	9%
29	Safety Comp	Operability	11%	1%	1%	1%	1%	8%	5%	1%	6%	6%	2%	6%	48%	3%	100%	66%	9%
30	All Comp 2FT	Operability	10%	1%	1%	1%	1%	8%	5%	1%	6%	6%	2%	6%	48%	3%	100%	67%	8%
31	Min Funct	2x Operability	9%	0%	3%	0%	6%	4%	2%	2%	5%	7%	4%	6%	46%	5%	100%	0%	29%
32	FoS	2x Operability	8%	0%	3%	0%	6%	4%	2%	2%	5%	9%	3%	7%	46%	4%	100%	6%	28%
33	LOC 1FT	2x Operability	10%	0%	2%	0%	4%	6%	2%	2%	5%	8%	3%	7%	46%	4%	100%	28%	23%
34	LOM 1FT	2x Operability	11%	0%	2%	0%	4%	7%	4%	1%	6%	8%	2%	7%	47%	3%	100%	35%	22%
35	Safety Comp	2x Operability	9%	1%	1%	0%	3%	7%	4%	1%	5%	7%	2%	6%	48%	5%	100%	45%	20%
36	All Comp 1FT	2x Operability	9%	1%	1%	0%	3%	7%	4%	2%	5%	7%	2%	6%	48%	5%	100%	46%	20%
37	LOC 2FT	2x Operability	10%	1%	1%	0%	2%	9%	4%	1%	6%	6%	2%	6%	48%	4%	100%	54%	18%
38	LOM 2FT	2x Operability	10%	0%	1%	0%	2%	8%	5%	1%	6%	6%	2%	6%	48%	3%	100%	58%	17%
39	Safety Comp	2x Operability	10%	1%	1%	1%	2%	9%	5%	1%	6%	6%	1%	6%	48%	3%	100%	60%	17%
40	All Comp 2FT	2x Operability	10%	1%	1%	1%	2%	9%	5%	2%	6%	6%	1%	6%	48%	3%	100%	61%	17%

Table H.5: ESAS subsystem mass at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	464	0	617	29	0	344	259	184	276	1359	980	654	4505	269	9939	0	0
2	FoS	Minimum Time	464	0	617	29	0	344	259	184	276	1923	1003	816	5136	269	11320	1381	0
3	LOC 1FT	Minimum Time	776	0	617	29	0	818	389	205	478	2045	1011	1150	6495	307	14320	4381	0
4	LOM 1FT	Minimum Time	924	0	617	29	0	905	786	233	620	2127	1015	1241	7490	308	16295	6356	0
5	Safety Comp	Minimum Time	1013	0	617	29	0	1055	942	235	706	2215	1020	1726	10251	809	20617	10678	0
6	All Comp 1FT	Minimum Time	1068	0	617	57	0	1262	1097	315	819	2259	1022	1793	10998	809	22115	12176	0
7	LOC 2FT	Minimum Time	1423	0	617	57	0	2080	1333	336	1127	2450	1032	2114	13245	809	26624	16685	0
8	LOM 2FT	Minimum Time	1577	0	617	57	0	2205	2048	363	1347	2547	1036	2266	14960	809	29833	19894	0
9	Safety Comp	Minimum Time	1671	0	617	57	0	2404	2274	366	1459	2599	1039	2516	15897	809	31709	21770	0
10	All Comp 2FT	Minimum Time	1729	0	617	86	0	2701	2404	446	1587	2651	911	2579	16603	809	33123	23184	0
11	Min Funct	Nominal Time	464	0	617	29	0	419	274	190	296	1359	980	661	4610	269	10169	0	251
12	FoS	Nominal Time	464	0	617	29	0	419	274	191	297	1923	1003	826	5244	269	11556	1381	257
13	LOC 1FT	Nominal Time	776	0	617	29	0	923	412	212	507	2086	1013	1167	6684	307	14733	4381	433
14	LOM 1FT	Nominal Time	924	0	617	29	0	1023	812	241	653	2170	1017	1261	7701	308	16754	6356	481
15	Safety Comp	Nominal Time	1013	0	617	29	0	1191	973	243	743	2215	1020	1746	10482	809	21080	10678	485
16	All Comp 1FT	Nominal Time	1068	0	617	57	0	1429	1133	325	865	2304	1024	1823	11331	809	22783	12176	689
17	LOC 2FT	Nominal Time	1423	0	617	57	0	2277	1376	346	1181	2450	1032	2143	13577	809	27288	16685	686
18	LOM 2FT	Nominal Time	1577	0	617	57	0	2415	2094	375	1405	2597	1038	2303	15373	809	30661	19894	848
19	Safety Comp	Nominal Time	1671	0	617	57	0	2632	2325	377	1522	2650	911	2543	16198	809	32313	21770	624
