Assessing the Impacts of Fractionation on Pointing-Intensive Spacecraft

by

Michael Gregory O'Neill

S.B., Aerospace Engineering, Syracuse University, 2007

Submitted to the Department of Aeronautics and Astronautics
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Master of Science

in AERONAUTICS AND ASTRONAUTICS
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Abstract

Fractionated spacecraft consist of physically independent, "free-flying" modules composed of various subsystems. Thus, a fractionated spacecraft might consist of one-module responsible for the power generation and storage, another module responsible for the communications and computing, another module responsible for the attitude and guidance determination, another module responsible for the payload, and so on. Fractionated spacecraft are of particular interest for pointing-intensive, remote sensing mission spacecraft because of their ability to physically decouple subsystems and payloads that truly need precise pointing, thereby potentially reducing the lifecycle cost of fractionated spacecraft relative to a comparable monolithic spacecraft, for a given space mission. Additionally, using fractionation to decouple pointing-intensive subsystems and payloads may potentially reduce the mass and size of the module containing the payload in a fractionated spacecraft (*i.e.,* Payload Module) relative to that of a comparable monolithic spacecraft. If fractionated spacecraft prove to reduce the mass and size associated with the Payload Module, for a given pointing-intensive, remote sensing mission, it may enable pointing-intensive fractionated spacecraft to have longer space mission lifetimes than comparable monolithic spacecraft.

This research seeks to quantitatively assess the impacts of various fractionated spacecraft architecture strategies on the lifecycle cost, mass, propellant usage, and mission lifetime of pointing-intensive, remote sensing mission spacecraft. A dynamic lifecycle simulation and parametric model was used to assess the lifecycle cost impacts, while the mass, propellant usage, and mission lifetime impacts were assessed using a non-parametric, physics-based computer model. Results from the research demonstrate that fractionated spacecraft can be both more and less expensive than a comparable monolithic spacecraft performing the same space mission. Additionally, the results show that due to the ability of fractionated spacecraft to decouple subsystems and payloads that truly need precise pointing, the mass and propellant usage of the Payload Module can be appreciably less than that of a comparable monolithic spacecraft. Subsequently, fractionated spacecraft can attain longer mission lifetimes than a monolithic spacecraft, and in certain instances, do so with a lesser lifecycle cost than the monolith at its respective shorter mission lifetime.

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Title: Assistant Professor of Aeronautics and Astronautics

Biographical Note

Greg O'Neill grew up in Lewisburg, Pennsylvania and graduated from Lewisburg Area High School in June of 2003. Thereafter he began his engineering studies at Syracuse University in the Department of Mechanical and Aerospace Engineering. In May of 2007, he graduated from Syracuse University with a Bachelors of Science in Aerospace Engineering and a minor in Mathematics and Economics. Then in September of 2007, he began his graduate studies at Massachusetts Institute of Technology. At MIT, he has a socio-technical interdisciplinary education and research focus, primarily drawing from the fields of Aerospace Engineering, Systems Engineering, Space Systems Design (Architecture), Mathematics, and Economics. He is particularly interested in the notion of value-centric design and its implications for assessing innovative spacecraft architectures. After completion of his Master's degree at MIT in September of 2009, he intends on pursuing his doctorate at MIT in an area related to his current research interests.

Following the completion of his collegiate education, Greg plans to teach abroad for two years through a nonprofit organization and then upon his return to the United States, pursue a research-intensive, professional career in industry or academia.

Acknowledgements

The completion of a Master of Science degree at MIT was never part of the "plan" I had for my education and professional career when I began my engineering studies six years ago. In fact, my confidence level was low when I began studying engineering and I honestly never thought I would never make it past the first semester. I mention this because upon examining my educational achievements to date, it emphasizes the true significance of all those who have helped me throughout my education. Thankfully, due to the influence of these individuals, over time my educational "plan" changed radically, for the better, and I began to make the transition from an unsure engineering student to a highly motivated and successful engineering student who truly loved his education. Eventually, I knew that part of the "plan" meant attending graduate school and that this would form the necessary foundation for my continual pursuit of knowledge. It is for these reasons that I am deeply appreciative of those who have contributed to my growth as a person and as a student, ensuring that my educational dreams are continually realized. Each of these individuals has, in their own unique way, fostered my education and helped me become a more confident person and student – something I simply could not have done on my own.

I would like to offer my gratitude to a few individuals who have had a significant impact on my education and life. First, I would like to thank my academic advisor, **Prof. Annalisa L. Weigel**. She provided me with my first "home" at MIT and has since been supportive of all of my academic endeavors. Throughout my time at MIT, she has continually provided me with opportunities to excel in my field and has served as source of unwavering guidance and enlightenment for my education and research; and for this, I will forever be grateful.

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Finally, I would like to extend my gratitude and love to the most important individuals in my life: family and friends; specifically, **Shannon Lack, Katie Lack, Robert Lack, Isabella O'Neill, Michael Byerly, Jeremy Quant, Ryan Hollm, and Christopher North**. Although I have had a rather tumultuous life, my family and friends have been supportive of all my life and academic endeavors. Without my family and friends, I could not have risen above those moments in my life when the hardships seemed insurmountable. I am deeply appreciative and loving of my family and friends for helping me become a better person and perpetually strengthening in me the notion that in life I must work hard to achieve great things, because great things… well, would not be so great without all the hard work.

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AFRL	Air Force Research Laboratory
AFOSR	Air Force Office of Scientific Research
AC	Alternating Current
AIAA	American Institute of Aeronautics and Astronautics
ASAT	Anti-Satellite Attack
Arch	Spacecraft Architecture
as	Arcsecond
ANDE	Atmospheric Neutral Density Experiment
ACS	Attitude Control System
ADS	Attitude Determination System
BoL	Beginning-of-Life
BTS	Bernoulli Trial Sequence
C&DH	Command and Data Handling
Comm	Communications
CS	Computer System
CONOPS	Concept of Operations
COCOMOII	COnstructive COst MOdel: Second Edition
CERs	Cost Estimating Relationships
CMU	Cost Model Uncertainty
cdf	Cumulative Distribution Function
DARPA	Defense Advanced Research Projects Agency
DSM	Design Structure Matrix
DC	Direct Current
EoL	End-of-Life
EPS	Electrical Power System
EM	Electromagnetic
ESA	European Space Agency
ExEP	Exoplanet Exploration Program
FCal	Fence Calibration
FY2008$M	Fiscal Year 2008 United States Dollars (in millions)
F6	Future, Fast, Flexible, Fractionated, Free-Flying
GINA	Generalized Information Network Analysis
GOES	Geostationary Operational Environmental Satellites
GEO	Geosynchronous Orbit
GCS	Guidance Control System
GNS	Guidance Navigation System
HIL	Hardware-in-the-Loop
IMU	Inertial Measurement Unit
Infra	Infrastructure
ICE	Integrated Concurrent Engineering
IA&T	Integration, Assembly, and Testing
LBTI	Large Binocular Telescope Interferometer
LV	Launch Vehicle
LC	Lifecycle

LCC	Lifecycle Cost
LEO	Low Earth Orbit
MIT	Massachusetts Institute of Technology
MoE	Measures of Effectiveness (MoE)
mas	Milli-arcsecond
MCA	Monte Carlo Analysis (Simulation)
MCAU	Monte Carlo Analysis Uncertainty
MMD	Multimodal Distribution
MATE	Multiple (Multi-) Attribute Tradespace Exploration
MATE-CON	Multiple (Multi-) Attribute Tradespace Exploration with Concurrent Design
MAUT	Multiple (Multi-) Attribute Utility Theory
NASA	National Aeronautics and Space Administration
NOAA	National Oceanic & Atmospheric Administration
NRL	Naval Research Laboratory
NRE	Nonrecurring (Costs)
ND	Normal Distribution
Ops	Operations
PL	Payload
PI	Pointing-Intensive
pdf	Probability Density Function
PoIM	Probability of Infant Mortality
RW	Reaction Wheel
RE	Recurring (Costs)
RSM	Remote Sensing Mission
R&D	Research and Development
SCAMP	Secondary Camera and Maneuvering Platform
S/C	Spacecraft
SET	Spacecraft Evaluation Tool (SET)
SIM	Space Interferometer Mission
SK	Stationkeeping
SLCC	Stochastic Lifecycle Cost
SPHERES	Synchronize Position Hold Engage & Reorient Experimental Satellites
TPF-C	Terrestrial Planet Finder – Coronagraph
TPF-I	Terrestrial Planet Finder – Interferometer
TOS	Terrestrial Observer Swarm
TCS	Thermal Control System
TT&C	Tracking, Telemetry, and Control
UV	Ultraviolet
USD	United States Dollars
UNP	University Nanosatellite Program
USCM8	Unmanned Space Vehicle Cost Model, 8th Edition
VP	Value Proposition
V&V	Verification & Validation
VPS	Visual Positioning System
WPD	Wireless Power Distribution

1. Research Methodology

The research methodology serves as the foundation for the development and progression of this research. Subsequently, there were objectives set forth specifically regarding the research methodology, which ensured that each stage of the research development and progression was a demonstration of scholarly achievement. The four the research methodology objectives are:

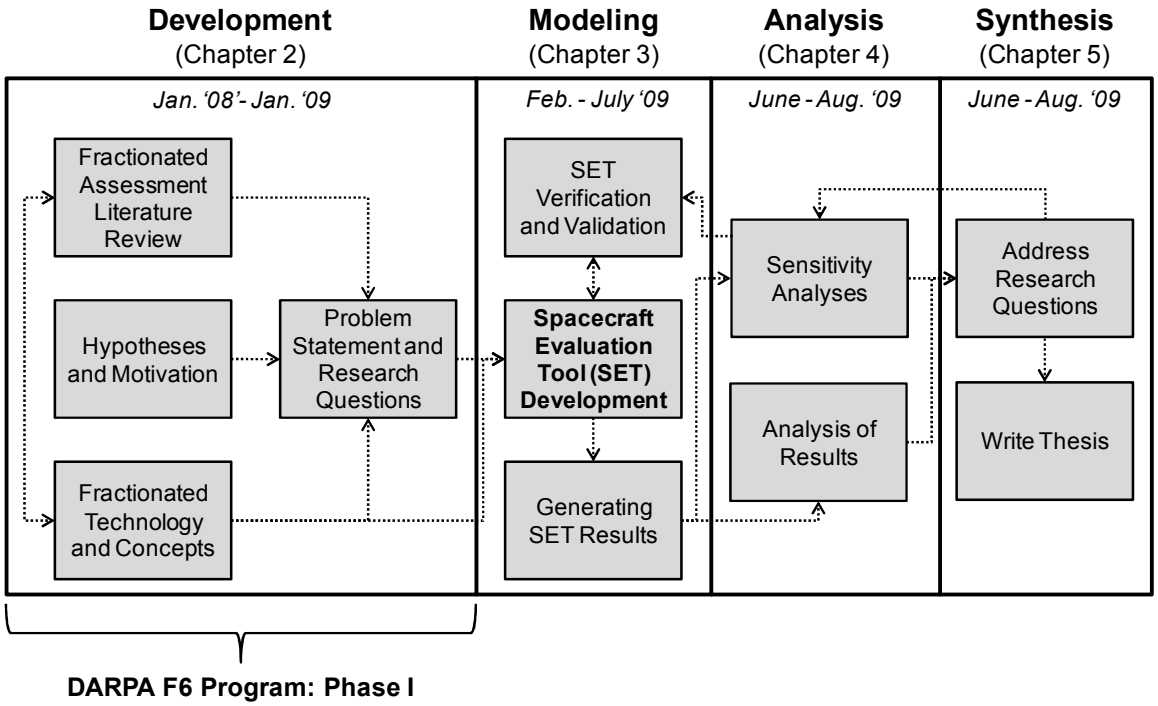
The research methodology must...

1. Formulate comprehensive, appropriate, and quantitative responses to the research questions (see Section 2.5).
2. Engender the unique contributions of this research (see Section 2.4).
3. Produce meaningful results reliably and repeatedly.
4. Be readily extendable to demonstrate broad applicability of the methodology to problems (*i*.e., questions) that were beyond the original scope of this research.

1.1. Research Methodology Overview

A conceptual overview of the research methodology is given in Figure 1-1. The research methodology was developed in January of 2008 and the subsequent execution of all research methodology constituents (shown as blocks in Figure 1-1) was completed by August of 2009. There are four phases of the research methodology, broadly categorized as development, modeling, analysis, and synthesis. Each of these phases has several constituents, where each constituent prescribes specific tasks to be completed that further mature the research. The dates assigned to each phase of the research methodology represent the time period in which the *majority* of the efforts put forth to address that phase took place; however, recognize that many aspects of this research were conducted concurrently.

Figure 1-1. The research methodology.



Conveniently, each of the four phases of the research methodology is the focus of one chapter.

- **Phase I: Development –** Chapter 2
- **Phase II: Modeling –** Chapter 3
- **Phase III: Analysis** – Chapter 4
- **Phase IV: Synthesis** – Chapter 5

1.1.1. Phase I: Development

The efforts put forth in addressing and developing the constituents of Phase I: Development, took place from January 2008 to January 2009. Chapter 2 discusses the outcomes from Phase I. The primary objectives of Phase I were to provide a foundation for understanding fractionated spacecraft as well as identify areas in which knowledge can be meaningfully contributed with regard to understanding the value propositions of fractionated spacecraft. As such, the first constituent of Phase I was the Fractionated Technology and Concept investigation (see Section 2.1). In this constituent, investigations of key concepts and technology related to fractionated spacecraft as well as demonstrations of fractionated spacecraft/technology were conducted. The second constituent of Phase I was the Hypotheses and Motivation development (see Section 2.2). In this constituent, important positive and negative hypotheses about fractionated spacecraft were researched and documented to serve as a source of motivation for the research. The third constituent of Phase I was performing a Fractionated Assessment Literature Review (see Section 2.3). The objective of this constituent was to gain an understanding of previous assessments of fractionated spacecraft. And the fourth and last constituent of Phase I was the formulation of the Problem Statement and Research Questions (see Sections 2.4 and 2.5). In this constituent, the synthesis of the other three constituents in Phase I were used to develop a problem statement and research questions; these being necessary for the modeling phase, that is, Phase II of the research methodology.

The execution of constituents in Phase I of the research methodology was done concurrently with working on Phase I of the DARPA System F6 Program (see Section 2.3.2). In Phase I of the F6 program, the author was part of the Northrop Grumman Space Technology Group and had the responsibility of performing value-centric assessments of monolithic and fractionated spacecraft. This responsibility specifically involved modeling monolithic and fractionated spacecraft and quantitatively assessing their costs and benefits relative to comparable monolithic spacecraft.

1.1.2. Phase II: Modeling

The majority of the efforts put forth in addressing and developing the constituents of Phase II: Modeling, took place from February 2009 to July 2009. Chapter 3 discusses the outcomes from Phase II. The primary objective of Phase II was to develop a computer-based tool for assessing monolithic and fractionated spacecraft value propositions. Subsequently, the first constituent of Phase II was the Spacecraft Evaluation Tool (SET) development (see Chapter 3). The SET is a Microsoft Excel® and Matlab® integrated software program that takes a set of inputs characterizing a particular problem (or research question) pertaining to a monolithic or fractionated spacecraft, and via a simulation, generates a set of outputs (*i.e.,* metrics to form the value proposition). Concurrently with the SET development, the second and third constituents of Phase II were performed: SET Verification and Validation and Generating SET Results. SET Verification and Validation (V&V) provided a means for ensuring appropriateness and accuracy of the SET relative to its respective inputs and outputs. Additionally, SET V&V occurred while generating the SET results and subsequently analyzing them while conducting Phase III of the research methodology, Analysis.

1.1.3. Phase III: Analysis

The majority of the efforts put forth in addressing and developing the constituents of Phase III: Analysis, took place from June 2009 to August 2009. Chapter 4 discusses the outcomes from Phase III. Through Phase III, the SET was applied to formulate quantitative monolithic and fractionated spacecraft value propositions. Based on the value propositions generated from the SET, Sensitivity Analyses and an Analysis of Results could commence (see Section 4.3 through 4.7), which are the two constituents of Phase III. The sensitivity analyses enabled further V&V of the SET as well as an understanding of the SET inputs of interest (*e.g.,* RSM payload ground resolution) on the SET outputs of interest (*e.g.,* lifecycle cost). Through the Analysis of Results constituent of Phase III, monolithic and fractionated spacecraft value propositions were quantified and used to formulate responses to the research questions. Subsequently, combining (synthesizing) all of these responses was the focus of Phase IV of the research methodology, Synthesis.

1.1.4. Phase IV: Synthesis

The majority of the efforts put forth in addressing and developing the constituents of Phase IV: Synthesis, took place from June 2009 to August 2009. Chapter 5 discusses the outcomes from Phase IV. The first constituent of Phase IV was Addressing the Research Questions (see Section 5.2), which involved synthesizing the large number of responses to the research questions generated from Phase III. Then following the formulation of succinct responses to the research questions, the second constituent of Phase IV: Write Thesis, commenced. The objective of this constituent of Phase IV was to document all outcomes of the research methodology phases and their respective constituents.

2. Problem Formulation

A research investigation into the impacts of fractionation on pointing-intensive spacecraft necessitates an understanding of the fundamental problem, which provides both context and motivation for the investigation. The problem is formulated along five successive dimensions: (1) relevant concepts and terminology, (2) motivation, (3) literature review, (4) problem statement and research contributions, and (5) research questions. The synthesis of these five successive dimensions ensures completeness of the problem formulation and subsequently embodies Phase I of the research methodology (see Section 1.1.1).

2.1. Relevant Concepts and Terminology

The first step in formulating the problem is to define and explore concepts and terminology that are essential to an understanding of this research investigation.

2.1.1. Spacecraft Performance

The performance of a spacecraft can be defined as an action (or lack thereof) that the spacecraft executes in the context of its mission that, in turn, provides value (or lack thereof) to at least one of the spacecraft's respective beneficiaries and/or beneficiary stakeholders. With regard to understanding the performance and value proposition of spacecraft, there are a few terms worth noting.

- Benefit: a service provided to an entity that is perceived as being advantageous or good.
- Value: benefit at cost, that is, benefit normalized by the cost of obtaining the benefit.
- Beneficiary: an individual, group of individuals, or organization that **does not expend resources** (*e.g.,* time, money, and regulations) for the development and/or operation of a system (*e.g.,* spacecraft), but **does benefit** from the system development and/or operation.
- Stakeholder: an individual, group of individuals, or organization that **expends resources** for the development and/or operation of a system, but **does not benefit** from the system development and/or operation.
- Beneficiary Stakeholder: an individual, group of individuals, or organization that **expends resources** for the development and/or operation of a system and **does benefit** from the system development and/or operation.

Performance can be thought of in both a static and dynamic (time dependent) sense. In a static context, performance is characterized by the instantaneous development and delivery of benefit/value, whereas in a dynamic context, performance is characterized by the accumulated development and delivery of benefit/value over a period of time. Common notions of a spacecraft performance include payload performance (*e.g.,* ground resolution) and mission lifetime. In terms of static performance, payload performance may be quantified as instantaneous ground resolution. Alternatively, in terms of dynamic performance, payload performance may be quantified as the average payload performance over the mission (this is particularly relevant for Earth observation spacecraft having highly elliptical orbits). However, in contrast to static and dynamic payload performance, mission lifetime (performance) is only a dynamic performance metric as it necessarily quantifies time. It should be noted that the term performance can also extend beyond traditional notions of performance, such as payload performance and mission lifetime, and used to describe specific subsystem functionality characteristics of spacecraft, for example, spacecraft use of shared resources (see Section 2.1.5).

Often spacecraft are compared on the basis of their performance and in the case when two or more spacecraft have an identical performance, it can be stated that they are *comparable*. If two or more spacecraft are comparable, this enables comparisons between the spacecraft to be made on an "equal" basis per their identical performance. Comparability is particularly important when assessing a large number of candidate spacecraft architectures to determine which is, for example, the least expensive in terms of lifecycle cost for a given level of performance.

2.1.2. Fractionation

Fractionation describes a *system* composed of physically independent (*i.e.,* structurally separate) *constituents* that can, but do not have to, collaborate to provide benefit/value to the beneficiaries and beneficiary stakeholders of that respective system. This definition for fractionation is based on a system's respective physical characteristics and functional relationships.

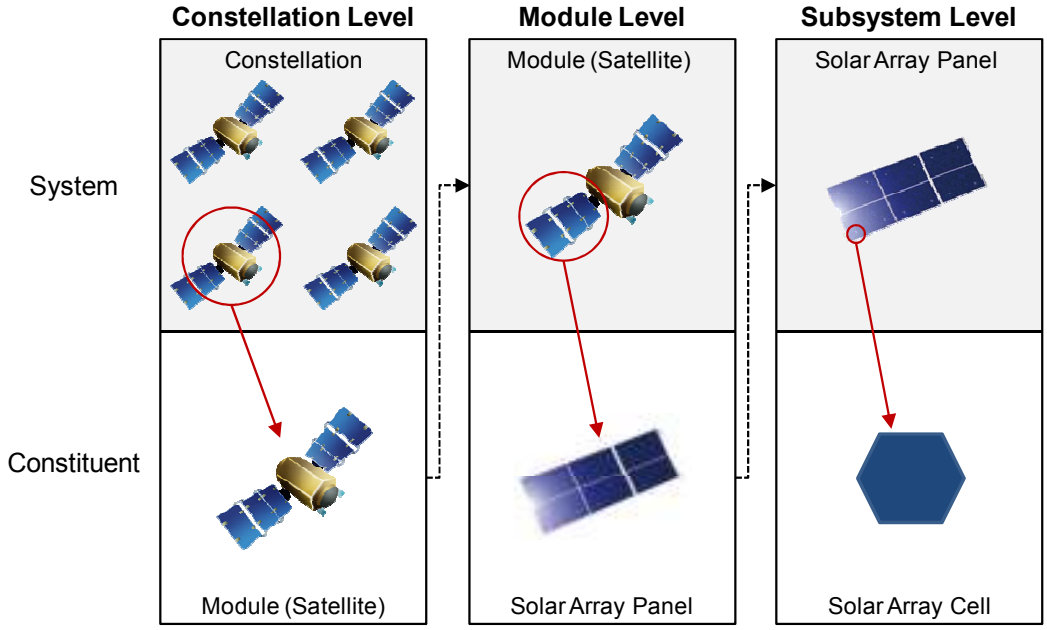
In literature, the terms modular and distributed often have the same connotation as fractionated as it is defined and employed hereafter. For the purposes of this research, distributed and fractionated are taken to have the same meaning, however, both of these terms differ from the term modular. A *fractionated (or distributed)* system has constituents are physically independent (*i.e.,* structurally separated) and that may or may not collaborate. A *modular* system has constituents that, in some capacity, can be designed, manufactured, integrated, tested, and/or assembled independently of one another (*e.g.,* concurrently) (Baldwin & Clark, 1997; Esper, 2005; Esper et al., 2004). A modular system therefore does not necessarily require that the system under consideration be fractionated (*i.e.,* have physically independent constituents), as many systems that are monolithic in nature can be modular as well (*e.g.,* laptop computers). However, it is worth acknowledging that fractionated systems are often perceived as being modular in nature. If a system is deemed to not be fractionated, thereby meaning that the system constituents are physically dependent (*i.e.,* a direct or indirect structural connection can be found between all constituents), the system is considered to be *monolithic.*

2.1.3. Fractionated Spacecraft

Designating a spacecraft as being fractionated or monolithic first requires a definition of the *system* under consideration so the *constituents* of the system can be identified. After the system is properly defined, the system's constituents can be examined to determine if the spacecraft is fractionated or monolithic based on the definition for fractionated systems provided in Section 2.1.2.

Due to the ambiguous nature of the term "system" and "constituent", this research suggests the adoption of three perspectives that provide a logical means for designating a spacecraft as being fractionated or monolithic based on its respective constituents. For a given spacecraft, each of the three perspectives prescribes a unique meaning as to the *system* and its respective *constituents* – which, as previously stated, are fundamental to determining whether the spacecraft is fractionated or monolithic. The first perspective is the "Constellation Level" perspective; here, the *system* is a spacecraft constellation and the *constituents* are each structurally connected grouping of subsystems in the constellation (aka satellites or *modules).* (The term module will be employed hereafter in place of the term satellite.) The second perspective is the "Module Level" perspective; here, the *system* is a single module in the constellation and the *constituents* are the individual subsystems present in that module. And the third perspective is the "Subsystem Level" perspective; here, the *system* is a given subsystem of a module and the *constituents* are the individual hardware components within that subsystem. Figure 2-1 notionally depicts these three perspectives.

Figure 2-1. The three perspectives for classifying a spacecraft as being fractionated or monolithic.



In Figure 2-1, if the Constellation Level perspective is considered, then the system is the constellation and the four constituents are each respective module (satellite) in the constellation. If the Module Level perspective is considered, then the system is a single module in the constellation, and the module subsystems are the constituents (in Figure 2-1 a solar array is a representative constituent). And if the Subsystem Level perspective is considered, then the system is a subsystem in a module and the constituents are the components of the subsystem (in Figure 2-1 a solar array cell is a representative constituent).

Given the definition for fractionated and monolithic spacecraft (*i.e.,* systems) provided in Section 2.1.2, the logic for determining whether a spacecraft is fractionated or monolithic is as follows:

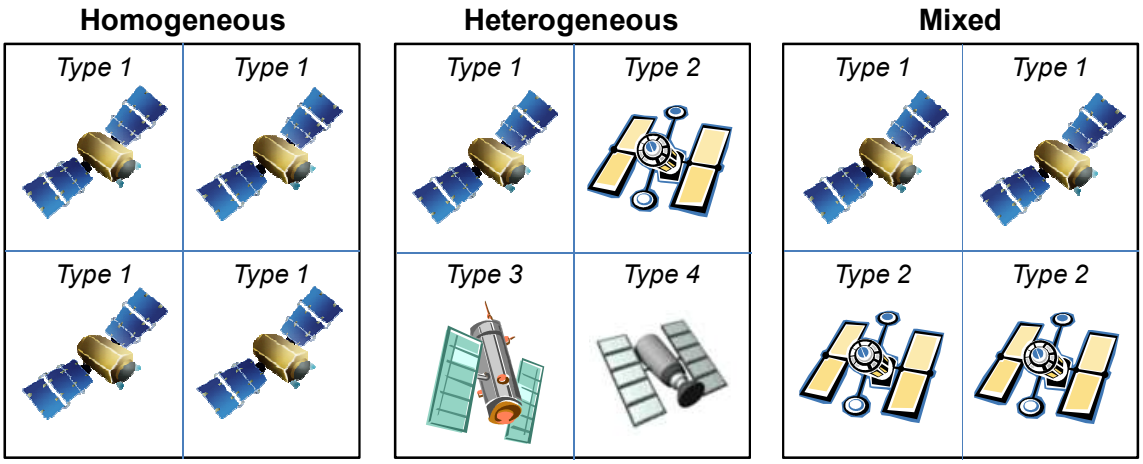
- If a spacecraft is *deemed* to be fractionated with regard to a*ny one of the three* perspectives, then it should be designated a *fractionated spacecraft*.
- If a spacecraft is *not deemed* to be fractionated with regard to *any one of the three* perspectives, then it should be designated a *monolithic spacecraft*.

Following the examination of a spacecraft with respect to each of the three perspectives, if the spacecraft is deemed to be fractionated, then it will belong to one of three classes of these respective spacecraft: homogeneous, heterogeneous, and mixed (*i.e.,* homogeneous & heterogeneous) (C. Mathieu & Weigel, 2005). Mixed fractionation is actually a subset of heterogeneous fractionation, however, there is value added in distinguishing between mixed and heterogeneous fractionation, as the term heterogeneous fails to convey strong traces homogeneity in a system. Classifying a fractionated spacecraft, from any one of the three perspectives (*i.e.,* Constellation, Module, Subsystem Level), is done as follows:

1. A fractionated spacecraft is *homogeneous* if <u>all</u> of the respective *constituents* of the *system* are identical in both form and function.
2. A fractionated spacecraft is *heterogeneous* if <u>not all</u> of the respective *constituents* of the *system* are identical in form and/or function.
3. A fractionated spacecraft is *mixed* if, in the *system*, there are <u>distinguishable groupings</u> of *constituents* that are homogeneous and heterogeneous.

Figure 2-2 provides instantiations of the three potential classifications of fractionated spacecraft from the Constellation Level perspective. In Figure 2-2, the *system* is the combination (*i.e.,* constellation) of all four modules where each module is considered a *constituent* that may be identical or different from the other *constituents* based on its respective shape, size, and color as shown in Figure 2-2.

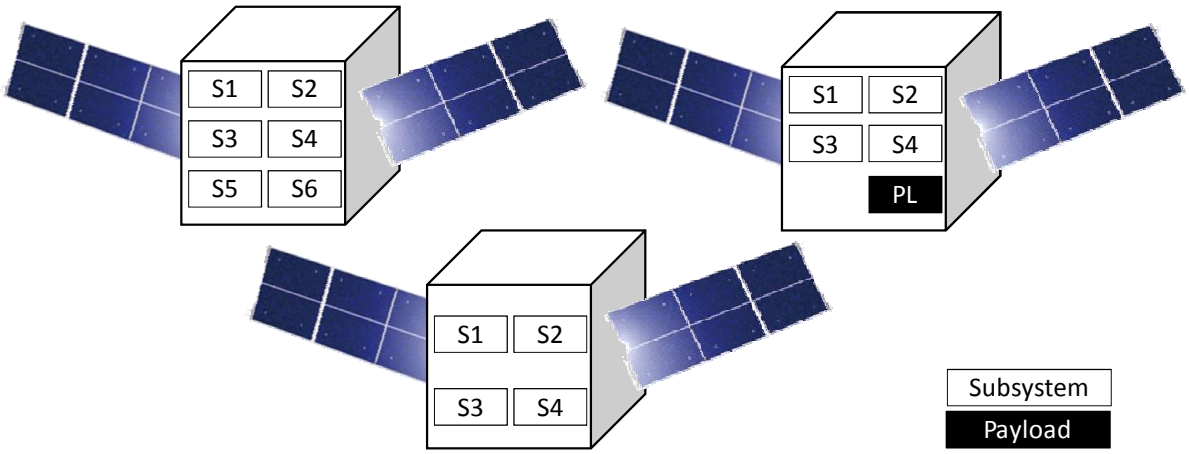
Figure 2-2. Homogenous, heterogeneous, and mixed fractionated spacecraft.



In Figure 2-2, a homogenous fractionated spacecraft contains four identical modules (satellites), whereas a heterogeneous fractionated spacecraft contains four different modules. Conversely, a mixed fractionated spacecraft contains two clear groupings of identical modules but that are subsequently different from one another. In the case of Figure 2-2, if the heterogeneous and mixed fractionated spacecraft were to both classified as heterogeneous, given that mixed fractionation is a subset of heterogeneous fractionation, constructive nuances between these two fractionated spacecraft would be lost.

To serve as an illustrative example in determining whether or not a spacecraft is fractionated from each of the three perspectives, and if so, which respective class of fractionation the spacecraft belongs to, consider the spacecraft represented in Figure 2-3.

Figure 2-3. A notional fractionated spacecraft.



The spacecraft shown in Figure 2-3 can be examined from each of the three perspectives to determine if it is fractionated and if so what respective class of spacecraft to which it belongs.

- **Constellation Level:** the *system* is the combination of all three modules in the constellation where each respective module is considered a *constituent*. The system is fractionated and heterogeneous. This is evident in that all the constituents are physically independent (*i.e.,* fractionated) but each module is different in form and functionality (*i.e.,* heterogeneous).
- **Module Level:** the *system* is any one of the three modules and the *constituents* are each of the respective subsystems in a given module. The system is not fractionated and is thereby monolithic. This is evident since each constituent is located inside the same structure/module (*i.e.,* physically dependent).
- **Subsystem Level:** the *system* is a given subsystem in one of the modules and the *constituents* (not shown in Figure 1-3) are the respective components of that subsystem. The system is not fractionated and is thereby monolithic. This is evident since each constituent is located inside the same subsystem (*i.e.,* physically dependent).

Based on this working example in determining whether the spacecraft in Figure 2-3 is fractionated or monolithic, since, from the Constellation Level perspective, the spacecraft in Figure 2-3 is deemed to be heterogeneous fractionated, the spacecraft should be designated as fractionated. Recall that it is only necessary to deem a spacecraft as begin fractionated from a minimum of *one of the three* perspectives for it to be considered a fractionated spacecraft.

Understandably, it may appear superfluous to examine a given spacecraft from the Constellation, Module, and Subsystem Level perspectives to determine whether the spacecraft is fractionated or monolithic. However, without the structured approach prescribed by these three perspectives (or for that matter any structured approach), there can be no consistency in the interpretation and subsequent assignment of the term fractionation (or lack thereof) to a given spacecraft. Therefore, adopting these perspectives offers one instantiation of maintaining a *consistent definition* of fractionated and monolithic spacecraft.

This research will investigate spacecraft that are perceived as being fractionated either the Constellation Level and/or Module Level. Additionally, from each of these two perspectives, fractionated spacecraft belonging to the homogeneous, heterogeneous, and mixed (*i.e.,* homogeneous & heterogeneous) classes of spacecraft will be investigated.

2.1.4. Spacecraft and Spacecraft Architecture

In the discussion of this research, specifically its respective outcomes in Chapter 4, the terminology spacecraft architecture are employed instead of the term spacecraft; for example, fractionated *spacecraft architecture* instead of fractionated *spacecraft*. It should be noted that the terminology *spacecraft architecture* are purposely used in discussions in which the intent is to emphasize the specific design (*i.e.,* structural hardware and subsystem composition) of spacecraft, as this is not as readily conveyed with the term *spacecraft*.

2.1.5. Shared Resources

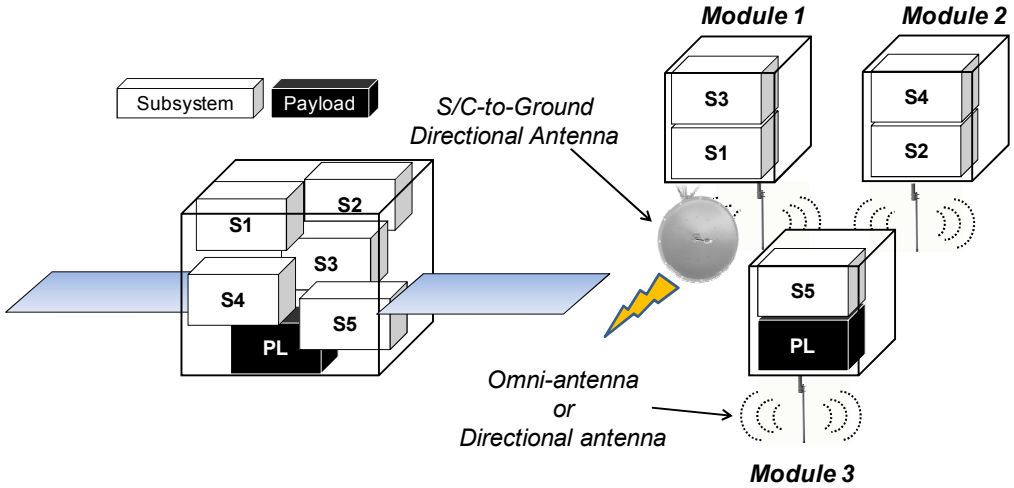
With regard to the Constellation Level perspective (see Section 2.1.3), fractionated spacecraft consist of physically independent, free-flying modules, each of which is composed of various "traditional" spacecraft subsystems. Therefore, an essential attribute of fractionated spacecraft is their ability to physically decouple (*i.e.,* separate) subsystems and payloads by placing them on different modules and, in doing so, enable the sharing of subsystem resources amongst modules via collaboration (*e.g.,* power, communications, data processing, attitude and guidance determination) (Brown & Eremenko, 2006a)[1]. Through the dispersion and subsequent sharing of certain subsystem resources, there is associated hardware required on the modules that provide shared resources (aka sources) as well as those modules that rely on/receive shared resources (aka recipients). The hardware associated with each shared resource may be simple instantiations of current technology, as is in the case when sharing the communications subsystem, or can require the application and demonstration of new(er) technologies, as is the case in sharing the power (generation and storage) subsystem.

It is hypothesized that the ability of fractionated spacecraft to share resources will specifically cause a reduction in their respective lifecycle cost to that of a comparable monolithic spacecraft, something of great value to spacecraft beneficiary stakeholders. This thereby provides motivation for this research and its assessment of the implications of sharing resources amongst modules in fractionated spacecraft on their respective value propositions (See Section 2.1.9). The shared resources specifically investigated through this research and subsequently enumerated hereafter are representative of potential subsystem resources that can be shared in fractionated spacecraft, given present spacecraft subsystem technology(ies).

- **Communications, Computer System, and Command & Data Handling (Comm_CS_C&DH)**

The Comm, CS, and C&DH subsystems can be shared amongst modules in a fractionated spacecraft (Brown & Eremenko, 2006a). Figure 2-4 provides a conceptual instantiation of sharing these three subsystem resources amongst modules in a fractionated spacecraft (a comparable monolith to the fractionated spacecraft is shown in the left of Figure 2-4).

Figure 2-4. Shared Resources: Comm_CS_C&DH.



[1] It should be noted that not all spacecraft subsystems are "fractionatable" (*i.e.,* they cannot be shared). Some subsystems, such as the thermal control system, must be present in all the respective modules of a fractionated spacecraft.

For a given fractionated spacecraft, in sharing the communications (Comm) subsystem resource, it is still necessary to have at least one module with a dedicated spacecraft-to-ground (S/C-ground) antenna for the uplink and downlink of mission and housekeeping data. However, it may not be necessary to have a S/C-ground antenna on every module, as this creates unwanted redundancy. Instead, as is depicted in Figure 2-4, omni- or small directional antennas can be used for inter-module communications to route data to and from *all* modules in the fractionated spacecraft. Then ultimately, any data needing to be sent to the ground can be routed to Module 1, since it has the dedicated S/C-ground antenna. Analogously, all of the data coming from the ground is sent to Module 1 and subsequently distributed by Module 1 to all other modules via omni- (or small directional) antennas. In sharing the Comm subsystem resource in this manner, a reduction in Comm redundancy is achieved, and this has the effect of reducing the Comm subsystem requirements for the modules without a dedicated S/C-ground antenna (*i.e.,* recipients).

Implicit to the fractionated spacecraft shown in Figure 2-4 is a shared computer system (CS) and command & data handling (C&DH) subsystem. The CS and C&DH can readily be shared amongst the modules in a fractionated spacecraft by having a non-uniform distribution of computing capabilities amongst the modules in a fractionated spacecraft. For example, the fractionated spacecraft in Figure 2-4 could have a high-performance computer system on Module 3 to process the payload (mission) data before it is transmitted to the ground via Module 1. Whereas, Modules 1 and 2 , due to the lack of a need to process payload data, can have much smaller, lower performance computer systems capable of only processing housekeeping, C&DH, and TT&C related data/tasks. As another example of sharing the CS and C&DH resource for the fractionated spacecraft shown in Figure 2-4, the computer onboard Module 3 could process the payload data as well as the housekeeping, C&DH, and/or TT&C data/tasks (in some capacity) for both Modules 1 and 2. This thereby further reducing the respective computing requirements and thus hardware for the CS and C&DH subsystems on Module 1 and 2. In sharing the CS and C&DH subsystem resources in this manner, a reduction in CS and C&DH redundancy is achieved that subsequently reduces the CS and C&DH subsystem requirements for those modules receiving the shared resource (*i.e.,* Comm_CS_C&DH recipients).

Sharing the Comm, CS, and C&DH subsystems (referred to hereafter as the Comm_CS_C&DH shared resource) relies on existing, and in most cases, well vetted technology (*e.g.,* omni- and directional antennas). Therefore, the hardware associated with this shared resource, for both the sources and recipients of this shared resource, is relatively mature with respect to the technical hardware involved. However, there are still notable challenges to be addressed in employing this shared resource amongst modules in a fractionated spacecraft, which include techniques, methods, algorithms, and protocols for successfully managing (1) data delivery, (2) command and data handling, (3) housekeeping and mission data processing, and (4) tasking, scheduling, and control.

- **Attitude Determination System and Guidance Navigation System (ADS_GNS)**

The ADS and GNS can be shared amongst the respective modules in a fractionated spacecraft. Relative to a specific frame of reference (*e.g.,* Earth or spacecraft inertial frame of reference), the ADS and GNS is responsible for determining the rotational and translational position/orientation of a body (*e.g.,* spacecraft, module) respectively. Similar to sharing the Comm_CS_C&DH resource, in sharing the ADS and GNS resource, reductions in redundancy amongst the modules is achieved, subsequently reducing the ADS and GNS related requirements for modules receiving the shared resource (*i.e.,* ADS_GNS recipients).

If the ADS and GNS are not shared, each module in a fractionated spacecraft must have dedicated hardware that can fulfill the ADS and GNS functional responsibilities. A representative set of this dedicated hardware is an Inertial Measurement Unit (IMU) and star tracker, which together, can fully determine the rotational and translational position of each spacecraft/module that they are present on (see Figure 2-5).

Figure 2-5. Honeywell IMU and Aero/Astro miniature star tracker.

Honeywell HG9900 IMU

Aero/Astro Miniature Star Tracker

However, in contrast, if sharing the ADS and GNS resource, a visual positioning system (VPS) can be used in place of a star tracker on recipient modules of this shared resource. The work of others has suggested a VPS to be a viable option for use in place of a dedicated GNS for determining an object's translational position (Mandy, Sakamoto, Saenz-Otero, & Miller, 2007; McGhan, Besser, Sanner, & Atkins, 2006). A VPS consists of a set of sensors that can be used to detect relative translational motion/positioning between two or more bodies, each of which has a VPS/sensors. Therefore, if the ADS and GNS are shared amongst modules in a fractionated spacecraft, often one-module is selected to be the "central" module that necessarily requires an IMU, star tracker, and visual positioning system (VPS). And the remaining modules in the fractionated spacecraft, which are recipients of the ADS and GNS shared resource, thereby only, require a VPS and IMU (the IMU is still required for determining rotational orientation). Given that the "central" module in a fractionated spacecraft can determine its absolute position with respect to a given frame of reference, the other modules can determine their respective relative position to the central module, via their respective VPS's, and subsequently their absolute positions with respect to the central modules' frame of reference. In sharing the ADS and GNS resource amongst modules in a fractionated spacecraft, it reduces the ADS and GNS related requirements for those modules relying on/receiving the shared resource (*i.e.,* ADS_GNS recipients).

There are two key decisions to be made with regard to the ADS and GNS. First, whether the modules in a fractionated spacecraft need to be in a *cluster* or *formation flying* on-orbit configuration. And second, how the ADS and GNS are to maintain that on-orbit configuration (*i.e.,* should the ADS and GNS resources be shared or should a dedicated ADS and GNS be on every module). Cluster flying describes the situation in which the relative positioning of the modules in a fractionated spacecraft is "approximate", thereby making it only necessary for each module to maintain roughly a certain relative position with respect to the other modules. And formation flying describes the situation in which the relative positioning of the modules is "exact" (*i.e.,* within an appreciably small margin), thereby making it necessary for each module maintain a precise relative position with respect to the other modules; this subsequently presents a much more difficult relative navigation problem than does cluster flying.

Regardless of whether the ADS and GNS subsystems are shared, the relative navigation of structures in space (*e.g.,* modules in a fractionated spacecraft) is an area of technology development that, in recent years, has gained appreciable momentum (Wu, Cao, & Xue, 2006). In relation to fractionated spacecraft, relative navigation is the process of keeping each of the respective modules in a fractionated spacecraft at a specific rotational orientation and translational position relative to the Earth and other modules, per the desired on-orbit configuration (*i.e.,* cluster flying or formation flying configuration). To achieve this, relative navigation requires determining the relative state variables (*i.e.,* position, velocity, acceleration) of a given module in fractionated spacecraft with respect to the other modules and Earth. For a given module, and based on the ADS and GNS hardware discussed herein, relative navigation relies on the use of two components. These components are **(1)** an IMU and VPS (if ADS and GNS are shared), *or* an IMU and star tracker (if ADS and GNS are not shared); and **(2)** relative navigation control algorithms which compute state variables based on IMU and VPS/star tracker information. Relative navigation is not a trivial challenge, especially in the case in which the inter-module separation distances are in the range of tens of meters, which is a candidate inter-module separation distance for fractionated spacecraft.

Relative navigation is an extremely complex problem given the nature of objects in space being highly susceptible to changes in rotational and translational position from interactions with the surrounding environment (*e.g.,* solar pressure, magnetic fields). Leading research and development efforts for relative navigation systems and control algorithms is being conducted at the Massachusetts Institute of Technology (MIT) and the University of Maryland. These two universities are currently developing *autonomous* relative navigation systems, which achieve relative navigation with limited input from the ground (system operator). At MIT, autonomous relative navigation systems are being developed as part of the Synchronize Position Hold Engage & Reorient Experimental Satellites (SPHERES) program (Mandy, Sakamoto, Saenz-Otero, & Miller, 2007), and at the University of Maryland, autonomous relative navigation systems are being developed as part of the Secondary Camera and Maneuvering Platform (SCAMP) program (Mandy et al., 2007; McGhan et al., 2006). In addition to work done in academics, the feasibility of (autonomous) relative navigation has been, in differing capacities, demonstrated through several spacecraft/programs including Space Technology 5, Cluster II, Atmospheric Neutral Density Experiment (ANDE), and the University Nanosatellite Program.

From a hardware perspective, sharing the ADS and GNS subsystem resource (referred to hereafter as the ADS_GNS shared resource) is fairly mature owing to the fact that much the hardware associated with this shared resource has been proven in space, or is a relatively simple extension of existing technologies that are space qualified. However, there are still challenges to be addressed in employing this shared resource in a fractionated spacecraft, which include developing relative navigation control algorithms and protocols for maintaining on-orbit spacecraft/module configurations to within very small tolerances, while avoiding catastrophic on-orbit collision(s).

- **Power (Power)**

The Power subsystem consists of two main elements: power generation and power storage (power regulation and control is assumed implicit). It is possible to share both of these elements of the Power subsystem amongst modules in a fractionated spacecraft. In the case of sharing power generation, a module in a fractionated spacecraft can produce its own power, but additionally, some amount (or all) of the power required by one or more other modules in the fractionated spacecraft. Subsequently, these modules can each now produce less (or none of the) power than they require, thereby reducing their respective power generation requirements. Similarly in sharing power storage, a module can store power for itself, but

additionally, some amount (or all) of the power required during eclipse periods by one or more other modules. Subsequently, these modules can each now store less (or none of the) power than they require during eclipse periods. The key to sharing the Power subsystem resource amongst modules in a fractionated spacecraft, in terms of both power generation and storage, is wirelessly distributing (routing) the generated and stored power – this is called wireless power distribution (WPD). As was the case in sharing the Comm_CS_C&DH and ADS_GNS resources, sharing the Power resource must be done wirelessly. This is because modules in a fractionated spacecraft are physically separated and they may need to be replaced on-orbit throughout the lifecycle of the spacecraft; hence, there cannot be "power lines" strung between fractionated spacecraft modules.

Excluding the hardware associated with power regulation and control, the major hardware constituents of the power subsystem are solar arrays (power generation) and primary and secondary batteries (power storage). Although there are numerous options for power generation, it is assumed that solar arrays are used for power generation since most remote sensing mission spacecraft (see Section 2.1.7), which are the subject of this research, employ solar arrays to generate power.

At a conceptual level, sharing power generation is relatively straightforward. If a module is producing its own power as well as power for other modules, then that respective module's solar array will need to increase in size and mass. Subsequently, the modules that are producing less power than they need can have a reduced solar array size and mass for their own power generation. Similarly sharing the power storage is relatively straightforward at a conceptual level: distribute the secondary (not primary) batteries amongst the modules as desired, just ensure that the aggregate power stored (and that can be transmitted wirelessly) in all modules is enough to supply the power needed by all modules during the orbit eclipse periods. The confluence of sharing power generation and storage relies on wirelessly distributing (transmitting/sending) the power amongst modules. The method employed for the distribution of power amongst modules will drive all decisions pertaining to the allocation of power generation and storage amongst modules in a fractionated spacecraft, as each method requires a unique set of hardware for the distribution (sending and receiving) of power amongst modules. Several methods for (wireless) power distribution are enumerated hereafter and are best categorized relative to the distance over which they can distribute power.

Short Distance (≤ ~0.5 meters)

1. Electromagnetic Induction: power distribution via mutual induction (see Figure 2-6)
   - *Associated Hardware:* metallic coils, ionized medium
   - *Type of Power Distribution:* wireless

Moderate Distance (≤ ~100 meters)

1. Evanescent Wave Coupling: electromagnetic (EM) waves (*i.e.,* power/energy) are distributed via an evanescent waveguide, that is, a material structure that funnels EM waves from a source to sink with minimal energy dissipation (loss). At the receiving end of a waveguide (*i.e.,* the sink), the EM waves are converted into DC power. Figure 2-6 shows a flexible evanescent waveguide.
   - *Associated Hardware:* EM transmitter/receiver, energy-DC converter, waveguide
   - *Type of Power Distribution*: wired (due to reliance on waveguide structure)

2. AC/DC Cables: power distribution via power cables with alternating or direct currents (better for longer and shorter distances respectively) (Kerslake, 2008).
   - *Associated Hardware:* power cables
   - *Type of Power Distribution:* wired (due to cables)

Figure 2-6. Electromagnetic conduction (short distance) and a flexible waveguide (moderate distance).

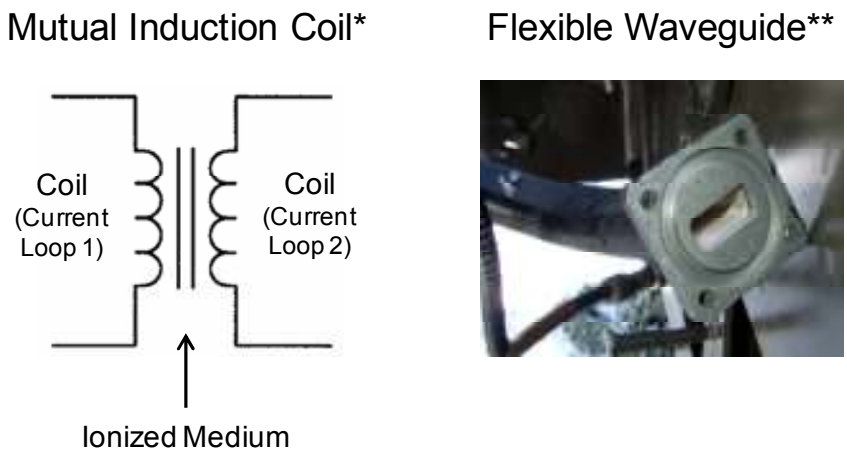


Image Source: * *pyroelectro.com* and ** *Open Source – wikipedia.org*

Long Distance (order of kilometers)

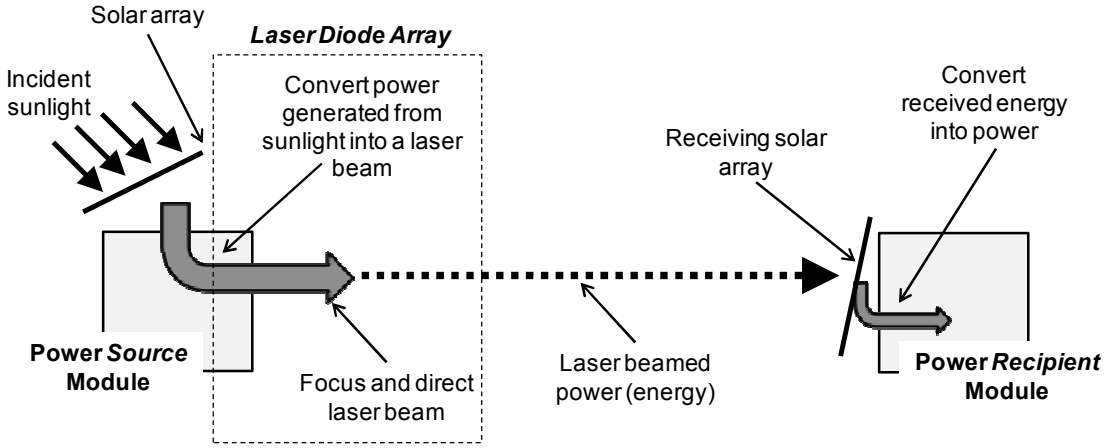
1. Radio and Microwave Power Transmission: wireless power distribution via beaming of radio and microwaves (energy) using a conventional antenna aperture (Kerslake, 2008; Landis, 1989; McSpadden & Mankins, 2002).