20	All Comp 2FT	Nominal Time	1729	0	617	86	0	2961	2460	459	1658	2703	913	2622	17098	809	34115	23184	1012
21	Min Funct	Operability	483	291	617	180	300	422	275	200	464	1463	988	732	5560	269	12245	0	2327
22	FoS	Operability	483	291	617	180	300	422	275	201	464	2045	1011	918	6227	269	13704	1381	2405
23	LOC 1FT	Operability	799	291	617	180	300	928	412	222	675	2175	1019	1256	7646	307	16828	4381	2528
24	LOM 1FT	Operability	946	291	617	180	300	1028	813	252	821	2262	1023	1350	8667	308	18857	6356	2584
25	Safety Comp	Operability	1036	291	617	180	300	1196	973	254	911	2309	1026	1849	11628	809	23379	10678	2783
26	All Comp 1FT	Operability	1090	291	617	208	300	1434	1133	335	1033	2402	1030	1925	12480	809	25089	12176	2995
27	LOC 2FT	Operability	1449	291	617	208	300	2284	1377	356	1350	2553	1038	2246	14739	809	29620	16685	3018
28	LOM 2FT	Operability	1603	291	617	208	300	2423	2095	385	1574	2654	912	2389	16345	809	32607	19894	2794
29	Safety Comp	Operability	1697	291	617	208	300	2639	2326	388	1691	2708	915	2641	17313	810	34544	21770	2856
30	All Comp 2FT	Operability	1754	291	617	237	300	2968	2556	469	1848	2815	918	2737	18402	810	36724	23184	3622
31	Min Funct	2x Operability	502	582	617	331	601	563	303	231	671	1576	996	818	6721	269	14781	0	4863
32	FoS	2x Operability	502	582	617	331	601	563	303	234	671	2174	1019	1030	7426	269	16322	1381	5023
33	LOC 1FT	2x Operability	821	582	617	331	601	1124	454	255	898	2312	1028	1376	8937	308	19643	4381	5343
34	LOM 1FT	2x Operability	968	582	617	331	601	1248	860	287	1051	2404	1031	1472	10001	308	21762	6356	5489
35	Safety Comp	2x Operability	1058	582	617	331	601	1447	1029	289	1150	2454	1034	1995	13263	809	26660	10678	6064
36	All Comp 1FT	2x Operability	1112	582	617	359	601	1743	1198	373	1286	2503	1036	2075	14157	809	28453	12176	6359
37	LOC 2FT	2x Operability	1475	582	617	359	601	2648	1455	394	1619	2711	916	2397	16435	810	33020	16685	6418
38	LOM 2FT	2x Operability	1628	582	617	359	601	2810	2179	426	1850	2818	919	2556	18237	810	36393	19894	6581
39	Safety Comp	2x Operability	1722	582	617	359	601	3058	2418	428	1976	2875	922	2813	19260	810	38443	21770	6754
40	All Comp 2FT	2x Operability	1780	582	617	388	601	3445	2657	513	2148	2931	924	2912	20384	810	40691	23184	7589

Table H.6: ESAS subsystem mass fractions at node configurations.

Node Number	Safety Level	Operability Level	Avionics (lbm)	EVA (lbm)	Crew (lbm)	Crew Accommodations (lbm)	Payloads (lbm)	Power (lbm)	Thermal (lbm)	ECLSS (lbm)	Secondary Structures (lbm)	Primary Structures (lbm)	TPS/MMOD (lbm)	Propulsion (lbm)	Main Engine Propellant (lbm)	RCS Propellant (lbm)	Total Spacecraft Mass (lbm)	Safety Mass	Operability Mass
1	Min Funct	Minimum Time	5%	0%	6%	0%	0%	3%	3%	2%	3%	14%	10%	7%	45%	3%	100%	0%	0%
2	FoS	Minimum Time	4%	0%	5%	0%	0%	3%	2%	2%	2%	17%	9%	7%	45%	2%	100%	12%	0%
3	LOC 1FT	Minimum Time	5%	0%	4%	0%	0%	6%	3%	1%	3%	14%	7%	8%	45%	2%	100%	31%	0%
4	LOM 1FT	Minimum Time	6%	0%	4%	0%	0%	6%	5%	1%	4%	13%	6%	8%	46%	2%	100%	39%	0%