- *Associated Hardware:* transmitting antenna, rectenna (EM wave-to-DC converter)
- *Type of Power Distribution:* wireless

2. Laser Power Beaming: wireless power distribution via beaming of electromagnetic waves (energy) at laser wavelengths (Kerslake, 2008; Russell J. De Young, Michael D. Williams, Walker, Greg L. Schuster, & Lee, 1991; Howell, Mark J. O'Neill, & Fork, 2004; M.D. Williams et al., 1993; Landis, 1989). See Figure 2-7 for a conceptual schematic of laser power beaming/distribution.

- *Associated Hardware:* solar arrays, laser diode array, microlens'
- *Type of Power Distribution:* wireless

Figure 2-7. Conceptual representation of a laser power beaming wireless power distribution system.



3. Electric Conduction: wireless power distribution via a thermo-ionic converter that transmits/creates a high-power ultraviolet beam (UV), which serves as an ionized channel through which current can travel analogous to current traveling through a wire.

- *Associated Hardware:* emitter, collector, ionized medium
- *Type of Power Distribution:* wireless

4. Concentrated, Reflected Sunlight: wireless power distribution via reflected, concentrated sunlight from one structure to another. The sunlight reflected to (*i.e.,* received by) another structure is converted into electricity with a heat engine (Landis, 1989; Turner, 2006).

- *Associated Hardware:* primary capture mirrors, beam steering mirrors, thermal reservoir, heat engine (energy-DC converter)
- *Type of Power Distribution:* wireless

For fractionated spacecraft, the short and moderate distance WPD systems are not practical given the likely case when the modules in a fractionated spacecraft are separated by more than 10 meters; subsequently, these WPD systems were not chosen for the Power shared resource. And in terms of the long distance WPD systems, all four options were considered for wirelessly distributing power amongst fractionated spacecraft modules. Although not enumerated herein, these four long distance WPD systems each have advantages, disadvantages, end-to-end efficiencies, and levels of technical maturity, all of which are paramount to selecting the "best" WPD system to employ. Based on these factors, laser power beaming was chosen to be the WPD system employed in fractionated spacecraft that share the Power resource. The selection of the laser beaming WPD system was predominantly made on the basis of the technical maturity and efficiency of the laser power beaming WPD system as compared to the other long distance WPD systems. Subsequently, the laser power beaming WPD system is modeled in the Spacecraft Evaluation Tool (see Chapter 3).

Revisiting the Module Level of Fractionation

In Section 2.1.3, the homogeneous, heterogeneous, and mixed classes of fractionation were discussed relative to the three perspectives adopted (*i.e.,* Constellation, Module, and Subsystem Level) for determining whether a spacecraft is fractionated or monolithic. Since shared resources are instantiations of physically independent (*i.e.* separated) subsystems, fractionated spacecraft employing shared resources provide an instantiation of a homogeneous/heterogeneous/mixed fractionated spacecraft with respect to the Module Level perspective.

2.1.6. Fractionated Spacecraft Technology Demonstrations

There have been numerous demonstrations of fractionated spacecraft, through which key technologies for these spacecraft have been developed and/or tested (*e.g.,* relative navigation and inter-module communication). These technology demonstrations serve several purposes. First, they mature the fractionated spacecraft concept, facilitating the transition of fractionated spacecraft from a notional design concept to a space-capable and ready system. Second, these technology demonstrations identify weak areas of fractionated spacecraft design, areas which can subsequently be improved and thus require more research and development. Lastly, these technology demonstrations provide, in a particular capacity, tangible instantiations of fractionated spacecraft, which raises the greater spacecraft community's awareness with regard to the capabilities and applicability of fractionated spacecraft. A discussion of the most significant technology demonstrations of fractionated spacecraft in academia, industry, and government is enumerated hereafter, organized by date of conception.

<u>1993 - Present</u>

SCAMP: University of Maryland

The Secondary Camera and Maneuvering Platform (SCAMP) has been designed and built by the University of Maryland for developing and demonstrating an autonomous relative navigation system (McGhan et al., 2006). The two broader objectives of SCAMP-related research are (1) to develop a relative navigation system that will enable fully autonomous station keeping and (2) to develop an autonomous rendezvous and docking system, both of which lead to the ability for spacecraft to traverse autonomously between specified waypoints in space. To date, two identical SCAMPs have been built. Each SCAMP is a 26-sided structure with six thrusters aligned with the primary SCAMP axes, in order to control its respective rotational and translational position (see Figure 2-8). Each SCAMP also has an onboard camera (needed for the VPS it employs) and an IMU. The VPS consists of a single camera on each SCAMP that can recognize the SCAMP with respect to other objects and sensors. The IMU consists of a tri-axial magnetometer, accelerometer, and three angular rate gyroscopes. The concurrent use of the SCAMPs respective IMU, VPS, and onboard computer enables the autonomous identification and control its relative position, velocity, and acceleration (Smithanik, Atkins, & Sanner, 2006). The SCAMPs are tested in a neutral buoyancy (*i.e.,* water) environment, manifested in the form of a 50' diameter, 25' deep pool, to simulate a zero gravity environment. Since SCAMP testing occurs in neutral buoyancy environment, it is necessary to communicate with the SCAMPs via an underwater fiber optic cable, rather than a conventional antenna transmitter/receiver system used for testing in an atmospheric or space environment.

Figure 2-8. A SCAMP in its neutral buoyancy environment.

Image Source: The University of Maryland, Space Systems Laboratory

<u>1995 - Present</u>

Darwin: ESA

The European Space Agency (ESA) has been developing a space interferometer called Darwin (European Space Agency, 2009). A space interferometer combines the electromagnetic waves captured by each of its respective telescopes (*e.g.,* optical mirrors) to form a single image of an object in space (Makins, 2002; Steel, 1967). As compared to a single telescope observation system, the benefit an optical space interferometer is that the images created by it have significantly more resolution than those created by the single telescope system. The Darwin space interferometer is to have four or five physically independent telescopes that make observations at infrared wavelengths. Due to the separated nature of its constituents, a space interferometer (*e.g.,* Darwin) is an instantiation of fractionated spacecraft, and subsequently the design of Darwin has contributed an understanding of many key technologies required for fractionated spacecraft such as relative navigation.

<u>1999 - Present</u>

SPHERES: MIT

The Synchronized Position Hold Engage and Reorient Experimental Satellites (SPHERES) are being developed by the Space Systems Laboratory at the Massachusetts Institute of Technology (MIT) (Mandy et al., 2007; Saenz-Otero & Miller, 2007; Mohan, Saenz-Otero, Nolet, Miller, & Sell, 2007). The broad objectives of SPHERES-related research are to demonstrate autonomous relative navigation systems and techniques that enable formation flying, which is directly extensible to the operation of fractionated spacecraft. The autonomous relative navigation system currently being developed and tested using the SPHERES will enable new types of space maneuvers (*e.g.,* autonomous rendezvous and docking). The SPHERES research group began in 1999 as part of an undergraduate design course at MIT that ultimately resulted in the design and manufacture of three identical SPHERES by MIT and Payload Systems Inc. Each of the respective SPHERES is roughly 0.4 x 0.4 x 0.4 meters in size and weighs about 3.5 kilograms (see Figure 2-9). The SPHERES can communicate and identify their position relative to one another using sensors (*i.e.,* a VPS) and a relative navigation system called the "global metrology system." The global metrology system consists of an infrared, ultrasound navigation, and gyroscope system. The global metrology system in each of the SPHERES can autonomously determine the relative position, velocity, and acceleration of the respective SPHERES with respect to one another. To date, testing by the SPHERES research group has shown the global metrology system to be a successful autonomous relative navigation system. And, in addition to the original three SPHERES built in 1999, three more SPHERES have since been manufactured. The ground testing for the SPHERES is done at the MIT Space Systems Laboratory, specifically on their SPHERES test bed. The test bed simulates a zero gravity space environment by elevating the SPHERES slightly off a low-friction surface via a CO^2 cold gas propulsion system, thereby enabling them to move around freely in two dimensions. Thus far, seven SPHERES test sessions have been successfully conducted in space. These tests occurred between March 2003 and March 2007 and were all performed onboard the International Space Station (ISS). The primary objective of the ISS tests was to demonstrate the successful use of the SPHERES hardware and relative navigation system. The SPHERES research group is still active and will continue to conduct tests in the future to improve the global metrology (autonomous relative navigation) system.

Figure 2-9. One of the SPHERES on the low-friction test bed at MIT.

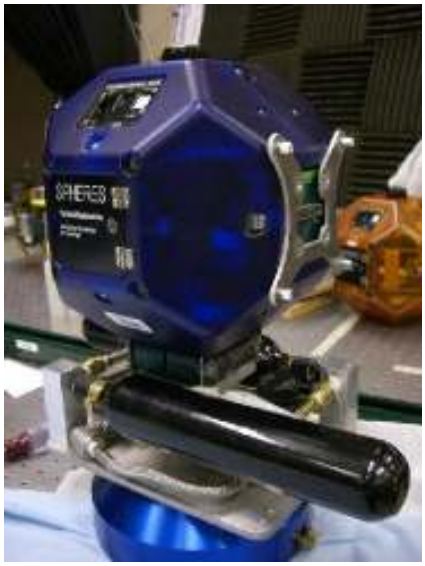


Image Source: MIT, Space Systems Laboratory

<u>1999 - Present</u>

University Nanosatellite Program: AFRL

The University Nanosatellite Program (UNP) began in January 1999 and has since consisted of two-year competitions in which universities compete against one another in the design and development of nanosats (Air Force Research Laboratory, 2008). The UNP is lead by the Air Force Research Laboratory (AFRL) and supported by the American Institute of Aeronautics and Astronautics (AIAA) and the Air Force Office of Scientific Research (AFOSR). Although not explicitly tasked with developing "fractionated spacecraft", the UNP has become the leading research and development effort in academia for the development of fractionated spacecraft (technologies). Presently, about 3,500 students across 25 universities participate in the UNP, which is in the middle of its fifth competition.

The UNP has two distinct stages. The first stage is a two-year competition in which any university can participate, and subsequently design and develop nanosats; there is a specified set of competition guidelines for the nanosats. Since the start of the UNP, the competition guidelines have been primarily focused on demonstrating key technologies required for the development of spacecraft that collaborate on-orbit, for example, technologies enabling formation flying via (autonomous) relative navigation). The first stage of the UNP concludes with a design review for each university, during which each university presents their final nanosat design(s). From these design reviews, normally one university team is selected to participate in the second stage of the UNP. During the second stage, the winning team(s) is(are) required to further increase the fidelity of their nanosats' design(s), hold additional design reviews with the AFRL, and ultimately manufacture their nanosats and send them to the AFRL. Once the AFRL receives the nanosats, the nanosats may enter a fast-track flight integration & testing program if they are to be launched into space.

The UNP has been a significant source of innovation, progress, and motivation for the development of fractionated spacecraft, albeit indirectly. To date, the nanosats designed and developed through the UNP have demonstrated in form and functionality some of the key technologies required for fractionated spacecraft. Due to interest and participation in the UNP constantly growing since its conception in 1999, the UNP is expected to continue to have an appreciable impact on maturing fractionated spacecraft.

<u>2000 - 2003</u>

TechSat-21: AFRL

The Technology Satellite 21 (TechSat-21) program began in 2000 and was lead by the Air Force Research Laboratory (AFRL); however, it is no longer active (Martin, Klupar, Kilberg, & Winter, 2001). The first objective of the TechSat-21 program and subsequent mission was to demonstrate formation flying with three micro-satellites (microsats) by having them operate together on-orbit as a "virtual satellite." And the second objective of the TechSat-21 program was to use the virtual satellite created by the three microsats to operate as an effectively unlimited aperture size radar imaging system (this is done via radar image interferometry). Each of the TechSat-21 microsats has an identical design and radar imaging system onboard, and mass and cost of 130 kg 17 $M respectively.

The TechSat-21 mission was to have a length of 1 year (with a possible extension to 1 or 2 years beyond that), during which the microsats would demonstrate a formation flying on-orbit configuration in a 550 km circular parking orbit, with the distance between the microsats varying between 100 to 5000 m. The TechSat-21 mission hoped to provide a "real-life" instantiation of sparse aperture processing and formation flying. As such, the two primary experiments intended for the TechSat-21 mission were to demonstrate: (1) autonomous formation flying maintenance, to achieve non-linear flying formations with multiple satellites; and (2) sparse aperture processing, through the combination of radar images from multiple

satellites into one holistic image via the development and application of innovative waveforms and signal processing. The TechSat-21 microsats were designed with a relative navigation system that would rely on periodic communication with the ground to determine the absolute and relative position of the TechSat-21 microsats (Chien et al., 2002). The TechSat-21 microsats were supposed to be launched in 2006 on an Atlas-5 launch vehicle, but the TechSat-21 program was ultimately cancelled in 2003 due to "technical difficulties" and numerous cost overruns, as was stated by the AFRL. However despite the cancellation of the TechSat-21 program, the TechSat-21 microsat research and development contributed to maturing techniques for the relative navigation of physically disperse (*i.e.,* independent) systems.

2000 - Present

Cluster II: NASA and ESA

The Cluster II program is jointly run by NASA and ESA. Cluster II consists of four satellites; the first two satellites were launched on July 16, 2000 and the second two were launched on August 9, 2000 (European Space Agency, 2008). The Cluster II mission has been extended three times since these launches in 2000 and it is presently intended keep the Cluster II mission active until the end of 2009. Each Cluster II satellite is identical in design with a mass of 1,200 kg and shape of a right circular cylinder with a 2.9 meter diameter and a 1.3 meter height, as is shown in Figure 2-10. The objective of the Cluster II mission is to investigate the Earth's magnetosphere through the concurrent operation of four identical satellites. The mission intends on keeping the four satellites in a tetrahedron-shaped, cluster flying, on-orbit configuration between the Sun and Earth. Throughout the mission, the distance between the Cluster II satellites is varied between 100 and 1000 km. The tetrahedron-shaped cluster flying configuration used in the Cluster II mission allows for the accurate determination of (1) three-dimensional and time varying phenomenon in the Earth's magnetosphere; (2) the study of small-scale plasma structures in space and time, in key plasma regions; and (3) magnetosphere phenomenon visualization. The Cluster II satellites employ a relative navigation system in order to maintain a tetrahedron-shaped cluster formation that relies on periodic communication with the ground for absolute and relative positioning.

Figure 2-10. An artist's rendition of a Cluster II satellite orbiting the Earth.

Image Source: ESA, Cluster Mission

2000 - Present

ExEP/TPF-I: NASA

The Exoplanet Exploration Program (ExEP) is lead by NASA and focused on developing and supporting technologies required for future exoplanet (*i.e.,* planets other than Earth) missions (Fischer et al., 2008). The specific objective of the ExEP is study aspects of certain planets outside our solar system in terms of their respective formation and development, physical and non-physical features, and ability to sustain life. The ExEP includes the development, manufacture, and operation of spacecraft that will help to fulfill the objectives of the program. Subsequently, the ExEP includes missions from the Navigator Program such as the Space Interferometer Mission (SIM), Terrestrial Planet Finder (TPF) Mission, Keck Interferometer Mission, Large Binocular Telescope Interferometer (LBTI) Mission, and the Michelson Science Center (MSC) Mission. The shorter-term goals of the ExEP primarily deal with developing technologies, for exploring and understanding planets, that enable moderate-scale "planet finding" missions; and one of the long-term goals of the ExEP is to perform a full-scale TPF mission.

The TPF consists of two distinct, but complimentary observatories: the coronagraph (TPF-C) and the formation flying infrared interferometer (TPF-I). The proposed TPF-C is a monolithic spacecraft having roughly a 4 x 6 meter telescope, capturing images at visible wavelengths. And the proposed TPF-I consists of multiple spacecraft, each of which has a single infrared telescope/mirror with a diameter of several meters (see Figure 2-11 for a conceptual example of one of the proposed TPF-I concepts). The TPF-I spacecraft is a space interferometer and, subsequently, will maintain a cluster flying or flying formation, on-orbit configuration to create an effectively unlimited aperture via interferometry (Makins, 2002; Steel, 1967). Therefore, the TPF-I mission will be a demonstration of key technologies required for fractionated spacecraft. The specific objectives of the TPF-C and TPF-I observatories are to measure the size, temperature, and placement of planets, as small as Earth, in far distant solar systems. Additionally, the TPF observatories will measure relative amounts of gases on a given planet to determine if it can support life.

The TPF mission and observatory designs are still in a pre-formulation study phase, so they are not expected to be deployed for another 10 to 15 years, provided that funding for the TPF program is sufficient to do so. In terms of deploying the TPF-C and TPF-I, the plan is to launch TPF-C first, and then the TPF-I five years later. The most recent news with regard to the TPF program is that NASA and ESA have discussed the possibility of collaborating on the development of one space interferometer mission to avoid redundancy and reduce costs, given the similar objectives of the TPF-I and Darwin missions.

Figure 2-11. A proposed concept for the TPF-I called the Emma X-Array.

Image Source: NASA, Jet Propulsion Laboratory

2006

ST5: NASA

The Space Technology 5 (ST5) spacecraft was developed by NASA and launched on March 22, 2006 - subsequently performing a 90-day mission (Hupp & Chandler, 2008). The objective of the ST5 spacecraft development and mission was to demonstrate the benefits of using a constellation of spacecraft to perform scientific studies. The ST5 system consisted of three nanosats, each weighing 25 kg and being roughly the shape of cube with equal length sides of 0.33 m. While in space, the ST5 nanosats had to perform maneuvers to maintain a formation flying configuration. The configuration chosen for the ST5 mission entailed the nanosats staying in a straight line while maintaining a 350 km distance between each other (*i.e.,* lead-trail configuration). The relative navigation system employed on the ST5 nanosats required each nanosat to communicate with the ground in order to identify and adjust its respective position relative to the other nanosats. Ultimately, the ST5 90-day mission was successful and subsequently all mission objectives were achieved.

2006 – 2007/2008

ANDE: NRL

The Atmospheric Neutral Density Experiment (ANDE) program was lead by the Naval Research Laboratory (NRL) (Naval Research Laboratory, 2005; Nicholas et al., 2003). The ANDE program involved developing two microsats, the Mock ANDE Active (MAA) and Fence Calibration (FCal). The specific objectives of the ANDE spacecraft and subsequent mission were to: (1) measure total neutral density along a predetermined orbit to improve orbit determination of space objects from Earth; (2) monitor the spin rate and orientation of spacecraft to better understand on-orbit flight dynamics; and (3) to provide a test subject for polarimetry studies. The MAA microsat was spherical with a 0.48 m diameter and mass of 50 kg whereas, the FCal satellite was slightly smaller but more massive than the MAA satellite, being spherical with a 0.44 m diameter and mass of 75 kg. The Space Shuttle Discovery deployed the MAA and FCal into space on December 22, 2006 and December 21, 2006 respectively. Once in space, the MAA and FCal satellites held a lead-trail configuration; the MAA satellite was leading the FCal satellite. The MAA satellite actively monitored its position relative to FCal satellite to gather data about atmospheric density and drag. The data gathered by the MAA and FCal satellites during their mission has enabled the NRL to better determine the ballistic coefficients of objects in LEO as well as improve their ability to model the space atmosphere. The MAA and FCal satellites did not have an autonomous relative navigation system but did share onboard data with one another through a set of modulated retro-reflectors (MRR). The ANDE mission therefore demonstrated coordinated formation flying and inter-satellite communication, both desirable attributes of fractionated spacecraft.

June 2009

ANDE-2: NRL

Directly following the completion of the ANDE mission, the NRL began the ANDE-2 program and subsequently, development of two new microsats, which were launched on July 15, 2009 aboard the Space Shuttle Endeavour (Naval Research Laboratory, 2009). The ANDE-2 mission is similar to the 2006 ANDE mission in that it employs two microsats that are both similar in mass and size to the ANDE microsats, and each ANDE-2 microsat has a relative navigation system. The ANDE-2 mission objectives are also similar to the ANDE mission objectives, which are to: (1) monitor atmospheric density, and (2) provide a test object for radar surveillance.

2.1.7. Remote Sensing Missions and Pointing-Intensive Spacecraft

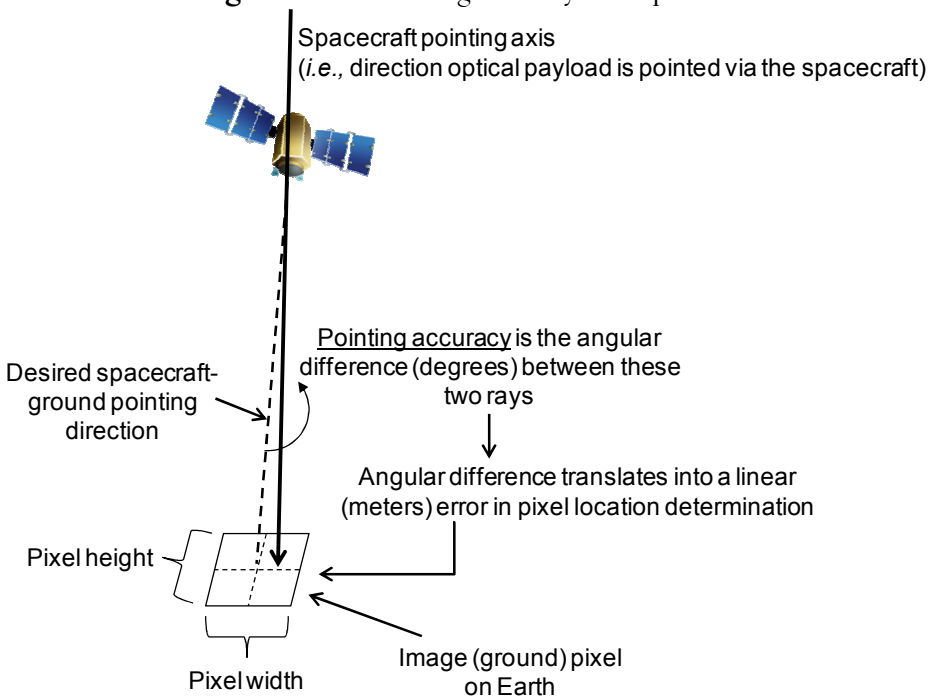
Remote sensing missions (RSMs) have the objective of making Earth observations over a specified range of the electromagnetic spectrum, most often this range falls in the visible, infrared, and near-infrared portion of the electromagnetic spectrum. The observations made in a RSM will pertain to the particular Earth-land and/or target coverage statistics defined by the RSM objectives. Such requirements may include, for example, observing all or part of the Earth's surface, oceans, atmosphere, magnetosphere, weather, resources, health of crops, and/or pollution. Spacecraft performing RSMs will therefore operate in orbits with altitudes and inclinations sufficient in meeting the observation requirements for the RSM. Positive attributes of spacecraft performing RSMs may include:

1. Continuous coverage over specific regions and/or targets on the Earth
2. Long spacecraft mission lifetime
3. High level of spacecraft autonomy and data processing
4. Ability to upgrade system software (and hardware) periodically
5. High availability during a given epoch or orbital period
6. Sufficient resolution of ground targets, as dictated by the RSM objectives
7. Mission/payload (image) data relay to ground results in little (or no) loss of quality
8. Ability to relay mission/payload data to the ground and distribute data in near real-time
9. Consistent equator cross-over time
10. Ability to correlate mission/payload data from multiple RSM instruments
11. Sufficient end-of-life RSM instrument mission/payload data gathering capability
12. Sufficient RSM instrument bandwidth for mission/payload data collection
13. Sufficient RSM payload instrument channels throughout the mission to accommodate demand
14. Operation of RSM instruments at optimal frequency(ies)
15. Desirable revisit rate (*i.e.,* the time between passes over the same location/target on Earth)

Earth-based RSMs have the objective of capturing images of the Earth's surface at a certain resolution, and most often at visible wavelengths. Spacecraft performing visible-wavelength, RSMs have optical mirror systems (aka telescopes) as their payload instruments. The major tradeoff for spacecraft performing visible-wavelength RSMs is the image resolution, optical mirror diameter (size), and the altitude and inclination of the spacecraft's respective orbit. As a result, Earth-imaging RSM spacecraft often have orbits with altitudes and inclinations of around 700 km and 98° respectively (consider GeoEye-1, Landsat-7, and EOS Aqua).

Inherent to the tradeoff between image resolution, mirror diameter, and orbit altitude and inclination is the pointing accuracy required by the spacecraft. Pointing accuracy is the difference between the desired pointing direction and the actual pointing direction (*i.e.,* spacecraft pointing axis), relative to a spacecraft inertial frame of reference (See Figure 2-12). Pointing accuracy can be measured in angular degrees relative to a spacecraft inertial frame of reference, or translated to the accuracy (in meters) in which targets/images on the Earth's surface can be discriminated. It is strongly desirable for the pointing accuracy of a spacecraft to be less than the image resolution; otherwise, it greatly diminishes the value of the images taken because the accuracy of their respective location on Earth cannot be guaranteed to within the detail of the image. High resolution, visible-wavelength, RSM spacecraft, as are being considered in this research, are required to be pointing-intensive due to their high pointing accuracy (*i.e.,* low tolerance for pointing error). Pointing tolerance is a set allowance for a certain amount of pointing error, that is, lack of pointing accuracy for a spacecraft. Pointing-intensive spacecraft often have pointing tolerances of approximately 36 milli-arcseconds (1e-4°).

Figure 2-12. Pointing accuracy description.



Fractionated spacecraft are of particular interest for pointing-intensive, RSM spacecraft because it is hypothesized that their ability to physically decouple subsystems and payloads that truly need precise pointing from the non-pointing-intensive spacecraft subsystems, will enable them to have lesser lifecycle costs and longer mission lifetimes than that of a comparable monolith. This hypothesis serves as motivation for this research effort, which specifically investigates pointing-intensive, RSM monolithic and fractionated spacecraft.

2.1.8. Lifecycle Uncertainties and Consequent Risks

The value delivery of a spacecraft over its respective lifecycle to beneficiary stakeholders (see Section 2.1.9) is a product of its continual performance (see Section 2.1.1). There are, however, risks that may proliferate throughout a spacecraft's respective lifecycle that have the effect of diminishing or destroying the spacecraft's ability to achieve or maintain a certain level of value delivery. These risks are the product of lifecycle uncertainties and there are multitudes of these, which can adversely affect the operation of a spacecraft throughout its respective lifecycle. For the purpose of better understanding lifecycle uncertainties and their consequent risk(s), it is useful to decompose the lifecycle uncertainties into distinct categories. Each category subsequently characterizes a unique type/class of lifecycle uncertainty and its respective risk(s), which adversely affect a spacecraft's respective lifecycle (*i.e.,* ability to deliver value). Table 2-1 provides a comprehensive listing of lifecycle uncertainty categories for a spacecraft; this table further expands upon the lifecycle uncertainties enumerated by Owen Brown et al. and Hugh McManus et al. (Brown & Eremenko, 2006b; McManus & Hastings, 2006).

Table 2-1. Spacecraft lifecycle uncertainties and consequent risks.

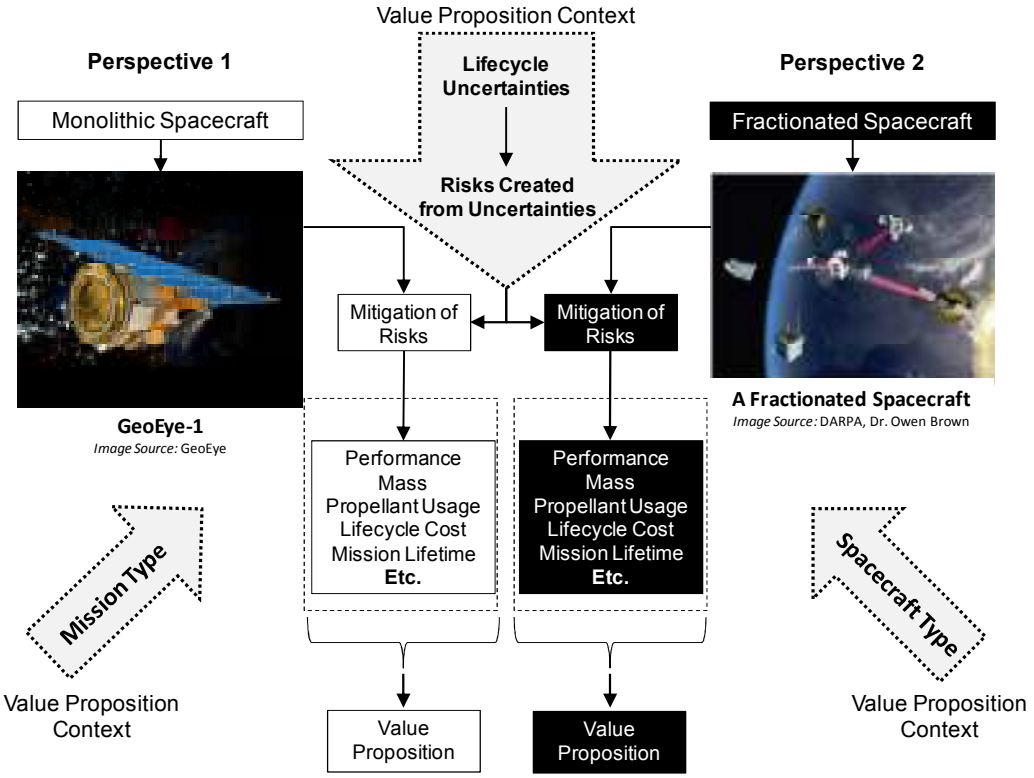
Lifecycle Uncertainties			
Uncertainty Category	**Uncertainty**	**Potential Risk(s)**	**Example**
Market Demand	Uncertainty due to fluctuations in demand for a spacecraft's supply (value delivery)	A spacecraft that cannot meet the demand or provides too much supply given the demand	User demand for a spacecraft's service increases 50% past the capacity for the spacecraft to supply that service
Market Supply	Uncertainty due to fluctuations in supply (value delivery) by a spacecraft	A spacecraft that supplies to little or supplies too much given the demand	A spacecraft supplies 50% more service than the market currently demands
National Security	Uncertainty due to exogenous, explicit, and hostile threats to a spacecraft	Change in mission profile/operations, reduction in spacecraft functionality, or complete loss of a spacecraft	An anti-satellite (Asat) weapon destroys a spacecraft
Technical	Uncertainty due to hardware and software reliability (probability of failure)		TCS electric heaters short (fail), thereby causing a spacecraft to effectively be destroyed during eclipse due to heat loss
Environmental	Uncertainty due to factors introduced via the environment in which a spacecraft exists		The aerodynamic drag produced on a spacecraft due to the atmospheric density causes it to increase in altitude
Operational	Uncertainty due to factors introduced via human-operator error		Ground operations sends incorrect command data, thus causing the payload to point to undesired targets
Launch	Uncertainty due to risk of launch vehicle failure or schedule slip		The launch vehicle fails right before orbit insertion, thereby destroying the spacecraft in the payload fairing
Funding	Uncertainty due to changes in the funding profile or allocation of funding at a specific point in time in a program	Change in spacecraft design, mission, and/or operations	During development, funding for a spacecraft is 10% lower than expected, thus requiring design changes
Programmatic	Uncertainty due to changes in a spacecraft's program (not funding related)		A labor shortage causes a delay in the manufacture of a spacecraft, thereby requiring the launch date to be pushed

One of the purported benefits of fractionated spacecraft is their ability to mitigate the risks (*i.e.,* maintain value delivery) resulting from the lifecycle uncertainties, shown in Table 2-1, more effectively than a comparable monolithic spacecraft. This in turn, is alleged to be the result of several of the purported positive attributes of fractionation in Section 2.2.1 (*e.g.,* ability to add and replace modules on-orbit).

2.1.9. Spacecraft Value Proposition

Spacecraft beneficiaries and beneficiary stakeholders derive benefit from a spacecraft through its respective performance (see Section 2.1.1). If the benefit provided by a spacecraft is quantified relative to the cost of obtaining that benefit, then it should be interpreted as value. Value delivery, that is, the supply of benefit relative to its cost, is often of more interest to spacecraft beneficiary stakeholders than benefit alone, this due to value being a cardinal metric that encapsulates benefit *and* cost. As such, this research adopts the use of a *value proposition* in attempt to both understand and quantify a spacecraft's respective value delivery (Richards, Szajnfarber, M. Gregory O'Neill, & Weigel, 2009). The value proposition, as is characterized through Figure 2-13, contains cardinal metrics of both benefit and cost in an attempt to provide an easy manner in which to view benefits relative to their respective costs, and hence derive/quantify value. The value proposition therefore provides an appropriate manner in which two or more spacecraft can be compared based on differences in their respective value delivery.

Figure 2-13. Spacecraft value proposition.



As is enumerated in Figure 2-13, the value proposition can be examined from any number of perspectives. In particular relevance to this research, there are two perspectives: monolithic spacecraft and fractionated spacecraft. The monolithic and fractionated spacecraft (*i.e.,* perspectives) each yield unique value propositions, given the context for the value proposition formulation. (The process of generating value propositions for a given perspective, given the context, is called an assessment or simulation.) The three key elements of the context are the (1) type of mission a spacecraft is performing (*e.g.,* RSM), (2) type of spacecraft (*e.g.,* pointing-intensive), and (3) the lifecycle uncertainties that adversely affect the value delivery of a spacecraft. Given the monolithic and fractionated spacecraft (perspectives) and a particular context for determining the value proposition for these spacecraft, the metrics comprising the value proposition can be any set of metrics appropriately and accurately characterizing the benefits and costs of these spacecraft (*e.g.,* mission lifetime, lifecycle cost, and power consumption). Therefore, the formation of the value proposition is completely at the discretion of the individual performing the assessment.

2.1.10. Confidence in the Value Proposition

The value propositions for the monolithic and fractionated spacecraft (perspectives) can be complemented by quantifying the confidence associated with those metrics in the value proposition that have values not known with certainty, this being the result of the manner in which the value proposition metrics are quantified via the assessment. Of particular relevance to this research is the confidence associated with the lifecycle cost (LCC) metric of a value proposition.

There are two types of uncertainty in LCC that can be quantified; however, please note that these uncertainties are entirely dependent on the manner in which LCC is quantified. The first source of uncertainty in LCC results from uncertainty in the cost model used to quantify LCC, this in turn being the

result of the cost model employing parametric cost estimating relationships (CERs). And the second source of uncertainty in LCC is due to the use of a Monte Carlo Analysis (MCA) to quantify LCC.

The first type of LCC uncertainty can be quantified from the statistical standard error (or deviation) corresponding to each of the respective CERs in the cost model employed to quantify LCC. The standard error for each CER will depend on, and thus change with, the LCC value. The result of the aggregation of statistical cost model error, for a given LCC value, is the ability to place confidence bounds on that LCC value. A common approach for representing this confidence is to quantify a given LCC value relative to its respective 5th and 95th percentile confidence interval. The quantification of LCC uncertainty, due to uncertainty in the cost model, is referred to as cost model uncertainty (CMU).

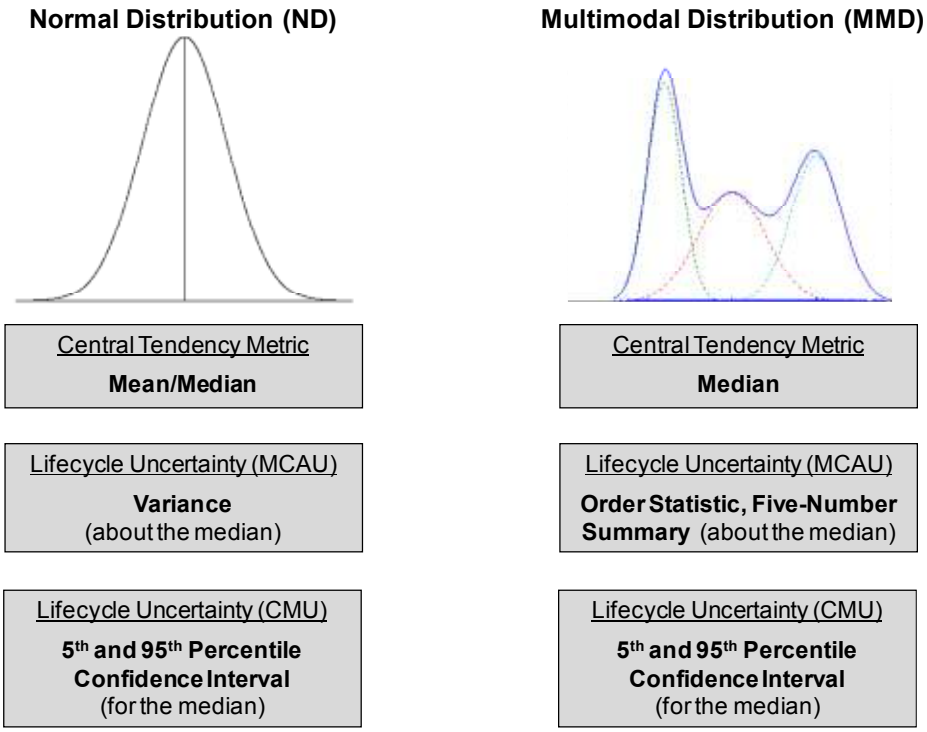
The second type of LCC uncertainty results from the use of a Monte Carlo Analysis (aka Monte Carlo Simulation) in a spacecraft assessment and is referred to as MCA uncertainty (MCAU) (Caflisch, 1998; Lim & Nebus, 2007). In employing a MCA, numerous lifecycles of a spacecraft are simulated, where one trial of the MCA simulates one lifecycle. Therefore, for *n* MCA trials there will be *n* values for LCC, some identical and some different, but each corresponding to one trial (lifecycle simulation) in the MCA. Given that lifecycle uncertainties and their consequent risks are not known *a priori* for a spacecraft's lifecycle, the motivation for using a MCA is that it better encapsulates all possible lifecycle's the spacecraft may experience, and hence LCCs for the spacecraft. To reinforce the benefits of a MCA with regard to quantifying a spacecraft's LCC consider the case in which only one MCA trial is used to quantify a spacecraft's LCC. Given all the potential variations in a spacecraft's lifecycle, due to lifecycle uncertainties, using only one MCA trial offers just a *single* sample of the lifecycle the spacecraft may experience, and hence LCC of the spacecraft. However, in contrast, if 2,500 MCA trials are used, this offers 2,500 samplings of the lifecycle the spacecraft may experience, and hence LCCs of the spacecraft - and in doing so provides a more holistic understanding of the potential LCC values for the spacecraft.

Performing a large number of MCA trials appropriately quantifies the relationship between a spacecraft's potential LCC values and its ability to mitigate the adverse LCC implications of lifecycle uncertainties to maintain value delivery. Through attaining an understanding of this relationship, a spacecraft architecture (design) can be made more robust with respect to lifecycle uncertainties, thereby leading to the design of more LCC-robust spacecraft.

However, the drawback of employing a MCA to quantify the LCC of a spacecraft is that there will be associated uncertainty in that LCC value. Specifically, this is because each MCA trial does not necessarily yield the same LCC for a given spacecraft since lifecycle uncertainties are stochastic by their very nature, and hence a spacecraft can have a different lifecycle (and thus LCC) in each MCA trial. Therefore, if there are *n* trials, there will be *n* LCC values for a spacecraft that are not necessarily the same. These LCC values can be represented as a distribution (*i.e.,* probability density function) to illustrate the statistical central tendency and variability of LCC over all the MCA trials. *The second type of LCC uncertainty is therefore manifested in the variation of the LCC distribution relative to the distributions respective measure of central tendency.* The most common LCC distributions resulting from a MCA are normal distributions (NDs) and multimodal distributions (MMDs) (see Figure 2-14). (See Appendix D for a discussion of the multimodal LCC distributions observed in this research and the implications of these distributions for the appropriate number of MCA trials to use.)

Figure 2-14 summarizes the quantification of LCC uncertainty for a given spacecraft due to cost model uncertainty (CMU) and MCA uncertainty (MCA), for both normal and multimodal LCC distributions.

Figure 2-14. Quantifying LCC uncertainty relative to the central measure of tendency.



For a given distribution of LCCs, as created from the MCA, the measure of central tendency and variability corresponding to that distribution can be quantified. Although it is important to preserve a LCC distribution in form, quantifying the measure of central tendency and variability of a LCC distribution is necessary to keep the comparison of LCC distributions tractable. The measure of central tendency and variability quantifications differ for ND and MMD and are as follows:

> If a LCC distribution is a **ND** then the most appropriate measure of central tendency and variability is the **mean** and **variance** of that LCC distribution respectively.
>
> If a LCC distribution is a **MMD** then appropriate measures of central tendency are the **mean, median, mode, and two-sided quartile weighted median** of that LCC distribution; and appropriate measures of variability are the **(1) number of modes, (2) number of dominant modes, (3) skewness, (4) kurtosis, and (5) order statistic, five-number summary (aka box-and-whisker plot)**[2] of that LCC distribution.

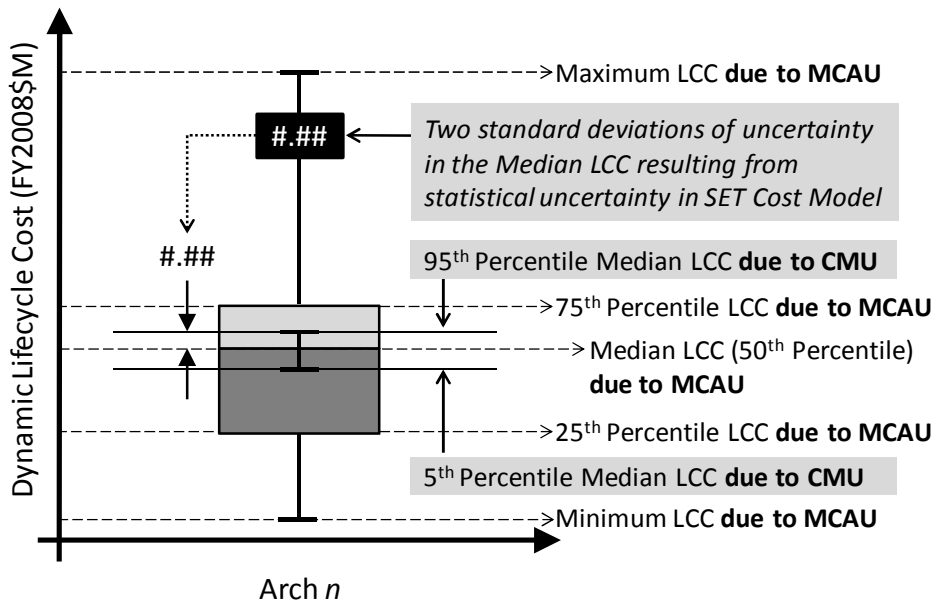
Quantifying the central tendency and variability of LCC distributions that form NDs is straightforward because there are statistically proven best metrics for central tendency and variability. However, unfortunately, for the LCC distributions that form MMDs, there is simply not a correct answer as to the

[2] The order statistic, five-number summary is only applicable if the measure of central tendency is the median. An order statistic, five-number summary consists of five values characterizing a given LCC distribution: maximum, 75th percentile, 50th percentile (median), 25th percentile, and minimum LCC

best measure(s) of central tendency and variability. Therefore, to quantify the uncertainty in LCC due to the MCA (*i.e.,* MCAU), this research investigation employs the median and order statistic, five-number summary as the measure of central tendency and variability metrics respectively. This decision is based on the appropriateness of these two metrics with respect to the LCC MMDs observed in this research (see Figure 4-31, Figure 4-32, and Figure D-1).

As is shown in Figure 2-14, the LCC due to CMU is computed for the median LCC value since uncertainty due to CMU and MCAU is assumed to be relative to the LCC distribution central measure of tendency (*i.e.,* median). This therefore implies that the order statistic, five-number summary is also computed and represented relative to the median LCC. For a given spacecraft, assuming that there is uncertainty in the LCC due to both CMU and MCAU and that the LCC distribution for the spacecraft is a MMD, the uncertainty in the LCC value can be visually quantified/represented as shown Figure 2-15. *Note that hereafter the word Dynamic precedes LCC if it was quantified by a MCA, if not; LCC is preceded by the word Static.*

Figure 2-15. Representing Dynamic LCC uncertainty relative to the measure of central tendency.



In Figure 2-15, the x-axis of denotes the monolithic or fractionated spacecraft architecture under consideration, whereas the y-axis denotes the Dynamic LCC value for that spacecraft with respect to each element of the order-statistic, five-number summary. Additionally, the y-axis denotes the Dynamic LCC of the spacecraft with respect to the elements of CMU about the Median Dynamic LCC.

The order statistic, five-number summary in Figure 2-15 characterizes the central tendency and variability of the Dynamic LCC distribution (*i.e.,* probability density function) for a given spacecraft through the following Dynamic LCC values: maximum, 75th percentile, 50th percentile (median), 25th percentile, and minimum. The 25th-75th percentile range is the inter-quartile range. In addition to these five numbers, the uncertainty in the Median Dynamic LCC due to CMU is shown about the Median Dynamic LCC value. The 5th and 95th percentile confidence values in the Median Dynamic LCC due to CMU are shown by the lower and upper bars in Figure 2-15 about the Median Dynamic LCC, respectively. Due to all Dynamic LCC values depicted in Figure 2-15 being an order statistic they can accurately compare any number of spacecraft Dynamic LCC distributions, regardless of statistical nuances between the distributions.

Reflection on the SET Cost Model Uncertainty (CMU)

If parametric cost models are used to estimate the (median dynamic lifecycle) cost of a spacecraft during the early stages of its design (*e.g.,* conceptual design phase), it may be inappropriate to quantify the uncertainty in that cost estimate due to uncertainty in the parametric cost model (*i.e.,* CMU). The basis for this hypothesis is that in parametric cost models, the CMU is determined by the respective standard error (or deviation) of the CERs employed in the cost model, and these do not quantify a critical source of error in a spacecraft cost estimate made during the early stages of design. This critical source of cost error addresses the fact that spacecraft programs and designs are susceptible to change, and therefore a spacecraft cost estimate made during the early stages of its design is not likely to be the cost of the spacecraft at the time of its deployment. This unaccounted cost error is therefore of significant importance for cost estimations made during the early stages of a spacecraft's design as it quantifies the cost implications due to the natural (cost) evolution and maturity of a spacecraft design over the course of its respective program . As such, CERs fail to capture this critical source of error because CERs are formulated based on the cost of real (*i.e.,* actual) spacecraft, assuming they were built as designed from the very beginning. Subsequently, CERs do quantify uncertainty due to differences/variations in cost estimates for spacecraft made during their early design stages relative to their actual respective costs at time of deployment.

Therefore, the standard error in a CER is accurate when the design of a spacecraft is certain, but not accurate/appropriate for estimating the cost uncertainty of a spacecraft at an early stage of its program (*e.g.,* conceptual design phase). In this sense, the estimates of cost uncertainty, due to standard errors in the CERs within a parametric cost model (*i.e.,* CMU), represent an absolute lower bound of uncertainty in spacecraft cost estimates made during the early stages of their design. Therefore, the relevant question is, how appropriate is it to quantify the cost uncertainty associated with a spacecraft cost estimate if the design of the spacecraft is hardly certain?

2.2. Motivation

The second essential element of the problem formulation is providing motivation for this research investigation and its respective outcomes. Subsequently, Section 2.2.1 develops motivation based on positive and negative hypotheses made about fractionated spacecraft, which enumerate important questions regarding the potential benefits and costs of fractionated spacecraft relative to comparable monolithic spacecraft. Section 2.2.2 then proceeds to develop motivation based the unknown nature of monolithic and fractionated value propositions, specifically for pointing-intensive, remote sensing missions.

2.2.1. Positive and Negative Hypotheses about Fractionated Spacecraft[3]

Positive and negative hypotheses about fractionated spacecraft relative to comparable monolithic spacecraft serve as a fundamental source of motivation for the assessment of fractionated spacecraft, and subsequently this research. The positive and negative hypotheses enumerate important purported benefits and costs of fractionated spacecraft as well as some of the most heated debates about fractionated spacecraft. Numerous positive and negative hypotheses about fractionated spacecraft are enumerated hereafter, which further expands upon those hypotheses enumerated by Matt Richards et al. and Owen Brown et al. (Richards et al., 2009; Brown, Long, Shah, & Eremenko, 2007).

[3] The terms positive and negative hypotheses are used instead of the terms advantages and disadvantages respectively, as the latter two terms convey a sense of certainty (*i.e.,* having been proved already), which is not appropriate.

Positive Hypotheses

As compared to monolithic spacecraft, fractionated spacecraft...

1. Diversify launch risk.
2. Diversify risk of on-orbit failure due to uncertainties in operational environment, exogenous threats to the system (*e.g.,* ASAT attacks), and flight hardware and software performance.
3. Enhance reliability through emergent sharing of subsystem resources and on-orbit redundancy.
4. Provide scalability in response to service demand fluctuations and need for new applications and functionality via on-orbit module replacement and addition.
5. Are more readily upgradable in response to technological obsolescence.
6. Enable incremental deployment of capability to orbit (*i.e.*, staged deployment).
7. Enable graceful degradation of on-orbit capability via removal of modules.
8. Provide more robustness in response to development delays, funding fluctuations, changes in requirements, and programmatic issues.
9. Have a lesser design, manufacture, integration, assembly, and testing time due to spacecraft system decoupling (*i.e.,* modularity).
10. Have lesser risk (*i.e.*, variability) associated with lifecycle cost for a given space mission.
11. Better facilitate production learning across multiple similar modules/spacecraft.
12. Enable spacecraft to be launched on smaller, less expensive launch vehicles with shorter lead times.
13. Have shorter design cycle times.
14. Can more easily employ product platforming that leads to economies of scale as the number of modules/spacecraft built increases.
15. Can perform new space missions or improve the applicability of existing space missions.
16. Will lead to cheaper access to space and a reduction in the present economic barriers that prevent many organizations from accessing space.
17. Better facilitate the development of cost-effective, multi-mission spacecraft.
18. Are more cost-effective in highly uncertain ("dangerous") operational environments.
19. Provide more value over the respective lifecycle of a spacecraft.
20. Can better meet the needs of the beneficiaries and beneficiary stakeholders.
21. Have a higher probability of maintaining value delivery in highly uncertain ("dangerous") operational environments
22. Are more likely to lead to the creation of new space technologies.
23. Can have longer effective mission (operational) lifetimes.

Negative Hypotheses

As compared to monolithic spacecraft, fractionated spacecraft...

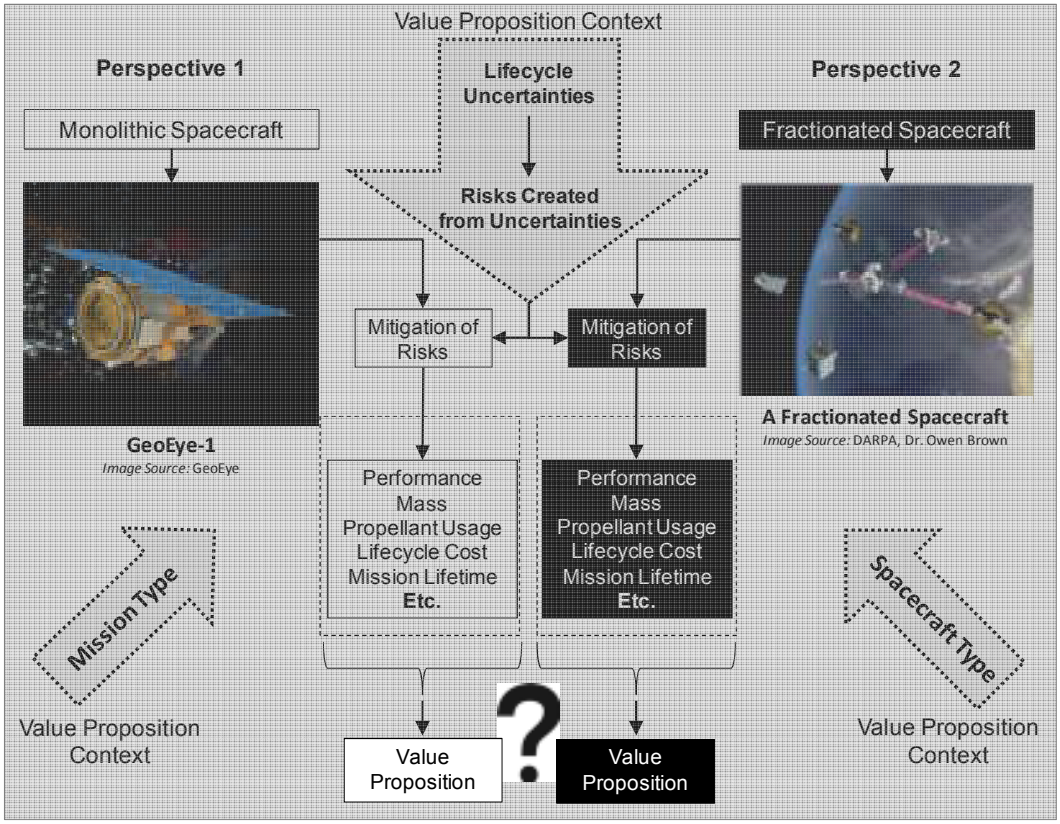
1. Have larger lifecycle costs.
2. Are more complex due to their inherent physically disperse nature.
3. Have more mass, which can have adverse implications for launch vehicle selection and cost.
4. Are more likely to be sub-optimal for spacecraft missions due to their use of standardized elements because of product platforming.
5. Have a higher technology risk and subsequent probability of on-orbit failure due to their requirement and incorporation of new, unproven technologies.
6. Have a much higher probability of on-orbit failure due to the collision of modules.
7. Have a significant risk of on-orbit failure if inter-module dependencies are present, which is the case when resources are shared amongst modules.
8. Have more complexity/difficulty associated with their on-orbit concepts of operations (CONOPS).
9. Will require new manufacturing, IA&T, and launch site operations.
10. Change the lifecycle cost (*i.e.,* cost expenditure) profile adversely.
11. Are less likely to see a positive return on investment.
12. Will have more difficulty being accepted and used by the industry and government.
13. Will have a higher probability of not being developed if program funding is delayed.
14. Have diffuse benefits and concentrated costs, that is, require significant upfront costs while having benefits that typically emerge late in the lifecycle.
15. Represent a technology "push" initiative, which is not favored by spacecraft acquisition programs.
16. Have benefits understood best through the adoption of a value-centric perspective – something new and not readily accepted by the government and commercial spacecraft industry.
17. Development requires investing heavily in reliability and mission insurance rather than launch insurance and this does not resonate well with some beneficiary stakeholders.
18. Perform missions not tailored to the present national priorities for the U.S. space program.
19. Are a source of disruptive innovation, which is not readily accommodated by the government and commercial space industry.
20. Introduce a new spacecraft paradigm, which most spacecraft developers are reluctant to adopt in their current practices given their large stake/investment in the status quo.
21. Introduce new and difficult challenges associated with manufacturing, launch vehicle procurement, and launch site preparations due to the likely need of more than one launch vehicle for deployment.
22. Rely on new spacecraft technologies that subsequently cannot be developed because of current government-lead spacecraft initiatives focused on sustaining current space capabilities rather than developing new ones.