5	Safety Comp	Minimum Time	5%	0%	3%	0%	0%	5%	5%	1%	3%	11%	5%	8%	50%	4%	100%	52%	0%
6	All Comp 1FT	Minimum Time	5%	0%	3%	0%	0%	6%	5%	1%	4%	10%	5%	8%	50%	4%	100%	55%	0%
7	LOC 2FT	Minimum Time	5%	0%	2%	0%	0%	8%	5%	1%	4%	9%	4%	8%	50%	3%	100%	63%	0%
8	LOM 2FT	Minimum Time	5%	0%	2%	0%	0%	7%	7%	1%	5%	9%	3%	8%	50%	3%	100%	67%	0%
9	Safety Comp	Minimum Time	5%	0%	2%	0%	0%	8%	7%	1%	5%	8%	3%	8%	50%	3%	100%	69%	0%
10	All Comp 2FT	Minimum Time	5%	0%	2%	0%	0%	8%	7%	1%	5%	8%	3%	8%	50%	2%	100%	70%	0%
11	Min Funct	Nominal Time	5%	0%	6%	0%	0%	4%	3%	2%	3%	13%	10%	7%	45%	3%	100%	0%	2%
12	FoS	Nominal Time	4%	0%	5%	0%	0%	4%	2%	2%	3%	17%	9%	7%	45%	2%	100%	12%	2%
13	LOC 1FT	Nominal Time	5%	0%	4%	0%	0%	6%	3%	1%	3%	14%	7%	8%	45%	2%	100%	30%	3%
14	LOM 1FT	Nominal Time	6%	0%	4%	0%	0%	6%	5%	1%	4%	13%	6%	8%	46%	2%	100%	38%	3%
15	Safety Comp	Nominal Time	5%	0%	3%	0%	0%	6%	5%	1%	4%	11%	5%	8%	50%	4%	100%	51%	2%
16	All Comp 1FT	Nominal Time	5%	0%	3%	0%	0%	6%	5%	1%	4%	10%	4%	8%	50%	4%	100%	53%	3%
17	LOC 2FT	Nominal Time	5%	0%	2%	0%	0%	8%	5%	1%	4%	9%	4%	8%	50%	3%	100%	61%	3%
18	LOM 2FT	Nominal Time	5%	0%	2%	0%	0%	8%	7%	1%	5%	8%	3%	8%	50%	3%	100%	65%	3%
19	Safety Comp	Nominal Time	5%	0%	2%	0%	0%	8%	7%	1%	5%	8%	3%	8%	50%	3%	100%	67%	2%
20	All Comp 2FT	Nominal Time	5%	0%	2%	0%	0%	9%	7%	1%	5%	8%	3%	8%	50%	2%	100%	68%	3%
21	Min Funct	Operability	4%	2%	5%	1%	2%	3%	2%	2%	4%	12%	8%	6%	45%	2%	100%	0%	19%
22	FoS	Operability	4%	2%	5%	1%	2%	3%	2%	1%	3%	15%	7%	7%	45%	2%	100%	10%	18%
23	LOC 1FT	Operability	5%	2%	4%	1%	2%	6%	2%	1%	4%	13%	6%	7%	45%	2%	100%	26%	15%
24	LOM 1FT	Operability	5%	2%	3%	1%	2%	5%	4%	1%	4%	12%	5%	7%	46%	2%	100%	34%	14%
25	Safety Comp	Operability	4%	1%	3%	1%	1%	5%	4%	1%	4%	10%	4%	8%	50%	3%	100%	46%	12%
26	All Comp 1FT	Operability	4%	1%	2%	1%	1%	6%	5%	1%	4%	10%	4%	8%	50%	3%	100%	49%	12%
27	LOC 2FT	Operability	5%	1%	2%	1%	1%	8%	5%	1%	5%	9%	4%	8%	50%	3%	100%	56%	10%
28	LOM 2FT	Operability	5%	1%	2%	1%	1%	7%	6%	1%	5%	8%	3%	7%	50%	2%	100%	61%	9%
29	Safety Comp	Operability	5%	1%	2%	1%	1%	8%	7%	1%	5%	8%	3%	8%	50%	2%	100%	63%	8%
30	All Comp 2FT	Operability	5%	1%	2%	1%	1%	8%	7%	1%	5%	8%	3%	7%	50%	2%	100%	63%	10%
31	Min Funct	2x Operability	3%	4%	4%	2%	4%	4%	2%	2%	5%	11%	7%	6%	45%	2%	100%	0%	33%
32	FoS	2x Operability	3%	4%	4%	2%	4%	3%	2%	1%	4%	13%	6%	6%	45%	2%	100%	8%	31%
33	LOC 1FT	2x Operability	4%	3%	3%	2%	3%	6%	2%	1%	5%	12%	5%	7%	46%	2%	100%	22%	27%
34	LOM 1FT	2x Operability	4%	3%	3%	2%	3%	6%	4%	1%	5%	11%	5%	7%	46%	1%	100%	29%	25%
35	Safety Comp	2x Operability	4%	2%	2%	1%	2%	5%	4%	1%	4%	9%	4%	7%	50%	3%	100%	40%	23%
36	All Comp 1FT	2x Operability	4%	2%	2%	1%	2%	6%	4%	1%	5%	9%	4%	7%	50%	3%	100%	43%	22%
37	LOC 2FT	2x Operability	4%	2%	2%	1%	2%	8%	4%	1%	5%	8%	3%	7%	50%	2%	100%	51%	19%
38	LOM 2FT	2x Operability	4%	2%	2%	1%	2%	8%	6%	1%	5%	8%	3%	7%	50%	2%	100%	55%	18%
39	Safety Comp	2x Operability	4%	2%	2%	1%	2%	8%	6%	1%	5%	7%	2%	7%	50%	2%	100%	57%	18%
40	All Comp 2FT	2x Operability	4%	1%	2%	1%	1%	8%	7%	1%	5%	7%	2%	7%	50%	2%	100%	57%	19%