The relationship between these negative hypotheses and implementation challenges with regard to the lifecycle of a (fractionated) spacecraft is treated in Appendix A.

2.2.2. Unknown Spacecraft Value Propositions

In addition to the positive and negative hypotheses about fractionated spacecraft, the value proposition introduced in Section 2.1.9 and shown again in Figure 2-16, provides further motivation for this research. The motivation specifically originates from a general lack of understanding between the value propositions for monolithic and fractionated spacecraft performing pointing-intensive, remote sensing missions. The context for these value propositions is enumerated hereafter and it emphasizes how little is known about nuances in monolithic and fractionated spacecraft value propositions relative to the cited context.

Figure 2-16. Unknown spacecraft value propositions.



The monolithic and fractionated spacecraft (*i.e.,* perspectives) each yield unique value propositions, given the context for the value proposition formulation, and as is discussed in Section 2.1.9, this context is composed of three elements. These are the (1) type of mission a spacecraft is performing (*e.g.,* RSM), (2) type of spacecraft (*e.g.,* pointing-intensive), and (3) the lifecycle uncertainties that adversely affect the value delivery of a spacecraft. For the purposes of this research, the three respective elements of the context are:

Value Proposition: Context

- Mission Type: Remote Sensing Mission
- Spacecraft Type: Pointing-Intensive
- Lifecycle Uncertainties to Consider: Launch, Technical, Environmental, and Operational

Remote sensing missions (RSMs) are the mission type due to their criticality to the commercial space industry and government. In today's world of an ever-increasing reliance on satellite images of the Earth, at higher and higher resolutions, the relevance of RSMs cannot be disputed.

Higher performance RSM payloads are pointing-intensive and therefore selecting pointing-intensive spacecraft as the spacecraft type for the context is a logical decision. Additionally, there is a hypothesis that the ability of fractionated spacecraft to physically decouple subsystems and payloads that truly need precise pointing from the non-pointing-intensive spacecraft subsystems, will enable them to have lesser lifecycle costs and longer mission lifetimes than that of a comparable monolith.

The lifecycle uncertainties being considered for the value proposition context are launch, technical, environmental, and operational – this leaves out market demand, market supply, national security, funding, and programmatic (see Section 2.1.8). The inherent tradeoff in defining the lifecycle uncertainty aspect of the context is that as the number of uncertainties considered increases, the resources (*e.g.,* time) required to evaluate the implications of those uncertainties on the value proposition increases. Subsequently based on this tradeoff, there is no optimal combination of the nine lifecycle uncertainties listed in Table 2-1 for which to compose the value proposition context. Subsequently, the four lifecycle uncertainties comprising the context for the purposes of this research are thought to adequately capture the lifecycle uncertainties of interest, as elicited in the literature (see Section 2.3), while keeping the evaluation of the lifecycle uncertainties tractable.

Therefore, with the three elements of the value proposition context now defined, the critical question motivating this research becomes:

> ***How do monolithic and fractionated spacecraft value propositions compare for pointing-intensive, remote sensing missions during which the spacecraft are subjected to launch, technical, environmental, and operational lifecycle uncertainties?***

This critical question drives at an area of spacecraft design that, in a quantitative sense, is mostly unknown, and thereby provides a meaningful source of motivation for this research investigation. This critical question also serves the purpose of guiding the research question development based on previous assessments of fractionated spacecraft, as are discussed in the next section.

2.3. Literature Review

The third step in formulating the problem statement is a review of pertinent literature. There have been ten notable research efforts conducted in academia, industry, and government that have served as the most significant sources for understanding fractionated spacecraft. In the literature review, academia is treated separately from industry and government because, to date, the latter two entities have always collaborated in assessing fractionated spacecraft. The literature review represents an appropriate compilation of research that has assessed, in at least some capacity, fractionated spacecraft. The literature review thereby ensures the adequate provision of context for understanding the unique contributions of this research (Section 2.4).

Thus far, in academia, detailed investigations of fractionated spacecraft have predominantly been conducted at the Massachusetts Institute of Technology (MIT), and in industry and government, nearly all investigations of fractionated spacecraft have been lead by Owen Brown. In the literature review, research is organized by seminal (or only) publication date.

2.3.1. Academia

2001, 2002

TOS Studies: MIT Course 16.89, MIT, CalTech, and Stanford

In academia, the first investigations of fractionated spacecraft were the A, B, C, and X-TOS (Terrestrial Observer Swarm) studies (Diller, 2002). All of these studies were focused on the design of TOS's that contained between 1 and 29 satellites, referred to as mothers and daughters, which collaborated in order to make observations of Earth's ionosphere. The A-TOS study was conducted in 2001 through a government-sponsored consortium, which involved the Massachusetts Institute of Technology, California Institute of Technology, and Stanford University. The A-TOS study employed the use of the Generalized Information Network Analysis (GINA) methodology for the TOS satellite designs. The B-TOS study then began in the spring 2001 semester at MIT as part of Course 16.89: Space Systems Engineering. B-TOS employed Multiple Attribute Utility Theory (MAUT) to assess various TOS designs and in doing so established the critical first step towards the development of the MATE methodology[4]. Following the B-TOS completion, the C-TOS study began in the summer of 2001, and like A-TOS, was a MIT, CalTech, and Stanford collaboration. Through C-TOS, integrated concurrent engineering (ICE) was used to assess the TOS designs. The last TOS study, X-TOS, began in the spring of 2002 at MIT as part of Course 16.89. During X-TOS, the Multiple Attribute Tradespace Exploration (MATE) methodology was employed.

In terms of assessing fractionated spacecraft, the major contributions of the TOS studies are providing an instantiation of a (fractionated) spacecraft model and, in addition, how to develop and apply an alternative method for evaluating fractionated spacecraft called MATE. And in terms of an understanding of fractionated spacecraft value propositions, through the use of the MAUT element of MATE, the TOS designs, which recall are fractionated spacecraft, are compared on the basis of *static* lifecycle cost (see Section 3.3.3) and utility (benefit). Subsequently, the TOS studies became the earliest investigation of fractionated spacecraft in academia, albeit indirectly.

2002

Diller: MIT

Directly following the TOS studies, another assessment of fractionated spacecraft was conducted by Nathan Diller, the most comprehensive summary of his research being published in his Master's thesis at MIT (Diller, 2002). The focal objective his research was to develop and demonstrate a methodology that builds on MATE by combining MATE with integrated concurrent engineering (ICE). (ICE was used during the C-TOS study.) The methodology Diller created is called MATE-CON (Multiple Attribute Tradespace Exploration with Concurrent Design) and it exploits the benefits of MATE and those gained by employing ICE. In order to demonstrate MATE-CON in the domain of aerospace systems, X-TOS was used as a case study in Diller's work. The major contribution of this research with regard to an understanding of fractionated spacecraft is a cost-benefit quantification of fractionated spacecraft (*i.e.,* X-TOS), where cost and benefit are quantified via the metrics of *static* lifecycle cost and utility respectively.

[4] As compared to cost-centric methodologies, the MATE methodology more appropriately incorporates the needs/objectives/preferences of beneficiary stakeholders in spacecraft design (Diller, 2002).

2003

Ross: MIT

As a follow on to both the TOS studies and Diller's research, Adam Ross further matured and then applied the MATE-CON methodology to the X-TOS study as part of his Mater's research at MIT (Ross, 2003). Through his research, a systematic demonstration of the MATE-CON methodology was performed using numerous candidate X-TOS designs. The major contributions of this research with regard to an understanding of fractionated spacecraft is the application of MATE-CON to fractionated spacecraft assessments as well as a further investigation into the *static* lifecycle costs and benefits of fractionation in the context of Earth observation systems.

2006

Ross: MIT

As part of his doctoral research at MIT, Adam Ross further extended the MATE, not MATE-CON, methodology to be able to evaluate spacecraft dynamically (Ross, 2006). Through his research and subsequent development of an extended MATE methodology, called Dynamic MATE, the benefit provided by a (fractionated) spacecraft over a period of time (*e.g.,* a lifecycle) can now be quantified. The implications of Ross's research are to provide a means for understanding spacecraft value delivery, not in a static context, but rather in a dynamic (time-dependent) context. As with his Master's research, Ross used X-TOS as a case study to demonstrate the extended MATE methodology. The major contribution of this research with regard to an understanding of fractionated spacecraft is quantifying certain diffuse benefits of fractionated spacecraft, that is, benefits that emerge and/or change over the course of a lifecycle.

2006

Mathieu: MIT

All research in academia discussed thus far used fractionated spacecraft to demonstrate (prove) a methodology, thereby emphasizing the methodology rather than explicitly forming insights with regard to fractionated spacecraft. The only exception to this is Charlotte Mathieu's work. Mathieu's research is therefore unique and subsequently represents the most important research effort to date in academia for understanding the implications of fractionated spacecraft. The general focus of her research was assessing several fractionated spacecraft relative to a comparable monolithic spacecraft (Charlotte Mathieu, 2006; C. Mathieu & Weigel, 2005, 2006). Mathieu's research specifically evaluated 1 monolithic and 11 fractionated spacecraft architectures, each of which is different from the others either by the number of modules and/or use of shared resources (see Section 2.1.5). The costs and benefits of all 12 spacecraft were primarily assessed for two different missions: communications and navigation. Mathieu's research, similar to all previous research in academia cited herein, employed the use of the utility metric for quantifying the benefit of spacecraft as well as the use of *static* lifecycle cost (see Section 3.3.3) to quantify the cost associated with spacecraft. The major conclusions of her research are that fractionated spacecraft can, in certain situations, provide more benefit than monolithic spacecraft; however, regardless of the benefit provided, fractionated spacecraft are always more costly and massive.

2.3.2. Industry and Government

1984

Molette: MATRA Espace

The first research directly assessing fractionated spacecraft was published in 1984 (Molette, Cougnet, Saint-Aubert, Young, & Helas, 1984). This research and subsequent publication specifically compared monolithic and fractionated satellites from a technical and economical perspective via a predominantly qualitative cost-benefit analysis. For the purposes of this research, the monolithic and fractionated were assumed to be performing a GEO telecommunications mission. The major conclusions from this research are that the cost of a fractionated satellite is around 1.44 times that of monolithic satellite, but the technical benefits (*i.e.,* ability for the system to adapt to new missions, growth potential, flexibility, orbit maintenance, and availability) of fractionated satellites is higher than a monolithic satellite.

2004

Brown: DARPA

The first of several Owen Brown-lead investigations of monolithic and fractionated spacecraft was published in 2004 (Brown, 2004). The motivation for his research and subsequent analysis is the hypothesis that fractionated spacecraft will have a lesser lifecycle cost *risk* than comparable monolithic spacecraft. Through his research, specific elements of monolithic and fractionated spacecraft lifecycle costs were quantified (*e.g.,* launch and NRE) along with their respective risk. For a given spacecraft, the lifecycle cost and its respective risks were determined by assuming statistical probability distributions (*e.g.,* a negative binomial distribution) for the failure of monolithic and fractionated spacecraft over their respective lifecycles. Therefore, the results of this research provided the first stochastic cost-benefit assessment of fractionated spacecraft, where cost and benefit are quantified as the lifecycle cost and risk (due to lifecycle cost) respectively. There are two major conclusions from Brown's research. First, the lifecycle cost of fractionated spacecraft will vary depending on the cost savings gained from production learning in building multiple fractionated spacecraft/modules. And second, fractionated spacecraft can, but do not always, have smaller lifecycle costs and lifecycle cost risks relative to a comparable monolithic spacecraft, especially as the number of modules produced for the fractionated spacecraft increases. Subsequently, economies of scale were identified through Brown's work as a crucial factor in enabling fractionated spacecraft to be less expensive than monolithic spacecraft.

2006

Brown: DARPA

Shortly after his assessment of fractionated spacecraft published in 2004, Owen Brown collaborated on a new, albeit slightly different, investigation of the costs and benefits of monolithic and fractionated spacecraft (Brown, Eremenko, & Roberts, 2006). Through this research a cost-value (note, not cost-benefit) analysis was conducted for monolithic and fractionated spacecraft performing a communications mission. In contrast to Brown's 2004 study, this research did not account for the stochastic nature of spacecraft lifecycles in the cost-value analysis. In this study, the lifecycle cost is broken down into NRE, RE, launch, and operations costs; and the value is quantified in dollar terms via assumptions regarding the dollar worth a several attributes of monolithic/fractionated spacecraft (*e.g.,* flexibility). In the results, the cost and value of monolithic and fractionated spacecraft are quantified over the mission lifetime; subsequently these spacecraft can be compared based on their respective cumulative cost incurred, and cumulative value delivered profiles. There are two major conclusions from Brown's research. First, the cumulative costs incurred for monolithic spacecraft always exceeds its cumulative value delivered. And second, for the majority of the mission lifetime, the cumulative cost incurred by fractionated spacecraft

exceeds the cumulative value delivered - but - there is a point, late in the mission lifetime, in which their cumulative value delivered exceeds their cumulative costs incurred. Therefore, if long mission lifetimes are considered (*i.e.,* missions in excess of 14 years), fractionated spacecraft may prove to deliver more value than cost over their respective lifetime; something a comparable monolithic spacecraft is less capable of.

2007

Brown: DARPA

Building on his other two investigations of fractionated spacecraft, Owen Brown conducted yet another collaborative investigation in which he performed a stochastic assessment of monolithic and fractionated spacecraft. This research was published in 2007 (Brown et al., 2007). In this research, monolithic and fractionated spacecraft were compared based on a metric called the Stochastic Lifecycle Cost (SLCC). The SLCC is a dollar value metric that quantifies a spacecraft's inherent level of flexibility as well as its lifecycle cost; here flexibility, like Brown's 2006 research is converted to dollar terms using a set of assumptions. In this research, three types of spacecraft were considered: monolithic, fractionated, and hybrid (*i.e.,* a spacecraft that is initially monolithic in nature until it reaches the destination orbit, at which point it "breaks apart" and becomes a fractionated spacecraft). Each of these spacecraft designs and their subsequent SLCC were quantified stochastically (via a MCA) twice, once for a National Oceanic & Atmospheric Administration (NOAA) mission, and then a second time for a Geostationary Operational Environmental Satellites (GOES) mission. There are three major conclusions from this research. First, the SLCC of fractionated spacecraft is comparable to the SLCC of monolithic spacecraft. Second, the SLCC of fractionated spacecraft is inversely proportional to its respective degree of fractionation. And third, hybrid spacecraft are a very attractive option because despite their comparable SLCC to that of a monolithic spacecraft, they provide a much higher level of flexibility (in dollar terms).

2008-Present

DARPA: System F6 Program

Presently the most considerable assessment of fractionation, in terms of monetary resources and time, is being conducted through the System F6 (Future, Fast, Flexible, Fractionated, Free-Flying) Program lead by the Defense Advanced Research Projects Agency (DARPA) (Defense Advanced Research Projects Agency, 2008; Brown & Eremenko, 2008; Shah & Brown, 2008). The F6 program has a long-term objective of demonstrating that monolithic spacecraft can effectively be replaced by a set of smaller spacecraft modules, that is, a fractionated spacecraft (see Figure 2-17). Large-scale implementation of the work resulting from the DARPA F6 Program could lead to a large shift in the design, development, deployment, and subsequent operation of spacecraft for commercial and military space missions.

The first phase of the F6 Program was a year long, running from January 2008 until January 2009 and the DARPA program manager was Owen Brown. Four industry-lead teams participated in the first phase: The Boeing Company, Lockheed Martin Space Systems Company, Northrop Grumman Space & Mission Systems Corporation, and Orbital Sciences Corporation. There were four top-level objectives for the first phase of the F6 program. First, mature key technologies necessary for fractionated spacecraft development and operation. Second, develop a fractionated spacecraft design/concept that can accomplish a mission that is of value and importance to United States national security. Third, create an innovative and analytical approach to compare the risk-adjusted, net value of monolithic to fractionated spacecraft, for a given mission that makes use of econometric tools. And fourth, develop a hardware-in-the-loop (HIL) test bed to emulate the actual operation fractionated spacecraft in space.

For the second F6 program phase, which is set to begin in September 2009 and last roughly 18 months, only one of the four teams that participated in the first phase was selected to participate. The team selected for, and specific objectives of, the second phase of the F6 Program have not been publicly released yet.

Figure 2-17. A fractionated spacecraft as envisioned by DARPA.

Image Source: DARPA, Owen Brown

The aspect of the F6 program ensuring that it will be an exciting venture to follow in the future is that it is the single largest research and development effort having the objective of assessing and demonstrating the fractionated spacecraft concept. With DARPAs initial investment of 38.5 $M in the first phase of the F6 program alone, and the likelihood of the second phase producing more tangible instantiations of fractionated spacecraft, who knows what will come of the F6 program – perhaps a full-scale demonstration of fractionated spacecraft in space.

2.3.3. Limitations

In assessing (*i.e.,* quantifying) the value propositions of monolithic and fractionated spacecraft there is an inherent tradeoff between the (1) fidelity of the assessment, (2) appropriateness of the assessment (relative to the questions being addressed by the assessment), (3) and resources required for the assessment. As is in the case of all tradeoffs, there is not an optimal balance between these three objectives of the assessment of spacecraft. It is in lieu of this, recognize that all previous assessments of fractionated spacecraft discussed in the literature review (see Section 2.3.1 and 2.3.2) have implicitly made this tradeoff such that the most appropriate mix of fidelity, appropriateness, and resources have been chosen on the basis of the specific objectives of those respective assessments. Therefore, the limitations of these previous assessments that will subsequently be cited herein and, furthermore, serve as motivation for this research, are due to differences in the assessment objectives (*i.e.,* fidelity, appropriateness, and resources) of this research investigation from the objectives of previous assessments of fractionated spacecraft. Subsequently, the limitations of these previous assessments cited hereafter should not be perceived as a means of diminishing the value of these assessments, but rather as a constructive extension of them.

The objectives for the assessment of monolithic and fractionated spacecraft set forth for this research effort were made on the basis of assessing the fractionated spacecraft concept aggressively, that is, the fidelity and appropriateness of the assessment were weighted far heavier than the consequent resources (*i.e.,* time) required to achieve the fidelity and appropriateness. It is recognized that the assessment objectives set forth

by this research investigation are by no means optimal; significant insight can still be gained about monolithic and fractionated spacecraft value propositions for significantly less resources than were expended in this research, albeit with lesser fidelity and/or appropriateness. Therefore, relative to the specific objectives of this research assessment, each of the ten previous assessments of fractionation cited in the literature review has at least one of the following three crucial limitations.

1. **Narrow scope of the value proposition.**
    a. Low fidelity models
        i. Parametric models or design-by-analogy
    b. Small number of fractionated spacecraft architectures investigated

2. **Lack of dynamic lifecycle considerations**

3. **Minimal focus on cardinal measures of benefit and/or value**

As is discussed in Section 2.1.9, the value proposition is paramount to understanding the implications of fractionated spacecraft relative to comparable monolithic spacecraft. Forming the value proposition for a spacecraft depends entirely on the assessment of that spacecraft, and generally, the more fidelity and appropriateness in an assessment, the more plentiful the supply of metrics characterizing a spacecraft that can be used to form the value proposition. In the case when only a small number of metrics are yielded from an assessment, some of the potentially crucial costs and benefits of a spacecraft may be not accounted for in the value proposition. For example, in a few of the previous assessments of fractionated spacecraft cited in the literature review, the value proposition used to compare monolithic and fractionated spacecraft consisted of only *one* metric, lifecycle cost, thereby not capturing any of the non-cost related attributes of monolithic and fractionated spacecraft in the value proposition. Additionally, in a few of the previous assessments only one, or less than a handful, of fractionated spacecraft architectures (designs) were investigated, which further narrows the value proposition for fractionation. Therefore, with this <u>narrow scope of the value proposition</u> one has to wonder if the *costs and benefits* of monolithic and fractionated spacecraft are adequately captured with a single metric and by examining only one (or a few) fractionated spacecraft architectures. Therefore, it is desirable to have access to a larger, rather than smaller, number of metrics for which to form the value proposition for monolithic or fractionated spacecraft, and in doing so <u>broaden the scope of the value proposition</u>. Additionally, the value proposition <u>can be broadened</u> by having an assessment that can consider a large number of fractionated spacecraft architectures such that multiple value propositions for a certain class of spacecraft architectures (*e.g.,* fractionated spacecraft with two modules) can be generated.

While broadening the scope of the value proposition is ideal, be mindful that the broader the scope, the more fidelity often required of the assessment, which in turn increases the resources required for the assessment. Subsequently, there have been two specific reasons for the narrow scope of the value proposition observed in some of the previous assessments of fractionation. First, the employment of low fidelity models (*e.g.,* parametric or design-by-analogy models), which yield a limited number of metrics for which to compose the value proposition. And second, the consideration of a small number (≤ 6) fractionated spacecraft to compare against one monolithic spacecraft; this consequently fails to capture an appropriate cross-section of fractionated spacecraft value propositions, given the plethora of potential fractionated spacecraft architectures (designs).

The second limitation of many previous assessments of fractionation is the lack of dynamic lifecycle considerations in the assessment of spacecraft value propositions. Many previous assessments cited in the literature review have considered spacecraft lifecycles as existing in a static context, meaning that the lifecycle uncertainties and consequent risks present in any given spacecraft lifecycle are not accounted for. The implication of not quantifying the dynamic lifecycle for the assessment of monolithic and fractionated spacecraft value propositions is that the benefits and costs of these respective spacecraft are computed assuming a perfect lifecycle. Consequently, these assessments completely fail to quantify/address many of the purported benefits (and costs) of fractionated spacecraft (see Section 2.2.1).

And the third limitation of previous assessments of fractionation is a minimal focus on *cardinal* measures of benefit and/or value. The majority of the previous assessments of fractionation cited in the literature review, especially in academia, rely exclusively on the metric of utility to quantify and compare the benefit (or lack thereof) of monolithic and fractionated spacecraft. While utility is a useful metric for aggregating benefit it has two critical disadvantages. First utility is an ordinal measure, meaning that it has *no* statistical significance and thus utility can *only* be used rank designs (*e.g.,* design A is more beneficial than design B) (von Neumann & Morgenstern, 1953; Keeney & Raiffa, 1993). Second, because utility is an ordinal measure of benefit, it is statistically meaningless in quantifying value (*i.e.,* benefit normalized by cost). Therefore, comparisons made between of monolithic and fractionated spacecraft value propositions should focus on quantifying benefit and value using cardinal metrics (*i.e.,* not utility) so that statistically meaningful comparisons of benefit and/or value can be made between monolithic and fractionated spacecraft.

Based on the discussion of these three limitations,
Table 2-2 lists each of the previous assessments of (monolithic) and fractionated spacecraft cited in the literature review in Section 2.3.1 and 2.3.2, the year the research was published, its major contribution(s), and inherent limitations.

Table 2-2. Previous assessments of fractionated spacecraft: contributions and limitations.

Author/Project	Publication Year	Contributions	Known Limitation(s)
Academia			
TOS Studies	2001, 2002	MATE/ICE development Application of utility to quantify benefit Static lifecycle cost assessment	Narrow scope of the value proposition Lack of dynamic lifecycle considerations Minimal focus on cardinal measures of benefit/value
Diller	2002	Development of MATE-CON Application of utility to quantify benefit Static lifecycle cost assessment	Narrow scope of the value proposition Lack of dynamic lifecycle considerations Minimal focus on cardinal measures of benefit/value
Ross	2003	Further development and application of MATE-CON Application of utility to quantify benefit Static lifecycle cost assessment	Narrow scope of the value proposition Lack of dynamic lifecycle considerations Minimal focus on cardinal measures of benefit/value
Ross	2006	Development of Dynamic MATE Application of utility to quantify benefit Static lifecycle cost assessment	Narrow scope of the value proposition Minimal focus on cardinal measures of benefit/value
Mathieu	2006	Assessment of 11 different fractionated spacecraft Investigation and assessment of shared resources Application of utility to quantify benefit	Narrow scope of the value proposition Lack of dynamic lifecycle considerations Minimal focus on cardinal measures of benefit/value
Industry and Government			
Molette	1984	First formal assessment of fractionated spacecraft Qualitative-based conclusion of fractionation benefits Investigation of several fractionated architectures	Narrow scope of the value proposition Lack of dynamic lifecycle considerations Minimal focus on cardinal measures of benefit/value
Brown	2004	Probabilistic cost-risk assessment of fractionation Risk due to cost used as a proxy for benefit Consider production learning effects	Narrow scope of the value proposition Minimal focus on cardinal measures of benefit/value
Brown	2006	Quantitative breakdown of lifecycle cost elements Quantified value of fractionated spacecraft Cumulative cost/value comparisons	Narrow scope of the value proposition Minimal focus on cardinal measures of benefit/value
Brown	2007	Examination of hybrid spacecraft Development of the SLCC metric for comparison Stochastic simulation via MCA	Narrow scope of the value proposition Minimal focus on cardinal measures of benefit/value
DARPA	2008-Present	Extensive risk-adjusted investigation of fractionation Investigation of key fractionated technologies Development of HIL test bed for fractionated spacecraft	Unknown

2.4. Problem Statement and Research Contributions

The problem statement and research contributions comprise the fourth aspect of the problem formulation for this research. The problem statement addresses the limitations of previous assessments of monolithic and fractionated spacecraft value propositions and in doing so explicitly enumerates the unique contributions of this research. Subsequently, the problem statement and research contributions are interwoven and thus presented in sequence.

Problem Statement

Given the limitations of previous research efforts in assessing the value propositions for monolithic and fractionated spacecraft, there is a need for…

Research Contributions

1. A high fidelity, bottom-up, dynamic quantitative assessment of monolithic and fractionated spacecraft value propositions.
2. An understanding of the monolithic and fractionated spacecraft value propositions using cardinal, "traditional" measures of effectiveness (MoE)
3. The ability to explore monolithic and fractionated spacecraft value propositions in both breadth and depth.

In achieving these contributions, this research mitigates being susceptible to all three crucial limitations possessed in the previous assessments of fractionation cited in the literature review (see Section 2.3).

2.5. Research Questions

The remaining aspect of the problem formulation is to define a set of research questions to guide the remaining three phases of the research methodology: Modeling, Analysis, and Synthesis. Based on the four elements of the problem formulation, with emphasis on the limitations of previous assessments of fractionation (Section 2.4), the research questions are as follows:

Research Questions

1. How do the value propositions for monolithic and fractionated spacecraft compare **across alternative spacecraft architectures (designs)**?

2. How do the value propositions for monolithic and fractionated spacecraft compare **relative to changing payload requirements** (*i.e.,* ground resolution)?

3. How do the value propositions for monolithic and fractionated spacecraft compare **relative to risks resulting from spacecraft lifecycle uncertainties** (*e.g.,* on-orbit failure)?

The first research question motivates the need for a high fidelity, bottom-up, quantitative assessment (*i.e.,* the first research contribution) of fractionation. The objective of this question is to explore the value propositions for a large number of fractionated spacecraft architectures that vary in module number and/or module subsystem composition. This is only possible with a high fidelity, non-parametric, model in which monolithic and fractionated spacecraft architectures are built from the "ground up", meaning they are built (and modeled) component-by-component, subsystem-by-subsystem, and module-by-module.

The second research question motivates the need to understand how the value propositions for monolithic and fractionated spacecraft change relative to their payload requirements. Since RSMs are driven by the design of the spacecraft payload, this research question has the objective of quantifying the extent to which payload requirements influence monolithic and fractionated spacecraft value propositions.

The third research question pinpoints one of the vital aspects of fractionated spacecraft, that is, the nature in which the value proposition changes relative to differing severities of the spacecraft lifecycle; here severity is defined as the probability that the risks resulting from lifecycle uncertainties will occur, hence a more severe lifecycle will have risks that occur more often. One hypothesis with regard to the third research is that as the lifecycle becomes more severe, the value propositions for fractionated spacecraft will become stronger (*i.e.,* be perceived as more valuable by beneficiary stakeholders) relative to that of a monolithic spacecraft. The responses formed with regard to the third research question are perhaps of most interest to individuals who want fractionated spacecraft to be "put through its paces", that is, placing fractionated spacecraft in the worst of situations to see how it fairs relative to monolithic spacecraft.

With respect to previous assessments of fractionated spacecraft cited in the literature review (see Section 2.3); the first research question has been addressed in part, albeit at a conceptual level. However, in contrast, none of previous assessments of fractionated spacecraft have explicitly addressed the second and third research questions posed herein, thereby making quantitative responses to these questions, as is done through this research effort, a unique contribution of knowledge.

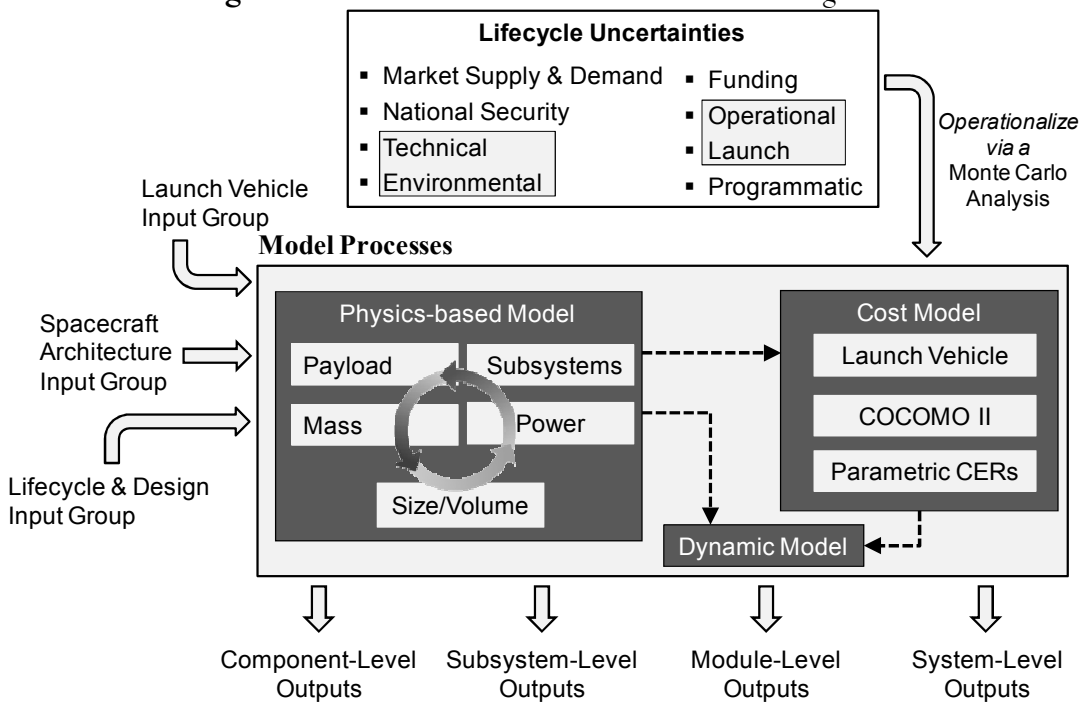
3. Modeling: The Spacecraft Evaluation Tool

Modeling is the second phase of the research methodology (see Section 1.1.2). Subsequently, Chapter 3 is devoted to a discussion of the modeling phase, more specifically the physical instantiation of the modeling phase, the Spacecraft Evaluation Tool (SET). The research questions are motivated by the problem statement (see Section 2.4), and appropriate quantitative responses to these research questions are formulated by applying the SET. Therefore, the SET forms the crucial link between the research methodology's Phase I: Development, and Phases III & IV: Analysis & Synthesis. As such, Chapter 3 will focus on discussing the development and applicability of the SET. There are three distinct functional partitions of the SET: inputs, model processes, and outputs; these are the subject of Section 3.2, 3.4, and 3.5 respectively. However, before a discussion of these three SET partitions commences, an overview of the SET is given.

3.1. SET Overview

The SET is a software program developed entirely by the author, thereby not employing models developed by others, the only exception being the parametric cost model embedded in the SET. The SET is embodied in a software program that uses a Microsoft Excel® and Matlab® integrated programming language platform and graphical user interface (GUI). The three distinct functional partitions of the SET are inputs, model processes (simulation), and outputs. An overview the SET functional flow is given in Figure 3-1.

Figure 3-1. SET overview: functional flow block diagram.



The SET inputs, as shown in Figure 3-1, are informed by the three research questions. Recall, that research questions seek to understand the value propositions of monolithic and fractionated spacecraft relative to specific spacecraft architectures (designs), payload requirements, and lifecycle uncertainties. Therefore, the SET inputs must provide adequate metrics (degrees of freedom) such that the research questions can be appropriately characterized in the SET. A detailed discussion of the SET inputs is provided in Section 3.2.

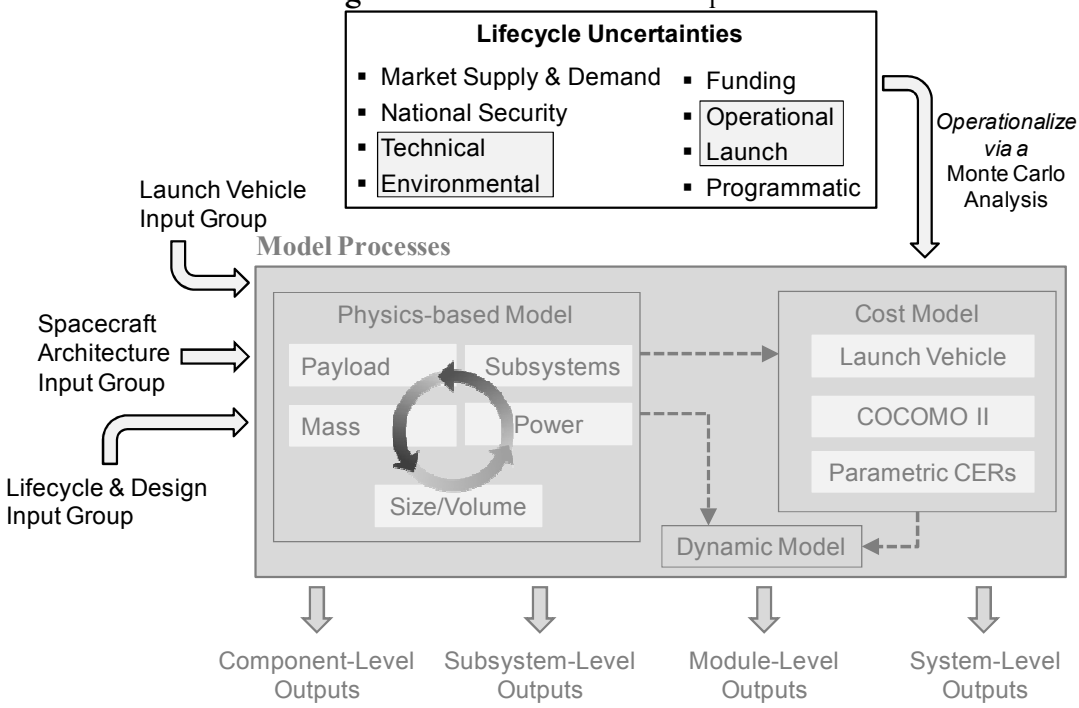
The SET outputs shown in Figure 3-1 are based on the desirable metrics for which to compose the value proposition for monolithic and fractionated spacecraft. Subsequently, the SET outputs must provide adequate metrics (information) for the monolithic and fractionated spacecraft value propositions so that appropriate, quantitative responses to the research questions can be formulated. A detailed discussion of the SET outputs is provided in Section 3.4.

The SET transforms the SET inputs into SET outputs via model processes (simulation) that consists of three models. The first model in the SET is a physics-based model and it models a spacecraft's hardware constituents. The second model in the SET is the cost model and it contains three major model processes that quantify the static lifecycle cost (LCC) of a given spacecraft. And the third model in the SET is the dynamic simulation model, which uses information from the physics-based and cost models to quantify the dynamic LCC of a spacecraft. A detailed discussion of the SET model processes is given in Section 3.3.

3.2. SET Inputs

The objective of the SET inputs is to characterize the context for monolithic and fractionated spacecraft value propositions (see Section 2.1.9). Therefore, the context, and thus SET inputs drive the resulting value proposition for a given spacecraft. The SET inputs were selected on the basis of providing adequate metrics (degrees of freedom) such that the research questions can be appropriately characterized by the SET inputs. For example, the second research question addresses the implications of monolithic and fractionated spacecraft value propositions relative to changing payload requirements; hence, there will need to be SET inputs that enable the changing of payload requirements. In this sense, the SET inputs can be thought of as "knobs" that can be "turned" to alter the value proposition (context) for a given spacecraft. The SET inputs are highlighted relative to the SET model processes and outputs in Figure 3-2.

Figure 3-2. Overview of SET inputs.



The SET inputs are most easily synthesized and discussed by categorizing them as belonging to one of three groups of inputs: Launch Vehicle, Lifecycle & Design, and Spacecraft Architecture. In total, for a given SET simulation, it requires the specification of 155 independent SET input values: 22 Launch Vehicle, 21 Lifecycle & Design, and 112 Spacecraft Architecture inputs. The Launch Vehicle, Lifecycle & Design, and Spacecraft Architecture SET inputs groups are discussed in Section 3.2.1, 3.2.2, and 3.2.3 respectively.

3.2.1. Launch Vehicle

The Launch Vehicle group of SET inputs specifies the number and type of launch vehicles that can be used for initial deployment and subsequent replenishments of spacecraft/modules throughout the lifecycle. Presently, the SET allows for the selection of up to 22 candidate launch vehicles for a given lifecycle simulation, a maximum of three can be used at any one time for deployments/replenishments. Table 3-1 provides a listing of the launch vehicles available to select from in the SET as well as their respective country of origin. The pertinent information for each respective launch vehicle, as is required for the launch vehicle selection model embedded in the SET, includes launch vehicle cost, stage masses/mass fractions, payload fairing dimensions, launch site latitude, and reliability (success rate). All of this data pertaining to each launch vehicle was obtained directly from launch vehicle manufacturers if possible and if not, from Steven Isakowitz's *International Reference Guide to Space Launch Systems* (Isakowitz, Hopkins, & Hopkins Jr., 2004).

Table 3-1. SET launch vehicle database[5].

Launch Vehicle	Country of Origin
Athena I	**United States**
Falcon I	
Minotaur	
Pegasus XL	
Scorpius	
Start-1	**Russia**
Athena II	**United States**
Delta II	
Commercial Taurus	
Angara 1.1	**Russia**
Kosmos 3M	
Rockot	
Strela	
Vega	**Italy/Europe**
Angara 1.2	**Russia**
Cyclone 2	**Ukraine and Russia**
Dnepr	
Long March 2C	**China**
PSLV	**India**
Atlas V 500	**United States**
Delta IV M+	
Titan IVB	

For a given launch vehicle payload (*i.e.*, module or set of modules) and the launch vehicles permitted for use, the launch vehicle selection model process selects between one and three launch vehicles that can collectively fit the launch payload constituents, both in terms of the launch vehicle payload mass and

[5] As of August 2009 the following launch vehicles are either no longer available, have no planned launches in the near future, or are still under development: Scorpius, Angara 1.1, Vega, Angara 1.2, Cyclone 2, Delta IV M+, and Titan IVB. These launch vehicles are still modeled in the SET, but as will be enumerated in Section 4.1.1, were ultimately not considered for the simulation of monolithic and fractionated spacecraft value lifecycles in the SET.

physical dimensions given the destination orbit and launch site latitude(s). The criteria for the launch vehicle(s) selection are to minimize and maximize the aggregate launch vehicle cost and reliability respectively. The launch vehicle selection model process therefore mimics the behavioral characteristics of a cost and risk-averse individual.

It is worth mentioning that fidelity of the launch vehicle selection model employed in the SET is likely beyond that of most previous assessments of fractionated spacecraft, for three specific reasons. The SET launch vehicle selection model (1) gives significant treatment to the physical dimension and mass implications of a spacecraft/module relative launch vehicle payload fairings; (2) accounts for the influence of launch vehicle launch site latitude and destination orbit altitude and inclination on the maximum mass that can delivered to the orbit; and (3) incorporates launch vehicle reliability. Therefore, the assessments of spacecraft in the SET do not assume that a certain launch vehicle will just "work" all the time, regardless of the spacecraft/modules design, mass, etc., and instead rigorously analyze the interaction between launch payloads and launch vehicles. By avoiding this assumption, the SET can effectively discriminate launch vehicle usage based on the respective size and mass of the respective spacecraft/modules to be deployed. And this proves, as is exemplified in Chapter 4, to be a crucial factor in fractionated spacecraft maintaining LCC-competiveness with monolithic spacecraft.

3.2.2. Lifecycle & Design

The second group of SET inputs, the Lifecycle & Design inputs (see Table 3-2), of which there are 21, define the lifecycle (mission) context in which a given monolithic or fractionated spacecraft operates as well as certain parameters governing the design of these spacecraft. Therefore, the Lifecycle & Design inputs drive the value proposition for a given spacecraft. Given that the Lifecycle & Design inputs characterize some of the major elements of a spacecraft's lifecycle, they include, for example, parameters describing a spacecraft mission (*e.g.,* orbit altitude), spacecraft's respective design (*e.g.,* autonomy level), a spacecraft's respective pointing requirements (*e.g.,* pointing tolerance), and the stochastic lifecycle of a spacecraft (*e.g.,* lifecycle uncertainties to consider).

Each of the 21 SET inputs shown in Table 3-2 is a "knob" that can be subsequently "turned" to change the context in which a spacecraft operates and subsequently change its respective value proposition. Specifically within the Lifecycle & Design group of inputs, there are 13 input categories: (1) orbital parameters, (2) CONOPS, (3) autonomy level, (4) lifetime, (5) sizing, (6) payload performance, (7) pointing requirements for the pointing-intensive modules, (8) pointing requirements for the *non-pointing-intensive* modules, (9) dynamic lifecycle simulation, (10) production, (11) lifecycle uncertainties, (12) launch, and (13) mission lifetime extension. A brief definition for each of the 21 SET Lifecycle & Design inputs spanning these 13 input categories is given hereafter.

Table 3-2. SET Lifecycle & Design inputs.

Input Category	-	Input	Units	Run Value
Orbital Parameters	1	Orbit Altitude	km	700
	2	Orbit Inclination	degrees	98
CONOPS	3	Cluster Separation Distance	m	20
Autonomy Level	4	Bus Autonomy Level	-	1
	5	Payload Autonomy Level	-	2
Lifetime	6	Mission Lifetime	years	7
Sizing	7	**Min** Packing Efficiency	m3/kg	0.014
Payload Performance	8	Linear Ground Resolution (i.e. Pixel Size)	m	0.5
Pointing Requirements for Pointing-Intensive Modules	9	Pointing Tolerance	deg.	0.00001
	10	Spacecraft Jitter	deg.	0.00014
	11	Slew Maneuver No.	-	12
	12	Slew Maneuver Magnitude	deg	90
Pointing Requirements for Non-Pointing-Intensive Modules	13	Pointing Tolerance	deg.	0.0005
	14	Spacecraft Jitter	deg.	0.00014
Dynamic Lifecycle Simulation	15	No. of MCA Trials	-	2,500
	16	Probability of Infant Mortality (PoIM)	-	0.015
Production	17	Build time learning factor	-	0.65
Lifecycle Uncertainties	18	Launch	-	1
	19	Technical, Environmental, and Operational	-	1
Launch	20	No. of LVs	-	3

Mission Lifetime Extension	21	Monolithic Mass (Datum)	kg	0.00

1. Orbit Altitude: distance between the surface of the Earth and a spacecraft
2. Orbit Inclination: angle between the orbital plane and Earth equator (measured clockwise from East)
3. Cluster Separation Distance: distance between fractionated spacecraft modules on-orbit
4. Bus Autonomy Level: capability of bus to perform automated processing and tasking
5. Payload Autonomy Level: capability of payload to perform automated processing and tasking
6. Mission Lifetime: length of the operational mission from BoL to EoL
7. Min Packing Efficiency: volume/mass metric governing launch vehicle packing density
8. Linear Ground Resolution: linear size of one pixel length for images captured of the Earth
9. Pointing Tolerance: a set allowance for an amount of pointing error (lack of pointing accuracy)
10. Spacecraft Jitter: high frequency pointing misalignment measured as an angular disturbance
11. Slew Maneuver No.: number of in-situ rotational displacements performed during each orbit
12. Slew Maneuver Magnitude: rotational angular displacement of slew maneuvers
13. Slew Maneuver No.: number of in-situ rotational displacements performed during each orbit
14. Slew Maneuver Magnitude: rotational angular displacement of slew maneuvers
15. No. of MCA Trials: number of Monte Carlo Analysis trails employed for the dynamic simulation
16. Probability of Infant Mortality: probability that a spacecraft fails within the first year of its mission
17. Build Time Learning Factor: learning factor that reduces the time if rebuilding a spacecraft/module
18. Launch: designate whether the launch vehicle lifecycle uncertainty should be considered
19. Technical, Environmental, and Operational: designate whether the technical, environmental, and operational lifecycle uncertainties should be considered
20. No. of LVs: specify the maximum number of launch vehicles that can be used for a given deployment
21. Monolithic Mass (Datum): value required for mission lifetime extension study (see Section 4.2.4)

The objective of enumerating the 21 Lifecycle & Design inputs in Table 3-2 is to emphasize the capability of the SET to explore a broad range a spacecraft value proposition contexts, thereby enabling the appropriate characterization of the three research questions (see Section 2.5) as well as questions beyond the scope of this research investigation.

3.2.3. Spacecraft Architecture

The Spacecraft Architecture group of SET inputs comprehensively defines the monolithic and fractionated spacecraft architectures (aka designs) to be assessed by the SET. To define (build) a given spacecraft architecture, 112 spacecraft architecture-related inputs must be specified. These inputs include (1) the number of modules; and then for each module (2) the composition of subsystems; (3) the use of shared resources (*i.e.*, is the module a shared resource source or recipient); (4) whether it has a RSM payload; and (5) whether it has a S/C-G directional antenna. As such, each spacecraft is built from the "ground up", that is, component-by-component, subsystem-by-subsystem, and module-by-module. The fidelity of the SET therefore enables the discrimination and subsequent understanding of nuances in monolithic and fractionated spacecraft architectures down to the component-level (sub-subsystem level). For fractionated spacecraft, if a module contains a RSM payload it is referred to as a *Payload Module*, otherwise it is referred to as an *Infrastructure Module*. The manner in which the Spacecraft Architecture inputs are specified in the SET to "build" a given monolithic or fractionated spacecraft is shown in Table 3-3.

Table 3-3. SET monolithic and fractionated spacecraft architecture specification (building).

Module Specification		Shared Resources			WPD	"Traditional" Bus Subsystems					Mission-Specific	
Type	**Module Number**	**Comm_CS _C&DH**	**ADS_ GNS**	**Power**	**Power Source?**	**Propulsion**	**ACS & GCS**	**TCS**	**Structures**	**Wiring**	**S/C-Ground Antenna**	**RSM PL**
Infrastructure	**1**	1	1	1	1	1	1	1	1	1	1	0
Payload	**2**	0	0	0.5	0	1	1	1	1	1	0	1
Infrastructure	**3**	0	1	1	1	1	1	1	1	1	0	0
Infrastructure	**4**	1	0	1	1	1	1	1	1	1	0	0
Infrastructure	**5**	0	1	1	1	1	1	1	1	1	1	0
-	**0**	0	0	0	0	0	0	0	0	0	0	0
-	**0**	0	0	0	0	0	0	0	0	0	0	0
-	**0**	0	0	0	0	0	0	0	0	0	0	0
	Value Range	0/1	0/1	[0, 1]	0/1	1	1	1	1	1	0/1	0/1
No. Modules	**5**											

1 = has/YES
0 = does not have/NO

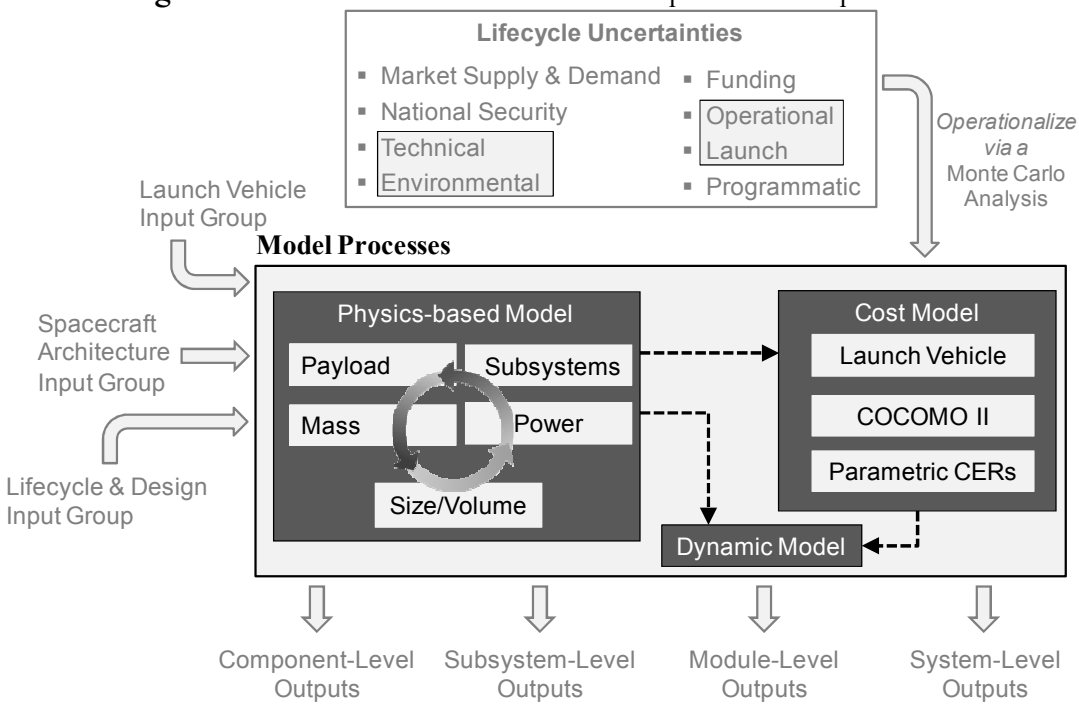
Table 3-3 illustrates how monolithic and fractionated spacecraft architectures are "built" in the SET using a five module fractionated spacecraft as an example. Moving from left to right in Table 3-3, along the top two rows. **(Type)** a "payload" module carries a mission payload whereas an "infrastructure" module supports the Payload Module(s) via shared resources, if applicable. **(Module Number)** There can be up to eight modules in a fractionated spacecraft (a monolithic spacecraft would obviously just have one-module). **(Shared Resources)** Following the specification of the module type, the use of shared resources can be specified (see Section 2.1.5). The Comm_CS_C&DH and ADS_GNS shared resources can either be used or not (yes/no), whereas the Power shared resource can be used in differing capacities; hence, the Power shared resource is a continuous input variable. For example, for a given module, specifying that the Power shared resource is 0.5 and 1 implies that the module produces and stores 50% and 100% of the power it requires respectively. **(Power Source?)** This designates modules as to whether or not they will generate and store power for any of the other modules in the fractionated spacecraft that are producing less power than they need, subsequently these modules will not be a power source. **("Traditional" Bus Subsystems)** are major subsystems that must be present on all modules. **(Mission-Specific Hardware)** Here the S/C-ground antenna and RSM payload are allocated amongst the modules in a fractionated spacecraft. There needs to be at least one S/C-ground directional antenna and RSM payload present in a monolithic or fractionated spacecraft; however, more than one S/C-ground antenna and RSM payload can be used if desired. Note that any module that does not have a dedicated S/C-ground antenna must be sharing the Comm_CS_C&DH resource, as it has no means of communicating directly with the ground. As a subsequent result of the manner in which the monolithic and fractionated spacecraft architectures are specified in the SET, as shown in Table 3-3, one spacecraft architecture is assessed at a time.

Given the fidelity at which monolithic and fractionated spacecraft architectures (designs) can be specified (built), the SET readily explores the implications of nuances in monolithic and fractionated spacecraft architectures with regard to the value proposition down to the component level (*i.e.,* sub-subsystem level). Additionally, an infinite number of fractionated spacecraft architectures can be explored given the Power shared resource being a continuous variable, thereby significantly increasing the scope of the value proposition that can be enumerated by the SET in terms of fractionated spacecraft.

3.3. Model Processes

Following the specification of the SET Launch Vehicle and Lifecycle & Design inputs, an assessment (simulation) of the monolithic or fractionated spacecraft, as specified by the Spacecraft Architecture input group, can commence. The SET simulation (*i.e.,* model processes) transform the SET inputs into the outputs through three successive models: physics-based, cost, and dynamic. An overview these three models and their major model processes relative to the SET inputs and outputs is given in Figure 3-3.

Figure 3-3. Overview of SET models and respective model processes.



Each of the three models in the SET contains several high-level (conceptual) model processes that in turn contain a multitude of sub-processes and sub-sub-processes, etc. not enumerated herein. However despite this, the three SET models and their respective model processes, as characterized by the design structure matrix (DSM) in Table 3-4, provide a succinct conceptual overview of the "inner workings" (flow of information) in the SET for a given simulation.

Table 3-4. Overview of SET model processes characterized in a design structure matrix[6].

Model		Model Process	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Inputs	1	SET Inputs																
Physics-Based Model	2	RSM Payload	X															
	3	Computer System, C&DH	X	X														
	4	Communications, TT&C	X		X													
	5	ADS, GNS	X															
	6	EPS	X								X							
	7	Propulsion, ACS, GCS	X					X				X	X					
	8	TCS	X	X	X	X	X	X	X				X					
	9	Power Required	X	X	X	X	X	X	X	X		X	X					
	10	Mass	X	X	X	X	X	X	X	X								
	11	Size, Volume	X	X	X	X	X	X	X	X		X						
Cost Model	12	LV Selection Model	X	X	X	X	X	X	X	X		X	X					
	13	COCOMO II	X	X														
	14	Parametric CERs	X	X	X	X	X	X	X	X		X						
Dynamic	15	Lifecycle Simulation (MCA)	X									X	X	X	X	X		
Outputs	16	SET Outputs		X	X	X	X	X	X	X	X	X	X	X	X	X	X	

Feedback Loops

Notation: command & data handling (C&DH); telemetry, tracking, & control (TT&C); attitude determination and guidance determination system (ADS, GNS); electric power system (EPS); attitude and guidance control system (ACS, GCS); launch vehicle (LV); cost estimating relationships (CERs); Monte Carlo Analysis (MCA)

There are five models in the SET. The first and fifth models are the SET Inputs (Section 3.2) and SET Outputs (Section 3.4). And the second, third, and fourth models are the Physics-Based Model (Section 3.3.1), Cost Model (Section 3.3.2), and Dynamic Lifecycle (Section 3.3.3).

As is evident by the DSM shown in Table 3-4, the SET model processes primarily occur in a feed-forward fashion, that is, most of the model processes rely on the outputs of previously completed processes. In developing the SET, it was attempted to keep the number of feedback loops to a minimum because feedback loops can greatly increase the amount of time required to run a simulation.

The 16 model processes shown in Table 3-4 are described hereafter relative to the specific aspect of a spacecraft design/lifecycle for which they are responsible.

1. SET Inputs: see Section 3.2.
2. RSM Payload: remote sensing mission payload design
3. Computer System, C&DH: design of a spacecraft's computer system and C&DH subsystem
4. Communications, TT&C: design of a spacecraft's communications and TT&C subsystem
5. ADS, GNS: design of a spacecraft's ADS and GNS
6. EPS: design of a spacecraft's EPS
7. Propulsion, ACS, GCS: design of a spacecraft's Propulsion System, ACS, and GCS
8. TCS: design of a spacecraft's TCS

[6] In a DSM, the X's in a given row represent the inputs required for the model process on that respective row, whereas the X's in a given column represent outputs from the model process in that respective column to other processes. For example, in Table 3-4 model process 7: propulsion and ACS/GCS requires inputs from processes 1, 6, 10, and 11, and outputs from the propulsion and ACS/GCS process are needed by processes 8-12 & 16.

9. Power Required: computing power profiles for daylight and eclipse periods
10. Mass: compute spacecraft and module masses
11. Size, Volume: compute spacecraft/module size & volume, and "packed" size and volume for LV
12. LV Selection Model: select launch vehicle combinations for lifecycle spacecraft deployments
13. COCOMO II: compute software development time and cost using a COCOMO II-based model
14. Parametric CERs: compute certain spacecraft cost elements on the basis of USCM8 CERs
15. Lifecycle Simulation: employ a MCA to simulate spacecraft lifecycle's and quantify lifecycle costs
16. SET Outputs: see Section 3.4

3.3.1. Physics-Based Model

One of the objectives of the SET was to provide a high-fidelity assessment of monolithic and fractionated spacecraft value propositions. Therefore, no aspect of the physics-based model relies on parametric models because the fidelity of the physics-based model must be appropriate to the level of fidelity at which spacecraft architectures are defined by the Spacecraft Architecture inputs (see Section 3.2.3). Subsequently, due to the fidelity of the Spacecraft Architecture inputs being down to the component level (*i.e.,* sub-subsystem level), as is the case when specifying the use of shared resources, the physics-based model similarly must have fidelity down to the component level. As such, within the physics based model spacecraft are built from the "ground up" component-by-component, subsystem-by-subsystem, and module-by-module.

The physics-based model consists of 10 high-level model processes, each of which contains numerous smaller scope processes that are not enumerated herein to keep the discussion tractable. The first four of these model processes proceed in a feed-forward fashion and are responsible for the design of the (2) RSM payload; (3) computer system and C&DH; (4) communications system and TT&C; and (5) the ADS and GNS. The remaining six physics-based model processes involve feedback loops, which were unavoidable given the dependency of those processes on the outputs of model processes occurring after them. These six model processes are responsible for the design of a spacecraft's respective (6) EPS; (7) Propulsion, ACS, GCS; (8) TCS; (9) Power Required; (10) Mass; and (11) Size and Volume. The six feedback loops were found to be the minimum achievable given the nature of the 10 physics-based model processes. To serve as a descriptive illustration of a feedback loop consider Model Process 8, that is, TCS. The TCS process relies on inputs from Processes 1-7 (these represent feed-forward relationships); however, the TCS process also relies on the output of model process 11: spacecraft size and volume determination, and this therefore represents a feedback relationship (aka loop).

A common approach to address feedback loops, as is employed in the SET, is employing an under and over-relaxed iterative procedure analogous to those employed in the field of Computational Fluid Dynamics for the convergence of flow field solutions (Lack, 2006). However, regardless of the approach used to deal with feedback loops it is always desirable to minimize them as they inevitably increase the time to execute the simulation, often significantly[7].

[7] A point of interest: The feedback loop portion of the SET simulation constitutes roughly 90% of the time required to run the simulation, with the remaining 10% begin consumed by all other feed-forward processes in the SET. This offers one instantiation of the adverse effects of feedback loops in which 40% of the modeling processes require 90% of the computation time.

3.3.2. Cost Model

Following the execution of the respective model processes in the physics-based model, the cost model computes the *static* lifecycle cost (LCC) of the spacecraft. The cost model consists of three model processes: launch vehicle selection model, COCOMO II, and parametric CERs. The *static* LCC assumes that the spacecraft will not fail or experience launch vehicle failures, due to lifecycle uncertainties, throughout its respective lifecycle. Therefore, the *static* LCC accounts for the development, launch, and operation of the spacecraft; however, it excludes the costs associated with having to replace modules due to on-orbit and launch vehicle failures. In contrast to *static* LCC, *dynamic* LCC accounts for these factors and is quantified in the Dynamic Model (see Section 3.3.3). Appendix B also further enumerates the difference between *static* and *dynamic* LCC.

Unlike the physics-based model, the cost model is a parametric model (the only one in the SET), and it is recognized that the cost model does not match the high fidelity of the physics-based model. However, given the proprietary nature of industry-employed cost models for spacecraft and the general lack of cost information available from manufacturers for spacecraft hardware, it was not feasible to build a comprehensive bottom-up (high fidelity) cost model for the SET. Therefore, with this in mind the cost model developed for the SET incorporated as much fidelity as possible to assimilate the fidelity of the physics-based model. This was specifically accomplished by employing CERs from the Unmanned Space Vehicle Model, 8th Edition (USCM8); actual hardware cost estimates for spacecraft subsystems and components whenever cost information from manufacturers could be obtained; and COCOMO II for the computer software cost and development time estimation (Tecolote Research Inc., 2009; Tieu, Kropp, & Lozzi, 2000; Boehm, 2000).

The SET cost model classifies spacecraft costs as either being nonrecurring (NRE), recurring (RE), or operations support (Ops). To quantify these three types of cost, the cost model employs 31 CERs for the determination of a spacecraft's NRE, RE, and Ops costs. Each CER is either parametric (the CER computes the cost based on mass), or is fixed (the CER is a known with certainty). Twenty-six of the 31 CERs are provided in Table 3-5 and 22 and 4 of those CERs are parametric and fixed respectively. The launch vehicle selection model discussed in Section 3.2.1 is embedded in the SET cost model. Therefore, for a given spacecraft and deployment/replenishment, the launch vehicle selection model chooses the cheapest and most reliable combination of launch vehicles. The cost of these launch vehicles is therefore treated separately from all the other CERs in the cost model.

To determine a spacecraft's respective *static* LCC, first, the CERs are used to compute the NRE and RE cost of each subsystem in a module. These are then aggregated to determine the NRE and RE cost of a module. Following this, the NRE and RE cost of all the modules is aggregated to determine the spacecraft cost (due to the CERs 1-24 only in Table 3-5), at which point the Ops costs can be computed (CERs 25 and 26 in Table 3-5). Finally, the *static* LCC of the spacecraft can be computed by summing the spacecraft cost due to (1) CERs 1-24, (2) CERs 25 and 26, and (3) launch vehicle(s) for initial (BoL) deployment of the spacecraft.

Table 3-5. SET cost model elements.

Ref No.	Non-Recurring Costs (NRE)	Acronym	Required Input	CER Type
1	Structure/Thermal	-	Mass	Parametric
2	Attitude Determination & Control System, and Guidance Navigation & Control System	ADCS, GNCS	Mass	Parametric
3				Parametric
4	Electrical Power Supply	EPS	Mass	Parametric
5	Telemetry, Tracking, and Command	TT&C	-	Fixed
6	Propulsion	-	-	Fixed
7	Payload	-	PL Dimensions	Parametric
8	Communications	Comm	Mass	Parametric
9	Computer Systems and Software	CS/C&DH	Mission/SK Data	Parametric
10	Integration, Assembly, & Test (IA&T)	IA&T	NRE sum	Parametric
11	Program Level (wrap)	-	NRE sum	Parametric
12	Aerospace Ground Equipment	AGE	NRE sum	Parametric

Ref No.	Recurring Costs (RE)	Acronym	Required Input	CER Type
13	Structure/Thermal	-	Mass	Parametric
14	Attitude Determination & Control System, and Guidance Navigation & Control System	ADCS, GNCS	Mass	Parametric
15				Parametric
16	Electrical Power Supply	EPS	Mass	Parametric
17	Telemetry, Tracking, and Command	TT&C	Mass	Parametric
18	Propulsion	-	-	Fixed
19	Payload	-	PL Dimensions	Parametric
20	Communications	Comm	Mass	Parametric
21	Computer Systems and Software	CS/C&DH	Mission/SK Data	Parametric
22	Integration, Assembly, & Test (IA&T)	IA&T	RE (spacecraft and comm)	Parametric
23	Program Level (wrap)	-	RE sum	Parametric
24	Launch Operations & Orbital Support	LOOS	-	Fixed

Ref No.	Operations Support	Acronym	Required Input	CER Type
25	Ground System Lease	GS	Time in View, ML	Parametric
26	Personnel Costs (GS employees, analyst, etc.)	Hu_supp	Total S/C cost	Parametric

3.3.3. Dynamic Model

The dynamic (lifecycle) model is responsible for quantifying a spacecraft's *dynamic* LCC, which is the cost of the spacecraft over the mission lifetime, accounting for the spacecraft's naturally stochastic lifecycle as characterized by lifecycle uncertainties (see 2.1.8). It is important to recognize the distinction between *static* and *dynamic* LCC. Static LCC only accounts for the cost of developing, launching, and operating a spacecraft. Therefore, static LCC is the lower-bound LCC for a given spacecraft since it does not account for the adverse LCC implications of lifecycle uncertainties. In contrast, for a given spacecraft, the dynamic LCC quantifies the totality of the static LCC, and in addition, accrues the RE and launch costs associated with replacing spacecraft/modules throughout the lifecycle due to risks resulting from lifecycle uncertainties (*e.g.,* on-orbit failure). Dynamic LCC will therefore be greater or equal to static LCC in magnitude. Moreover, since dynamic LCC accounts for the adverse LCC implications of lifecycle

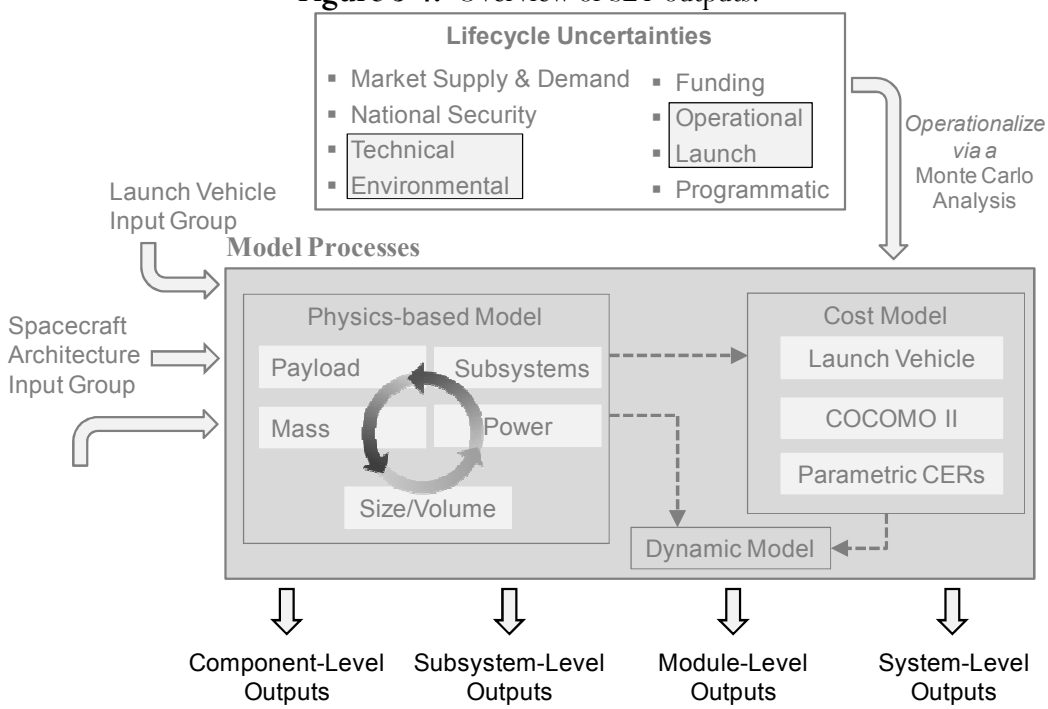
uncertainties, it is more appropriate than static LCC for quantifying the LCC of spacecraft having stochastic lifecycles[8].

Appendix B provides an extended discussion about static and dynamic LCC with respect to the manner in which they are quantified by the SET. Appendix C then complements Appendix B with a discussion pertaining to the propagation of lifecycle uncertainties throughout the lifecycle of a given monolithic or fractionated spacecraft, as simulated by the SET.

3.4. SET Outputs

The remaining functional partition of the SET is the outputs. Each output produced by the SET can be thought of as a potential element or dimension of the value proposition (see Section 2.1.9). The SET outputs enumerated hereafter emphasize the breadth (*i.e.,* capability) and depth (*i.e.,* fidelity) of the SET simulation, and subsequently, ability of the SET to populate monolithic and fractionated spacecraft value propositions.

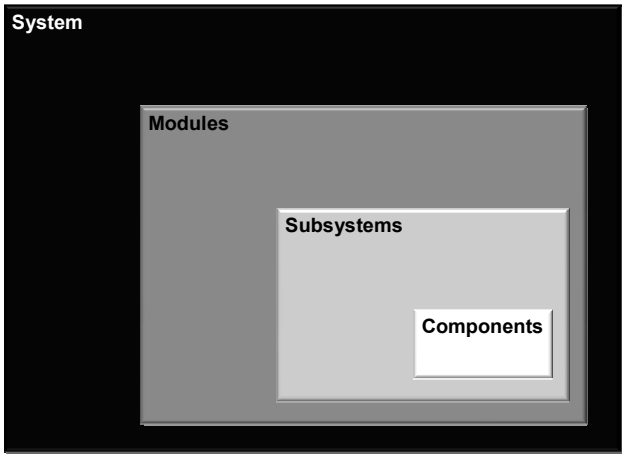
Figure 3-4. Overview of SET outputs.



are four levels of detail (fidelity) quantified by the SET outputs; from highest to lowest fidelity, these are designated as component, subsystem, module, and system-level outputs respectively. Each successively lower fidelity-level output aggregates the characteristics of the higher fidelity-level outputs; therefore, system-level outputs aggregate the characteristics of the outputs from its respective modules, module-level outputs in turn aggregate the characteristics of the outputs from its respective subsystems, and so forth. Figure 3-5 conceptually represents the relationship between the four SET output levels.

[8] In comparing static and dynamic LCC, one should note that dynamic LCC is *more appropriate* than static LCC because dynamic LCC accounts for lifecycle uncertainties. This, however, does not insinuate differences in accuracy between static and dynamic LCC because each is equally accurate relative to the perspective of a spacecraft's lifecycle they have chosen to adopt.

Figure 3-5. SET output-level relationships.



The system-level SET outputs aggregate the characteristics (*i.e.,* outputs) from the respective modules in a fractionated spacecraft[9]. The system-level SET outputs are preceded by the word *System* and include metrics such as *System* Static Lifecycle Cost, *System* Mass, and *System* Propellant Usage.

The module-level SET outputs aggregate the characteristics from all the subsystems within a respective module. The module-level SET outputs are preceded by the name of the *module* they correspond to and include metrics such as *(Payload) Module* Dynamic Lifecycle Cost, *(Payload) Module* Mass, and *(Payload) Module* Propellant Usage.

The subsystem-level SET outputs aggregate the characteristics from all of the components within a respective subsystem. The subsystem-level SET outputs are preceded by the name of the *subsystem* they correspond to and include metrics such as *Thermal Control Subsystem (TCS)* size/volume, *Communications* mass, and *RSM Payload* cost.

Lastly, the component-level SET outputs represent the lowest-order (*i.e.,* highest-fidelity) outputs produced by the SET. The component outputs do not aggregate their respective sub-components; rather, they are based off actual component/hardware data and empirically derived mathematical functions that characterize hardware (*e.g.,* battery densities). The component-level SET outputs are referred to by the specific names of components they characterize and include metrics such as *RSM Payload Mirror* heat dissipation, *Propellant Tank* length/radius/wall thickness, and *Reaction Wheel Unit (RWU)* electrical power consumption and spin rate (*i.e.,* angular velocity).

3.4.1. Subsystem and Component-Level SET

For a given module in a spacecraft, the metrics listed in Table 3-6 show all of the subsystem-level outputs, and a representative cross-section of the component-level SET outputs, as generated by the SET for that module. A representative cross-section of component-level outputs is presented in Table 3-6 because enumerating all of the SET component-level outputs is not tractable given that there are typically several hundred for just a single spacecraft module. It is important to recognize that the subsystem and component-level outputs listed in Table 3-6 are generated for *each* module in a spacecraft.

[9] For monolithic spacecraft, the spacecraft and module-level outputs will be identical since a monolith only has one-module.

Table 3-6. Subsystem and component-level SET outputs.

Subsystem and Component-Level SET Outputs for *each* Module		
Subsystem	**Subsystem Output**	**Component Output**
RSM Payload	**Power**	Power
		Data Rate
		Earth Coverage
	Mass	Mass
	Size/Volume	Volume
		Mirror Diameter
		Mirror Dimension 1
		Mirror Dimension 2
		Mirror Dim 3
	Cost	Cost
		Heat Dissipation
Computer System, C&DH	**Mass**	Mass
	Size/Volume	Volume
	Cost	Cost
	Power	Power
	Cost	Software Cost
		Software Development Time
		Data Rate
Communications and TT&C	**Mass**	Directional Antenna Mass
		Mass Inside Module
	Size/Volume	Directional Antenna Diameter
		Omni-antenna Length
		Volume Inside Module
	Power	Omni-antenna Power
		Directional antenna Power
ADS, GNS	**Mass**	Mass
	Size/Volume	Volume
	Power	Power
EPS	**Size/Volume**	Eclipse Battery Volume
		WPD Volume
		Mirror Dimension 1
		Total Battery Volume
		Solar Array Area
		Secondary Solar Array Area
	Mass	Eclipse Battery Mass
		Solar Array Mass
		WPD Mass
		Day Battery Mass
		Day Battery Volume
		Total Battery Mass
		WPD Heat Dissipation (day)
		WPD Heat Dissipation (eclipse)
Propulsion and ACS/GCS	**Mass**	Thruster Mass
		Propellant Tank Mass
		ACS Mass
	Propellant Usage/Mass	Propellant Mass
		Stationkeeping Propellant Mass
	Size/Volume	Propellant Tank Volume
		Thruster Volume
		ACS Volume
		Propellant Tank Length
		Propellant Tank Radius
		ACS RW Z-Radius
		ACS RW X/Y-Radius
	Power	ACS Max Power Draw
TCS	**Size/Volume**	Volume Inside Module
		Radiator Volume
	Mass	Mass Inside Module
		Radiator Mass
	Power	Power Required (during eclipse)

For a given subsystem shown in Table 3-6, all components in that respective subsystem are aggregated to quantify the subsystem size/volume, mass, and power requirement. With the exception of the RSM payload, the cost of a given subsystem is determined using the parametric cost model (see Section 3.3.2), which computes a subsystem's respective cost via CERs based on the mass of the subsystem.

In contrast to some of the system and module-level outputs, the subsystem and component-level outputs remain constant across all MCA trials employed in the SET. Therefore, these outputs only need to be computed and reported once for a given assessment of a spacecraft. The reason for this is that spacecraft architectures and their respective modules do not change over the lifecycle, and therefore quantifying these subsystem and component-level input values is independent of the dynamic lifecycle simulation aspect of the SET, as is evident in Table 3-4. However, the subsystem and component-level outputs are still needed for the dynamic (lifecycle) model in the SET, as is also evident in Table 3-4.

3.4.2. Module-Level SET

For *a given* spacecraft module, the module-level outputs aggregate the subsystem and component-level outputs shown in Table 3-6 corresponding to the respective subsystems and components present in that module. The module-level outputs given in Table 3-7 are therefore quantified for each module in a spacecraft. In contrast to the subsystem and component-level outputs, some of the module-level outputs vary with each MCA trial and hence they must be computed and reported for each trial; subsequently, these are called *dynamic* module-level outputs. And the module-level outputs that remain constant across all MCA trails, and hence only need to be computed and reported once, are called *static* module-level outputs.

Table 3-7. *Static* and *dynamic* module-level SET outputs.

***Static* Module-Level Outputs**	
Module Output	**Units**
Mass Propellant Mass Stationkeeping Propellant Mass	kg
Power Required (Eclipse) Power Required (Day)	watt
Inside Dimension 1 Inside Dimension 2 Inside Dimension 3 Launch Vehicle Dimension 1 Launch Vehicle Dimension 2 Launch Vehicle Dimension 3 Sizing (Inside) Volume (Inside) Sizing (Launch Vehicle) Volume (Launch Vehicle)	m, m3
NRE RE Ops	FY2008$M
Lifetime	years
Earth coverage per orbit	%
Fuel Efficiency	kg/year

***Dynamic* Module-Level Outputs**	
Module Output	**Units**
Module Type	Infra/Payload
Number of Replenishments	-
Module Number	-
Dynamic LCC	FY2008$M

In Table 3-7 the static module-level outputs primarily include elements pertaining to the mass, power, size/volume, and fixed costs of a spacecraft. In contrast, the dynamic module-level outputs include elements pertaining to the dynamic lifecycle cost of the module (see Section 3.5.1) and the number of times a given module needs to be replaced (replenished) over the lifecycle (for a given MCA trial/simulation of the lifecycle).

3.4.3. System-Level

For *a given* spacecraft, the system-level outputs shown in Table 3-8 aggregate the module-level outputs shown in Table 3-7 corresponding to each respective module in that spacecraft. Analogous to the module-level outputs, some of the system-level outputs vary with each MCA trial and hence they must be computed and reported for each MCA trial; these are called *dynamic* system-level outputs. And similar to module-level outputs, there are some system-level outputs that remain constant across all MCA trails. Hence, these only need to be computed and reported once and subsequently called *static* system-level outputs. However in contrast to the module-level outputs, there are certain system-level outputs which, for a given spacecraft, characterize lifecycle cost statistics due to the lifecycle cost distribution created from the MCA (*e.g.*, uncertainty in the median lifecycle cost value). Since these lifecycle cost statistics rely on the distribution of lifecycle cost created from the MCA, they are computed after the MCA trials are done being performed in the SET. Subsequently, the lifecycle cost statistics are only reported once for a given assessment of a spacecraft. The metrics pertaining to lifecycle statistics are called *lifecycle cost statistic* system-level outputs. (Please refer to Section 2.1.10 for a detailed treatment of the *lifecycle cost statistic* system-level SET outputs.) Table 3-8 presents the system-level SET outputs generated by the SET organized as *static, dynamic*, and *lifecycle cost statistic* system-level outputs.

Table 3-8. *Static, dynamic,* and, *lifecycle cost statistic* system-level SET outputs.

Static **System-Level Outputs**	
System Output	**Units**
No. of MCA Trials	-
Lifetime	years
Mass Propellant Mass Stationkeeping Propellant Mass	kg
NRE Cost RE Cost Operations Cost Launch Vehicle Cost Static LCC	FY2008$M
Fuel Efficiency	kg/year
Earth coverage per orbit pass	%

Dynamic **System-Level Outputs**	
System Output	**Units**
Dynamic LCC	FY2008$M
Cost Profile Cumulative Cost Profile	FY2008$M / year

Lifecycle Cost Statistic **System-Level Outputs**		
Statistic Description	**System Output**	**Units**
Uncertainty due to SET Cost Model (CMU)	LCC Variance	(FY2008$M)^2
	Standard Deviation of LCC LCC 5th Percentile LCC 95th Percentile LCC Mode LCC Median	FY2008$M
Uncertainty due to MCA (MCAU)	LCC Number of Modes LCC Number of Dominant Modes LCC Skewness LCC Kurtosis	-
Order Statistic, Five-Number Summary	Maximum Dynamic LCC 75th Percentile Dynamic LCC 50th Percentile Dynamic LCC 25th Percentile Dynamic LCC Minimum Dynamic LCC	FY2008$M
Lifecycle Cost Distribution Measures of Central Tendency	Median Dynamic LCC Mode Dynamic LCC Two-sided Quartile Weighted Median Dynamic LCC	FY2008$M

The *static* system-level outputs include metrics such as *system* lifetime, mass, static lifecycle cost (LCC), whereas the *dynamic* system-level outputs characterize the LCC related profiles and aggregate Dynamic LCC, for a given spacecraft, corresponding to each MCA trial (see Appendix B for a discussion pertaining to LCC profiles). Lastly, the *lifecycle cost statistic* system-level outputs characterize/quantify the LCC distribution resulting from the MCA (trials) in terms of the distributions respective measure of central tendency and variability (*i.e.,* uncertainty in the LCC measure of central tendency).

SET Output Perspective

For a given spacecraft architecture and its respective SET Lifecycle & Design and Launch Vehicle inputs, the total number of outputs produced by the SET can be determined by Eq. (1). Note that all the SET outputs are candidate metrics for which to form the value proposition corresponding to a given spacecraft.

Equation 1. Relationship for determining the number of SET outputs.

$$NoO = 30 + n \times 105 + m \times \left[(n \times 4) + 3 \right]$$

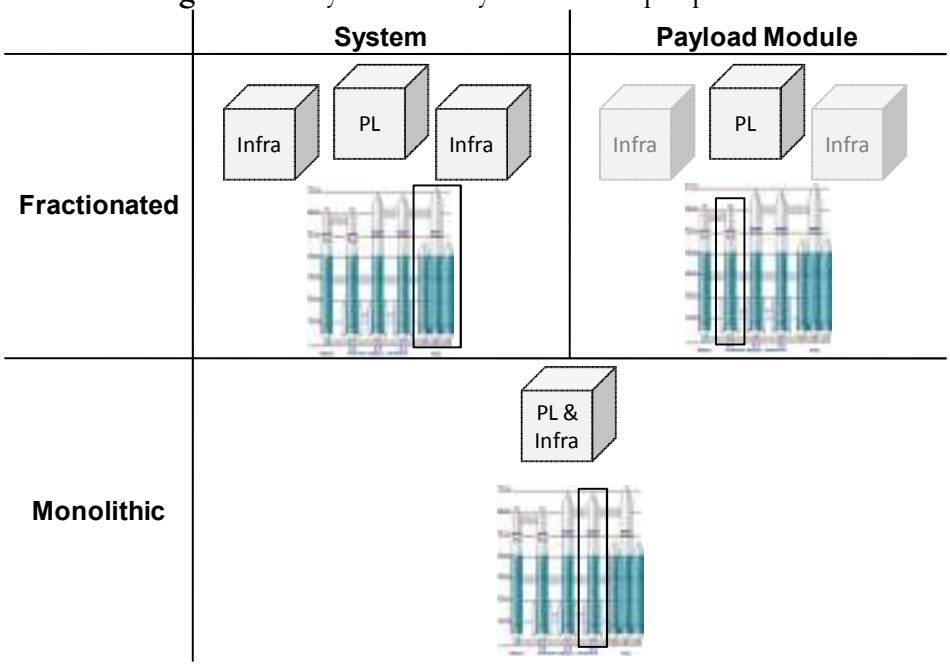
	NoO	number of SET outputs
$\forall$	n	number of modules
	m	number of MCA trials

For example, Eq. (1) yields 37,845 outputs for a three-module fractionated spacecraft assessed with 2,500 MCA trials. Each of these 37,845 outputs quantifies either a physical or a cost related characteristic of the fractionated spacecraft at the component, subsystem, module, or system SET output level.

3.5. SET Output Perspectives: System versus Payload Module

The SET system and module-level outputs generated by the SET (*e.g.,* dynamic lifecycle cost and mass) characterizing a given fractionated spacecraft provide two different perspectives with regard to the benefits and costs of fractionated spacecraft that are subsequently adopted in this research (see Section 4.6). The two perspectives, called the *System* and *Payload Module* perspectives, are depicted in Figure 3-6.

Figure 3-6. System and Payload Module perspectives.



In adopting the *System* perspective, a spacecraft developer is assumed responsible for the development, deployment, and operation of the spacecraft throughout the mission lifetime. For the case of a fractionated spacecraft, this implies that the developer is responsible for the development, deployment, and operation of all spacecraft modules. This perspective thereby quantifies the implications of fractionated spacecraft based on the system-level SET outputs such as *System* mass, *System* dynamic LCC, and *System* propellant usage.

Conversely, in adopting the *Payload Module* perspective, a spacecraft developer is assumed responsible for the development, deployment, and operation of only the Payload Module (*i.e.,* the module that contains the mission payload) throughout the mission lifetime. For the case of a fractionated spacecraft, this implies that the developer is responsible for the development, deployment, and operation of *only* the spacecraft modules carrying a mission payload. This perspective thereby quantifies the implications fractionated spacecraft based on the *Payload Module*-level SET outputs such as *Payload Module* mass, *Payload Module* dynamic LCC, and *Payload Module* propellant usage.

Note that if a monolithic spacecraft is being considered, the *System* and *Payload Module* perspectives are the same because a monolithic spacecraft has only one-module.

The motivation for understanding the implications of fractionation from both the *System* and *Payload Module* perspectives arises from the following questions:

> **1.** What are the implications of fractionation if a spacecraft developer is responsible for the entire spacecraft (*i.e.,* all of its respective modules)?
> **2.** What are the implications of fractionation if a spacecraft developer is only responsible for the Payload Module in the hypothetical situation in which there is already Infrastructure Modules on-orbit to support that Payload Module?

The first question motivates the need to understand how much, in terms of aggregate resources, it will cost an entity or set of entities to develop and operate a fractionated spacecraft (and all of its respective modules). The response to this question would be of interest to a spacecraft developer assuming either all or the majority of the responsibility in developing and operating a fractionated spacecraft. Subsequently, the implications of fractionated spacecraft for this type of spacecraft developer would be relative to the *System* perspective (*i.e.,* System-level SET outputs).

However, as the second question alludes to, it can be hypothesized that in the future there may be organizations that supply Infrastructure Modules on-orbit to serve as shared resource sources (see Section 2.1.5) for Payload Modules that are owned and operated independently of the Infrastructure Modules. This hypothesis is therefore functionally equivalent to automobiles relying on gas stations for fuel rather than carrying the entirety of the fuel they will require for their respective lifetimes. If this hypothetical situation were to occur, a spacecraft developer would only need to supply the module(s) with the mission payload, rather than all of the fractionated spacecraft modules. Of course, there would be a fee for the use of (*i.e.,* reliance on shared resources from) the Infrastructure Modules. Moreover, since relying on shared resources can reduce the requirements of the Payload Module (development and operation), the *Payload Module* perspective of fractionated spacecraft may radically change (lower) the cost barriers to entry in the spacecraft market. Subsequently, the implications of fractionated spacecraft, for a spacecraft developer only responsible for the Payload Module development and operation, would be relative to the *Payload Module* perspective (*i.e.,* Payload Module-level SET outputs).

3.5.1. System and Payload Module Lifecycle Cost

To serve as an illustrative example of the System and Payload Module perspectives (SET outputs) consider the *System* and *Payload Module* Dynamic lifecycle cost (LCC). The System Dynamic LCC quantifies the cost (development, deployment, operation, and replenishment) of the entire spacecraft (*i.e.,* all modules) over the lifecycle (mission lifetime). In contrast, the Payload Module Dynamic LCC quantifies the cost (development, deployment, operation, and replenishment) of the *only* the Payload Module over the lifecycle (mission lifetime).

Therefore, to compute the *System* and *Payload Module* Dynamic LCC, an entire stochastic dynamic lifecycle simulation is performed twice in the SET, once for the entire spacecraft and then a second time for only the Payload Module. However, to maintain consistency between the System and Payload Module Dynamic LCC value corresponding to a given fractionated spacecraft, in the Payload Module dynamic lifecycle simulation, the number of on-orbit (not launch vehicle) failures is identical to those experienced in the System dynamic lifecycle simulation. (The launch vehicle failures are not held constant because the spacecraft (System) and Payload Modules are likely to employ different launch vehicles that subsequently have different reliabilities.) Therefore, in maintaining a consistent number of on-orbit failures when

computing the System and Payload Module Dynamic LCC, it enables for the comparison of these two LCC values (see Figure 4-34) on the basis in which the System and Payload Modules experience the same technical, environmental, and operational lifecycle uncertainties. This equal basis for comparison is necessary because changing from the System to Payload Module perspective does not change the physical design of the Payload Module in a fractionated spacecraft.

3.5.2. Forming the Value Proposition from the SET Outputs

Given the (tens of) thousands of SET outputs characterizing a given spacecraft at the component, subsystem, module, and system-level, keeping the value proposition representative while still tractable is not a trivial task. Understandably, the value proposition must be tractable in the sense that meaningful comparisons between two or more value propositions can readily be made – something accomplished through using fewer, rather than more, of the (tens of) thousands of metrics (*i.e.,* SET outputs) quantified for a given spacecraft. The challenge is thus in deciding which of the SET outputs to use to populate monolithic and fractionated spacecraft value propositions.

Given the potential SET outputs, this research investigation opted to use outputs in the value proposition that most appropriately and succinctly quantify direct quantitative responses to the three research questions (see Section 2.5). Subsequently on the basis of the research questions, the value proposition, for the purposes of this research investigation, is composed of the following 11 metrics (SET outputs) characterizing monolithic and fractionated spacecraft architectures[10]:

- ***System*** (1) Mission Lifetime, (2) Static Lifecycle Cost, (3) Dynamic Lifecycle Cost, (4) Payload Performance (*i.e.,* Ground Resolution), (5) Mass, and (6) Propellant Usage.
- ***Payload Module*** (7) Mission Lifetime, (8) Dynamic Lifecycle Cost, (9) Payload Performance (*i.e.,* Ground Resolution), (10) Mass, and (11) Propellant Usage.

3.6. SET Limitations and Implications for the Research Contributions

Section 3.1 through 3.5 enumerated the breadth (capability) and depth (fidelity) of the SET, but these two attributes do not come without limitations. It is recognized that there is an inherent tradeoff between the fidelity of a spacecraft model, its accuracy, and the subsequent resources required for its development; the higher the fidelity of the model, the more accurate it will be at the cost of more resources required to develop (and run) the model. This tradeoff informs the most significant limitation of the SET. The SET was intended to be a high fidelity, accurate spacecraft model. Consequently, the high fidelity and accuracy of the SET resulted in it its respective development and verification & validation dominating the time allotted for this research effort, which in turn, allowed less time to analyze and synthesize the results from the SET. Additionally, the fidelity of the SET required appreciable amounts of time (as compared to fully parametric spacecraft models) to run a single assessment of a monolithic or fractionated spacecraft. Therefore, in hindsight, the SET did achieve its objectives of being a high fidelity, accurate spacecraft model but it may have sacrificed too many resources for the sake of achieving these two attributes of the SET. In

[10] A potential future extension of this research would be to examine the value propositions of monolithic and fractionated spacecraft using other SET outputs from the component, subsystem, module, and/or system-level. Changing the metrics comprising the value proposition will lead to the observation new and valuable nuances between monolithic and fractionated spacecraft not observed in the respective outcomes of this research effort.

conclusion, there is likely a better (more efficient) balance between fidelity, accuracy, and resources than was employed for developing the SET.

Despite the SET having limitations, the development and application of the SET helps to fulfill the unique contributions of the research. These research contributions are repeated again for convenience.

Research Contributions

1. Provide a high fidelity, bottom-up, dynamic quantitative assessment of monolithic and fractionated spacecraft value propositions.
2. Enable an understanding of the monolithic and fractionated spacecraft value propositions using cardinal, "traditional" measures of effectiveness (MoE)
3. Provide the ability to explore monolithic and fractionated spacecraft value propositions in both breadth and depth.

Based on the discussion of the SET provided in Chapter 3, using the SET for spacecraft assessments "provides a high fidelity, bottom-up, dynamic quantitative assessment of monolithic and fractionated spacecraft value propositions." This is evident from the SET inputs and model processes (see Sections 3.2 and 3.3). In addition, by using the SET for spacecraft assessments it "provides an understanding of the monolithic and fractionated spacecraft value propositions using cardinal, "traditional" measures of effectiveness." This is evident from the very nature of the SET outputs being cardinal measures that may be perceived as a MoE by beneficiary stakeholder (see Section 3.4). Lastly, by using the SET for spacecraft assessments it "enables the ability to explore monolithic and fractionated spacecraft value propositions in both breadth and depth." This is evident from the number of SET inputs (degrees of freedom) and outputs (see Sections 3.2 and 3.4).

Chapter 3 has been devoted to discussing the development of the SET. Accordingly, the results from the applying the SET, in terms of monolithic and fractionated spacecraft value propositions, is the subject of the Chapter 4, the Analysis.

4. Analysis: SET Results

The analysis is the third phase of the research methodology (see Section 1.1.3). The intent of the analysis is to employ the Spacecraft Evaluation Tool (SET) to generate sufficient data with regard to monolithic and fractionated spacecraft value propositions to form quantitative responses to the three research questions[11]. In the analysis, each research question became the focus of one case study. However, the insights gained from each case study are not treated as mutually exclusive responses to each research question; rather, the collective insight gained from *all three* case studies is used to form responses to *all three* research questions (see Chapter 5). Subsequently, in Chapter 4, the results corresponding to the case studies are organized with respect to their commonality in forming monolithic and fractionated spacecraft value propositions, not by case study. By categorizing the results in this manner, it provides a logical and succinct "story" about monolithic and fractionated spacecraft value propositions that avoids becoming a seemingly endless and unconnected cluster of information. The three case studies are briefly described below and thereafter an outline of the presentation of the results in Chapter 4 is given.

The first case study, *Investigation of Nuances in Spacecraft Architecture*, is focused on exploring responses to the first research question:

1. How do the value propositions for monolithic and fractionated spacecraft compare **across alternative spacecraft architectures (designs)**?

The emphasis of the first case study is to explore the implications of significant and subtle nuances (changes) in monolithic and fractionated spacecraft architectures (aka designs) in terms of their effect on the value proposition. The architectural nuances investigated include the number of modules, use of shared resources, and separation distance between the modules. Comparisons in this case study are often made on the basis of classes of spacecraft architectures (*e.g.,* two vs. three-module spacecraft architectures) and comparable spacecraft architectures (*e.g.,* one and two-module spacecraft architectures employing the same shared resources or having the same payload performance).

The second case study, *Implications of Payload Performance and Mission Lifetime*, is focused on exploring responses to the second research question:

2. How do the value propositions for monolithic and fractionated spacecraft compare **relative to changing payload requirements** (*i.e.,* ground resolution)?

The emphasis of the second case study is to expand upon the value proposition for monolithic and fractionated spacecraft quantified through the first case study with regard to payload performance and mission lifetime. In this case study, the payload performance investigation considers four payload (*i.e.,* telescope) ground resolutions, and the mission lifetime investigation explores the implications of leveraging the purported benefits of fractionation to extend the mission lifetime of fractionated spacecraft relative to that of a comparable monolith.

[11] In the discussion of this research, specifically its respective outcomes in Chapter 4, the terminology spacecraft architecture are employed instead of the term spacecraft; for example, fractionated *spacecraft architecture* instead of fractionated *spacecraft*. It should be noted that the terminology *spacecraft architecture* are purposely used in discussions in which the intent is to emphasize the specific design (*i.e.,* structural hardware and subsystem composition) of spacecraft, as this is not as readily conveyed with the term *spacecraft*.

The third case study, *Exploring Spacecraft Lifecycle Uncertainties*, is focused on exploring responses to third research question:

> **3.** How do the value propositions for monolithic and fractionated spacecraft compare **<u>relative to the risks resulting from spacecraft lifecycle uncertainties</u>** (*e.g.*, on-orbit failure)?

The emphasis of the third case study is further expanding the value propositions quantified through the first and second case study by enumerating the implications of the risks resulting from lifecycle uncertainties on monolithic and fractionated spacecraft value propositions. Specifically, this case study informs the value propositions for a selected set of monolithic and fractionated spacecraft architectures with respect to four distinct levels of probability of infant mortality (PoIM) and three mission lifetimes.

Chapter 4 is partitioned into seven sections, Section 4.1 - 4.7. Section 4.1 is devoted to presenting the SET Launch Vehicle, Lifecycle & Design, and Spacecraft Architecture inputs used to generate the results shown in Section 4.3 through 4.7. Section 4.2 has a discussion pertaining to the manner in which the results are presented (formatted) in Section 4.3 through 4.7, thereby providing a guide to understanding the analysis data. The remaining five sections in Chapter 4, Section 4.3 through 4.7, cover each of the five respective divisions of the results presentation as shown in Figure 4-1.

Figure 4-1. Analysis outline: presentation of results.

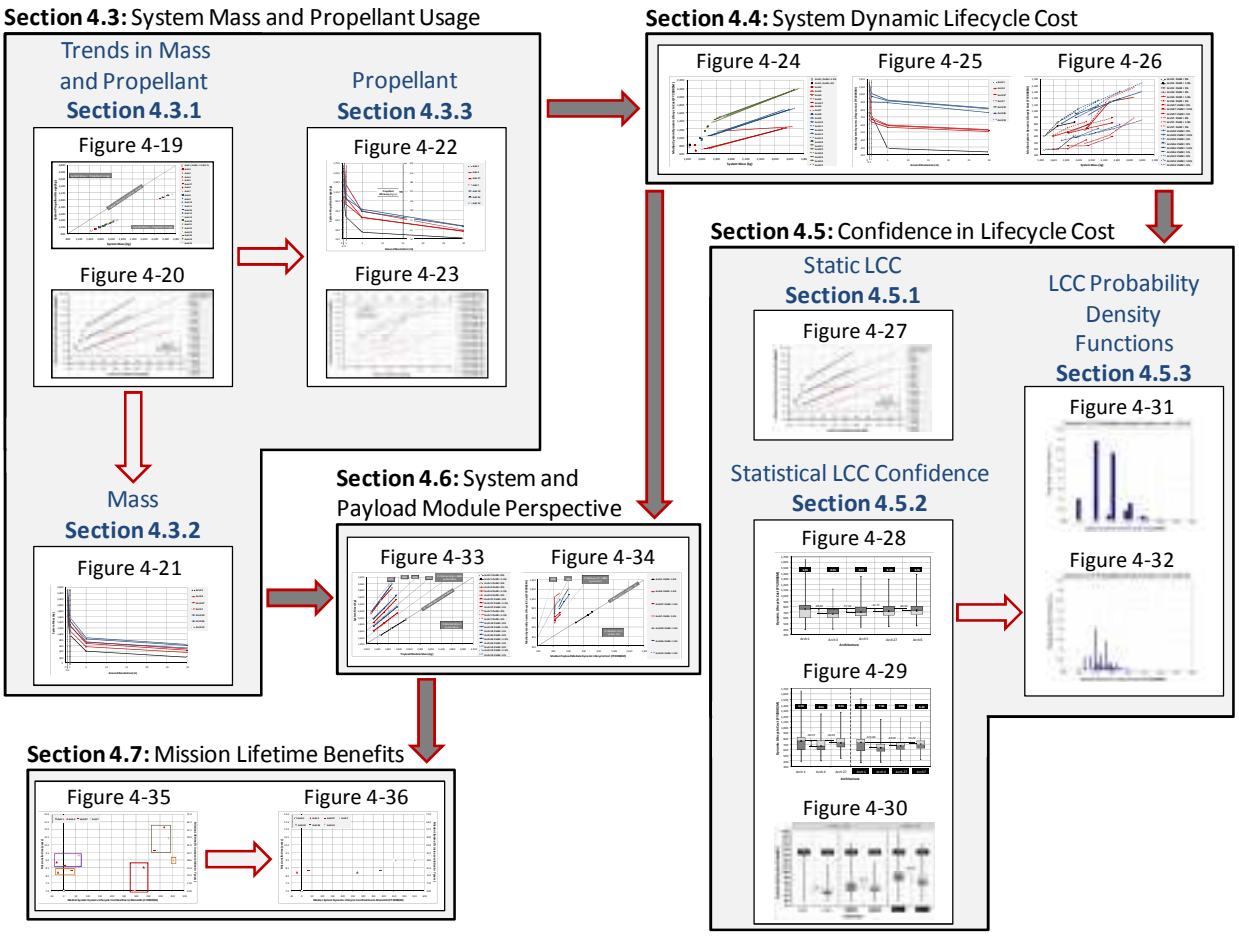


Figure 4-1 presents a structured outline describing the organization and subsequent presentation of the results/data generated for all three case studies from the SET. The flow of results in the analysis, as presented in Figure 4-1, provides a logical and succinct means for understanding the value propositions of monolithic and fractionated spacecraft. The five main divisions (Sections) of the results presentation in Chapter 4, can be thought of as parts of a "story", each of which successively contributes to an understanding (dimensions) of monolithic and fractionated spacecraft value propositions. Section 4.3 begins informing monolithic and fractionated spacecraft value propositions by enumerating the relationship between System Mass and Propellant Usage (Section 4.3.1, 4.3.2, and 4.3.3). Section 4.4 then provides an understanding of the Dynamic Lifecycle Cost implications of monolithic and fractionated spacecraft. Thereafter, in Section 4.5, the confidence in the Dynamic Lifecycle Cost values presented in Section 4.4 is quantified, specifically through a discussion of Static versus Dynamic Lifecycle Cost (Section 4.5.1), statistical uncertainty in the SET cost model (Section 4.5.2), and MCA lifecycle cost probability density functions (Section 4.5.3). Section 4.6 then introduces a new dimension to monolithic and fractionated spacecraft value propositions by considering the Mass and Dynamic Lifecycle Cost implications of these spacecraft architectures with respect to the *System* and *Payload Module* perspective (see Section 3.5). Lastly, Section 4.7 enumerates the implications of the Payload Module mass (and size) disparity between comparable monolithic and fractionated spacecraft with regard their respective mission lifetime capability.

For purposes of reference, Table 4-1 provides a listing of the figures found in Section 4.3 through 4.7 corresponding to the respective results generated from the SET for each of the three case studies

Table 4-1. Case study results guide.

Case Study					
1		2		3	
Figure 4-	Page No.	Figure 4-	Page No.	Figure 4-	Page No.
19	108	21	112	23	114
20	109	22	113	26	119
24	115	25	118	30	126
27	121	29	124	33	129
28	123	35	133	34	131
31	127	36	134		
32	127				

4.1. Analysis (Case Study) SET Inputs

Each set of results in the analysis will be presented hereafter as a figure (plot) relative to one of the three case studies. Each of the three case studies has a unique set of SET inputs that will be specified in Section 4.1. Subsequently, on each set of results (*i.e.,* figure) presented in Section 4.3 through 4.7, the case study which the set of results belongs to will be denoted. This thereby enables the specific SET Launch Vehicle, Lifecycle & Design, and Spacecraft Architecture inputs corresponding to each respective set of results presented hereafter to be readily discerned. Section 4.1 therefore provides the SET Lifecycle & Design, Launch Vehicle, and Spacecraft Architecture inputs for each of the three case studies and subsequently their respective results provided in Section 4.3 through 4.7.

4.1.1. Launch Vehicle

In terms of the Launch Vehicle group of SET inputs, of the 22 possible launch vehicles, for each case study, 14 were selected as candidate launch vehicles for the dynamic lifecycle simulations of spacecraft. For all three case studies, the launch vehicles selected are identical and given in Table 4-2.

Table 4-2. Case Study Launch Vehicle inputs.

Small Class (LEO push mass < 1000 kg)						
Launch Vehicle	Athena I	Falcon I	Minotaur	Pegasus XL	Scorpius	Start-1
Country of Origin	*United States*					*Russia*

Medium Class (LEO push mass 1000-3000 kg)								
Launch Vehicle	Athena II	Delta II	Commercial Taurus	Angara 1.1	Kosmos 3M	Rockot	Strela	Vega
Country of Origin	*United States*			*Russia*				*Italy/Europe*

Large Class (LEO push mass 3000-7000 kg)					
Launch Vehicle	Angara 1.2	Cyclone 2	Dnepr	Long March 2C	PSLV
Country of Origin	*Russia*	*Ukraine and Russia*		*China*	*India*

Heavy Class (LEO push mass > 7000 kg)			
Launch Vehicle	Atlas V 500	Delta IV M+	Titan IVB
Country of Origin	*United States*		

KEY
Launch vehicle considered
Launch vehicle not considered

4.1.2. Lifecycle & Design

The SET Lifecycle & Design input values used for the three case studies are presented in Table 4-3.

Table 4-3. Case Study Lifecycle & Design inputs.

				Case Study 1	**Case Study 2**	**Case Study 3**
Input Category	**-**	**Input**	**Units**	**Run Value**		
Orbital Parameters	1	Orbit Altitude	km	700	700	700
	2	Orbit Inclination	degrees	98	98	98
CONOPS	3	**Cluster Separation Distance**	m	**20, 1000, 5000**	1,000	1,000
Autonomy Level	4	Bus Autonomy Level	-	1	1	1
	5	Payload Autonomy Level	-	2	2	2
Lifetime	6	**Mission Lifetime**	years	7	7	**5, 7, 9**
Sizing	7	**Min** Packing Efficiency	m3/kg	0.014	0.014	0.014
Payload Performance	8	**Linear Ground Resolution (i.e. Pixel Size)**	m	0.5	**0.5, 1, 5, 30**	0.5
Pointing Requirements for Pointing-Intensive Modules	9	Pointing Tolerance	deg.	0.00001	0.00001	0.00001
	10	Spacecraft Jitter	deg.	0.00014	0.00014	0.00014
	11	Slew Maneuver No.	-	12	12	12
	12	Slew Maneuver Magnitude	deg	90	90	90
Pointing Requirements for Non-Pointing-Intensive Modules	13	Pointing Tolerance	deg.	0.0005	0.0005	0.0005
	14	Spacecraft Jitter	deg.	0.00014	0.00014	0.00014
Dynamic Lifecycle Simulation	15	No. of MCA Trials	-	2,500	2,500	2,500
	16	**Probability of Infant Mortality (PoIM)**	-	**0.01, 0.015 (monolith), 0.015 (fractionated)**	0.015	**0, 0.015, 0.05**
Production	17	Build time learning factor	-	0.65	0.65	0.65
Lifecycle Uncertainties	18	Launch	-	1	1	1
	19	Technical, Environmental, and Operational	-	1	1	1
Launch	20	No. of LVs	-	3	3	3

Mission Lifetime Extension	21	Monolithic Mass (Datum)	kg	N/A	1668.74, 962.91, 404.37, 196.56	N/A

All Case Studies

In terms of pointing requirements and spacecraft jitter, these are kept at extremes meaning the pointing tolerance provides appreciable accuracy about all three, principle axes of a spacecraft given the orbit altitude, inclination, payload resolution, and spacecraft jitter. The spacecraft jitter value was selected to represent a near-worst case scenario and is equal to one of the more extreme jitter values experienced by the Hubble Space Telescope during its in-situ slewing maneuvers (Pavlovsky, Koekemoer, Mack, Karakla, & Rose, 2006). The number and magnitude of slews is based off the orbital profile of GeoEye-1, which is capable of performing at least 60° (net) in-situ slew maneuvers. Subsequently, given the low pointing tolerance, large spacecraft jitter, and high number and magnitude of in-situ slews, the spacecraft investigated in this case study are considered to be pointing-intensive.

For SET inputs of orbit altitude, orbit inclination, and mission lifetime, these were found to be representative of satellites performing remote sensing missions. Specifically, GeoEye-1, RapidEye, Landsat-7, EOS Aqua, and NPOESS were used as points of reference to determine the values for these three respective SET inputs.

For all three case studies, all lifecycle uncertainties were considered and their consequent risks, that is, launch, technical, environmental, and operational lifecycle uncertainties. Additionally, the time required to rebuild spacecraft/modules (due to learning effects) is 65% more efficient, in terms of time, than manufacturing the spacecraft/modules the first time. Lastly, three is the maximum number of launch vehicles that can be used simultaneously to deploy spacecraft/modules.

First Case Study Only

The SET inputs of interest in the first case study were the inter-module (*i.e.,* cluster) separation distance, and exclusively for monolithic spacecraft architectures, probability of infant mortality (PoIM). As such, three inter-module separation distances were investigated: 20, 1000, and 5000 m (0.02, 1, and 5 km). The 20 m separation distance was treated as the lower bound inter-module separation distance, this being due to the practical accuracy limits of ADS's and GNS's in operation today[12]. The 1000 and 5000 m separation distances were selected on the basis of insights gained from preliminary and current designs for the TPF-I, which proposes inter-module separation distances for the TPF-I of similar magnitudes (Lawson & Dooley, 2005; Lawson, Lay, Johnston, & Beichman, 2007).

The PoIM was held constant at 1.5% for all fractionated spacecraft architectures, but for the monolithic spacecraft architectures a PoIM of 1.5 and 5% was investigated. These PoIM values are taken from typical "bathtub" hazard functions for spacecraft reliability as well as cited PoIM values from the work of others (Brown et al., 2007; Larson & Wertz, 1999). A PoIM of 1.5 % is considered nominal, whereas a PoIM of 5% represents an extremely high rate of infant mortality and is subsequently considered the worst-case PoIM scenario.

[12] JPLs Blackjack GPS receiver is an example of a high precision on-orbit determination system and it has an accuracy of 1 to 2 m. A heuristic often employed in the area of attitude determination and guidance navigation is to keep a minimum of ten times the control accuracy distance between objects in space – hence 20 m was selected as the lower bound separation distance on the basis of keeping at least ten times the Blackjack's control accuracy between modules on-orbit.

Second Case Study Only

In the second case study, the SET inputs of interest were the payload performance, that is, the optical imaging system (*i.e.,* telescope) ground resolution capability. Specifically four values of ground resolution were investigated: 0.5, 1, 5, and 30 m. These four values were chosen given their representative nature of the performance of current satellite imaging systems and, additionally, they span three notional classes of ground image resolutions: high, medium, and low. Images at a resolution of 0.5 m belong to the high-resolution image class, and this number was specifically chosen because the highest resolution images *commercially available* in the world are currently at a 0.5 m resolution (these are captured by GeoEye-1). The 1 and 5 meter resolutions belong to the high and medium resolution image classes respectively. These two resolution values are based on surveying the ground resolution capabilities of four notable Earth observation satellites: GeoEye-1, RapidEye, Landsat-7, and NPOESS. Across these four Earth observation satellites, the ground image resolutions (at visible wavelengths) vary from 0.5 to 5000 m, but due to emphasis of this research on investigating pointing-intensive fractionated spacecraft, which are subsequently reminiscent of RSM satellites with high ground resolutions, the resolutions of 1 and 5 m were selected. A 30 m ground resolution (low resolution) was selected as a representative upper bound resolution for pointing-intensive spacecraft, thereby rounding out this case study's respective sampling of high, medium, and low-resolution RSM payloads.

The PoIM was held constant at 1.5% for all fractionated spacecraft architectures, but for the monolithic spacecraft architectures a PoIM of 1.5 and 5% was investigated. These PoIM values are taken from typical "bathtub" hazard functions for spacecraft reliability as well as cited PoIM values from the work of others (Brown et al., 2007; Larson & Wertz, 1999). A PoIM of 1.5 % is considered nominal.

Third Case Study Only

For the third case study, the SET inputs of interest were the mission lifetime and probability of infant mortality (PoIM). With regard to mission lifetime, three values were selected: 5, 7, and 9 years. These mission lifetime values are based on the mission lifetimes of representative RSM satellites, namely, GeoEye-1, RapidEye, Landsat-7, EOS Aqua, and NPOESS. Among these RSM satellites, 7 years was found to be the average mission lifetime and 5 to be the lower bound mission lifetime. And although rarely achieved (or designed to achieve), 9 years was reasoned as a pseudo maximum mission lifetime for RSM spacecraft.

The intent of the third case study is to investigate the effects of the naturally stochastic lifecycle for a pointing-intensive, RSM spacecraft. As such, the combination of varying the mission lifetime and PoIM and considering all lifecycle uncertainties effectively achieves this. Through these influences on the stochastic lifecycle, numerous combinations of lifecycle severities (*i.e.,* probabilities of lifecycle uncertainties adversely affecting a spacecraft) are simulated. For example, a 9 year mission and a PoIM of 5% represents an extremely "severe (harsh) environment" meaning that there is likely to be numerous on-orbit failures (due to technical, environmental, and operational lifecycle uncertainties) and more, rather than less, launch vehicle failures per the need to replenish spacecraft/modules more often. Conversely, a mission lifetime of 5 years with a PoIM of 0% represents a "benign environment" in which there are likely to be very few on-orbit failures (due to technical, environmental, and operational lifecycle uncertainties) and less, rather than more, launch vehicle failures per the need to replenish spacecraft less often.

4.1.3. Spacecraft Architecture

For the three case studies considered in the analysis, the remaining SET inputs to be specified pertain to the specific monolithic and fractionated spacecraft architectures investigated in each case study. For the first case study, there were 22 spacecraft architectures investigated and each has a unique number of modules and utilization of shared resources. For the second and third case study, there were 7 spacecraft architectures investigated in total; the same 7 architectures are investigated in the second and third case study. The spacecraft architectures investigated in the three case studies are summarized in the next nine figures, Figure 4-2 through Figure 4-10. The first six figures, Figure 4-2 through Figure 4-7, depict the one, two, three, and four-module spacecraft architectures considered in the three case studies, respectively. The last three figures, Figure 4-8 and Figure 4-10, provide a condensed representation of the spacecraft architectures considered in the three case studies based on their distinguishing characteristics, namely, number of modules and use of shared resources.

In terms of all 22 unique spacecraft architectures collectively investigated across the three case studies, it is important to note that a given module must have a TT&C, ACS, GCS, Propulsion, TCS, Structures, and Wiring subsystem as these subsystems, for the purposes of this research effort, cannot be shared amongst the modules (see Section 2.1.5). The first module in a given spacecraft architecture is designated an *Infrastructure Module* because it always contains every subsystem, is a source for shared resources (if employed) and, has a S/C-ground antenna. The second module is always designated a *Payload Module*, because it has the RSM payload (there is only one used). And all remaining modules in a given spacecraft architecture, if applicable, are designated *Infrastructure Modules* and mirror the shared resource usage of the *Payload Module*, unless power is being shared at which point they support the first *Infrastructure Module* in generating and storing power for the *Payload Module*. Since a monolithic spacecraft architecture consists of only one-module, which contains the RSM payload and S/C-ground antenna, a monolithic spacecraft is referred to as an *Infrastructure/Payload Module*.

The visual depiction of the spacecraft architectures investigated in the case studies provided in Figure 4-2 through Figure 4-7 is complemented by the following brief discussion of the hardware associated with each shared resource that is required on the modules providing and receiving/relying on the resource (*i.e.,* sources and recipients respectively)[13]. (Section 2.1.5 will be a useful reference for the following discussion.)

Comm_CS_C&DH

<u>Shared Resource *Source*</u>: The module will need a dedicated S/C-ground antenna and one directional antenna *for each* module in the fractionated spacecraft architecture receiving the shared resource. Additionally, the module will require a higher performance computer than what it needed when not sharing this resource to handle the mission (payload) data, TT&C, and C&DH processing from the other modules that are recipients of this shared resource.

<u>Shared Resource *Recipient*</u>: The module no longer requires a dedicated S/C-ground antenna. As compared to when it did not rely on this shared resource, the module now can make use of a lower performance computer for real-time TT&C and C&DH processing because all other processing is sent to the shared resource source(s). Additionally, the module will need one directional antenna for inter-module communication with a module in the fractionated spacecraft that has a dedicated S/C-ground antenna.

[13] While a shared resource affects all subsystems in a spacecraft architecture, only the hardware constituents relating to the specific subsystem(s) involved with the shared resource are enumerated herein.

ADS_GNS

Shared Resource *Source*: The module will require an IMU, star tracker, and one Visual Positioning System (VPS) *for each* module that is a recipient of this shared resource.

Shared Resource *Recipient*: The module no longer requires a dedicated star tracker, and instead requires an IMU and one VPS for determining its relative (and ultimately absolute) position with regard to a module in the fractionated spacecraft that has a star tracker, IMU, and VPS.

Power

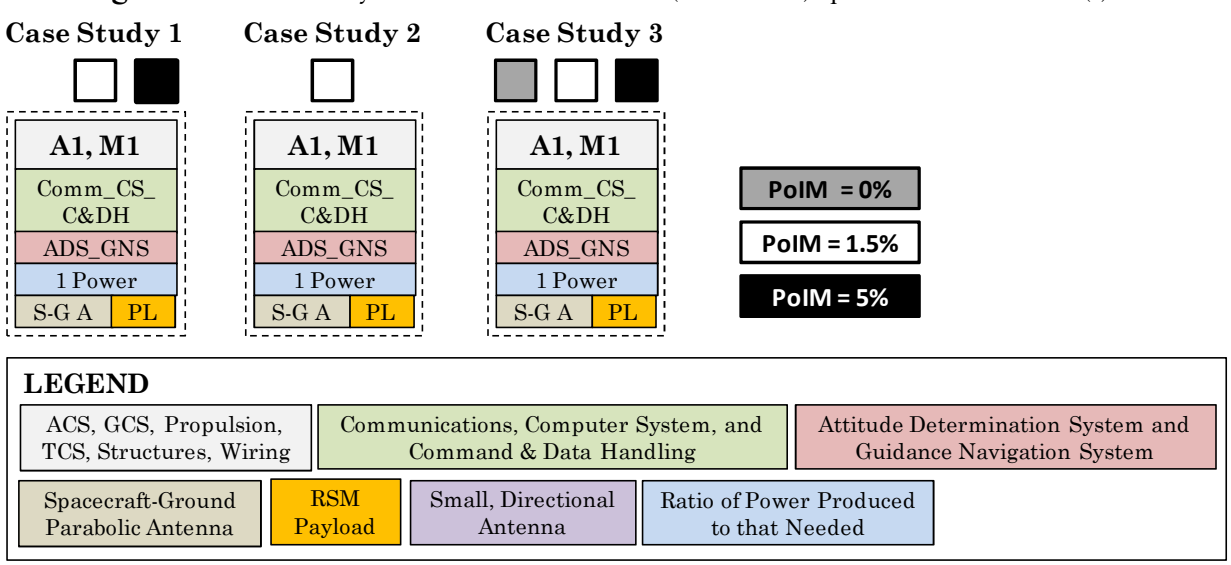
Shared Resource *Source*: As compared to when the module is not a source for this shared resource, the module will need a larger solar array area for power generation, laser diode array with microlens', and additional secondary batteries for storing power during eclipse periods for the modules that are recipients of this shared resource[14].

Shared Resource *Recipient*: As compared to when the module is not a recipient of this shared resource, the module will require a smaller solar array area for producing its own power, smaller secondary batteries for storage of the power it generates, and a solar array for receiving laser beamed/transmitted power from the modules that are sources. If no power is produced and stored by the module, then no solar array is needed for generating power and only very small secondary batteries are needed (for potential disparity between power supply and demand).

Notation

In Figure 4-2 through Figure 4-7, the notation: *n Power* quantifies the ratio of power produced by a module to the power required by the module. If the Payload Module is sharing power, meaning it is receiving power from one or more of the other Infrastructure Modules in the fractionated spacecraft, the Infrastructure Module power will 1 + X, where X is the ratio of the power not being produced by the Payload Module to the number of modules serving as power sources. (It is assumed that supplying power to the Payload Module is evenly distributed amongst Infrastructure Modules in a fractionated spacecraft that are sources for the Power shared resource.)

Figure 4-2. Case Study 1, 2, and 3: one-module (monolithic) spacecraft architecture(s).



[14] For thermal control reasons it is necessary to have the module beam/transmit power continuously throughout a given orbit, daylight and eclipse periods included.

Figure 4-3. Case Study 1: two-module fractionated spacecraft architectures.

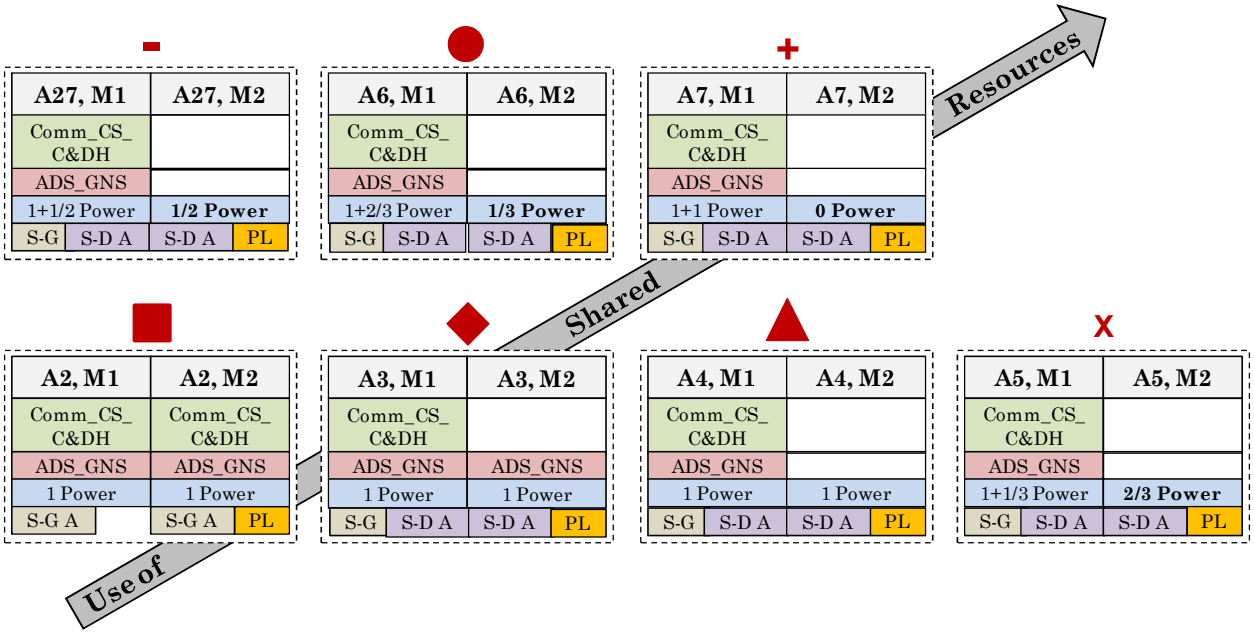


Figure 4-4. Case Study 2 and 3: two-module fractionated spacecraft architectures.

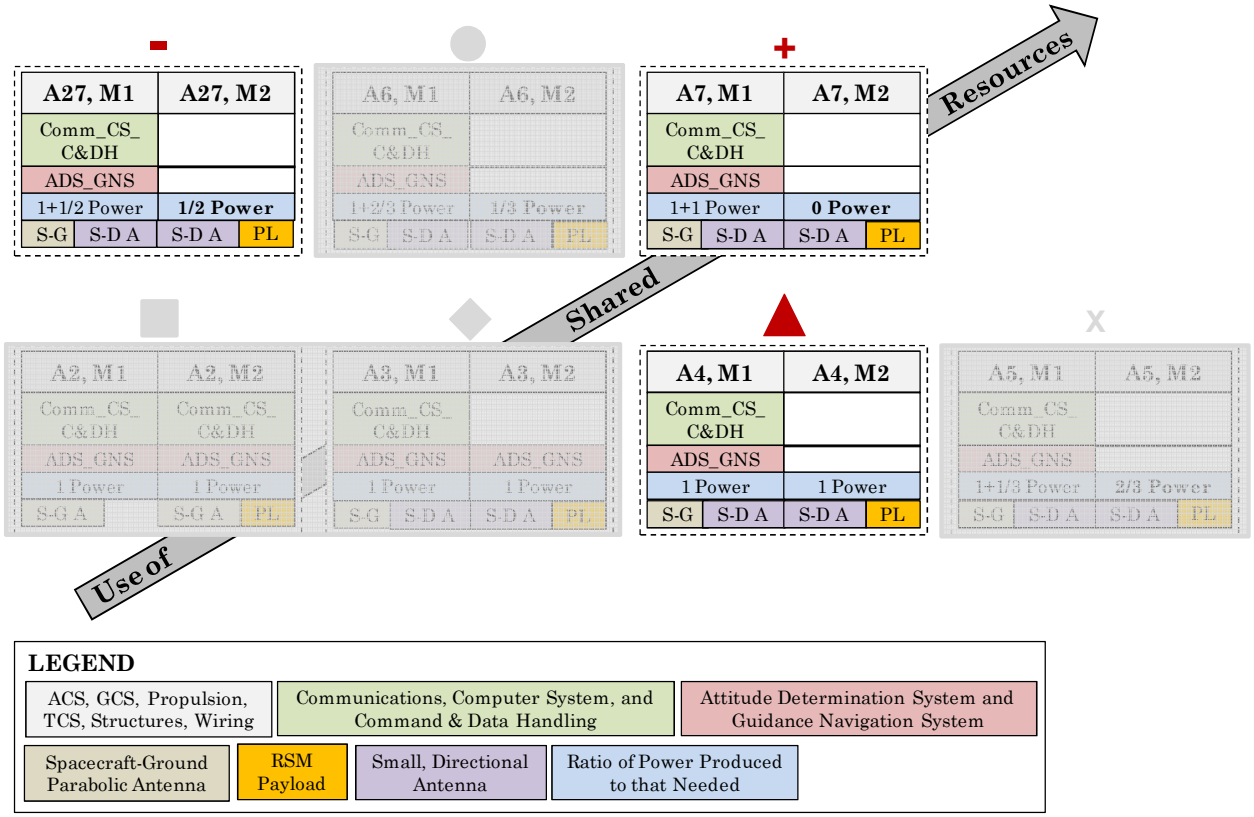


Figure 4-5. Case Study 1: three-module fractionated spacecraft architectures.

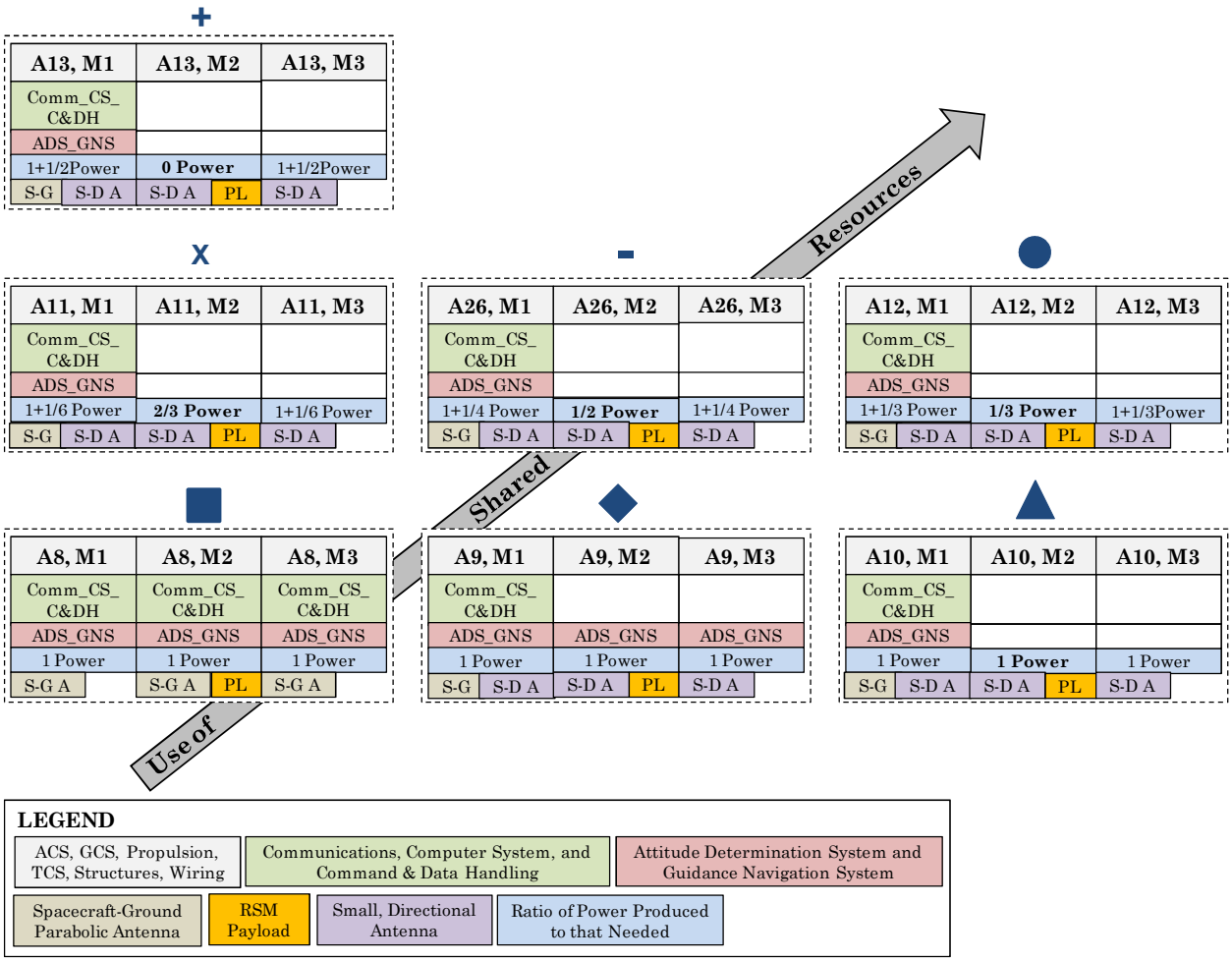


Figure 4-6. Case Study 2 and 3: three-module fractionated spacecraft architectures.

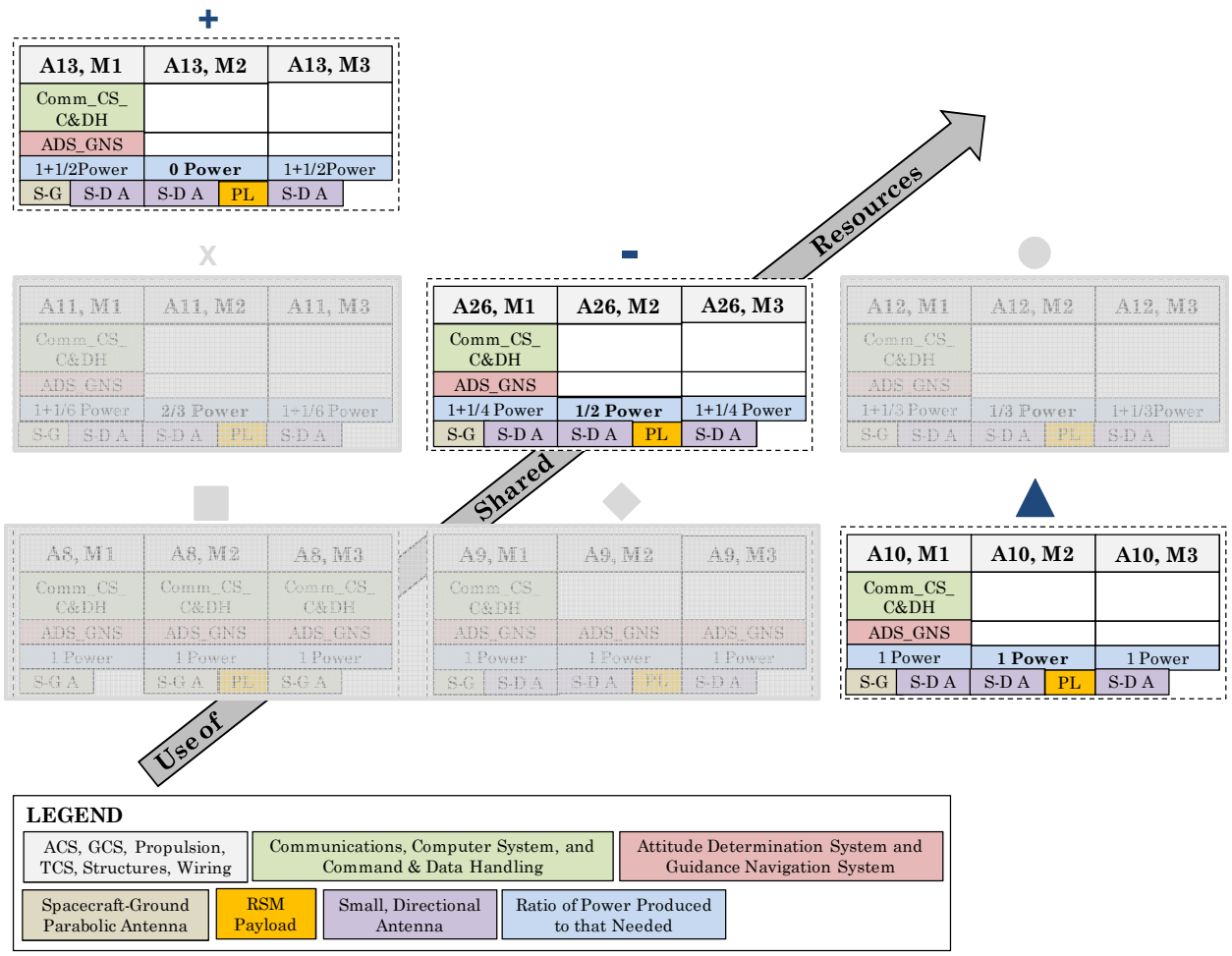
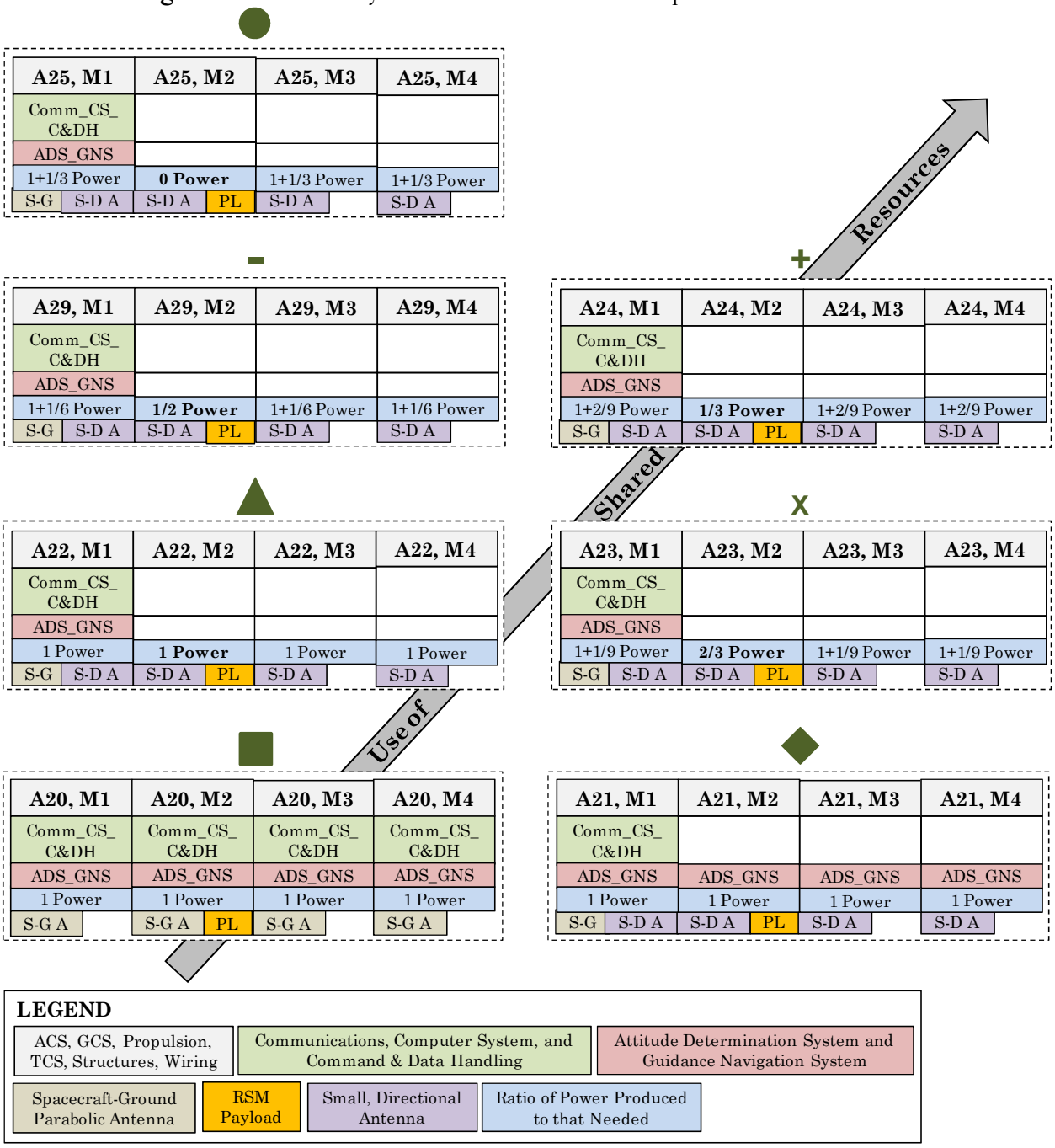


Figure 4-7. Case Study 1: four-module fractionated spacecraft architectures[15].



To serve as an illustrative example for understanding the spacecrafts as they are visual depicted in Figure 4-2 through Figure 4-7, consider Spacecraft Architecture 23 (Arch 23) in Figure 4-7. Module 1 (M1) in this architecture is not relying on any shared resources; hence, it has every subsystem and is a source for all shared resources employed. Since M1 is a source for the Comm_CS_C&DH shared resource it has a dedicated S/C-ground antenna. Additionally, M1 must have three small directional antennas for inter-

[15] No four-module fractionated spacecraft architectures are investigated in the second and third case study.

module communication with the other three modules in the spacecraft as well as have a high performance computer to handle mission (payload) data, TT&C, and C&DH processing from the other modules that are recipients of this shared resource. Due to M1 also being a source for the Power shared resource, M1 must not only produce and store 100% of its own power but also 1/9th of the power required by M2, as M2 is only producing and storing 2/3rd of the power it requires. Subsequently, the solar array for M1 will need to increase in size and it has secondary batteries for storing 1/9th of the power needed by M2 during eclipse periods. And due to M1 also being a source for the ADS_GNS shared resource, M1 requires a dedicated star tracker, IMU, and three VPS's, one for each of the other three-modules in the spacecraft.

Module 2 (M2) is the Payload Module in Arch 23 as contains the (only) RSM payload. Due to M2 being a recipient of the Comm_CS_C&DH and ADS_GNS shared resource, it has a low performance onboard computer system for real-time TT&C and C&DH processing related activities and replaces a dedicated star tracker with a VPS. Additionally, M2 no longer needs a dedicated S/C-ground antenna and instead only needs a small directional antenna for inter-module communication. Moreover, due to M2 being a recipient of the power shared resource, it is now only producing and storing 2/3rd of the power it requires. Subsequently, M2 can generate 2/3rd of its power with a solar array that is smaller than that needed to produce all of its power. And in terms of power storage, the day and eclipse period secondary batteries in M2 can be smaller than they are if M2 was producing all of its power. However, due to M2's reliance on shared power, it does require a receiving solar array for each module that beams/transmits laser power (energy) to it such that each of its respective receiving arrays can capture the airy disk projected by the laser beam directed towards it.

Module 3 and 4 (M3 and M4) are both Infrastructure Modules and are identical. Due to M3 and M4 being recipients of the Comm_CS_C&DH and ADS_GNS shared resource, they each have a low performance onboard computer system for real-time TT&C and C&DH processing related activities and replace a dedicated star tracker with a VPS. Additionally, M3 and M4 no longer need a dedicated S/C-ground antenna and instead only require a small directional antenna for inter-module communication. However since M3 and M4 are sources for the shared Power resource, they must each produce and store 100% of their own power plus 1/9th of the power required by M2, as M2 is producing and storing only 2/3rd of the power it requires. Subsequently, the solar array for both M3 and M4 will need to increase in size and they must each have secondary batteries for storing 1/9th of the power M2 needs during eclipse periods.

The description of Spacecraft Architecture 23 can be applied by analogy to all the other spacecraft architectures considered in the case studies and depicted in Figure 4-2 through Figure 4-7. Across all 22 unique spacecraft architectures considered in the three case studies, for a given use of shared resources, Module 1 is identical and the Payload Module and remaining Infrastructure Modules (if applicable) in the spacecraft architecture are identical in terms of the shared resource profile (*i.e.,* recipient and source distribution). Hence, the two key differentiators between all 22 spacecraft architectures are the number of modules and use of shared resources. As such, the monolithic and fractionated spacecraft architectures considered in the three case studies can be represented in a more condensed form as is shown in Figure 4-8 (Case Study 1), Figure 4-9 (Case Study 2), and Figure 4-10 (Case Study 3).

Figure 4-8. Case Study 1: condensed representation of spacecraft architectures.

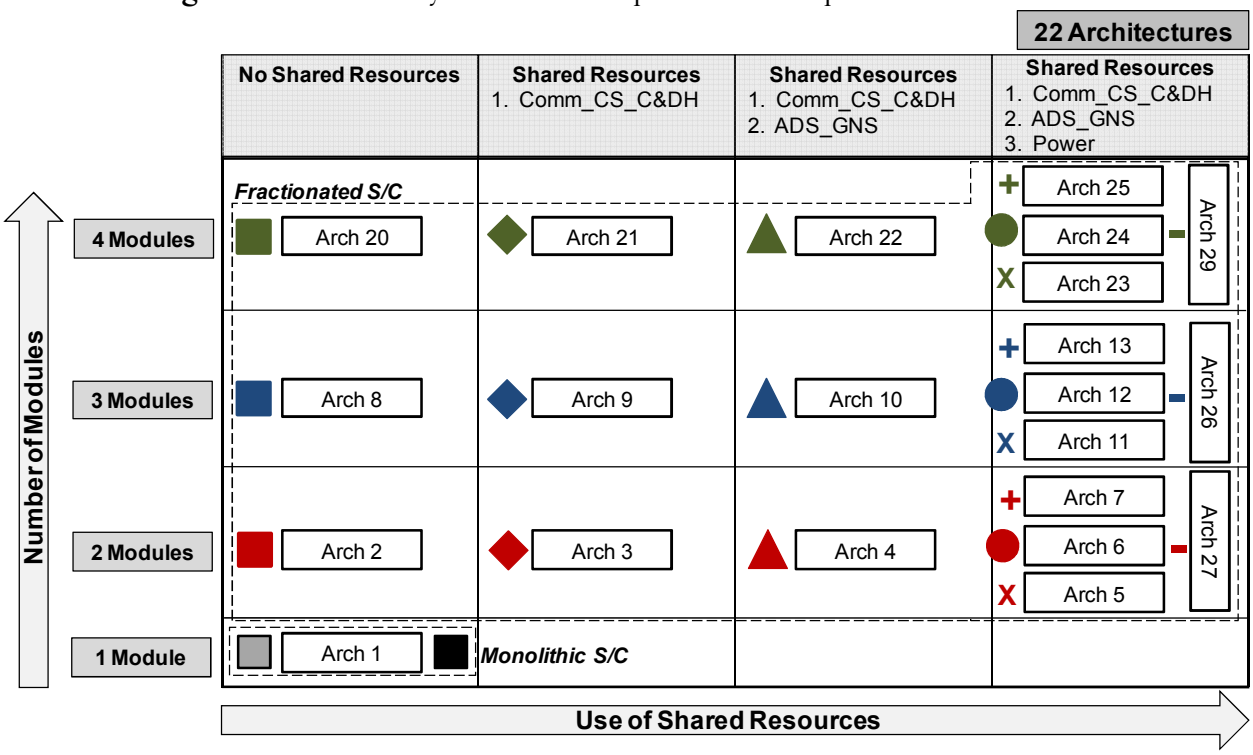


Figure 4-9. Case Study 2: condensed representation of spacecraft architectures.

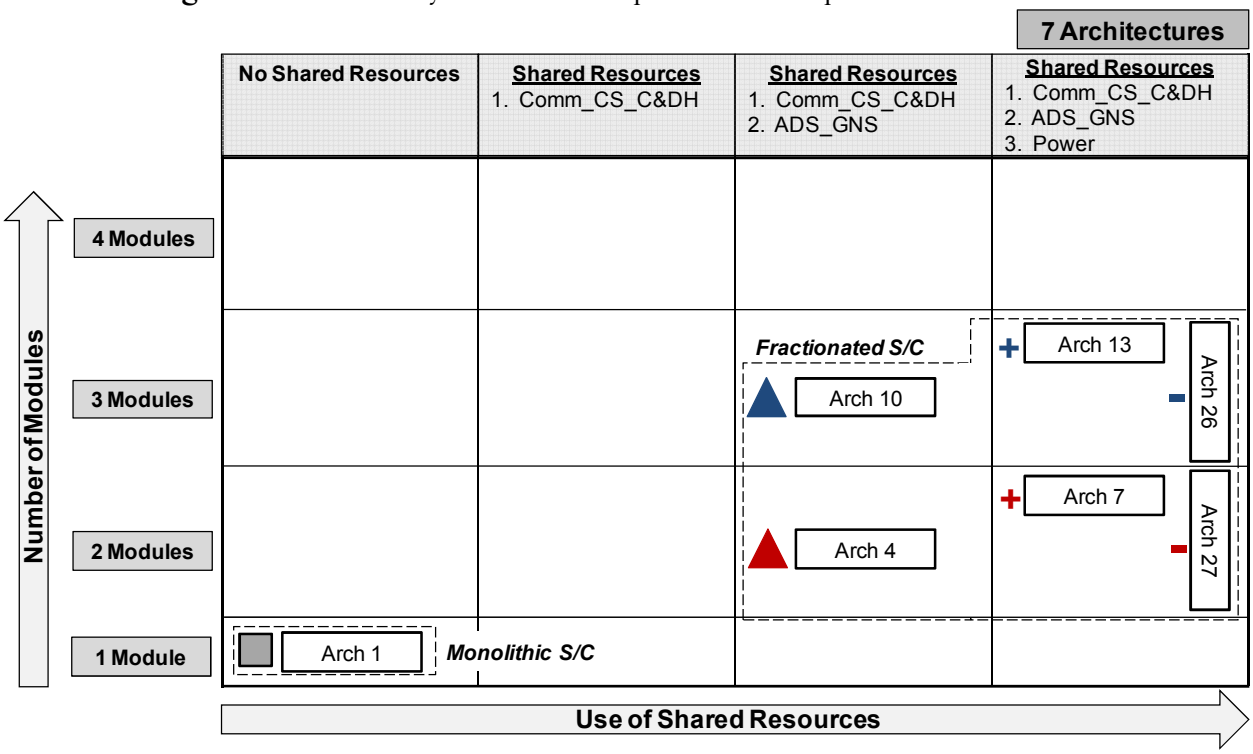
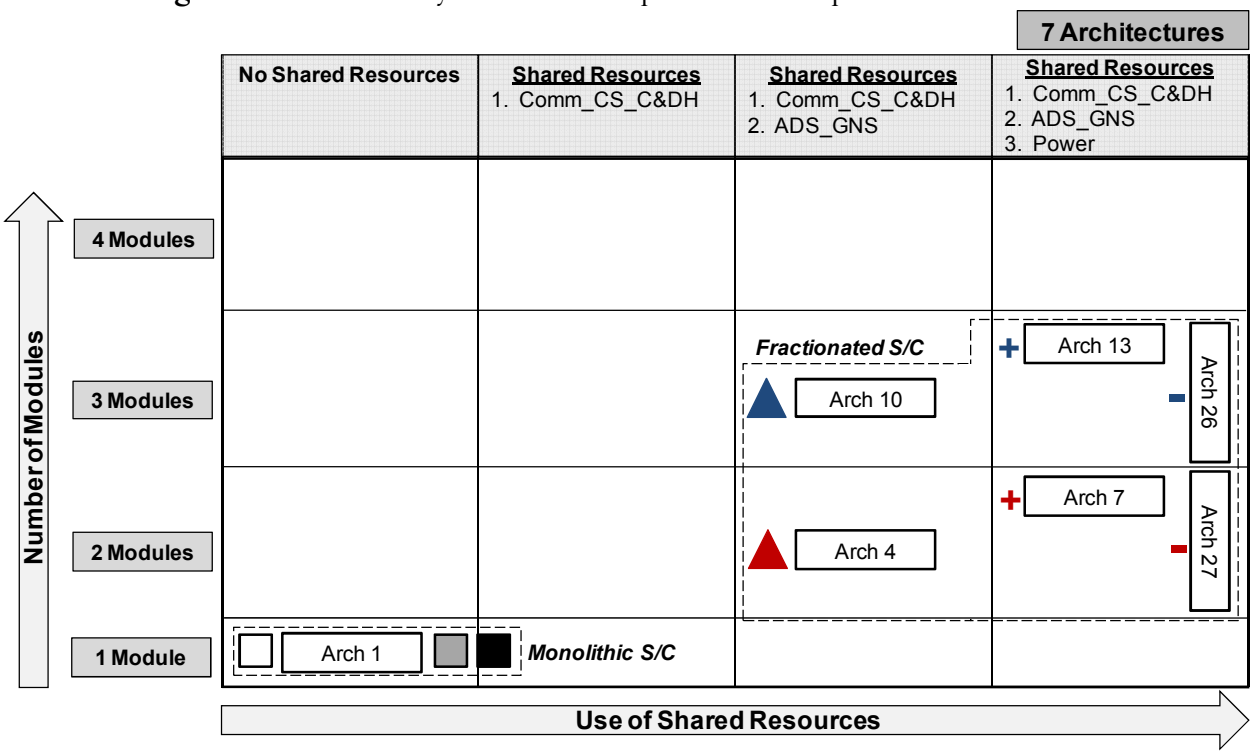


Figure 4-10. Case Study 3: condensed representation of spacecraft architectures.



The unique symbol color and shape displayed next to each of the monolithic and fractionated spacecraft architectures considered in the three case studies, as shown in Figure 4-8 through Figure 4-10, is used to differentiate the spacecraft architectures throughout the results presented in Section 4.3 through 4.7.

4.2. Results Format

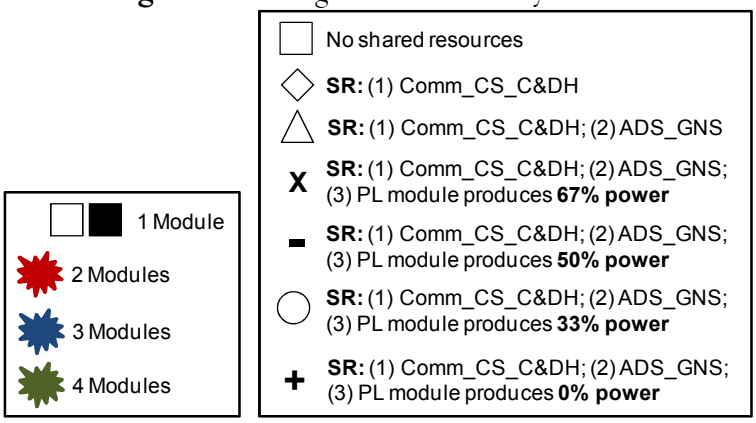
Each set of results in the analysis will be presented hereafter as a figure (plot) relative to one of the three case studies. Subsequently, there are specific formats used to convey the information (dimensions) of data, pertaining to the spacecraft architectures in each set of results, which requires explanation. Section 4.2 therefore will provide an overview of the formatting of the results in Section 4.3 through 4.7. Section 4.2 begins with presenting the legends corresponding to the results for each of the three case studies in the analysis (Section 4.2.1) and following this, a discussion pertaining to lines of data points found in the results (plots) is treated (Section 4.2.2). Thereafter, a description with respect to quantifying the confidence in the dynamic lifecycle cost value of monolithic and fractionated spacecraft is given (Section 4.2.3). Lastly, a discussion is provided pertaining to the manner in which fractionated spacecraft can provide mission lifetime benefits (Section 4.2.4).

4.2.1. Case Study Legends

The results presented in **Section 4.3, 4.4, 4.5.1, 4.6, and 4.7** are visually organized in two-dimensional plots that convey up to six dimensions of information with regard to a given spacecraft architecture, each plot in turn pertaining to one of the three case studies considered in the analysis. In a given set of results, each spacecraft architecture is always characterized by at least its respective (1) x-axis value (*e.g.,* System Mass), (2) y-axis value (*e.g.,* Dynamic LCC), (3) number of modules, and (4) use of shared resources. And then depending on the specific set of results, the architecture may be characterized by its respective (5) inter-module module separation distance, ground resolution, or mission lifetime; and (6) probability of

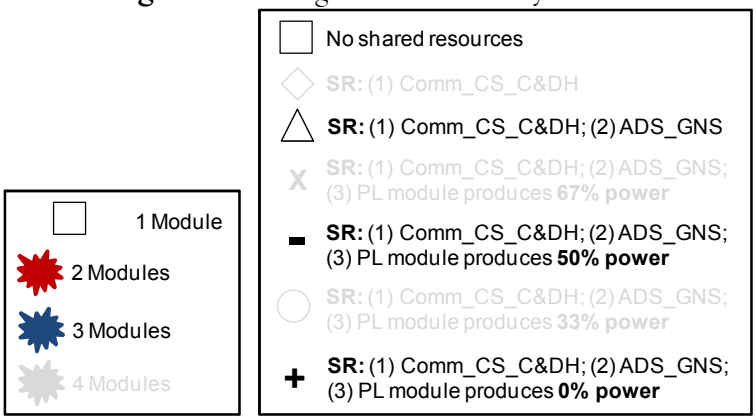
infant mortality. For a given set of results, the (1) x-axis and (2) y-axis values for a given spacecraft architecture can be found along the bottom horizontal and left vertical axes of the results (plot) respectively. And the (3) number of modules, (4) use of shared resources, and (6) probability of infant mortality (if applicable), corresponding to an architecture is designated by the symbolism given in Figure 4-11, Figure 4-12, and Figure 4-13 for Case Study 1, 2, and 3 respectively. Lastly, for a given architecture, the (5) module separation distance, ground resolution, or mission lifetime is characterized with lines passed through data points corresponding to that architecture in the results (see Section 4.2.2).

Figure 4-11. Legend for Case Study 1 results.



In terms of number of modules, WHITE and BLACK squares represent one-module (monolithic) spacecraft architectures; and RED, BLUE, and GREEN shapes represent two, three, and four-module fractionated spacecraft architectures respectively. In terms of shared resources, there are seven unique shapes used to convey the specific shared resources employed in a given spacecraft architecture. Therefore, the legend in Figure 4-11 uniquely defines each spacecraft architecture found in the results pertaining to the first case study. For example, a RED "circle" is Spacecraft Architecture 6 (see Figure 4-3 and Figure 4-4) whereas a GREEN "X" is Spacecraft Architecture 23 (see Figure 4-7).

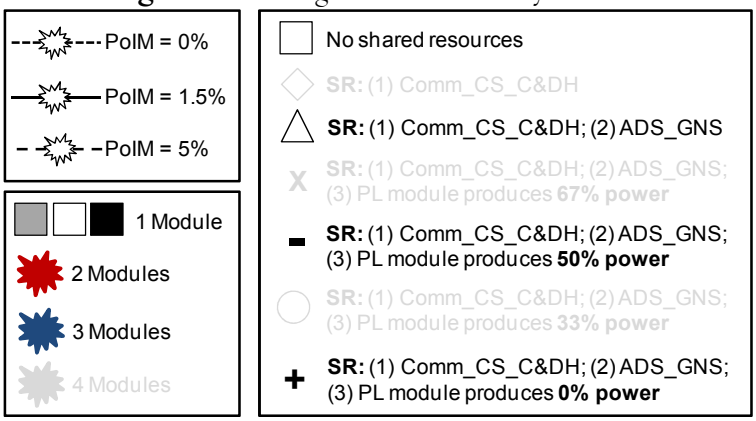
Figure 4-12. Legend for Case Study 2 results.



In terms of number of modules, WHITE squares represent one-module (monolithic) spacecraft architectures; and RED and BLUE shapes represent two and three-module fractionated spacecraft architectures respectively. In terms of shared resources, there are seven unique shapes used to convey the specific shared resources employed in a given spacecraft architecture, four of which are needed to describe the shared resources employed in the seven spacecraft architectures considered in the second case study.

Therefore, the legend in Figure 4-12 uniquely defines each spacecraft architecture found in the results pertaining to the second case study. For example, a RED "triangle" is Spacecraft Architecture 4 (see Figure 4-3 and Figure 4-4), whereas a BLUE "+" is Spacecraft Architecture 13 (see Figure 4-5 and Figure 4-6).

Figure 4-13. Legend for Case Study 3 results.



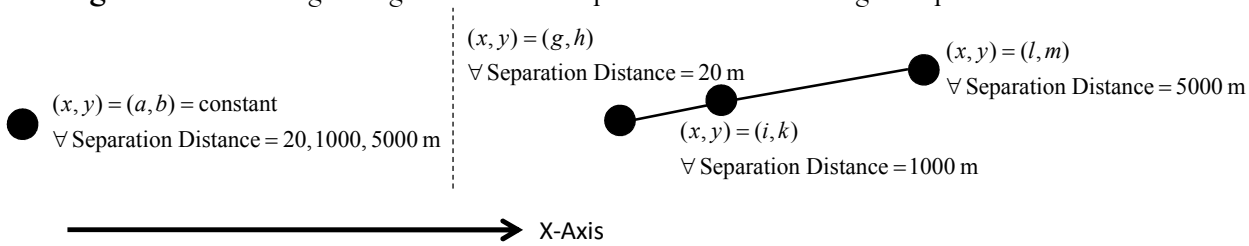
In terms of number of modules, GRAY, WHITE, and BLACK squares represent one-module (monolithic) spacecraft architectures; and RED and BLUE shapes represent two and three-module spacecraft architectures respectively. In terms of shared resources, there are seven different shapes used to convey the specific shared resources employed, four of which are needed to describe the shared resources employed in the seven spacecraft architectures considered in this case study. Therefore, the legend in Figure 4-13 uniquely defines each spacecraft architecture found in the results pertaining to the third case study. For example, a RED "triangle" with a long dashed line through it is Spacecraft Architecture 4 at a PoIM of 5% (see Figure 4-3 and Figure 4-4), whereas a BLUE "+" with a short dashed line through it is Spacecraft Architecture 13 at a PoIM of 0% (see Figure 4-5 and Figure 4-6).

4.2.2. Interpreting Lines of Data Points

The results presented in **Section 4.3, 4.4, 4.5.1, 4.6, and 4.7** may have lines passed through them that characterize either the inter-module separation distance, ground resolution, or mission lifetime of the spacecraft architectures considered in Case Study 1, 2, and 3 respectively.

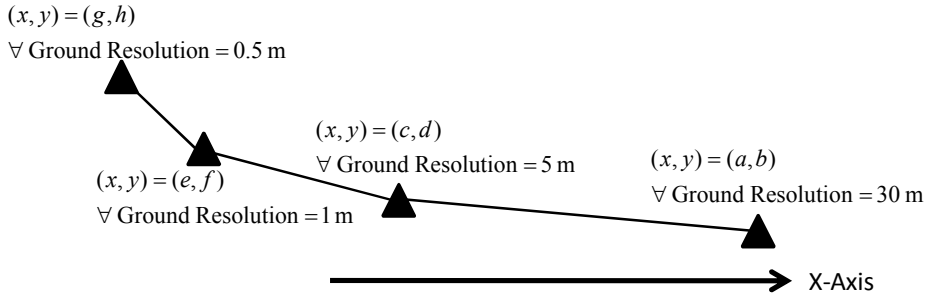
The results generated in **Case Study 1** characterize fractionated spacecraft architectures having three different inter-module separation distances. Therefore, for these result, lines are passed through each of the three data points corresponding to a spacecraft architecture having the three inter-module separation distances considered. For a given line corresponding to a spacecraft architecture, the furthest to the left, middle, and furthest to the right point on the line (with respect to the x-axis), represents the spacecraft architecture with a 20, 1000, and 5000 m separation distance respectively. If there appears to be no line for a spacecraft architecture then the single data point corresponding to that spacecraft architecture represents all three separation distances; this implies that for 20, 1000, and 5000 m separation distances, the y and x values for the spacecraft architecture are (close to) identical. Figure 4-14 notionally depicts the two possible scenarios corresponding to a spacecraft architecture at the three inter-module separation distances considered in Case Study 1.

Figure 4-14. Distinguishing inter-module separation distance for a given spacecraft architecture.



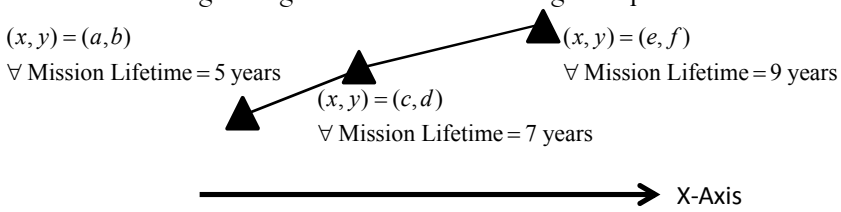
The results generated in **Case Study 2** characterize fractionated spacecraft architectures having four different payload (image) ground resolutions. Therefore, for these results, lines are passed through each of the four data points corresponding to a spacecraft architecture, where each data point signifies the spacecraft architecture at a particular ground resolution. For a given line, in going from left to right along the line (relative to the x-axis), the data points on the line represent the spacecraft architecture with a 0.5, 1, 5, and 30 m ground resolution respectively. Figure 4-15 notionally depicts a line passed through data points corresponding to a spacecraft architecture at the four ground resolution values considered in Case Study 2.

Figure 4-15. Distinguishing ground resolution for a given spacecraft architecture.



The results generated in **Case Study 3** characterize fractionated spacecraft architectures having three different mission lifetime values. Therefore, for these results, lines are passed lines through each of the three data points corresponding to a spacecraft architecture, where each data point corresponds to the spacecraft architecture at one of the three mission lifetime values. For a given line corresponding to a spacecraft architecture, the furthest to the left, middle, and furthest to the right point on the line (with respect to the x-axis), represents the spacecraft architecture with a 5, 7, and 9 year mission lifetime respectively. Figure 4-16 notionally depicts a line passed through data points corresponding to a spacecraft architecture with the three mission lifetime values considered in Case Study 3.

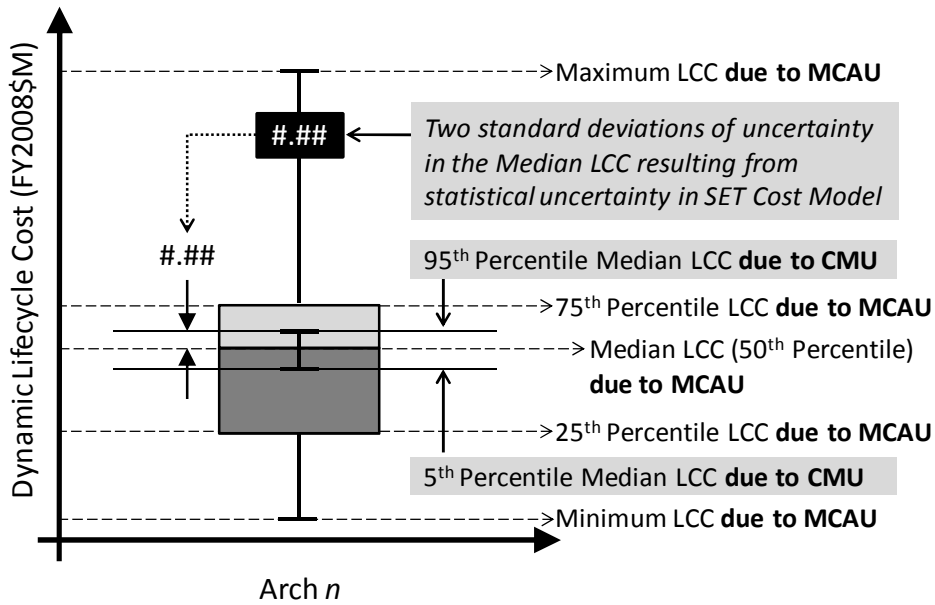
Figure 4-16. Distinguishing mission lifetime for a given spacecraft architecture.



4.2.3. Dynamic Lifecycle Cost Order Statistic, Five-Number Summary

The results presented in **Section 4.5.2** correspond to all three case studies, and specifically quantify the confidence associated with the dynamic lifecycle cost values presented in Section 4.4. The confidence is quantified using an order statistic, five-number summary (aka box-and-whisker plot). The x-axis of these "confidence plots" denotes the spacecraft architecture, whereas the y-axis denotes the Dynamic LCC value for that spacecraft architecture corresponding to each metric used to characterize the uncertainty in its respective Median Dynamic LCC. The uncertainty in the Median Dynamic LCC for a given architecture is notionally represented in Figure 4-17.

Figure 4-17. Representing Dynamic LCC uncertainty relative to the measure of central tendency.



Please refer to Section 2.1.10 for more information describing the origins of each element in Figure 4-17. The order statistic, five-number summary in Figure 4-17 characterizes the central tendency and variability of the Dynamic LCC distribution (*i.e.,* probability density function) for a given spacecraft through the following Dynamic LCC values: maximum, 75th percentile, 50th percentile (median), 25th percentile, and minimum. The 25th-75th percentile range is the inter-quartile range. In addition to these five numbers, the uncertainty in the Median Dynamic LCC due to CMU is shown about the Median Dynamic LCC value. The 5th and 95th percentile confidence values in the Median Dynamic LCC due to CMU are shown by the lower and upper bars in Figure 4-17 about the Median Dynamic LCC, respectively. Due to all Dynamic LCC values depicted in Figure 4-17 being an order statistic they can accurately compare any number of spacecraft Dynamic LCC distributions, regardless of statistical nuances between the distributions.

4.2.4. Mission Lifetime Capability

In Section 4.7, the benefit of extended mission lifetime (*i.e.,* longer mission lifetime capability) relative to that of a comparable monolith, due to employing fractionation, is specifically investigated. The logic for this analysis and subsequent quantification of the mission lifetime benefits provided by fractionation is as follows. Consider a monolithic spacecraft at a given ground resolution, designed for a 7-year mission; subsequently given the mass of the monolith, it will require the use of a certain launch vehicle for

deployment (*e.g.,* Atlas V 500). Now consider the hypothetical situation in which the launch vehicle used for the monolithic spacecraft, performing the 7-year mission, is the biggest launch vehicle that can be used and that the monolith, due to its physical mass and size, is the largest payload that can "fit" in the launch vehicle and still be deployed to the desired operational orbit. Hence, given these constraints, the monolith cannot achieve greater than a 7-year mission lifetime.

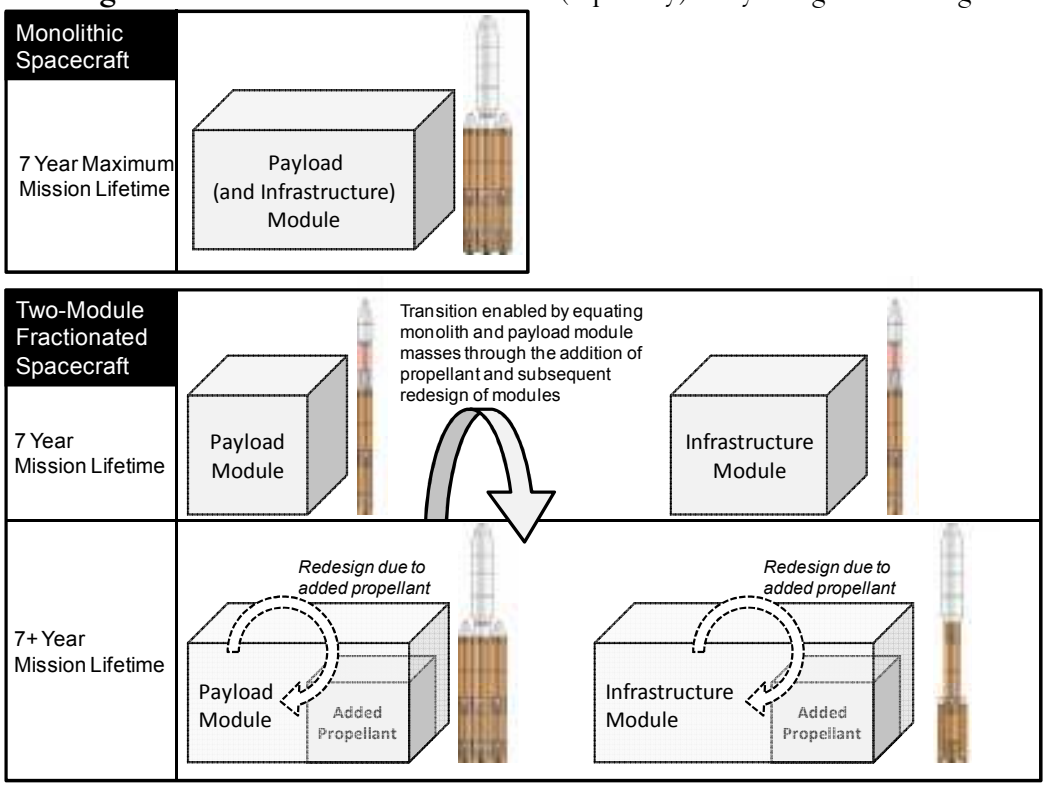
Therefore, in terms of extending the mission lifetime (capability) beyond the 7-year datum (*i.e.,* monolithic) mission lifetime, one option is to use fractionation. Since fractionation can structurally "decompose a monolith" into separate modules, the modules in a fractionated spacecraft can be individually smaller and less massive than the comparable monolith. Therefore, in this hypothetical situation, the potentially smaller size and lesser mass of the modules in a fractionated spacecraft, along with the use of more than one launch vehicle for deployment, can be used to extend the mission lifetime (capability). Specifically, this is done by taking the smaller mass of the Payload Module in a fractionated spacecraft, which is designed for a 7-year mission lifetime, and increasing its mass by adding more propellant. This addition of mass to the Payload Module is done until its respective mass is identical to the mass of the 7-year monolith – thus equating launch payload fairing masses but not exceeding the size and mass constraints hypothetically imposed by the monolith in this study[16]. Therefore, the fractionated spacecraft Payload Module, and subsequently all other modules, has more propellant than is required for the 7-year mission, thereby enabling the fractionated spacecraft to have a longer mission lifetime (capability) than 7 years (*i.e.,* the monoliths maximum mission lifetime). Additionally even though the aggregate size and mass of the fractionated spacecraft modules will be larger than the 7-year monolith, given that the modules in a fractionated spacecraft can be launched on multiple launch vehicles, the fractionated spacecraft can be deployed without violating the hypothetical maximum size and mass constraints imposed by the monolith for this analysis. The logic for this analysis is depicted visually in Figure 4-18 for a two-module fractionated spacecraft architecture and a 7 year datum mission lifetime (*i.e.,* the monolith cannot achieve more than a 7 year mission lifetime).

The motivation for understanding monolithic and fractionated spacecraft value propositions in terms of mission lifetime benefits, is to provide an instantiation of the advantages provided fractionated spacecraft through their ability to effectively reduce the size and mass of the launch vehicle (*i.e.,* payload fairing) elements relative to a comparable monolith. Through the quantitative assessments of potential mission lifetime extensions performed through this analysis, it forms a response to the suppliant question; is the potentially longer mission lifetime capability of fractionated spacecraft, relative to that of a comparable monolith, worth the potentially larger LCC of these fractionated spacecraft? In terms of applicability of this hypothetical situation to the "real world", one can readily envision a situation in which for political, financial, territorial, or preferential reasons that the launch of a spacecraft may be limited to a certain set of launch vehicles and hence largest launch vehicle. Subsequently, there will be limits as to the largest

[16] The extended mission lifetime of the fractionated spacecraft is a function of both the amount of propellant added but also the physical architecture of the spacecraft (modules) because as you add more mass, the architecture changes. As such, the analysis does not simply (and incorrectly) take a fractionated spacecraft architecture for a 7-year mission lifetime and add "X" kg of propellant to it so the Payload Module mass equals that of a comparable monolith. This would bias fractionated spacecraft that share the most resources and are subsequently the most massive, as always having the longest mission lifetime capabilities. Therefore, the mission lifetime extension analysis fully accounts for the lifetime being a function of both propellant added and the architecture. Subsequently, the analysis as performs successive iterations over a fractionated spacecraft architecture so that the mass balance between the Payload Module and monolith is achieved and the architecture (design) of all modules in the fractionated spacecraft are accurate to this mass balance. Therefore, the extended mission lifetime capability of a given fractionated spacecraft architecture shown in Section 4.7 is entirely accurate to that mission lifetime value.

structure(s) (*i.e.,* spacecraft or modules), in terms of size and mass, which can "fit" in the largest launch vehicle and still get the structure(s) to the desired destination orbit. This therefore imposes a "cap" on the longest mission lifetime that can be achieved with a monolith. Subsequently, the results from this analysis quantitatively explore the implications of fractionation, specifically with regard to LCC, for extending the mission lifetime capability of fractionated spacecraft relative to that of a comparable the monolith, which in turn, is only able to achieve an *n*-year mission lifetime, given the launch vehicle payload fairing size and mass constraints ("cap").

Figure 4-18. Mission lifetime extension (capability) analysis logic/reasoning.



4.3. System Mass and Propellant Usage

System Mass and Propellant Usage are the first two elements of monolithic and fractionated spacecraft value propositions to be enumerated in the Analysis. First, the relationship between the System Mass and Propellant Usage is quantified and discussed (see Section 4.3.1). Following this, the System Mass of monolithic and fractionated spacecraft is quantified with an emphasis on understanding the relationship between System Mass and Dynamic LCC (see Section 4.3.1 and 4.3.2). Lastly, the System Propellant Usage of monolithic and fractionated spacecraft is quantified with an emphasis on understanding the relationship between System Propellant Usage and Dynamic LCC (see Section 4.3.1 and 4.3.3).

4.3.1. Trends in System Mass and Propellant Usage

For monolithic and fractionated spacecraft architectures, System Propellant Usage is always less than System Mass and additionally System Mass and Propellant Usage have a positive, linear correlation. Intuitively, for a given monolithic and fractionated spacecraft architecture, the System Propellant Usage will be less than the System Mass because the Propellant is only one constituent of the System Mass. Therefore, as is shown in Figure 4-19 and Figure 4-20, the System Mass is always less than Propellant Usage for monolithic and fractionated spacecraft. The lesser magnitude of System Propellant Usage than Mass is a consistent trend observed, but additionally, the relationship between System Mass and Propellant Usage for a given monolithic or fractionated spacecraft is linearly positive (see Figure 4-19). The specific reason for this linearly positive relationship is that Propellant Usage scales linearly with the inertia of a given monolithic or fractionated spacecraft architecture (*i.e.,* modules). The inertia in turn, is a linear function of a spacecraft architecture's physical size (*i.e.,* dimensions) as well as its respective System Mass. Therefore, Propellant Usage should always scale in a linear-positive manner with respect to System Mass, as is demonstrated in Figure 4-19.

Figure 4-19 characterizes the 22 spacecraft architectures investigated in Case Study 1 with respect to their (y) System Propellant Usage (kg) and (x) System Mass (kg).

Figure 4-19. Trends in System Mass and Propellant Usage.

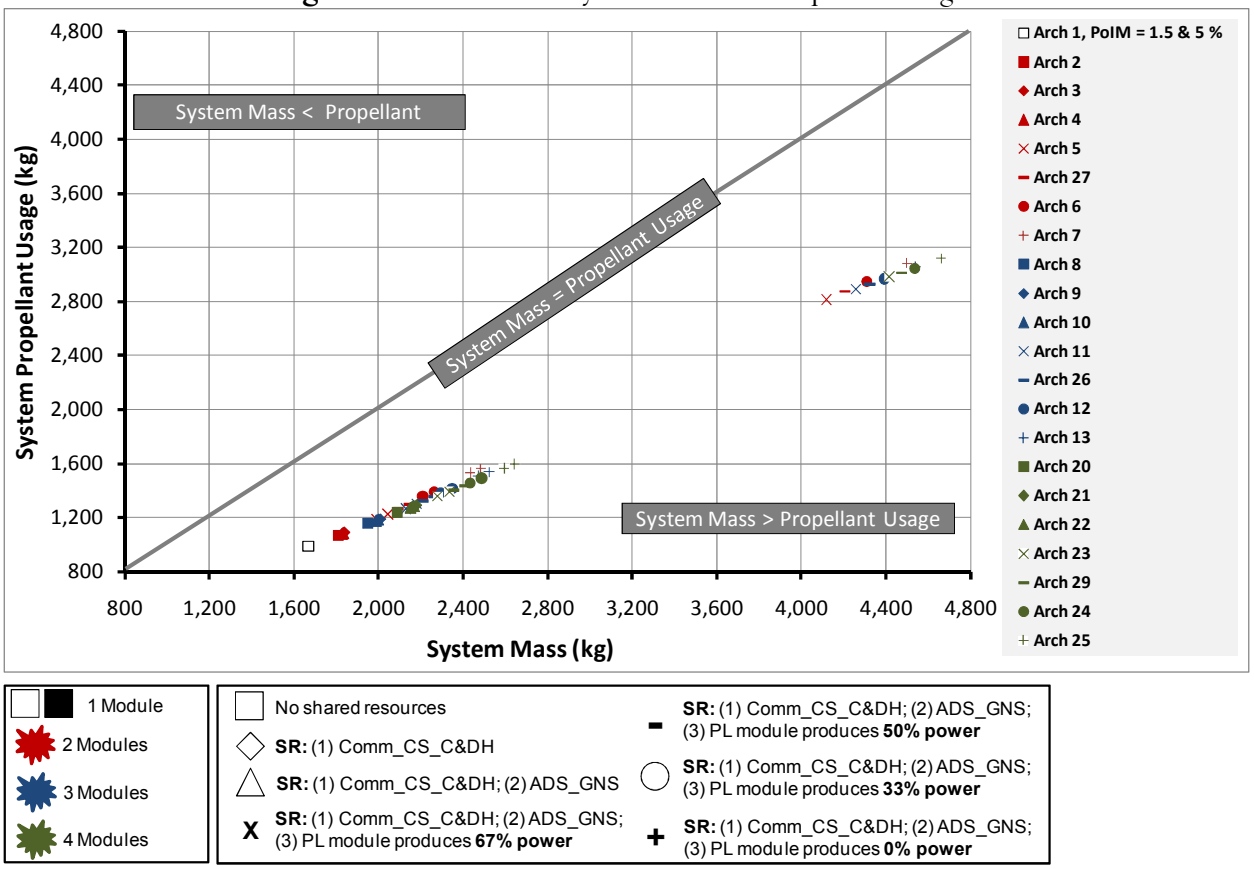
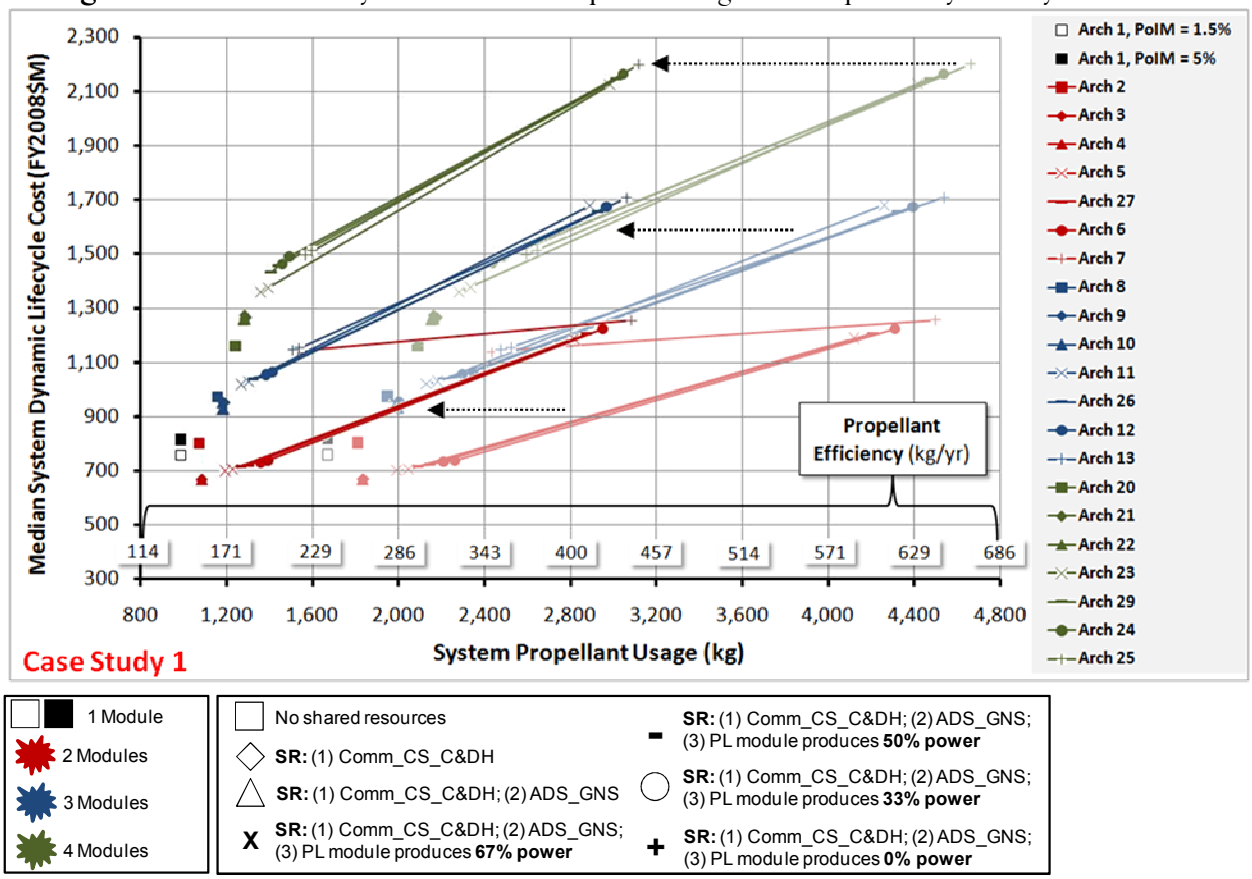


Figure 4-20 characterizes the 22 spacecraft architectures investigated in Case Study 1 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Propellant Usage (kg). In Figure 4-20, the *light colored* data points and lines represent the System Dynamic LCC v. *System Mass* trends.

Figure 4-20. Trends in System Mass and Propellant Usage with respect to System Dynamic LCC.



Due to the linearly positive relationship between System Mass and Propellant Usage, all *relative* trends and insights elicited from results with regard to System Dynamic LCC v. System Mass and Propellant Usage will be identical. A specific instantiation of this is the linear relationship between System Mass and Propellant Usage shown in Figure 4-20. In Figure 4-20, the results with respect to Dynamic LCC and System Mass (light colored points and lines) are shown to be a linear transformation of the results with respect to Dynamic LCC and System Propellant Usage (dark colored points and lines) and vice versa. Subsequently, the *relative* trends and insights enumerated in Section 4.3.1, 4.3.2, and 4.3.3 with regard to results plotted (positioned) with respect to System Dynamic LCC and Mass can be used to infer *relative* trends and insights with regard to the same set of results plotted with respect to System Dynamic LCC and Propellant Usage. However, note that trends and insights with regard to *absolute* measures of System Mass and Propellant Usage are not the same because on an absolute scale, System Propellant Usage is different in magnitude from System Mass, as is evident in Figure 4-19 and Figure 4-20.

As shown in Figure 4-20, the System Mass (and aggregate physical size) of fractionated spacecraft will always be larger than the System Mass (and size) of comparable monolithic spacecraft. The reason as to the consistently larger mass of fractionated spacecraft is that they have a higher *system-wide redundancy*. System-

wide redundancy is defined as the total number of a given subsystem present in a fractionated spacecraft relative to one, as a monolith only has one of these subsystems. Consequently system-wide redundancy has a mass (and size) penalty associated with it, manifested in the form of mass and structural overhead that is, the extra mass and size incurred when replacing one subsystem with a set of smaller subsystems, holding functionality roughly constant. For example, a fractionated spacecraft with three modules will have three thermal control Systems (TCS's) instead of one TCS that a comparable monolith has, and therefore the fractionated spacecraft has system-wide redundancy. Thus, for a given subsystem, due to the mass and structural overhead associated with that subsystem on each of the *n* modules in a fractionated spacecraft, the aggregate mass (and size) of those *n* subsystems will be larger than the respective mass and size of that subsystem on the monolith. In extrapolating this logic to every subsystem in a spacecraft, the aggregate mass (*i.e.,* System Mass) of a fractionated spacecraft, on the basis of its respective subsystems, will always be larger than that of a comparable monolith. Moreover, as the number of modules increases, the mass and structural overhead associated with system-wide redundancy proportionally increases. Therefore, increasing the number of modules only further exacerbates the mass disparity between comparable fractionated and monolithic spacecraft.

Employing shared resources also contributes to the System Mass disparity between comparable monolithic and fractionated spacecraft, albeit adversely for fractionated spacecraft. Employing shared resources in a fractionated spacecraft changes the distribution of mass (and size) throughout a fractionated spacecraft's respective modules. And through this change in mass (and size) distribution, sharing resources always increases the System Mass of fractionated spacecraft relative to that of a comparable monolith; hence, sharing resources does not provide *System* Mass savings. The reason for this is that hardware associated with sharing resources (see Section 2.1.5 and 4.1.3) always has a larger aggregate mass (and size) than that of the collective hardware associated with not sharing those resources. However, despite this, the motivation for employing shared resources lies in the fact that they can reduce the effective mass (and size) of the Payload Module (see Section 4.6 and 4.7).

Fractionated spacecraft that (1) do not share any resources, (2) share only the Comm_CS_C&DH resource, and (3) share the Comm_CS_C&DH and ADS_GNS resource, have appreciably similar (if not identical) System Masses, which, in addition, are independent of separation distance, as is shown in Figure 4-20. The reason for this is that the hardware associated with sharing the Comm_CS_C&DH and ADS_GNS resource (see Section 2.1.5 and 4.1.3) does not generate an appreciable System Mass penalty for a fractionated spacecraft employing either or both of these shared resources, as compared to a fractionated spacecraft not employing these shared resources. For example, consider the subsystem hardware (and hence mass) composition differences in Arch 3 and 4 in Figure 4-3 as compared to that of a fractionated spacecraft not employing shared resources, Arch 1 in Figure 4-2. Additionally, the mass of the hardware associated with sharing the Comm_CS_C&DH and ADS_GNS resources does not change much, if at all, in response to changing inter-module separation distance. Hence, the respective System Mass of fractionated spacecraft that (1) do not share any resources, (2) share only the Comm_CS_C&DH resource, and (3) share the Comm_CS_C&DH and ADS_GNS is nearly (if not) constant with respect to inter-module separation distance.

In contrast, once fractionated spacecraft share the Power (generation and storage) subsystem resource, this increases their System Mass relative to that of a comparable monolith. Additionally, for fractionated spacecraft with the same number of modules, the System Mass increases uniformly with respect to the amount of Power shared (*i.e.,* power not produced or stored by the recipient modules) and inter-module separation distance. The main hardware constituents associated with sharing Power (see Section 2.1.5 and

4.1.3), in terms of driving System Mass differences between fractionated spacecraft architectures, are a laser diode array (for modules that are sources) and a solar array (for modules that are sources and recipients). If the separation distance is constant, then the receiving solar array on the recipient module(s) will remain constant in size and mass, regardless of the amount of power shared; the size and mass of this solar array is not dictated by power shared, but rather by the size of the airy disk[17], which is constant if separation distance is constant. However, for a given separation distance, if the power being transmitted to the recipient module(s) (*i.e.,* shared) increases, this increases the size and thus mass of the solar array for generating power on the source (infrastructure) modules. The solar array on the source modules, in terms of size and mass, ends up scaling close to linearly with the power being transmitted to the recipient module(s); by a factor of about 1.5 for every 1/3 increase in power transmitted to the recipient module(s). Therefore, if the inter-module separation distance is constant but the power being shared (transmitted) increases, this causes a uniform increase in System Mass of fractionated spacecraft, as can be observed in comparing fractionated spacecraft in Figure 4-20 that share different amounts of the Power resource but all have the same inter-module separation distance and number of modules.

Alternatively, if the amount of Power shared in a given fractionated spacecraft is held constant but the separation distance between its respective modules increases, this uniformly increases its System Mass. The size and mass of the laser diode and solar array on the modules that are sources for the Power shared resource remains constant, regardless of the separation distance, as their size and mass only depends on the amount of power transmitted to the recipient module(s) (*i.e.,* shared). However, the solar array for receiving the transmitted power on the recipient module(s) does increase with separation distance; recall that this is necessary to capture the airy disk created by the laser beam, which in turn, grows in size as the separation distance increases. The size of the airy disk and subsequently the size and mass of the receiving solar array on the recipient module(s), scales uniformly with separation distance. Moreover, as the separation distance increases, the inter-module directional antennas employed for sharing the Comm_CS_C&DH resource, increase in mass (and size) proportionally, although, these do not have as dominant an effect as the receiving solar array on the recipient module(s) for increasing the System Mass. Therefore holding the amount of Power shared constant in a fractionated spacecraft, as the inter-module separation distances increases, the System Mass uniformly increases (in a linear fashion). This is evident in observing the uniform change in System Mass with inter-module separation distance for a given fractionated spacecraft sharing the Power resource in Figure 4-20.

4.3.2. System Mass

In addition to quantifying the System Mass of fractionated spacecraft with regard to shared resources and inter-module separation distance (see latter part of Section 4.3.1), the System Mass can be examined with regard to Ground Resolution (*i.e.,* payload performance). Subsequently, this is the topic of Section 4.3.2.

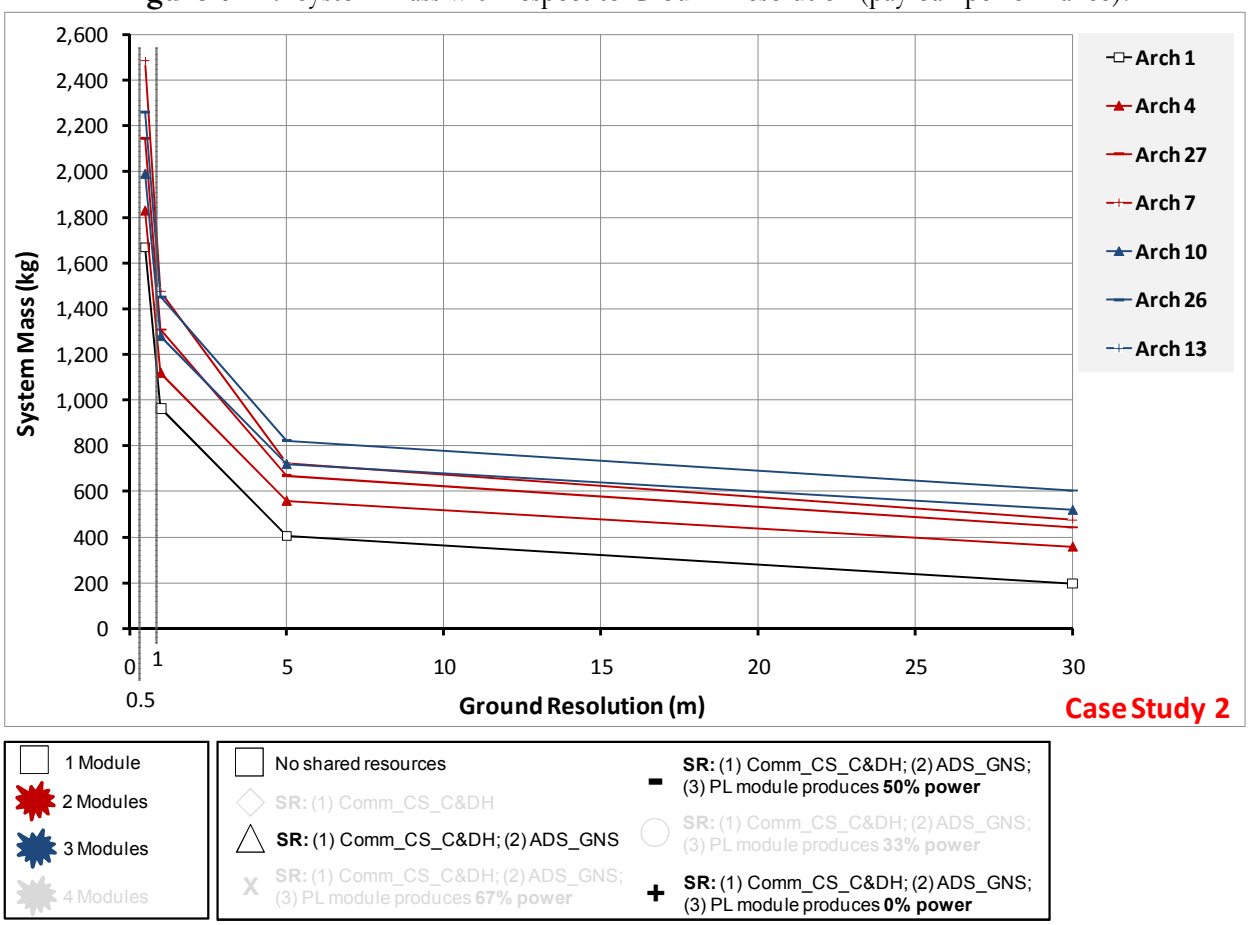
As is shown in Figure 4-21, regardless of Ground Resolution, monolithic spacecraft are always less massive than comparable fractionated spacecraft, which is consistent with the discussion pertaining to the results in Figure 4-20. Also consistent with the results in Figure 4-20 is the observation in Figure 4-21 that as the use of shared resources or number of modules increases in a fractionated spacecraft, their respective System Mass proportionally increases relative to that of a comparable monolithic spacecraft. Specifically, with regard to trends in System Mass, with respect to varying Ground Resolution, the System Mass of monolithic and fractionated spacecraft exponentially decays with a decreasing Ground Resolution. The

[17] The airy disk is region of "best" focused light emitted by a laser diode array. The airy disk forms a circular pattern and contains roughly 90% of the energy contained in a laser beam transmitted/sent to recipient modules.

reason for this exponential trend is that the RSM payload mass scales with the inverse of the payload Ground Resolution, as a subsequent result of the RSM payload model in the SET. And since the payload design drives the design (and hence size and mass) of monolithic and fractionated spacecraft architectures, there is a strong positive correlation between the System Mass and payload mass for a given spacecraft architecture. Therefore, it is expected that System Mass of the spacecraft architectures in Figure 4-21 will scale inversely with Ground Resolution.

Figure 4-21 characterizes the 7 spacecraft architectures investigated in Case Study 2 with respect to their (y) System Mass (kg) and (x) Payload Ground Resolution (m).

Figure 4-21. System Mass with respect to Ground Resolution (payload performance).



4.3.3. System Propellant Usage

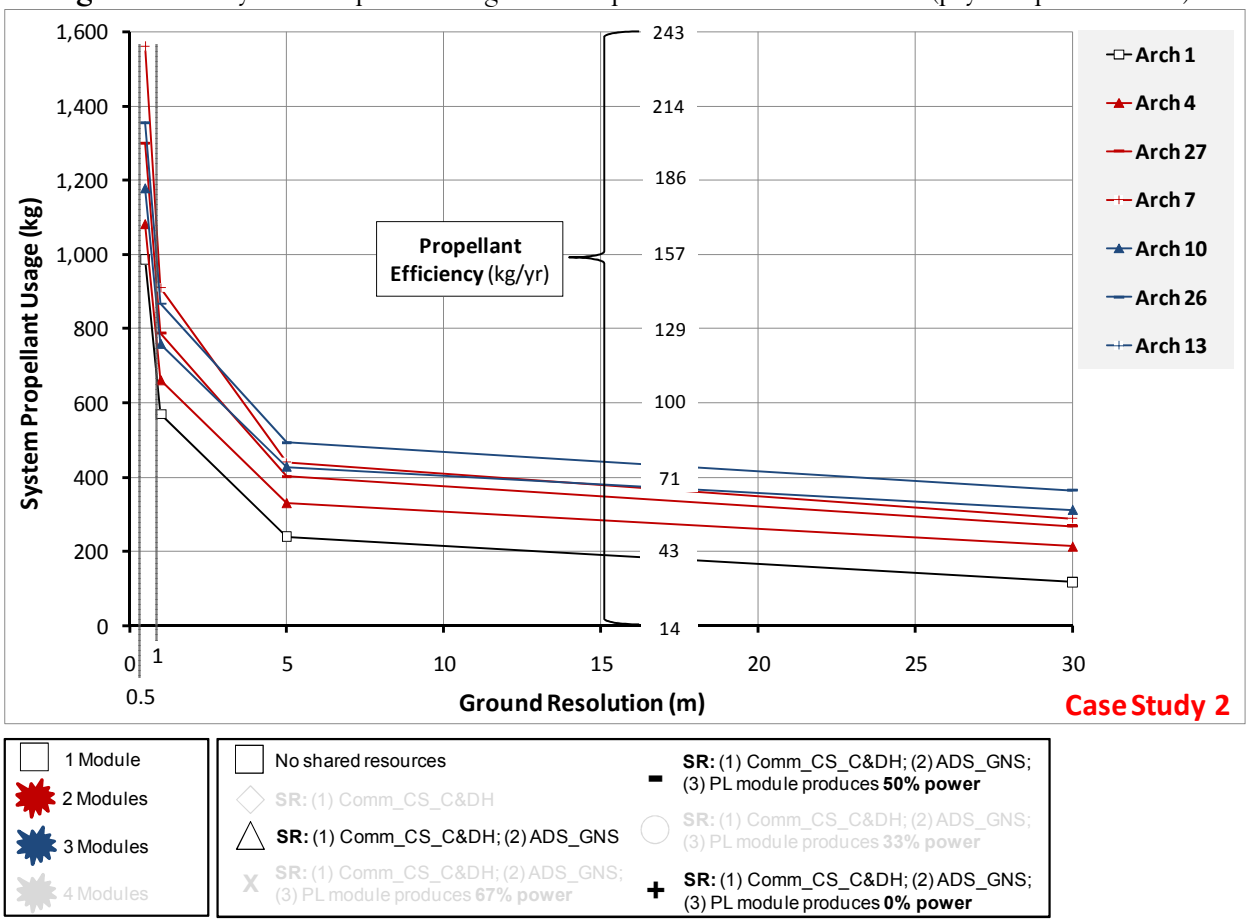
In addition to quantifying the System Propellant Usage of fractionated spacecraft with regard to shared resources and inter-module separation distance (see earlier part of Section 4.3.1), the System Propellant Usage can be examined with regard to Ground Resolution (payload performance) as well as mission lifetime and probability of infant mortality (PoIM). Subsequently, this is the topic of Section 4.3.3.

In Figure 4-22 and Figure 4-23, the consistently lower System Propellant Usage of monolithic spacecraft as compared to fractionated spacecraft follows the line of reasoning provided in the discussion of the results found in Figure 4-19 and Figure 4-20, since System Propellant Usage correlates linearly with System Mass.

Subsequently, this line of reasoning explains the trend observed in Figure 4-22 and Figure 4-23 that as the use of shared resources or number of modules increases, their System Propellant Usage proportionally increases relative to that of a comparable monolithic spacecraft. And in terms of Figure 4-22, the exponential decrease in System Propellant Usage with a decreasing Ground Resolution, follows the same line of reasoning supplied in the discussion pertaining to the results shown in Figure 4-21.

Figure 4-22 characterizes the 7 spacecraft architectures investigated in Case Study 2 with respect to their (y) System Propellant Usage (kg) and (x) Payload Ground Resolution (m).

Figure 4-22. System Propellant Usage with respect to Ground Resolution (payload performance).

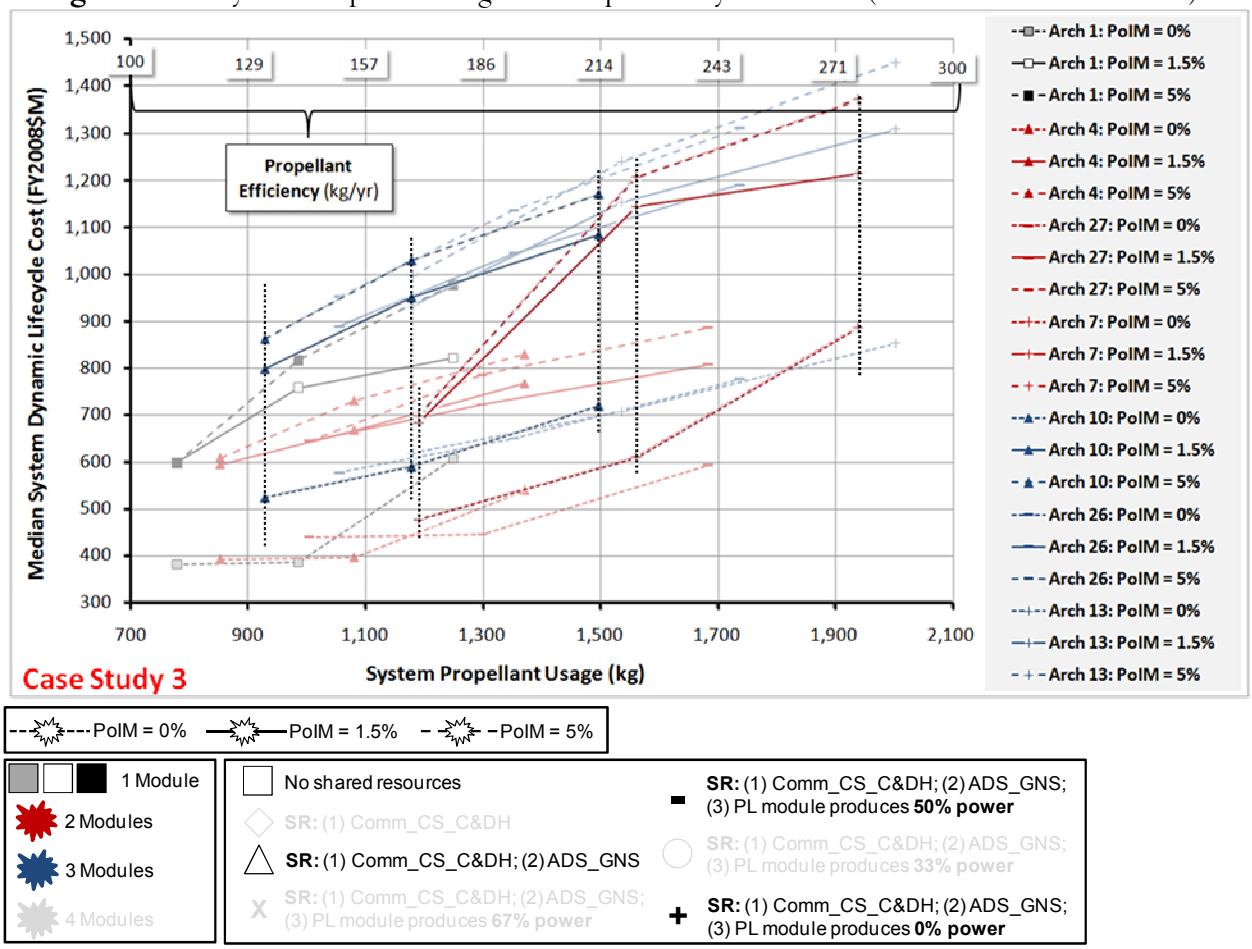


The results shown in Figure 4-23 on the following page quantify the implications of nuances in spacecraft architectures with regard to the risk resulting from lifecycle uncertainties and mission lifetime. Subsequently, the results in Figure 4-23 show the spacecraft architectures considered in the third case study with respect to all nine combinations of mission lifetime and PoIM values considered in the case study (*e.g.*, 7 years and 1.5% respectively). The severity (harshness) of a spacecraft lifecycle increases with an increasing probability of occurrence of risks resulting from lifecycle uncertainties, which in turn, increases due to an increase in mission lifetime and/or PoIM. The large number of data points and lines in Figure 4-23 may appear confusing at first; however, one manner to readily compare the monolithic and fractionated spacecraft in Figure 4-23 is to consider either a constant mission lifetime *or* PoIM. Considering a constant mission lifetime requires that the same data point (from left to right) on each spacecraft architectures' respective line of the data points be compared (*e.g.*, consider the middle data point on each

line which corresponds to a mission lifetime of 7 years). Alternatively, considering a constant PoIM requires that only the respective data points for lines having the same dash style (*i.e.,* constant PoIM) for spacecraft architecture's be compared.

Figure 4-23 characterizes the 7 spacecraft architectures investigated in Case Study 3 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Propellant Usage.

Figure 4-23. System Propellant Usage with respect to Dynamic LCC (Mission Lifetime and PoIM).



For a given spacecraft architecture and PoIM in Figure 4-23, as the mission lifetime increases, the System Propellant Usage, and hence System Mass, increases. (Recall that System Mass is a linear transformation of System Propellant Usage; see Figure 4-19.) The reason for this trend is that if the PoIM is constant, the longer the mission lifetime, the more Propellant required for on-orbit stationkeeping; hence System Propellant Usage and Mass increase with mission lifetime. Specifically, the propellant required for stationkeeping increases with an increasing mission lifetime because of having to operate longer in space, but it also increases in part, due to spacecraft architectures becoming more massive (aside from the increased propellant) as the mission lifetime increases. As the mission lifetime increases, the subsystems in monolithic and fractionated spacecraft architectures generally become more massive and power consuming, due to a "trickledown" effect, from the need to address degradation effects in a spacecraft. For example, a solar array responsible for generating a given amount of power will be larger and more massive for a 9-year

than 5-year mission. Subsequently, this "trickledown effect" has implications for increasing the size and mass of numerous subsystems within a spacecraft.

Another trend present in the results in Figure 4-23 is that for a given spacecraft architecture and mission lifetime, as the PoIM changes in magnitude, the System Mass of the architecture remains constant. The reason for this is that the physical characteristics (*e.g.,* Mass, power consumption, heat dissipation) of a given spacecraft architecture is independent of the PoIM. The PoIM only influences the risks resulting lifecycle uncertainties that can occur throughout a spacecraft's respective lifecycle, and thus PoIM is a factor exogenous to the design of a given spacecraft.

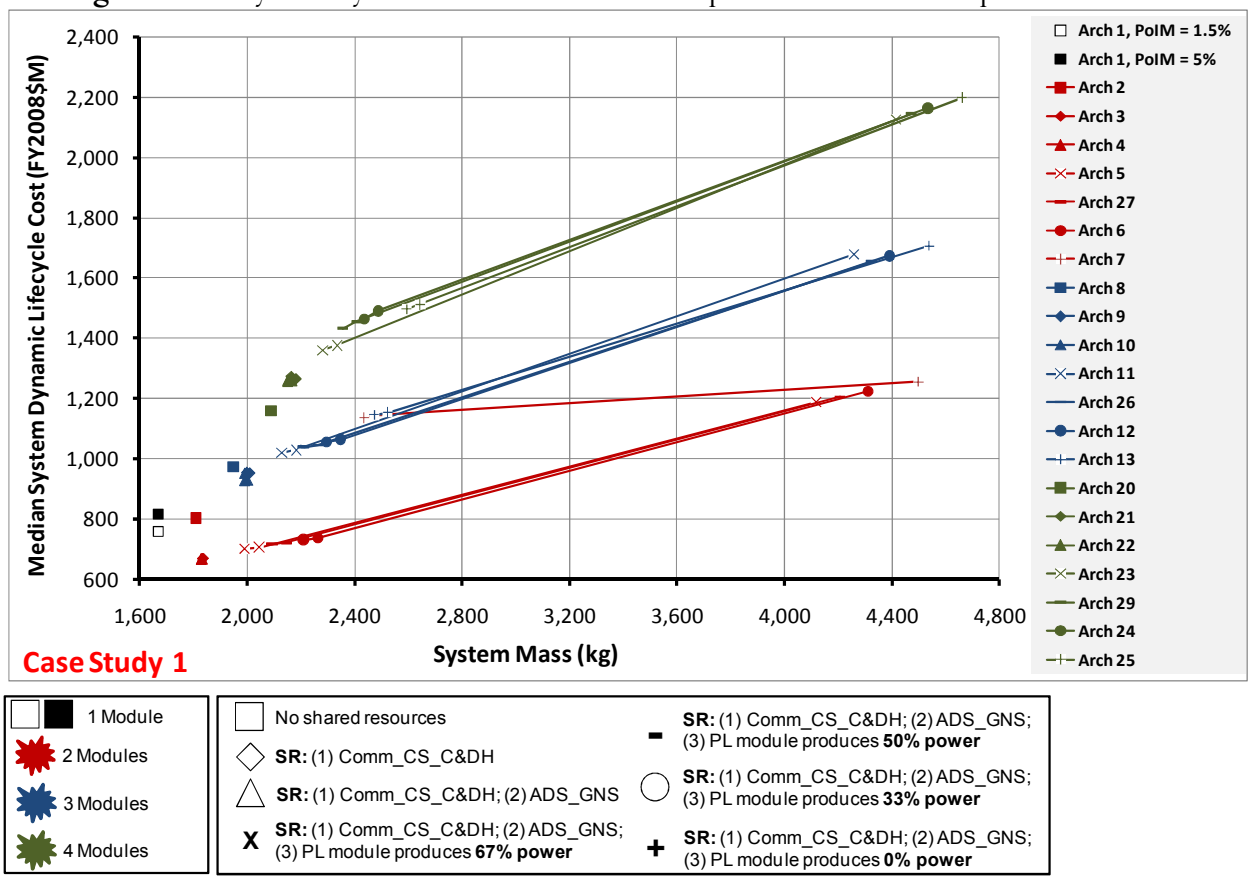
4.4. System Dynamic Lifecycle Cost

The monolithic and fractionated spacecraft value propositions formulated in Section 4.3 with regard to System Mass and Propellant can be further expanded through the consideration of Dynamic Lifecycle Cost (LCC). Therefore, Section 4.4 explores the Dynamic LCC aspect of monolithic and fractionated spacecraft value propositions with an emphasis on inter-module separation distance (see Figure 4-24), Ground Resolution (payload performance) (see Figure 4-25), and mission lifetime and PoIM (see Figure 4-26).

To begin, the Dynamic LCC of monolithic and fractionated spacecraft value propositions is quantified and discussed with respect to inter-module separation distance (see Figure 4-24).

Figure 4-24 characterizes the 22 spacecraft architectures investigated in Case Study 1 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Mass.

Figure 4-24. System Dynamic LCC and Mass with respect to inter-module separation distance.



In Figure 4-24, fractionated spacecraft architectures are generally more expensive than comparable monolithic spacecraft; 87.3% or 55 out of the 63 data points in Figure 4-24 serve as instantiations of this trend. As was demonstrated and concluded in Section 4.3, fractionated spacecraft architectures are always more massive than comparable monolithic spacecraft, regardless of the shared resources employed or the number of modules in the fractionated spacecraft. Therefore, the consistently higher System Mass of fractionated spacecraft architectures as compared to a monolith results in their having larger aggregate NRE, RE, and Operations Support costs; this in turn contributes to the generally larger Dynamic LCC of fractionated spacecraft relative to that of a comparable monolith.

However, this alone does not guarantee that fractionated spacecraft architectures will have a larger Dynamic LCC than a comparable monolith. The remaining Dynamic LCC factor is the aggregate cost of replenishments. For a given use of shared resources, as the number of modules increases in a fractionated spacecraft, not only does the aggregate (System) Mass tend to increase, as was discovered in Section 4.3, but the number of inter-module dependencies in a fractionated spacecraft increase as well. As such, inter-module dependencies exacerbate the implications of modules failing on-orbit because a dependent module (*i.e.,* shared resource recipient) will fail if a module it is dependent upon (*i.e.,* shared resource source) fails. Therefore, in terms of *aggregate number of replenishments*, the implication of an increase in inter-module dependencies in a spacecraft is to increase the aggregate number of replenishments. Consequently, this may increase the number of launch vehicle failures due to the provision of more opportunities in which launches, and thus launch failures, can occur throughout the lifecycle. However in contrast, in terms of *aggregate replenishment costs,* comprised of RE and launch vehicle costs, relative to the aggregate replenishment costs of a comparable monolith, as the number of inter-module dependencies in a fractionated spacecraft increases, the fractionated spacecraft may or may not have a larger aggregate cost of replenishments. This is due to differences in launch vehicle usage and thus launch vehicle costs between comparable monolithic and fractionated spacecraft and is subsequently discussed in the last paragraph on the page.

Therefore, in conclusion with regard to the consistently higher Dynamic LCC of fractionated spacecraft relative to a comparable monolithic spacecraft, this is the result of their larger System Mass and often-higher aggregate cost of replenishments as compared to that of the monolith. However, as is evident by the results presented in Figure 4-24, some fractionated spacecraft architectures (*i.e.*, Arch 3, 4, 5, 27, and 6) actually have a lesser Dynamic LCC than the comparable monolithic spacecraft despite their higher System Mass. The reason for this was cited at the end of the previous paragraph, namely, differences in launch vehicle usage and thus launch vehicle costs between comparable monolithic and fractionated spacecraft.

Through the creation of modules and sharing of subsystem resources, fractionated spacecraft can potentially make use of a smaller launch vehicle or set of smaller launch vehicles as compared to the single launch vehicle employed by a comparable monolith. This is possible because each of the respective modules in a fractionated spacecraft may be individually smaller in mass and size than a comparable monolith[18], thereby potentially enabling the modules to fit into any combination of up to three smaller launch vehicles as compared to the single launch vehicle employed by the monolith. And since the cost of launch vehicles tends to decrease with the size of the launch vehicle (*i.e.,* its push Mass to LEO), fractionated spacecraft can

[18] This is true for all high-resolution (*i.e.,* 0.5 m) fractionated spacecraft with 1000 m or less inter-module separation distances, regardless of the use of shared resources. (These spacecraft architectures also happen to be the most LCC-competitive fractionated spacecraft.) Only when the ground resolution is medium or low (*i.e.,* 5 and 30 m) and/or the separation distance is 5000 m, do the individual modules in a fractionated spacecraft exceed the size and mass of a comparable monolith.

potentially have lesser launch costs than that of a comparable monolith. Therefore, despite the higher aggregate (*i.e.,* System) Mass and often aggregate number of replenishments of fractionated spacecraft relative to that of a comparable monolith, due to the dominance of launch vehicle costs on the initial deployment and replenishment costs, fractionated spacecraft can have an equal or lesser Static and/or Dynamic LCC than the monolith. Therefore, an important attribute of fractionated spacecraft, in terms of Static and Dynamic LCC-competiveness, is their ability to attain potentially lesser launch vehicle costs than that of a comparable monolithic spacecraft. Subsequently, this attribute of fractionated spacecraft is the reason as to situations observed in Figure 4-24 and Figure 4-27 in which fractionated spacecraft have an equal or lesser Static and/or Dynamic LCC, respectively than a comparable monolithic spacecraft, despite their higher System Mass (and hence NRE, RE, and Operations Support costs) and often aggregate number of replenishments.

Differences or changes in launch vehicle usage and hence vehicle costs, due to a change in inter-module separation distance, Ground Resolution, or Mission Lifetime, also provides the reasoning as to situations in which monolithic or fractionated spacecraft architectures have inconsistent (abnormal) trends in LCC as compared to other monolithic and/or fractionated spacecraft. For example in Figure 4-24, at 20 and 1000 m separation distances, Arch 7 requires significantly more expensive launch vehicles than the other two-module fractionated spacecraft architectures; hence, its Dynamic LCC is significantly higher (abnormal) at these separation distances than that of the other two-module fractionated spacecraft.

Lastly, with regard to the results given in Figure 4-24, the System Mass and hence Dynamic LCC of fractionated spacecraft increases with an increasing number of modules, use of shared resources, and/or inter-module separation distance (see Section 4.3.1 and 4.3.2). Recall, that the Dynamic LCC correlates positively with the System Mass given the mass-based parametric nature of the SET cost model. Subsequently, the reasoning as to all System Dynamic LCC-Mass trends observed in Figure 4-24 can be readily surmised through extending the reasoning as to why the System Mass increases with an increasing number of modules, use of shared resources, and/or inter-module separation (Section 4.3), and the discussion pertaining to the aggregate number and cost of replenishments (Section 4.4).

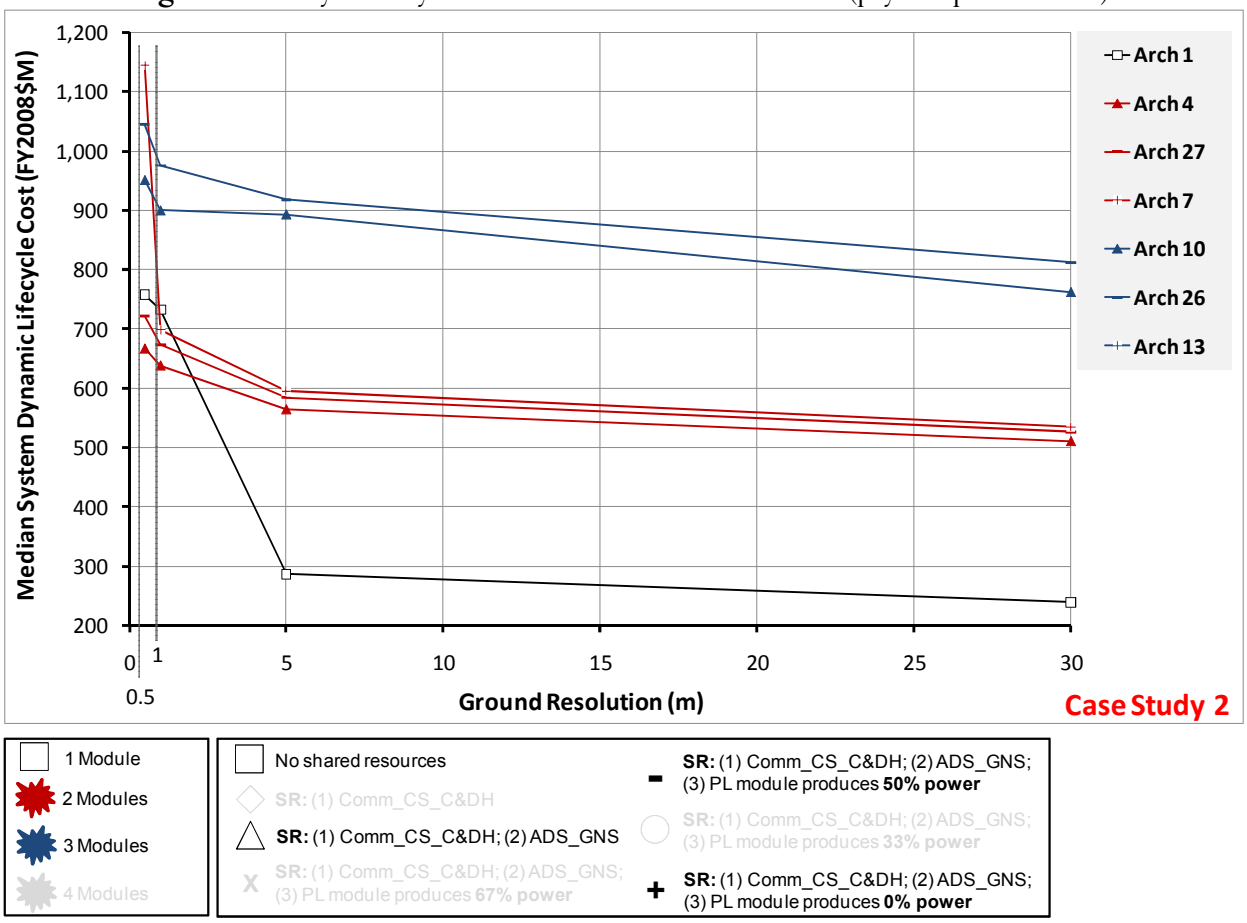
Figure 4-25 on the following page quantifies the System Dynamic LCC of monolithic and fractionated spacecraft value propositions with respect to varying Ground Resolution (payload performance). The first notable trend in the results shown in Figure 4-25 is that the Dynamic LCC of a given spacecraft architecture is nonlinear with respect to Ground Resolution. The reason for this follows from the Dynamic LCC correlating positively with the System Mass, which in turn scales inversely proportionally with respect to Ground Resolution (see discussion pertaining to the results in Figure 4-22). Additionally, changes in launch vehicle usage and hence launch costs do occur, for a given spacecraft architecture, in transitioning across the four Ground Resolutions considered, and this gives rise to some of the nonlinearity in the Dynamic LCC with respect to Ground Resolution seen in Figure 4-25.

Additionally in Figure 4-25, the spacecraft architecture with the least expensive Dynamic LCC is not always the monolithic spacecraft. At 0.5 and 1 m Ground Resolutions, certain two-module fractionated spacecraft are less expensive than the comparable monolith, despite their larger System Mass and aggregate number of replenishments. The reason for this follows from the critical attribute of fractionation cited in the discussion pertaining to Figure 4-24, namely, these fractionated spacecraft have appreciably lesser launch costs than the comparable monolith, and this enables them to have a lesser Dynamic LCC. In contrast, for the medium and low resolutions of 5 and 30 m respectively, the monolith is the least expensive spacecraft by a significant Dynamic LCC margin. The reason for this being that as the Ground Resolution decreases,

the RSM payload drives less of the physical design of a spacecraft. Subsequently, using fractionation to separate the pointing-intensive payload from the other subsystems/modules, provides proportionally less mass (and size) savings relative to the monolith; this in turn is due to the mass and structural overhead associated with system-wide redundancy proportionally increasing with decreasing payload dominance (*i.e.,* Ground Resolution)[6]. Consequently, due to the increasing mass (and size) penalties incurred by fractionated spacecraft with a decreasing Ground Resolution, the ability for fractionated spacecraft to have lesser launch vehicles costs than that of a comparable monolith diminishes or ceases altogether with a decreasing Ground Resolution. For the two-module fractionated spacecraft in Figure 4-25, this situation does occur when the Ground Resolution is 5 and 30 m because the respective launch vehicle costs of the monolithic and two-module fractionated spacecraft are equal. Hence, at 5 and 30 m resolutions, given the equal launch costs and the higher aggregate (*i.e.,* System) Mass of fractionated spacecraft relative to that of a comparable monolith, their Dynamic LCC is appreciably larger.

Figure 4-25 characterizes the 7 spacecraft architectures investigated in Case Study 2 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Mass.

Figure 4-25. System Dynamic LCC and Ground Resolution (payload performance).

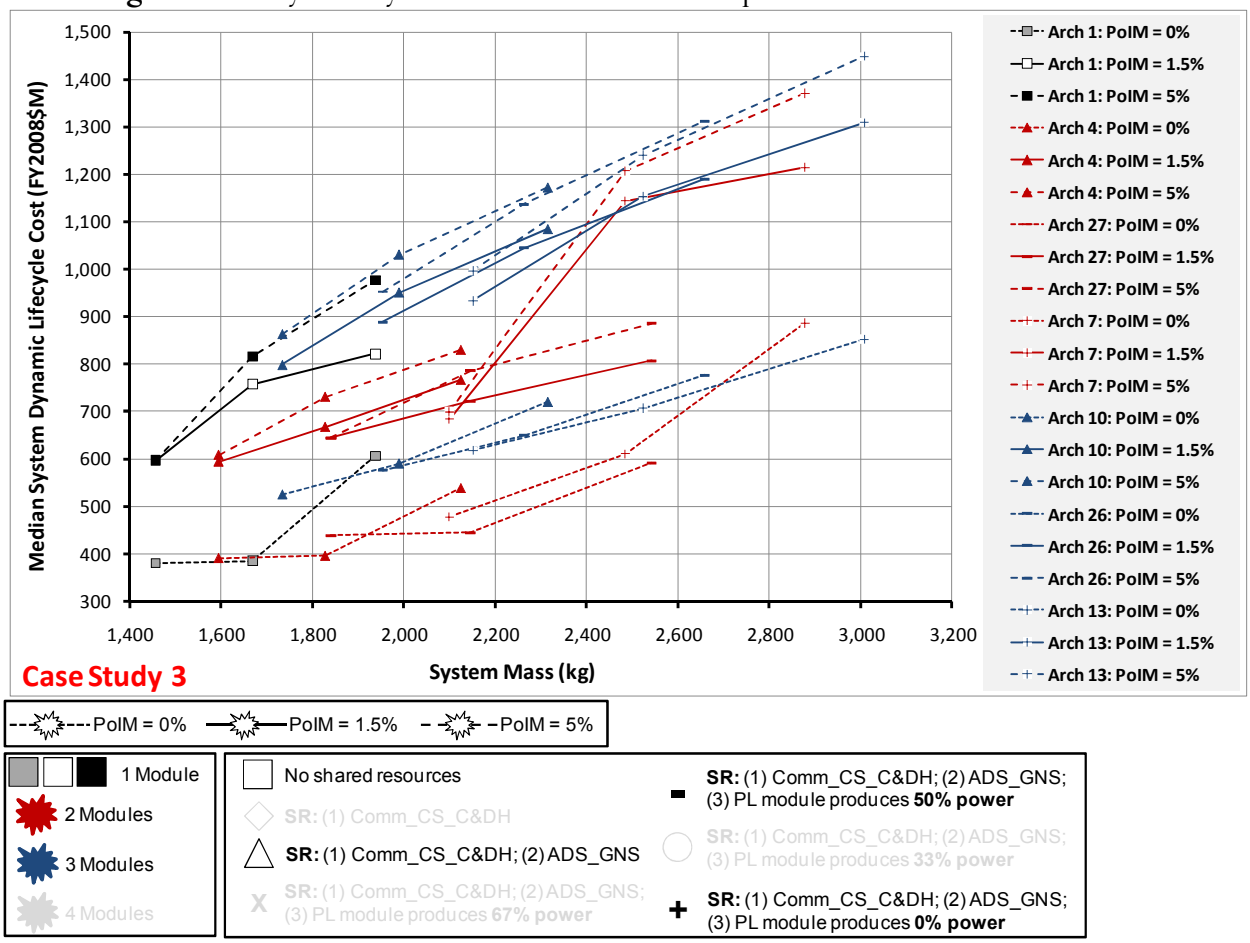


[6] Although not shown in the results in Section 4.3, this observation is true for fractionated spacecraft at any ground resolution.

With respect to the System Dynamic LCC dimension of monolithic and fractionated spacecraft value propositions, the remaining aspect of this value proposition dimension to be enumerated in Section 4.4 is quantifying and understanding System Dynamic LCC with regard to mission lifetime and PoIM. As alluded to in Section 4.3.3, specifically Figure 4-23, the environmental severity (harshness), as dictated by the mission lifetime and PoIM, significantly influences the Dynamic LCC of a given spacecraft architecture. Subsequently, Figure 4-26 enumerates the implications of mission lifetime and PoIM for Dynamic LCC.

Figure 4-26 characterizes the 7 spacecraft architectures investigated in Case Study 3 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Mass.

Figure 4-26. System Dynamic LCC and Mass with respect to Mission Lifetime and PoIM.



The first notable trend in Figure 4-26 is that for a given spacecraft architecture and PoIM, the Dynamic LCC increases as the mission lifetime increases. The reason for this follows from the fact that System Dynamic LCC positively correlates with System Mass, which in turn was found to increase with an increase in mission lifetime (see discussion pertaining to Figure 4-23). Additionally, as the mission lifetime increases it increases the aggregate number and thus cost of replenishments for a given spacecraft architecture, which subsequently increases its respective Dynamic LCC. For a given PoIM, the trend of an increasing aggregate number of replenishments with an increasing mission lifetime is due to a longer mission lifetime simply provides more opportunities (*i.e.,* time) for on-orbit and subsequently launch vehicle failures to occur; hence the aggregate number of replenishments increases. Consequently, for a given PoIM, the trend of an

increasing aggregate cost of replenishments with an increasing in mission lifetime is due to an increased aggregate number of replenishments with mission lifetime. Additionally, Dynamic LCC increases with mission lifetime due to a longer mission lifetime leading to larger and more massive spacecraft and because the aggregate cost of replenishments tends to also increase with mission lifetime due to an increase in launch costs and rebuilds of a spacecraft architecture (*i.e.,* RE costs).

Another notable trend present in the results shown in Figure 4-26 is that for a given spacecraft architecture and mission lifetime, as the PoIM increases in magnitude, the Dynamic LCC increases as well. Following the line of reasoning supplied in the previous paragraph, for a given mission lifetime, an increase in PoIM increases the probability that on-orbit and subsequently launch vehicle failures will occur during the mission lifetime period, thereby increasing the aggregate number of replenishments for a spacecraft architecture. And due to their being a cost associated with replenishments (*i.e.,* the cost of rebuilding and launching spacecraft/modules), increasing the aggregate number of replenishments, due to an increase in PoIM, tends to increase the aggregate cost of replenishments and hence Dynamic LCC of spacecraft architectures. There are, however, certain instances in which for a given mission lifetime, increasing the PoIM does not change the Dynamic LCC of an architecture. These instances occur if, for a given spacecraft architecture and mission lifetime, the increase in PoIM is not large enough to "trigger" (*i.e.,* cause) an increase in the aggregate number of replenishments, noting that the number of replenishments is an integer value.

Lastly, with regard to the results in Figure 4-26, amongst the fractionated spacecraft architectures considered in Case Study 3, there is not a consistent optimum combination of mission lifetime and PoIM in terms of minimizing their respective Dynamic LCC to that of a comparable monolith. Alternatively stated, as the severity (harshness) of the lifecycle for monolithic and fractionated spacecraft increases, it does not necessarily mean that fractionated spacecraft become more desirable, in terms of Dynamic LCC, than comparable monolithic spacecraft. The results in Figure 4-26 therefore provide quantitative instantiations both supporting and refuting the notion that fractionated spacecraft are "better" than comparable monolithic spacecraft in more sever (harsh) mission lifecycles. Specifically, this is evident in examining the relative LCC disparity between two/three-module fractionated spacecraft architectures and the comparable monolith in Figure 4-26, for all nine combinations of mission lifetime and PoIM values. One mission lifetime and PoIM combination does not prove to be the best case (*i.e.,* yield the minimum Dynamic LCC) for all fractionated spacecraft architectures considered. Therefore, while for a specific spacecraft architecture there will be an optimum mission lifetime and PoIM value, in terms of minimizing its respective Dynamic LCC to that of a comparable monolith, there is simply not a clear optimum mission lifetime and PoIM within or across the two and three-module class of fractioned spacecraft architectures.
As an aside, it is worth noting that, in Figure 4-26, two spacecraft architectures are consistently the most Dynamic LCC-competitive to a comparable monolith across the nine combinations of mission lifetime and PoIM values. These are Arch 4 and 10, and they can be thought of as the most Dynamic LCC-robust fractionated spacecraft in terms of mitigating the adverse LCC implications of mission lifetime and PoIM.

4.5. Confidence in the System Dynamic Lifecycle Cost

The suppliant question with regard to the results presented and discussed in Section 4.3 and 4.4 is, with what level of confidence can it be concluded that certain fractionated spacecraft are less expensive than comparable monolithic spacecraft? Responses to this question are subsequently formulated in Section 4.5 through a three-part discussion. The first part pertains to instilling confidence in Dynamic LCC through the comparison of Static and Dynamic LCC values corresponding to various fractionated spacecraft architectures (Section 4.5.1). The second part of the discussion addresses the two elements of Dynamic

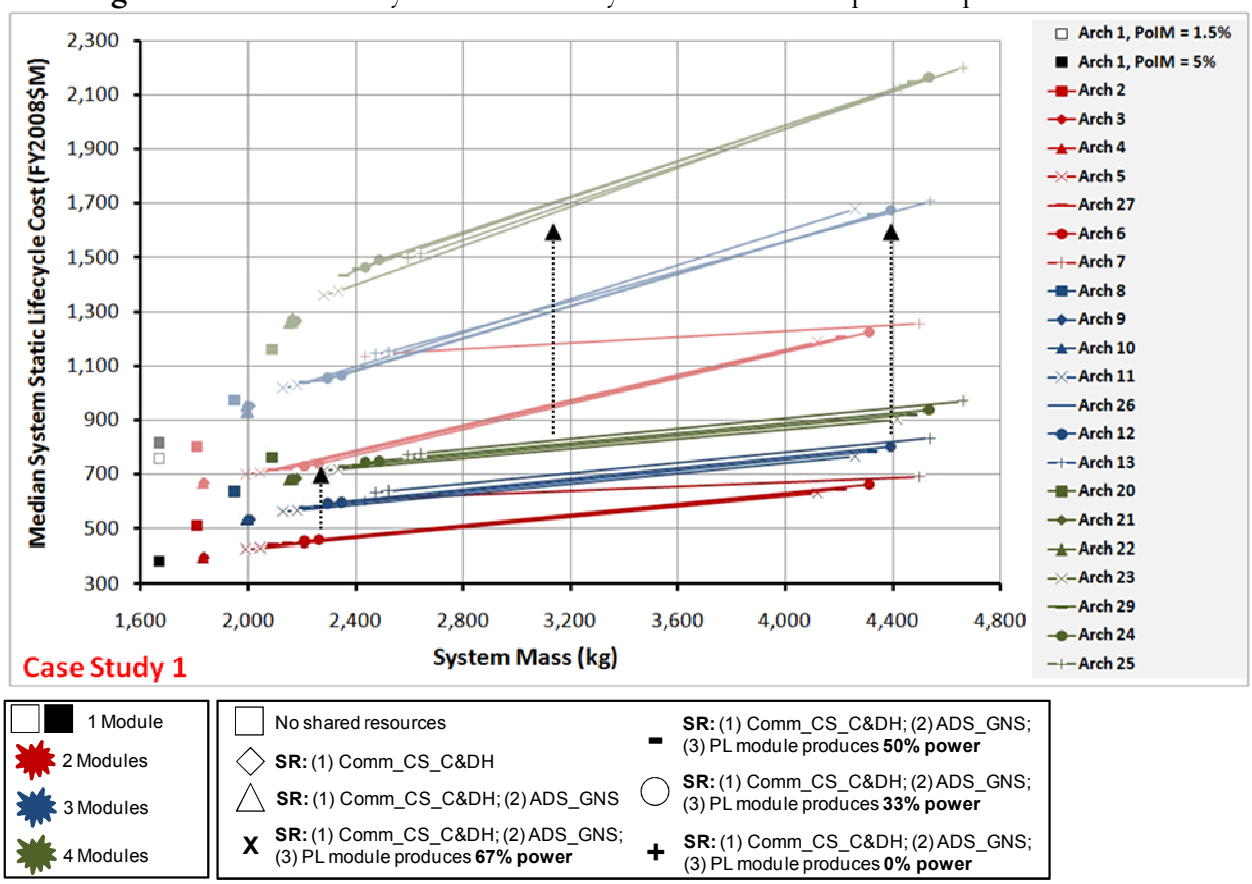
LCC uncertainty discussed in Section 2.1.10 and 4.2.3 (Section 4.5.2). And the third part of the discussion further enumerates statistical confidence in Dynamic LCC by presenting multimodal, Dynamic LCC distributions, as generated by the MCA in the SET, for select spacecraft (Section 4.5.3).

4.5.1. Static versus Dynamic Lifecycle Cost

Quantifying the Static LCC is beneficial for two reasons. First, Static LCC provides a lower-bound LCC estimate for a given spacecraft because it precludes the adverse LCC implications of lifecycle uncertainties in the spacecraft's lifecycle. And second, Static LCC instills confidence in the Dynamic LCC values, and hence dynamic lifecycle simulation model in the SET, if a logical path of reasoning can demonstrate differences in (or a transition between) the Static and Dynamic LCC for a given spacecraft. Subsequently, Section 4.5.1 enumerates trends between the Static and Dynamic LCC of the spacecraft architectures considered in Case Study 1 (see Figure 4-27).

Figure 4-27 characterizes the 22 spacecraft architectures investigated in Case Study 1 with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M) and (x) System Mass. In Figure 4-27, the *light colored* data points and lines represent the System *Dynamic LCC* v. System Mass trends.

Figure 4-27. Trends in System Static and Dynamic LCC with respect to separation distance.



The first trend in Figure 4-27 to be noted is that the Static LCC of monolithic and fractionated spacecraft architectures is always less than their respective Dynamic LCC; by a maximum, minimum, and average of 58%, 34%, and 45% respectively (*i.e.,* ratio of Static to Dynamic LCC of 42%, 66%, and 55% respectively). The specific reason for this trend is that the dynamic simulation and hence Dynamic LCC of a

spacecraft architecture, as performed and quantified within the SET respectively, intentionally accounts for lifecycle uncertainties and their consequent risks (*i.e.,* on-orbit and launch vehicle failures). In turn, the risk of on-orbit and launch vehicle failures throughout a lifecycle can necessitate the replenishment of spacecraft/modules, and these replenishments ultimately increase the (Dynamic) LCC of a spacecraft architecture relative to its respective Static LCC. Therefore, for a given spacecraft architecture, the Dynamic LCC quantifies the totality of the Static LCC, and in addition, accrues the RE and launch costs associated with replacing spacecraft/modules throughout the lifecycle due to risks resulting from lifecycle uncertainties (*e.g.,* on-orbit failure). Hence, this necessarily requires the Dynamic LCC of a given spacecraft architecture be (appreciably) greater or equal to its respective Static LCC.

Another noteworthy observation with regard to the results in Figure 4-27 is that for a given spacecraft architecture, the *relative* System Dynamic LCC-Mass trends are appreciably similar when viewed from the Static LCC and Dynamic LCC v. System Mass perspectives. The reason for this observation is that for a given spacecraft architecture, the System Mass remains unchanged with regard to Static and Dynamic LCC, given that mass is not dependent on, but is rather used to quantify, Static and Dynamic LCC. Subsequently, for a given spacecraft architecture, the cause for the (slight) variation in the *relative* Static and Dynamic LCC-Mass trends, as shown in Figure 4-27, is due to a translation in LCC (*i.e.,* y-axis) value between the Static and Dynamic LCC. For a given spacecraft architecture, the amount of LCC translation needed to transition from its respective Static to Dynamic LCC is dictated by the aggregate cost (and hence number) of replenishments for that spacecraft architecture, and this in turn is dictated by the number of inter-module dependencies in that spacecraft architecture. For a given use of shared resources, as the number of modules and/or use of shared resources increases in a fractionated spacecraft, the aggregate number and cost of replenishments will increase proportionally (see discussion pertaining to Figure 4-24). And since the aggregate cost of replenishments is responsible for differences between Static and Dynamic LCC values, for a given spacecraft architecture and use of shared resources, as the number of modules increases, the proportional disparity (both in magnitude and form) between the Static LCC and Dynamic LCC-Mass trends ("curves") will increase proportionally. Therefore, amongst the two, three, and four-module fractionated spacecraft in Figure 4-27, the two-module spacecraft architectures show the least deviation, in terms of both translation and form, between their respective Static and Dynamic LCC-Mass "curves."

Lastly, Figure 4-27 provides a quantitative instantiation of Static LCC inappropriately accounting for the costs associated with sharing resources amongst modules in fractionated spacecraft architectures (*i.e.,* inter-module dependencies). The reason for this is that the Static LCC does not account for the adverse LCC implications of an increased aggregate number and thus cost of replenishments, due to employing shared resources in a fractionated spacecraft, as the number of modules increases. Therefore, the Static LCC for fractionated spacecraft, most often shows that sharing resources increases the LCC-competiveness of fractionated spacecraft relative to a comparable monolith when in reality (*i.e.,* with respect to a stochastic lifecycle) this may not be the case (consider Arch 22 in Figure 4-27). Consequently, the Static LCC of fractionated spacecraft architectures forms inappropriate recommendations as to employing shared resources in fractionated spacecraft.

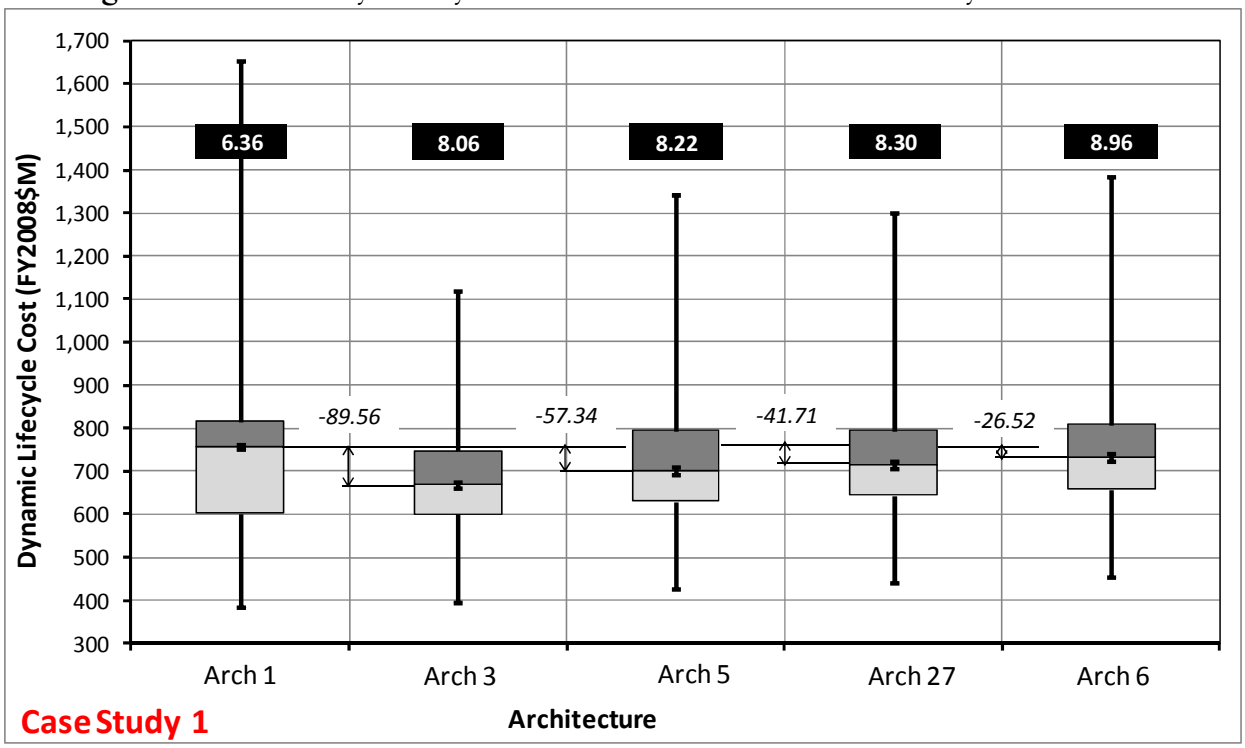
4.5.2. Statistical Confidence in the Dynamic Lifecycle Cost

The confidence in Dynamic LCC facilitated by the comparison of relative and absolute trends in Static and Dynamic LCC provided in Section 4.5.1 can be complemented by quantifying the two aspects of statistical confidence (*i.e.,* lack of uncertainty) in Dynamic LCC values produced by the SET (see Section 2.1.10 and 4.2.3). Subsequently, the statistical confidence in the Dynamic LCC is quantified for the most Dynamic

LCC-competitive fractionated spacecraft architectures found in the results generated for Case Study 1, 2, and 3 in Figure 4-28, Figure 4-29, and Figure 4-30 respectively.

Figure 4-28 characterizes Arch 1, 3, 5, 27, and 6 in Case Study 1 at 20 m separation distances with respect to their order statistic, five-number summary (aka box-and-whisker plot). See Section 2.1.10 and 4.2.3 for a discussion of the order statistic, five-number summary for a given spacecraft architecture.

Figure 4-28. Median System Dynamic LCC confidence for select Case Study 1 architectures.



The first notable trend with regard to the results shown in Figure 4-28 is that the minimum Dynamic LCC values for all five spacecraft architectures are relatively the same, whereas the maximum Dynamic LCC values vary appreciably in magnitude. The minimum Dynamic LCC shown in Figure 4-28, for each spacecraft architecture, characterizes the LCC of these architectures associated with not having any (*i.e.,* zero) replenishments throughout their respective lifecycle's – therefore, the minimum Dynamic LCC *is the* Static LCC. For a given PoIM and mission lifetime, the dynamic lifecycle simulation of a given spacecraft architecture in the SET does not preclude the possibility of attaining a "perfect" lifecycle in at least one of the 2,500 lifecycle simulations (MCA trials) used assess that architecture. Subsequently, for the 2,500 MCA trials used to assess Arch 1, 3, 5, 27, and 6, at least one of those trials yielded a perfect lifecycle for each of these architectures. Hence, the lower bound minimum Dynamic LCC for Arch 1, 3, 5, 27, and 6 shown in Figure 4-28 is the Static LCC of those respective spacecraft architectures. And since, in Figure 4-28, the respective Static LCC of Arch 1, 3, 5, 27, and 6 are so close in magnitude, the minimum Dynamic LCC values for Arch 1, 3, 5, 27, and 6 in Figure 4-28 are similarly close in magnitude.

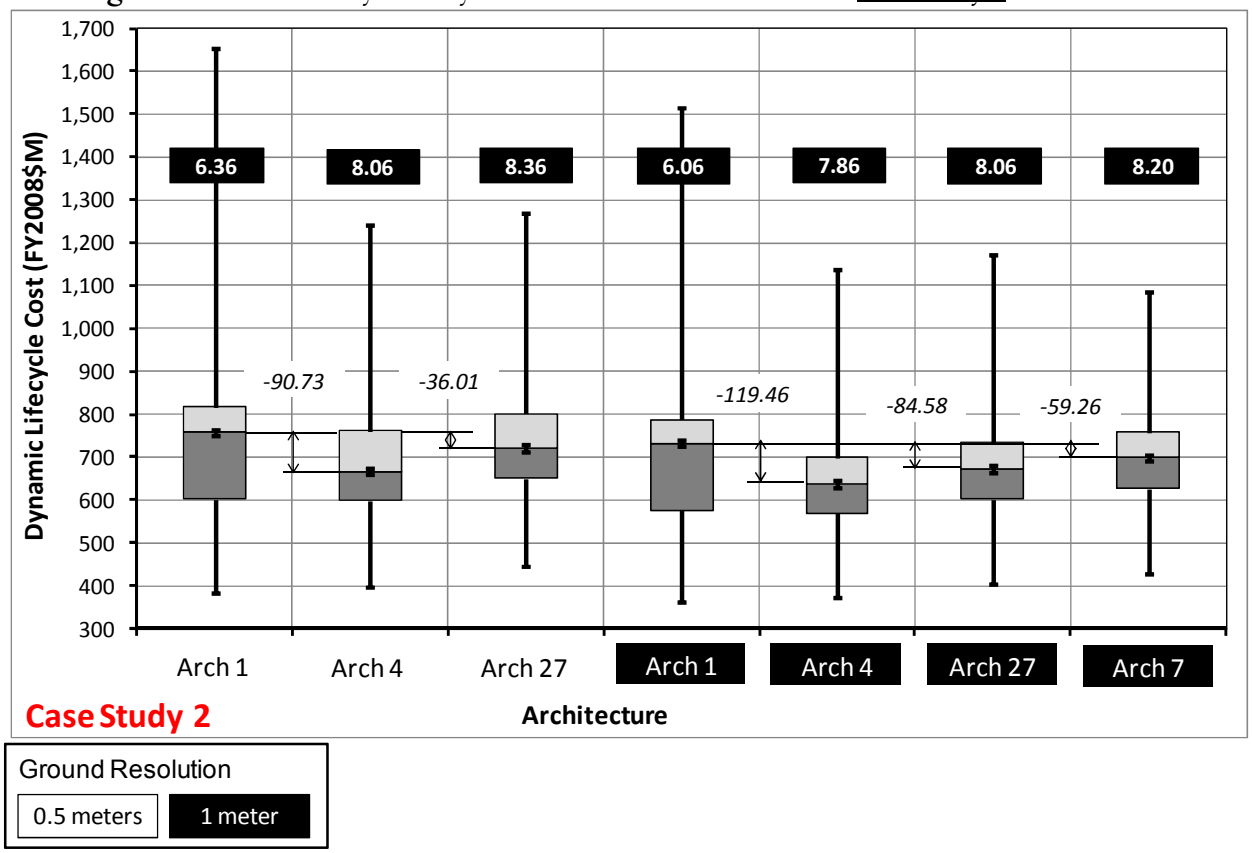
However, in contrast to the relatively consistent minimum Dynamic LCC values in Figure 4-28, the maximum Dynamic LCC values vary appreciably across Arch 1, 3, 5, 27, and 6. This is the direct result of the differences in launch vehicles used by Arch 1, 3, 5, 27, and 6 as well as differences in the number of inter-module dependencies between these spacecraft architectures. And the collective effect of these

differences is increasing the potential to have appreciable variance amongst the Dynamic LCC values corresponding to the worst-case scenarios (*i.e.,* maximum Dynamic LCC) for Arch 1, 3, 5, 27, and 6.

The remaining notable trend in Figure 4-28 is that the uncertainty in the Median Dynamic LCC due to uncertainty in the SET cost model (*i.e.,* CMU) (see Section 2.1.10) tends to, but does not always, increase with an increasing Median Dynamic LCC value. For example, consider Arch 1, which has a lower CMU than every fractionated spacecraft architecture even though its respective Median Dynamic LCC is larger than that of Arch 3, 5, 27, and 6. The reasoning for the lesser CMU of Arch 1 is that as the ratio of *the portion* of Median Dynamic LCC **due to** launch costs **to the** cost of building, rebuilding (for replenishments), and operation *increases*, the CMU for that Median Dynamic LCC value *decreases*. As such, this ratio is significantly higher for Arch 1 than for Arch 3, 5, 27, and 6. Therefore, even though Arch 1 has a higher Median Dynamic LCC value, a much larger portion of the Dynamic LCC value is the result of certain costs (*i.e.,* launch costs) than is in the case of the fractionated spacecraft architectures. Hence, Arch 1 has a larger Median Dynamic LCC than Arch 3, 5, 27, and 6 but still has a lesser CMU. Moreover, with regard to the CMU of Arch 3, 5, 27, and 6, the CMU increases successively with Arch 3, 5, 27, and 6 as a result of their successively larger Median Dynamic LCC values and their having similar launch costs relative to their respective Median Dynamic LCC values.

Figure 4-29 characterizes Arch 1, 4, and 27 in Case Study 2 (for 0.5 m Ground Resolution), and Arch 1, 4, 27, and 7 in Case Study 2 (for 1 m Ground Resolution) with respect to their order statistic, five-number summary (aka box-and-whisker plot). See Section 2.1.10 and 4.2.3 for a discussion of the order statistic, five-number summary for a given spacecraft architecture.

Figure 4-29. Median System Dynamic LCC confidence for select Case Study 2 architectures.

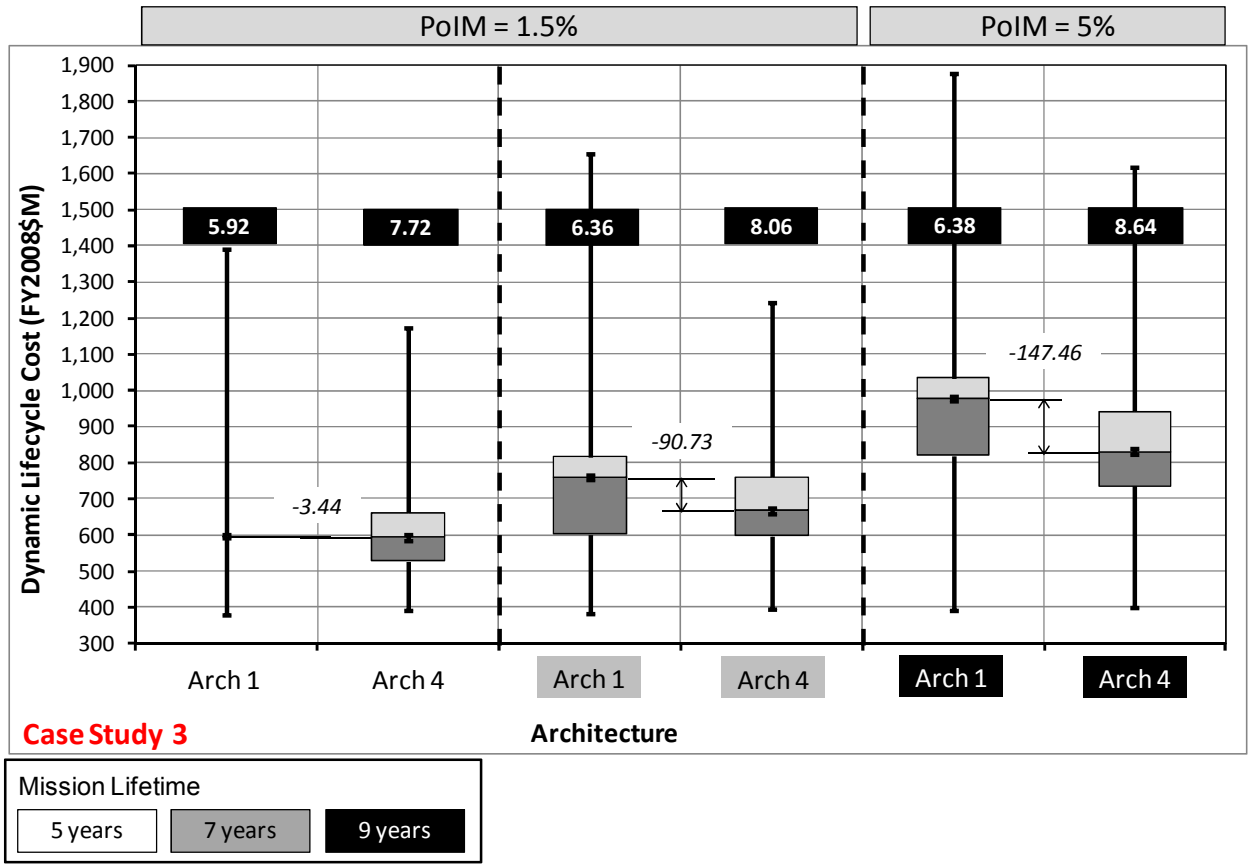


On the basis of the respective order statistic, five-number summary of Arch 1, 4, 27, and 7 in Figure 4-29, it can be posited that fractionated spacecraft architectures have less cost "risk" than a comparable monolithic spacecraft architecture. Here risk is assumed to be inversely proportionally to confidence (*i.e.,* lack of uncertainty) in the Median Dynamic LCC, for a given spacecraft architecture. In Figure 4-29, for the monolithic and fractionated spacecraft architectures having identical Ground Resolutions, in comparing their respective (1) maximum, (2) minimum, (3) 25th percentile, and (4) 75th percentile Dynamic LCC *relative* to their Median Dynamic LCC, the disparity between these four values and the Median Dynamic LCC is always the largest for the monolithic spacecraft (Arch 1). Specifically, this is evident by the five-number summary of the monolith (for both the 0.5 and 1 m resolution cases) in Figure 4-29, as it is less condensed around its respective Median Dynamic LCC value than is the case for the comparable fractionated spacecraft architectures. Therefore, with respect to an order statistic, five-number summary, Arch 1 can be said to have a higher cost "risk", that is, there is a higher uncertainty (*i.e.* lower confidence) that the LCC of Arch 1, for the space mission at hand, will end up equaling the Median (expected) Dynamic LCC value. Conversely, the five-number summary for the fractionated spacecraft architectures is consistently more condensed around their respective Median Dynamic LCC values than is the case for comparable monolithic spacecraft. Therefore, with respect to an order statistic, five-number summary, Arch 4, 27, and 7 can be said to have a lower cost "risk", that is, this there is a lower uncertainty (*i.e.* higher confidence) that the LCC of Arch 4, 27, and 7, for the space mission at hand, will end up equaling their respective Median (expected) Dynamic LCC values. In conclusion, if it is desirable to have a higher level of confidence in performing a given space mission, at the expected/estimated LCC, fractionated spacecraft are more desirable than monolithic spacecraft. Alternatively stated, fractionated spacecraft are more LCC-robust than monolithic spacecraft to perturbations in the stochastic lifecycles of these spacecraft.

Figure 4-30 on the following page characterizes the confidence in Dynamic LCC for select Case Study 3 spacecraft architectures. The first constructive observation with regard to the results shown in Figure 4-30 is that at a 5 year mission lifetime, Arch 1 has an indistinguishable inter-quartile Dynamic LCC range, that is, its respective 75th, 50th (*i.e.*, median), and 25th percentile Dynamic LCC values are identical. The reason as to this observation is that as the mission lifetime and PoIM decrease, there is less variation in the Dynamic LCC values for monolithic (and fractionated) spacecraft architectures. This is the result of lower mission lifetime and PoIM values increasing the convergence of the aggregate number and cost of replenishments for a given spacecraft architecture, across all MCA trials used for their respective assessment. The convergence is due to lower mission lifetime and PoIM values simply providing less occurrences of on-orbit and launch vehicle failures, and hence variation in the aggregate number and cost of replenishments. Subsequently, the lower the mission lifetime and PoIM, the higher the probability that the 75th, 50th (*i.e.*, median), and 25th percentile Dynamic LCC will be the same for a spacecraft architecture. Therefore, given the indistinguishable inter-quartile range of Arch 1, and the CMU of Arch 1 and 4, the 75th, Median, and 25th percentile Dynamic LCC of Arch 1 is in-differentiable to the Median Dynamic LCC of Arch 4.

Figure 4-30 characterizes Arch 1 and 4 in Case Study 3 at a 5, 7, and 9 year mission lifetime (for PoIM of 1.5%) lifetime and at a 9 year mission lifetime (for a PoIM of 5%) with respect to their order statistic, five-number summary (aka box-and-whisker plot). See Section 2.1.10 and 4.2.3 for a discussion of the order statistic, five-number summary for a given spacecraft architecture.

Figure 4-30. Median System Dynamic LCC confidence for select Case Study 3 architectures.



Another noteworthy observation with regard to results shown in Figure 4-30 is that across the mission lifetimes and PoIM combinations, the Median Dynamic LCC values of Arch 4 are always either in-differentiable or lower than the Median Dynamic LCC Arch 1. Additionally, Arch 4 has a similar CMU magnitude and less cost "risk" than that of Arch 1 because of its more condensed order statistic, five-number summary relative to its Median Dynamic LCC. This observation is readily apparent in Figure 4-30 in comparing the Dynamic LCC order statistic, five-number summaries between the monolith (Arch 1) and comparable two-module fractionated spacecraft (Arch 4).

4.5.3. Lifecycle Cost Probability Density Functions

Section 4.5.3 provides the reasoning as to the uncertainty in the Median Dynamic LCC of the spacecraft architectures observed in Figure 4-28 through Figure 4-30. As such, Section 4.5.3 provides representative Dynamic LCC probability density functions (pdfs), as generated by the SET MCA, for Arch 1 and 3 shown in Figure 4-28. (Please refer to Appendix D for a discussion of the implications of the Dynamic LCC pdfs produced by the SET with respect to the appropriate number of MCA trials to use for SET simulations.)

Figure 4-31 characterizes Arch 1 in Case Study 1 with respect to its pdf (aka histogram), which shows the Dynamic LCC distribution for Arch 1 populated by the 2,500 MCA trials employed to assess it.

Figure 4-31. Spacecraft Architecture 1 (monolithic) probability density function (histogram).

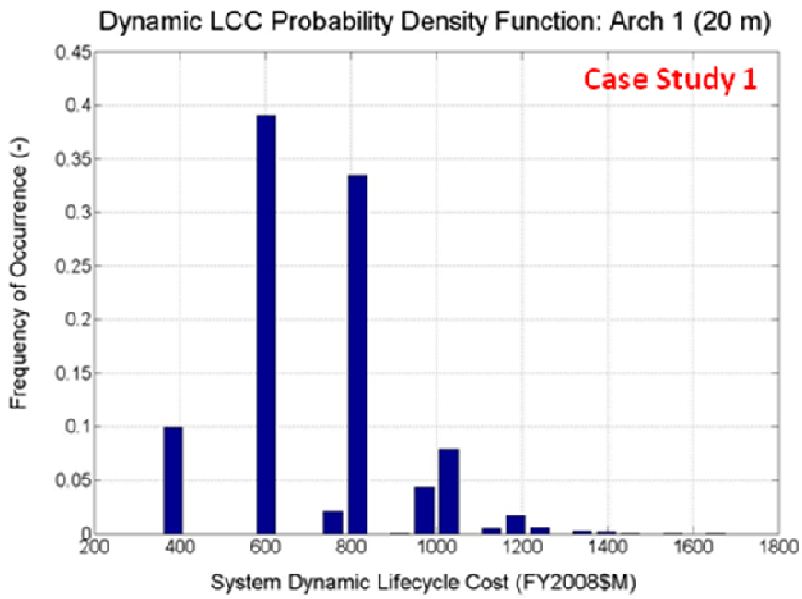
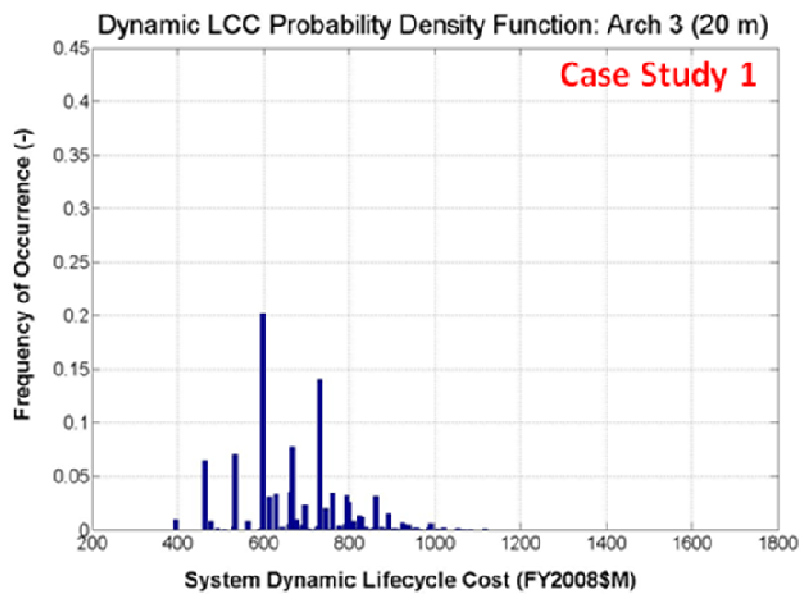


Figure 4-32 characterizes Arch 3 in Case Study 1 with respect its pdf (aka histogram), which shows the Dynamic LCC distribution for Arch 3 populated by the 2,500 MCA trials employed to assess it.

Figure 4-32. Spacecraft Architecture 3 (fractionated) probability density function (histogram).



The first observation to be noted with regard to the probability density functions (pdfs) shown in Figure 4-31 and Figure 4-32 is that Arch 1 is consists of **(1)** fewer, **(2)** more disperse, and **(3)** higher frequency LCC "bins" as compared to the pdf corresponding to Arch 3. This thereby provides the reasoning as to the more condensed nature of the five-number summary about the Median Dynamic LCC corresponding to Arch 3 as compared to that of Arch 1 in Figure 4-28. **(1)** The number of replenishments throughout a spacecraft architecture's respective lifecycle is an integer. Therefore, a histogram showing the Dynamic LCC distribution generated from the 2,500 MCA trials for a spacecraft architecture will contain a certain number of LCC "bins", each of which corresponds to a specific number and distribution of replenishments for the spacecraft architecture's respective modules (or module in the case of a monolith). Moreover, the number of bins increases with the number of modules and inter-module dependencies in a spacecraft architecture[19]. The reason for this is that as the number of modules and inter-module dependencies increases, there are more potential combinations of on-orbit and launch vehicles failures that occur in a lifecycle, and hence different LCC values and LCC bins. Therefore, the reason as to the fewer LCC bins for Arch 1 than for Arch 3 is that Arch 1 has no inter-module dependencies and hence a lesser potential to have a high number of LCC bins. **(2)** Additionally, the individual cost replenishments for two-module spacecraft architectures is often less than that of a monolith (as is the case with Arch 1 and 3), and hence the LCC disparity between the LCC bins tends to be larger for monolithic spacecraft than for fractionated spacecraft. **(3)** Lastly, given that the sum of the frequency values across all LCC bins in a given pdf must be equal to 1.0, because the monolithic spacecraft pdf has fewer LCC bins, the bins must generally have higher frequencies than those in the pdf for Arch 3, which in turn has more LCC bins.

And the remaining trend to be noted with regard to the pdfs presented in Figure 4-31 and Figure 4-32 is that the inter-quartile (25-50th percentile) range of fractionated spacecraft often falls within the inter-quartile range of the comparable monolithic spacecraft; for examples of this trend refer to Figure 4-28, Figure 4-29, and Figure 4-30. The implications of this trend are to suggest that fractionated spacecraft have a lesser Dynamic LCC "risk" than comparable monolithic spacecraft. This trend is substantiated by the noticeable difference between the pdfs shown in Figure 4-31 and Figure 4-32, namely, that the pdf for Arch 1 consists of (1) fewer, (2) more disperse, and (3) higher frequency LCC bins as compared to the pdf for Arch 3. These three characteristics of the pdf associated with Arch 1 causes it to exhibit a larger dispersion of Dynamic LCC values about its respective Median Dynamic LCC than Arch 3 does about its respective Median Dynamic LCC value. This therefore provides reason as to why the order statistic, five-number summary for Arch 1 often has an inter-quartile range that encapsulates the inter-quartile range of comparable fractionated spacecraft architectures in Figure 4-28 through Figure 4-30. Subsequently, with respect to the inter-quartile range, monolithic spacecraft often exhibit more Dynamic LCC "risk" than comparable fractionated spacecraft (see discussion pertaining to the results in Figure 4-29).

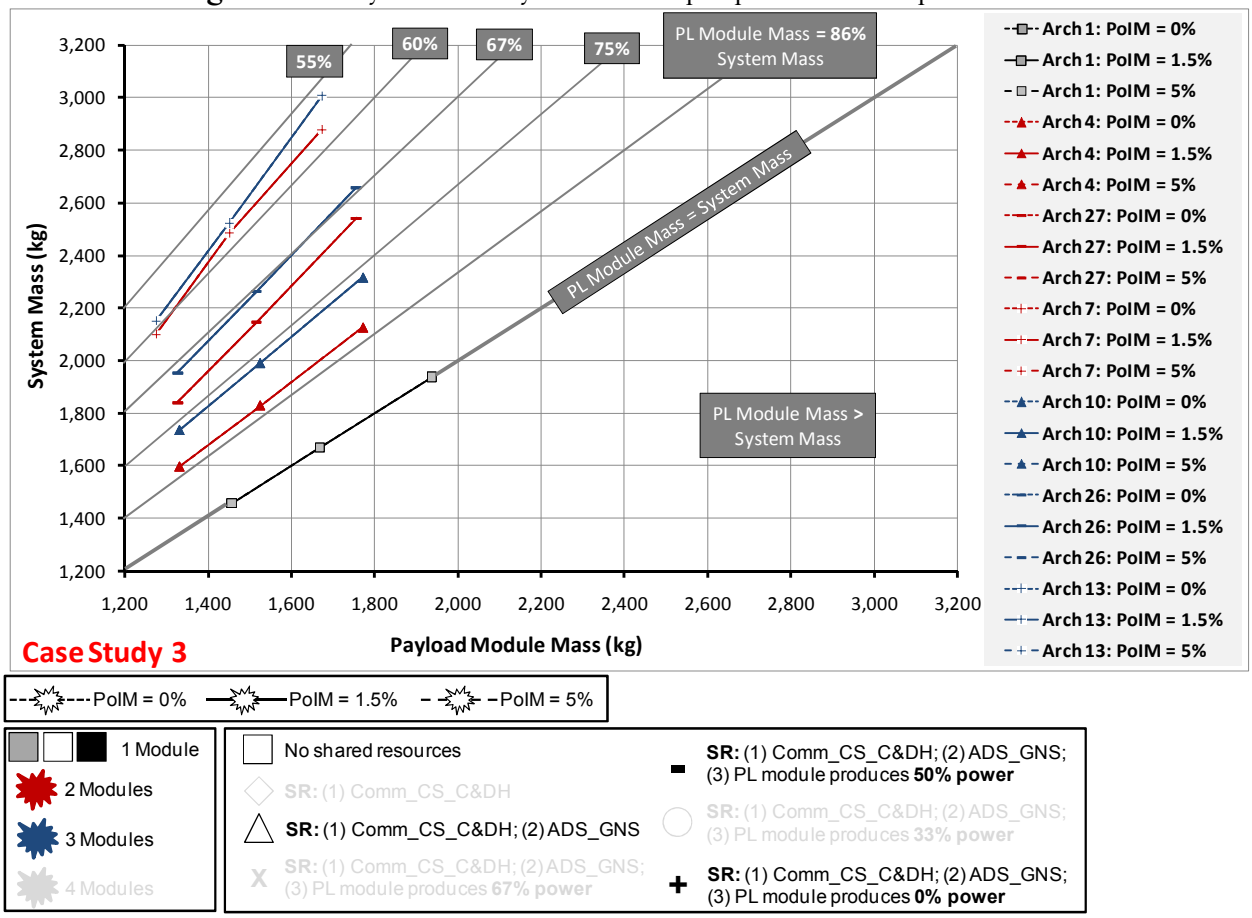
4.6. System and Payload Module Perspective

The value proposition for monolithic and fractionated spacecraft quantified and enumerated in Section 4.3, through 4.5 can be further expanded through examining the Mass and Dynamic LCC of these spacecraft with respect to the *System* and *Payload Module* perspective (see Section 3.5). Comparing the Mass and Dynamic LCC from these two perspectives provides a unique glimpse into the potential benefits offered by fractionation not enumerated in Section 4.3 through 4.5. Section 4.6 begins by quantifying the Mass for monolithic and fractionated spacecraft relative to the System and Payload Module perspectives.

[19] Although this is not the case in Figure 4-28, the bins corresponding to the median and 25th percentile (or any percentile for that matter) have a higher probability of being one in the same for monolithic spacecraft architectures than for fractionated spacecraft architectures as a result of the smaller number of bins in the monolith's Dynamic LCC pdf.

Figure 4-33 characterizes the 7 spacecraft architectures investigated in Case Study 3 with respect to their (y) System Mass (kg) and (x) Payload Module Mass (kg). This figure therefore characterizes the Mass tradeoff relative to the System and Payload Module perspectives (see Section 3.5).

Figure 4-33. System and Payload Module perspective with respect to Mass.



As can be discerned from the results shown in Figure 4-33, for a given mission lifetime and PoIM, the Payload Module always has a lesser Mass than the comparable monolithic spacecraft. The lesser Mass of the Payload Module as compared to the monolith is due the use of shared resources. Shared resources enable the Payload Module in a fractionated spacecraft to reduce its respective Mass relative to a comparable monolithic spacecraft by removing some of the subsystem hardware needed if the Payload Module does not rely on shared resources. Therefore, since monolithic spacecraft cannot share resources, fractionation can be used to effectively reduce the mass (and size) of the structure containing the payload (*i.e.,* Payload Module). Although employing shared resources does not appreciably reduce the Mass of the Payload Module relative to the monolith, it still does attain a lower mass (and size) which can have other benefits (see Section 4.7).

Another trend in Figure 4-33 worth noting is that for a given mission lifetime, as the use of shared resources increases, the absolute Payload Module Mass hardly changes but the ratio of Payload Module to System Mass decreases appreciably. The reason for this is that the 0.5 m Ground Resolution RSM payload dominates the design of the Payload Module (and monolith), and this has the effect of desensitizing the

Payload Module Mass savings to be gained from relying on shared resources. (For example, if you remove 25 kg from a 1400 kg structure this causes a proportionally smaller reduction in the objects Mass than if removing 25 kg from a 100 kg structure.) Therefore, for a given mission lifetime and regardless of the shared resources employed, the absolute Payload Module Mass remains nearly constant. In contrast, for a given mission lifetime, increasing the use of shared resources does appreciably increase the System Mass; this being due to the majority of the mass, power consumption, etc. associated with the hardware required for sharing resources being purposefully allocated to the Infrastructure Modules (see Section 4.3.1 for a further discussion with regard to this trend). Thus for a given mission lifetime, as the use of shared resources increases, the Payload Module Mass remains relatively constant but the System Mass increases appreciably, which leads to an appreciable decrease in the ratio of Payload Module to System Mass.

And the remaining observation pertaining to the results in Figure 4-33 is that across all spacecraft architectures considered in Case Study 3, the mass of the Payload Module is independent of PoIM, but is dependent on (varies with) the mission lifetime. This observation is similar to that based on the results presented in Section 4.3.3, specifically for Figure 4-23. For a given spacecraft architecture, the reasoning as to the independence of its respective Payload Module Mass and PoIM, is that PoIM is a factor exogenous to physical design (and thus modeling) of spacecraft architectures. However, in contrast, the mission lifetime positively correlates with Payload Module Mass because an increasing mission lifetime increases the Propellant Usage required for the mission, which in turn increases the Payload Module Mass (for a further insight with regard to this trend, the discussion for Figure 4-23.)

Analogous to quantifying the System and Payload Module perspectives for monolithic and fractionated spacecraft with respect to Mass, the results in Figure 4-34 quantify these perspectives with respect to the Median Dynamic LCC. For the particular set of results shown in Figure 4-34, only the spacecraft architectures at a PoIM of 1.5% are included to keep the results discernible.

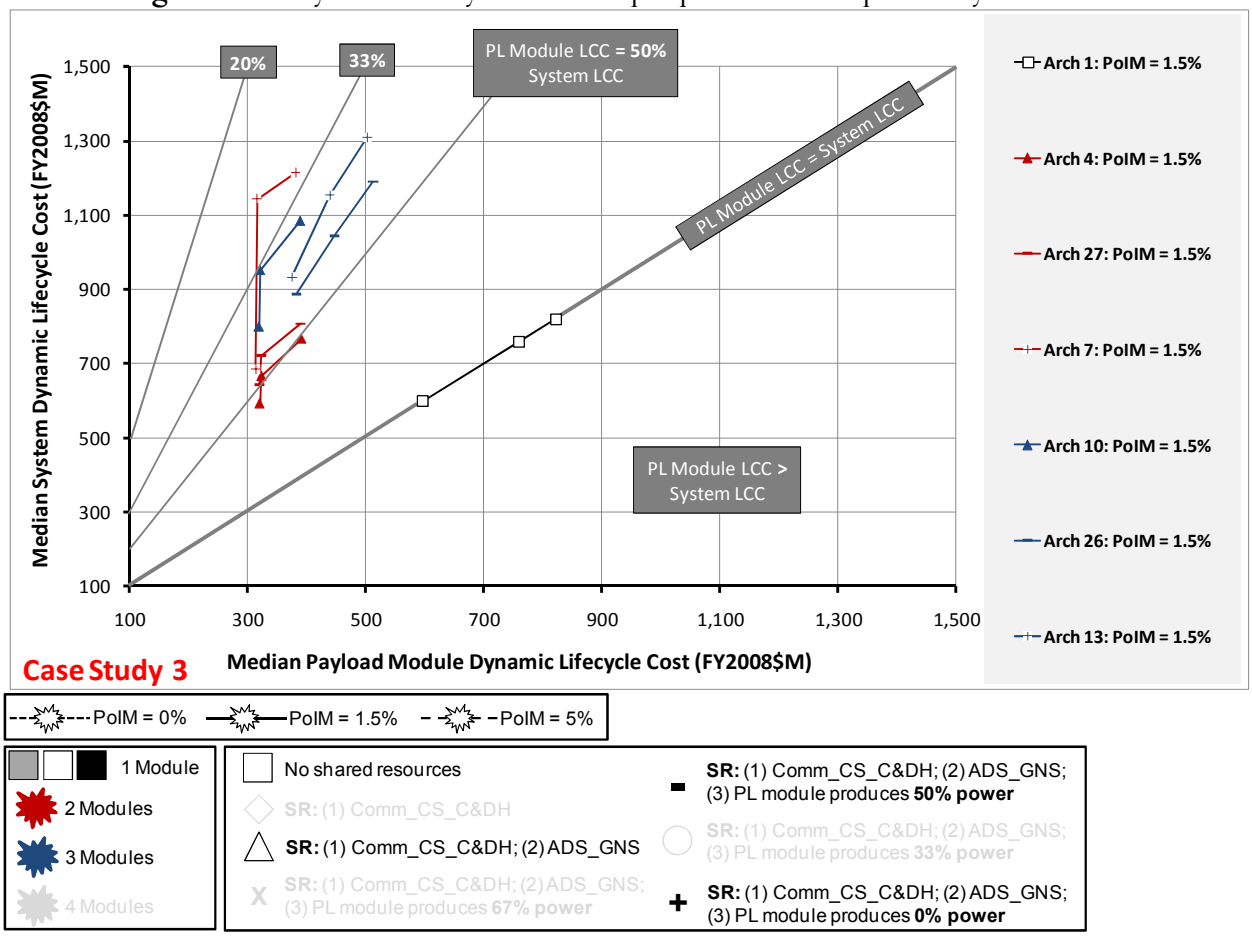
Figure 4-34 on the following page characterizes the System and Payload Module perspectives with respect Dynamic LCC for select architectures in Case Study 2. The first observation with regard to the results in Figure 4-34 is that for a given mission lifetime and PoIM, in terms of absolute LCC, the Payload Module LCC is always less than that of the comparable monolith. This is in part due to the lesser mass of the Payload Module relative to the monolith, as shown in Figure 4-33, but the majority of the reason for the lesser LCC of the Payload Module as compared to the monolith is due to differences in launch vehicle usage and hence replenishment costs (see Section 4.4). The Payload Modules corresponding to the fractionated spacecraft shown in Figure 4-33 are each able to use a single, smaller, and less expensive launch vehicle for replenishments than is employed by the monolith. Subsequently, this has appreciably decreases the LCC of the Payload Module as compared to the LCC of the comparable monolithic spacecraft.

Another trend worth noting from Figure 4-34 is that with the exception of the monolith, for a given fractionated spacecraft architecture, the Payload Module Dynamic LCC is appreciably less than the System Dynamic LCC. (Note that the monolithic spacecraft architecture, Arch 1, has an identical System and Payload Module Dynamic LCC because for the monolith these two perspectives are identical.) Across all of the spacecraft architectures and mission lifetimes depicted in Figure 4-34, the Payload Module Dynamic LCC is a maximum, minimum, and average 72%, 45%, and 59% less than the System Dynamic LCC, respectively (*i.e.,* ratio of Payload Module to System Dynamic LCC is 28%, 55%, and 41% respectively). These percentages demonstrate that in contrast to the System perspective, from the Payload Module perspective, fractionated spacecraft are significantly less expensive (see Section 3.5). Intuitively, for a given fractionated spacecraft architecture, the Payload Module Dynamic LCC should be less than the System

Dynamic LCC as it is but one constituent of the System Dynamic LCC; the remaining constituents being the respective LCCs of the other modules. More specifically, the lesser LCC of the Payload Module is the direct result of the smaller mass and aggregate number and cost of replenishments of the Payload Module relative to that of a comparable monolithic spacecraft.

Figure 4-34 characterizes the 7 spacecraft architectures investigated in Case Study 3, at a PoIM of 1.5% only, with respect to their (y) Median System Dynamic Lifecycle Cost (FY2008$M); and (x) Median Payload Module Dynamic Lifecycle Cost (FY2008$M). This figure therefore characterizes the Dynamic LCC tradeoff relative to the System and Payload Module perspectives (see Section 3.5).

Figure 4-34. System and Payload Module perspective with respect to Dynamic LCC.



Lastly, in Figure 4-34, on an absolute LCC scale, there are vertical "bands" of Payload Module LCC across the spacecraft architectures and mission lifetimes. The most prominent bands in Figure 4-34 occur around 320 and 390 (FY2008$M). (Since the number of replenishments for the Payload Module (and System) is an integer, the Payload Module LCC bands are discrete.) The reason for these bands is that the number of replenishments ultimately dictates the Payload Module LCC, not its respective mass as the mass increase due to an increase in mission lifetime is primarily due to increased propellant, which is relatively inexpensive (~10 $/kg). Hence, each LCC band represents Payload Modules that, through their respective mission lifetime and PoIM, end up having the same number of replenishments and roughly the same cost of replenishments, due to both RE and launch costs. Subsequently, the driver for differences in Payload

Module LCC values as seen in Figure 4-34 is the aggregate number of Payload Module replenishments with potential variations due to differences in launch vehicle failures between Payload Modules. It is important to note that for a given fractionated spacecraft architecture, increasing the mission lifetime does not necessarily increase the Payload Module LCC, if the increase in mission lifetime, at the given PoIM value, is not enough to "trigger" additional replenishments for the Payload Module throughout its respective lifecycle. If this is the case, the Payload Module LCC between two mission lifetimes will appear as a vertical line in Figure 4-34.

On the basis of the results in both Figure 4-33 and Figure 4-34, employing shared resources beyond the Comm_CS_C&DH causes a very little reduction in the absolute mass of a Payload Module but tends to increase its respective absolute Dynamic LCC. Consequently, in contrast, employing shared resources beyond the Comm_CS_C&DH causes an appreciable increase in the absolute System Mass and Dynamic LCC. This therefore emphasizes that, for a given mission lifetime, employing shared resources reaches a point of diminishing returns (benefit) in terms of mass and LCC for fractionated spacecraft, from both the System and Payload Module perspectives. Subsequently, in considering both the System and Payload Module perspectives, for a given fractionated spacecraft architecture, employing fewer rather than more shared resources proves to yield the optimum tradeoff between minimizing the Mass and Dynamic LCC of the System (spacecraft) and Payload Module.

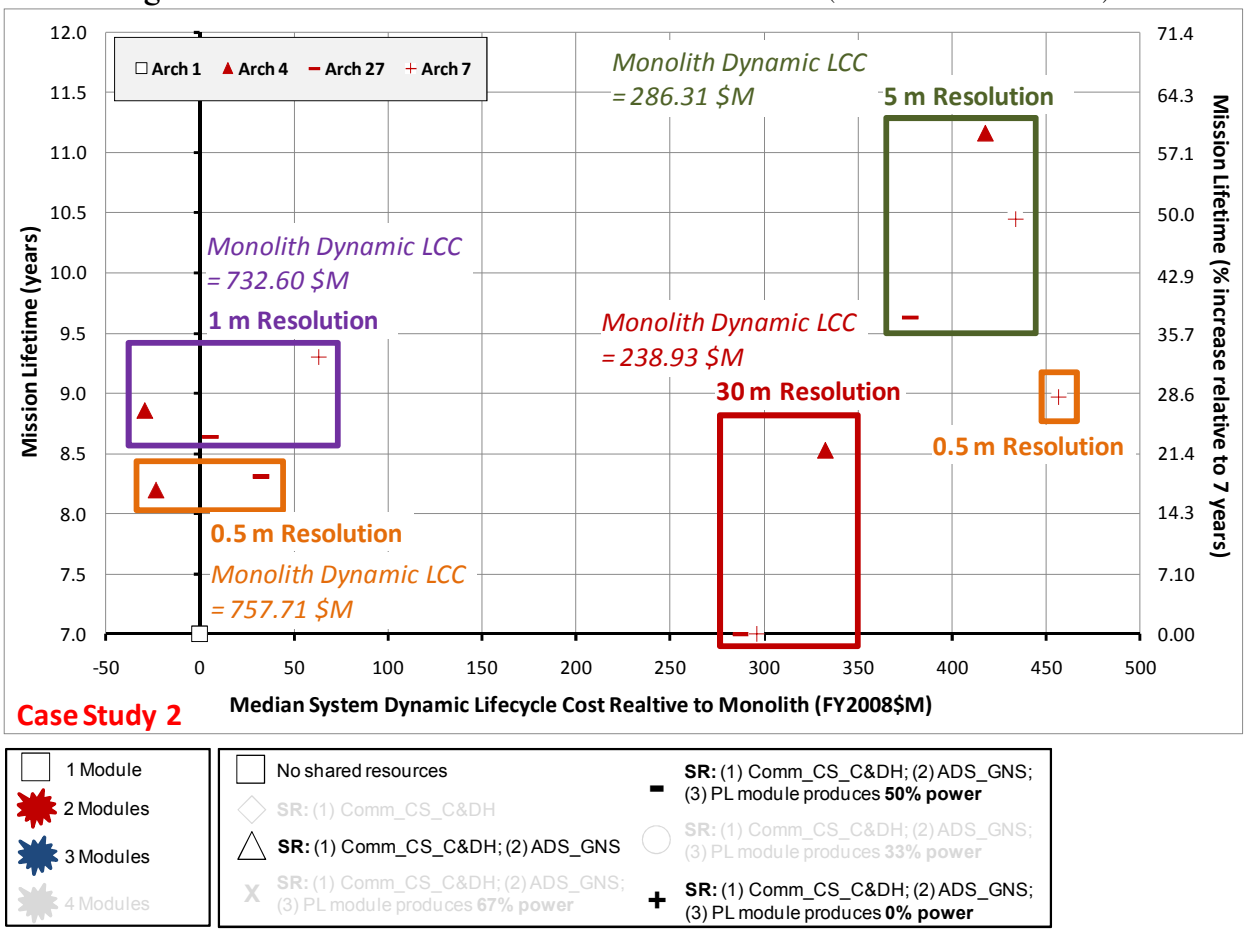
4.7. Mission Lifetime Capability

In Section 4.6, the Mass and Dynamic LCC benefits provided by fractionation relative to the Payload Module (*i.e.,* the structure containing the payload) perspective were quantified. Based on these quantifications, it was found that in terms of minimizing the System and Payload Module Mass and Dynamic LCC, employing fewer rather than more shared resources in fractionated spacecraft is desirable. This is the subsequent result of employing shared resources reaching a point of diminishing return (benefit) in terms of the Mass and Dynamic LCC of fractionated spacecraft. This conclusion however does not reflect all of the potential benefits to be gained through reducing the mass (and size) of the Payload Module via fractionation, specifically, through employing shared resources. One such hypothesized benefit resulting from the reduction in the mass (and size) of the Payload Module from using shared resources is to increase the mission lifetime capability of fractionated spacecraft relative to that of a comparable monolithic spacecraft (see Section 4.2.4). Subsequently this is the motivation for Section 4.7, which quantitatively enumerates the mission lifetime benefit provided by fractionation. More generally, Section 4.7 provides quantitative responses to the question; relative to a comparable monolithic spacecraft, what other benefits besides a reduction in Dynamic LCC (see Section 4.4) can fractionated spacecraft provide?

Interpreting Figure 4-35 and Figure 4-36: the x-axis value for each fractionated spacecraft is relative to the comparable monolithic spacecraft (*i.e.,* the monolith with the same Ground Resolution). Therefore, in Figure 4-35 and Figure 4-36, the Dynamic LCC corresponding to a monolithic spacecraft at each of the four Ground Resolutions is given and this information can subsequently be used to compute the absolute Dynamic LCC of all fractionated spacecraft in Figure 4-35 and Figure 4-36.

Figure 4-35 characterizes the two-module fractionated spacecraft investigated in Case Study 2 with respect to their (y1) mission lifetime capability (years); (y2) mission lifetime capability relative to 7 years (%); and (x) Median System *Dynamic* Lifecycle Cost relative to a comparable monolith (FY2008$M).

Figure 4-35. Mission Lifetime benefits due to fractionation (all Ground Resolutions).

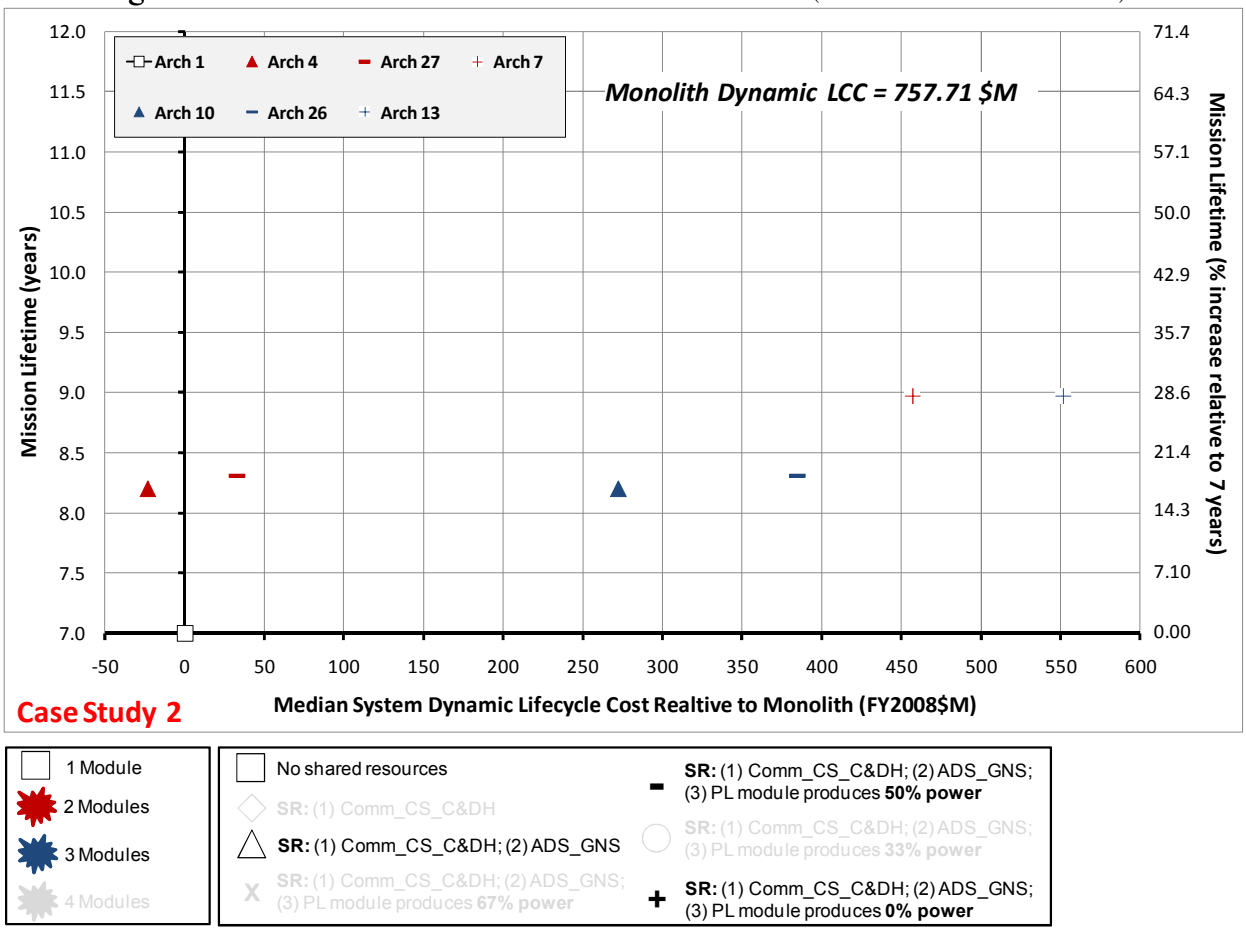


One observation with regard to the results presented in Figure 4-35 is that the mission lifetime capability of fractionated spacecraft is nonlinear with respect to Ground Resolution and use of shared resources, this being summarily evident by the distribution of data in Figure 4-35. The reason for this nonlinearity is that the mission lifetime capability of fractionated spacecraft, relative to the 7-year datum mission lifetime, is a function of the mass disparity between their respective Payload Modules and the comparable monolithic spacecraft as well as the physical design of those Payload Modules, as this dictates their respective propellant efficiency (and hence Propellant Usage). The mass disparity between a given Payload Module and monolith and the physical design of the Payload Module happen to be nonlinear with respect to Ground Resolution and use of shared resources – the physical modeling of spacecraft in the SET is, in general, highly nonlinear. Therefore, it is expected that the mission lifetime capability of fractionated spacecraft with respect to Ground Resolution and use of shared resources be nonlinear. And this nonlinear trend leads to the conclusion that across all Ground Resolutions considered there is no optimum use of shared resources to be employed in fractionated spacecraft, in terms of maximizing mission lifetime capability relative to a comparable monolithic spacecraft.

Of interest in Figure 4-35 are the 0.5 m Ground Resolution fractionated spacecraft, in particular because one of these architectures can provide more mission lifetime than the comparable monolith but at a lesser Dynamic LCC. Subsequently, 0.5 m Ground Resolution two *and* three-module fractionated spacecraft are depicted in Figure 4-36.

Figure 4-36 characterizes the 7 spacecraft architectures investigated in Case Study 2, at a 0.5 m Ground Resolution, with respect to their (y1) mission lifetime capability (years); (y2) mission lifetime capability relative to 7 years (%); and (x) Median System *Dynamic* Lifecycle Cost relative to a comparable monolith (FY2008$M).

Figure 4-36. Mission Lifetime benefits due to fractionation (0.5 m Ground Resolution).



A notable observation with regard to results shown in Figure 4-36 is that for a given Ground Resolution, comparable two and three-module fractionated spacecraft architectures have an identical mission lifetime capability, but the Dynamic LCC is always more for the three-module spacecraft architectures[20]. For a given Ground Resolution, comparable (*i.e.,* identical use of shared resources) two and three-module spacecraft architectures will have identical Payload Module designs (and hence Payload Module Mass) since the RSM payload mass, etc. is identical and, additionally, the hardware associated with sharing resources required on the Payload Module is independent of the number of modules in a fractionated spacecraft. As such, the mission lifetime capability will be the same for the two and three-module spacecraft architectures

[20] Although not shown in Section 4.7, this observation is true for fractionated spacecraft at any ground resolution.

because it is solely determined from mass differences between the Payload Module and comparable monolithic spacecraft. However, in contrast to the identical mission lifetime capability of comparable two and three-module fractionated spacecraft architectures, their respective Dynamic LCC values will different since the hardware associated with shared resources *is* dependent on all modules (*i.e.,* Payload and Infrastructure Modules) in a fractionated spacecraft. Subsequently, three-module fractionated architectures have a larger System Mass and aggregate number and cost of replenishments than that of comparable two-module fractionated architectures, thereby causing them to have a higher Dynamic LCC, despite their equivalent mission lifetime capability with comparable two-module fractionated architectures.

One of the most intriguing aspects of the results shown in Figure 4-36 is that Arch 4 is not only able to provide 1.20 more years of mission life (17.1%) than a comparable monolithic spacecraft, but it can do so for 23.41 (FY2008$M) less Dynamic LCC than the monolith performing its 7 year (shorter) mission. This result thereby provides one instantiation of the significant benefits, not enumerated in Section 4.6, to be gained from fractionation via (1) employing shared resources and (2) decoupling the payload and subsystems that truly need precise pointing from the other subsystems in a spacecraft. To place this specific result with respect to Arch 1 and 4 in context to the results presented earlier (see Section 4.4), for a 7-year mission lifetime, the Dynamic LCC disparity between Arch 1 and 4 is 90.73 (FY2008$M), Arch 4 being less expensive. Now if the mission lifetime of Arch 4 is increased to 8.2 years but Arch 1 still performs a 7 year mission, the Dynamic LCC disparity between Arch 1 and 4 is 23.41 (FY2008$M), Arch 4 still being less expensive. Subsequently, the lesser Dynamic LCC of Arch 4, at an 8.20-year mission lifetime, is the result of its ability to use a set of smaller and less expensive launch vehicles than the size and cost of the launch vehicle employed by Arch 1. (Recall that the context put forth for the analysis in Section 4.7 disallows the use of launch vehicles larger than that employed by a 7-year monolith, but it does not preclude the use of smaller launch vehicles.) The ramifications of the results in Figure 4-36 can be inferred to conclude that, given a fixed allowable cost for a spacecraft's respective development and ensuing operation, fractionated spacecraft can provide more mission lifetime than comparable monolithic spacecraft.

5. Synthesis: Discussion of the Results

Synthesis is the fourth and final phase of the research methodology (see Section 1.1.4). The objective of the synthesis is to enumerate the broader implications of the analysis in Chapter 1 as well as formulated succinct responses to the three research questions. The synthesis is subsequently parsed into two sections. Section 5.1 provides a discussion with regard to trends in monolithic and fractionated spacecraft value propositions that are consistent across the results in Chapter 4 (Section 5.1.1) as well as insights pertaining to the formation of the value proposition (Section 5.1.2). Section 5.2 then formulates succinct responses to each of the respective research questions based on the analysis in Chapter 1 and discussion in Section 5.1.

5.1. The Value Proposition: Trends and Formation

The analysis in Chapter 4 predominantly provides case study specific insights with regard to the value propositions for monolithic and fractionated spacecraft. Subsequently, very little of the analysis explicitly elicits general trends and relationships between fractionated and monolithic spacecraft value propositions that are consistent across the respective results of all three case studies. Nor is any of the analysis devoted to providing insight/reflection as to the formation of the value proposition, that is, the selection of the metrics for which to compose it. Therefore, the insights provided in Section 5.1 and Section 5.2 with regard to trends in, and the formation of, the value proposition respectively, is pertinent because they adopt a more holistic perspective of the results corresponding to the three case studies presented in Chapter 4. Therefore, the insights discussed in Section 5.1 and Section 5.2 may be constructive for future assessments of monolithic and fractionated spacecraft.

5.1.1. Value Proposition Trends: The Costs and Benefits of Fractionation

Fractionated spacecraft can be thought of as separating the constituents of a monolithic spacecraft into physically distinct structures (modules). The creation and ensuing operation of multiple modules in place of a single module (*i.e.,* monolithic structure) is the fundamental form (and often functional) difference between monolithic and fractionated spacecraft. The creation of modules via fractionation subsequently allows further nuances to exist between monolithic and fractionated spacecraft, namely, through the sharing of subsystem resources (see Section 2.1.5). Subsequently, because of the form and function differences between comparable monolithic and fractionated spacecraft, fractionated spacecraft will always have a larger aggregate (*i.e.,* all of the modules) mass and physical size. The reason as to the consistently larger mass of fractionated spacecraft is that they have a higher *system-wide redundancy*. System-wide redundancy is the total number of a certain subsystem in a fractionated spacecraft relative to one, as a monolith only has one of these subsystems. For example, a fractionated spacecraft with three modules will have three thermal control systems (TCS's) instead of one TCS, which a comparable monolith has, and therefore the fractionated spacecraft has system-wide redundancy. Consequently, system-wide redundancy has a mass and size penalty associated with it manifested in the form of mass and structural overhead respectively. The mass and structural overhead is the extra mass and size incurred when replacing one subsystem with a set of smaller subsystems, holding functionality roughly constant[21]. Therefore, as compared to a certain

[21] The mass penalty and hence larger aggregate mass of fractionated spacecraft as compared to a monolith resulting from system-wide redundancy is akin to comparing the mass of a Boeing 747-400 to *three* Boeing 707-320B (commercial aircraft). One Boeing 747-400 can carry roughly as many passengers as three Boeing 707-320B aircraft; hence, their respective functionality in terms of carrying passengers is roughly the same. However, despite their roughly equivalent functionality, the individual mass of each Boeing 707-320B is 40.3% the mass of a Boeing 747-400. Therefore, the aggregate mass of all three Boeing 707-320B aircraft is 120.9% the mass of a Boeing 747-400. Consequently, the structural overhead and subsequent mass penalty associated with

subsystem on a monolith, due to the mass and structural overhead associated with that subsystem on each of the *n* modules in a fractionated spacecraft, the aggregate mass (and size) of those *n* subsystems will be larger than that of the subsystem on the monolith. In extrapolating this logic to every subsystem, the aggregate mass and size of a fractionated spacecraft, based on its respective subsystems, will always be larger than that of a comparable (*i.e.,* functionally equivalent) monolith. Consequently, as the number of modules increases, it only further increases the mass disparity between fractionated and monolithic spacecraft.

System-wide redundancy also has implications for the hardware associated with sharing subsystem resources for modules that are both supplying a shared resource (sources) and receiving a shared resource (recipients) (see Section 2.1.5 and 4.1.3). Subsequently, employing shared resources changes the distribution of mass and size throughout a fractionated spacecraft that ultimately yields two consistent trends with regard to fractionated spacecraft and shared resources. The first trend is that for a given use of shared resources, as the number of modules in a fractionated spacecraft increases, the mass savings gained from sharing resources diminishes; hence, the aggregate mass of fractionated spacecraft tends to increase with use of shared resources. This is the result of an increase in the number of modules in a fractionated spacecraft causing proportionally larger increases in the mass penalties associated with system-wide redundancy. In turn, this is because an increasing number of subsystems (due to an increase in number of modules), proportionally increases the mass penalties due to structural overhead, holding subsystem functionality roughly constant. And the second trend is that the aggregate mass of a fractionated spacecraft **(1)** increases, **(2)** decreases, and **(3)** increases as its use of shared resources changes from **(1)** sharing no resources **to** sharing Comm_CS_C&DH; **(2)** sharing Comm_CS_C&DH **to** sharing Comm_CS_C&DH and ADS_GNS; and **(3)** sharing Comm_CS_C&DH **to** sharing Comm_CS_C&DH, ADS_GNS, and Power. It should be noted that (1) increase and (2) decrease as stated in the previous sentence are appreciably small as is evident by the very similar mass of, for example, Spacecraft Architecture 2, 3, and 4 in Figure 4-24.

Therefore, in terms of aggregate (*i.e.,* all modules) mass, fractionated spacecraft are always more massive than a comparable monolith when not sharing resources and sharing resources generally (*i.e.,* accounting for the last trend in the previous paragraph) exacerbates the mass disparity between comparable monolithic and fractionated spacecraft, in favor of the monolith. This observation is explicitly evident in the results for the three case studies in Chapter 4 in which fractionated spacecraft are always more massive than a comparable monolith. Additionally, in Chapter 4, it is readily apparent that, for a given use of shared resources, increasing the number of modules in a fractionated spacecraft increases its respective mass.

Consequently, the consistently higher aggregate mass of fractionated spacecraft relative to that of a comparable monolithic spacecraft leads to another consistent trend in fractionated spacecraft value propositions; specifically with regard to the Dynamic LCC. An appreciable portion of the Static and Dynamic LCC corresponding to a given spacecraft is due to NRE and RE costs, both of which correlate positively with mass. The NRE costs are incurred only for the initial (*i.e.,* BoL) development of a spacecraft. In contrast, the RE costs are incurred for the initial spacecraft development and for each subsequent replenishment of a spacecraft throughout the lifecycle. Therefore, the larger aggregate mass of fractionated spacecraft due to a larger number of modules and/or use of shared resources has the effect of increasing the Static and Dynamic LCC, based on NRE and RE costs, of fractionated spacecraft relative to that of a comparable monolith. Therefore, if the aggregate mass disparity between a fractionated spacecraft

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transporting the same number of passengers with three aircraft instead of one incurs a mass (and size) penalty analogous to system-wide redundancy in fractionated spacecraft incurring a mass (and size) penalty due to structural overhead.

and comparable monolith is large enough, it can ensure the Static and Dynamic LCC of the fractionated spacecraft will be larger based on NRE and RE costs alone.

Another consistent trend observed in the analysis in Chapter 4 is that as the number of modules present in a fractionated spacecraft increases, the aggregate (*i.e.,* across all modules) number of replenishments increases relative to that comparable monolithic spacecraft - but not necessarily the aggregate cost of replenishments. Given the significance of replenishment costs for the Dynamic LCC, this trend therefore strongly influences differences in the value proposition between comparable monolithic and fractionated spacecraft. As the number of modules and/or use of shared resources increases in a fractionated spacecraft, not only does the aggregate mass tend to increase as was discussed previously, but the number of inter-module dependencies in a fractionated spacecraft increases as well. For fractionated spacecraft, the number of inter-module dependencies increases with an increasing number of modules, holding the use of shared resources constant. As such, inter-module dependencies exacerbate the implications of modules failing on-orbit because a dependent module (*i.e.*, shared resource recipient) will fail if a module it is dependent upon (*i.e.*, shared resource source) fails. Therefore, in terms of aggregate number of replenishments, the implication of an increase in inter-module dependencies is to increase the aggregate number of replenishments. Consequently, this may increase the number of launch vehicle failures based on providing more opportunities for launch failures to occur. However in contrast, in terms of aggregate replenishment costs (comprised of RE and launch vehicle costs), relative to the aggregate replenishment costs of a comparable monolith, as the number of inter-module dependencies in a fractionated spacecraft increases, the fractionated spacecraft may or may not have a larger aggregate replenishment costs. The reason for this is differences in launch costs (usage) between comparable monolithic and fractionated spacecraft.

Additionally, the analysis in Chapter 4 provided quantitative instantiations both supporting and refuting the hypothesis that as the severity (harshness) of a spacecraft's respective lifecycle (mission) increases, fractionated spacecraft will perform "better" than a comparable monolithic spacecraft. This was specifically evident in the analysis in which it was shown that certain fractionated spacecraft are more Dynamic LCC-competitive to a comparable monolith if the spacecraft have *benign* lifecycles; whereas, other fractionated spacecraft were more Dynamic LCC-competitive if the spacecraft have *extremely harsh* lifecycles.

In contrast to the trends in the value proposition cited previously, which are generally disadvantageous to fractionated spacecraft, there are also trends in the respective value propositions for fractionated spacecraft that are advantageous. The first of these trends is that through the creation of modules and sharing of subsystem resources, fractionated spacecraft can potentially make use of a single smaller or set of smaller launch vehicles as compared to the launch vehicle employed by a comparable monolith. This is possible because each of the respective modules in a fractionated spacecraft may be individually smaller in mass and size than a comparable monolith.[22] This thereby enables the modules in a fractionated spacecraft potentially to fit into either a single smaller launch vehicle or any combination of up to three smaller launch vehicles as compared to the single launch vehicle employed by a comparable monolith. And since the cost of launch vehicles tends to decrease with their respective size (*i.e.,* push mass to LEO), fractionated spacecraft can potentially have lesser launch costs than that of a comparable monolith. Therefore, despite the higher aggregate mass and number of replenishments of fractionated spacecraft, due to the dominance of launch vehicle costs on the initial deployment and replenishment costs, the potentially lesser launch costs of

[22] For high-resolution fractionated spacecraft with 1000 m or less inter-module separation distances this is true. (As the case studies showed, these are the most Static and Dynamic LCC-competitive fractionated spacecraft to a comparable monolithic spacecraft.) The only situations in which the modules in a fractionated spacecraft are more massive and bigger than a comparable monolith are when the Power resource is shared and the inter-module separation distance is greater than or equal to 5000 m.

fractionated spacecraft provide an opportunity to reduce their respective Static and Dynamic LCC relative to that of a comparable monolith. The most important attribute of fractionated spacecraft in terms of Static and Dynamic LCC-competiveness is the ability to leverage their potentially lesser launch costs than that of a comparable monolithic spacecraft to attain a smaller Static and Dynamic LCC than the monolith. Subsequently, this attribute of fractionated spacecraft is the reason as to situations observed in the analysis in Chapter 1 in which a fractionated spacecraft has an equal or lesser Static and/or Dynamic LCC than that of a comparable monolith.

Another trend with regard to fractionated spacecraft that strengthens (and weakens) their respective value proposition relative to that of a comparable monolith has to do with varying payload performance. For high performance RSM payloads (*i.e.,* payloads with ground resolutions of 0.5 and 1 m), two-module fractionated spacecraft can be less expensive than a comparable monolith despite their larger aggregate mass and number of replenishments. The reason for this is due to the critical attribute of fractionation cited in the previous paragraph, namely, these fractionated spacecraft have appreciably lesser launch costs than that of the comparable monolith, and this enables them to attain a lesser Dynamic LCC. However in contrast, for the medium and low performance RSM payloads (*i.e.,* payloads with ground resolutions of 5 and 30 m respectively), a monolith is the least expensive spacecraft by a significant LCC margin. The reason for this is that as the resolution decreases, the payload drives less of the physical design of a spacecraft. Subsequently, using fractionation to separate (decouple) the pointing-intensive payload from the other modules/subsystems provides proportionally less mass and size reductions for the respective modules in a fractionated spacecraft relative to the mass and size of a comparable monolith. This is because the mass and structural overhead associated with system-wide redundancy and sharing resources proportionally increases with decreasing payload dominance (*i.e.,* ground resolution). Consequently, as the ground resolution decreases, the ability for fractionated to leverage lesser launch vehicles costs to reduce their respective Dynamic LCC relative to that of a comparable monolith diminishes or ceases altogether. For the two-module fractionated spacecraft in Figure 4-25, this situation does occur when the ground resolution is 5 and 30 m. At these two resolutions, the respective launch vehicle costs these fractionated spacecraft is equal to that of the monolith. Hence, given their already higher aggregate mass and lack of advantage in terms of launch costs, the Dynamic LCC of these fractionated spacecraft must be larger than that of the comparable monolith.

Lastly, a trend with regard to fractionated spacecraft that strengthens their respective value proposition relative to that of a comparable monolith is due to their potentially smaller Payload Module mass and size relative to the mass and size of a comparable monolith. The potential benefit resulting from the seclusion of the RSM payload onto its own module via fractionation is to enable fractionated spacecraft to have longer mission lifetimes than that of a comparable monolith. Consider the situation in which a monolithic spacecraft, performing an n year mission, reaches the maximum mass and size limits of the largest possible launch vehicle available for use; and additionally, the Payload Module (and the Infrastructure Modules) in a fractionated spacecraft are less massive and smaller than the monolith for that same n year mission. If this situation does occur, it enables fractionated spacecraft to have a mission lifetime longer than n years, and sometimes for a lesser Dynamic LCC than the n-year mission monolith (see Section 4.7, specifically Figure 4-35 and Figure 4-36). As such, a consistent trend is that as long as the inter-module separation distance is less than or equal to 1000 m, through the creation of modules and sharing of subsystem resources, fractionated spacecraft are able to reduce the mass and size of the Payload Module (and Infrastructure Modules) relative to the mass and size of a comparable monolith. Subsequently, this increases the mission lifetime capability (potential) of fractionated spacecraft relative to that of a comparable monolith while, in some cases, still keeping the Dynamic LCC of fractionated spacecraft less than that of the monolith.

The culmination of the trends enumerated in Section 5.1.1, with respect to the monolithic and fractionated spacecraft value propositions, is the explicit identification of the catalysts for differences between the value propositions of comparable monolithic and fractionated spacecraft. These catalysts can be interpreted as "leverage points" for demonstrating that fractionated spacecraft can have appreciably stronger *or* weaker value propositions relative to the value proposition of a comparable monolithic spacecraft.

1. **Lifecycle Uncertainties: (a)** As the probability of occurrence for the risks resulting from a lifecycle uncertainties (*e.g.,* on-orbit failure) decreases/increases, it provides less/more opportunities for fractionated spacecraft leverage potentially lower replenishment costs against a comparable monoliths respective replenishment costs to attain a lesser Dynamic LCC than the monolith. **(b)** The Mass, Propellant Usage, Payload Performance, and Mission Lifetime dimensions of the value proposition remain unchanged with regard to lifecycle uncertainties.
2. **Number of Modules and Shared Resources: (a)** As the number of inter-module dependencies in a fractionated spacecraft decreases/increases as the result of a decrease/increase in the number of modules and/or use of shared resources, the aggregate number of replenishments of the fractionated spacecraft will decrease/increase. **(b)** Relative to a comparable monolith, as the number of inter-module dependencies decreases/increase in a fractionated spacecraft, the aggregate costs replenishments costs relative to that of a comparable monolith may or may not decrease/increase based on differences in launch vehicle usage between the monolithic and fractionated spacecraft. **(c)** As the number of modules in a fractionated spacecraft decreases/increases, the Static and Dynamic LCC due to NRE and RE costs will decrease/increase.
3. **Mission Lifetime: (a)** As the mission lifetime decreases/increases, it proportionally decreases/increases the aggregate Mass, Propellant Usage, and NRE and RE costs of fractionated spacecraft relative to that of a comparable monolithic spacecraft. **(b)** As the mission lifetime decreases/increases, the opportunities for (*i.e.,* period of time in which) the risks resulting from lifecycle uncertainties (*e.g.,* on-orbit failure) to occur decreases/increases. **(c)** Thus, one implication of decreasing/increasing mission lifetime is a proportional decrease/increase in the NRE and RE cost contributions to a spacecraft's respective Static and Dynamic LCC. **(d)** And another implication of decreasing/increasing mission lifetime is to provide less/more opportunities for fractionated spacecraft to leverage their potentially lower replenishment costs against a comparable monolithic spacecraft's replenishment costs to attain a lower Dynamic LCC than that of the monolith. **(e)** Lastly, fractionated spacecraft can have a longer mission lifetime relative to that of a comparable monolithic spacecraft if their respective Payload Module mass and size is smaller than the monolith, which is the case for pointing-intensive, high-resolution payload RSM missions.
4. **Launch Vehicle Reliability and Access: (a)** For a given launch vehicle reliability, as access to smaller and less expensive launch vehicles decreases/increases, the probability that the aggregate cost of replenishments of fractionated spacecraft will be less than the cost of replenishments (and subsequently Static and Dynamic LCC) of a comparable monolith decreases/increases.
5. **Payload Performance: (a)** As the RSM payload Ground Resolution decreases/increases, in terms of Static and Dynamic LCC, it becomes more/less desirable to use a fractionated spacecraft instead of a monolithic spacecraft.

Based on the analysis presented in Chapter 1 and discussion in Section 5.1.1, the five catalysts listed above provide the most appropriate synthesis of the "leverage (or hinge) points" for which fractionated spacecraft can demonstrate to have stronger *or* weaker value propositions than comparable monolithic spacecraft.

5.1.2. Value Proposition Formation

In addition to the insights enumerated in Section 5.1.1 with regard to trends in monolithic and fractionated spacecraft value propositions, there are also two important insights to be noted with regard to the formation of the value proposition, that is, the selection of the metrics for which to compose the value proposition. The discussion of these two respective insights about the value proposition formation is based on the metrics comprising the value proposition employed in this research (see Section 3.5.2).

The first insight regarding the value proposition formation is that forming the value proposition with the System and Payload Module-level metrics is beneficial for understanding monolithic and fractionated spacecraft value propositions. Recall that the value proposition is composed of the *System* Median Dynamic LCC, Static LCC, Mass, Propellant Usage, Mission Lifetime, Payload Performance; and *Payload Module* Median Dynamic LCC, Mass, Propellant Usage, Mission Lifetime, Payload Performance. As is reasoned in Section 3.5.2, these 11 metrics in the value proposition are appropriate given their representative nature; this being evident by their accurate encapsulation of the component and subsystem-level characteristics of a spacecraft while observing the need to keep the comparison of value propositions tractable. Subsequently, the discussion in the analysis in Chapter 1 confirmed that the 11 metrics in the value proposition are comprehensive enough, but still small enough in number, to readily compare two or spacecraft value propositions. Additionally through the analysis in Chapter 4, it was confirmed that the 11 metrics are also detailed and accurate enough for differences in two or more value propositions to be identified/traced to specific nuances in spacecraft architecture designs down to the component and subsystem-level. In this sense, the value proposition and its respective 11 metrics employed herein for the comparison of monolithic and fractionated spacecraft provided a reasonable balance of fidelity, tractability, and scope.

The remaining notable comment with regard to the value proposition formation specifically addresses the incorporation of System Static *and* Dynamic LCC in the value proposition. Nearly every previous assessment of fractionated spacecraft (see Section 2.3) opted to quantify the cost of fractionation and implications of shared resources (if applicable) using Static LCC. However, in contrast, the analysis in Chapter 4 and synthesis in Chapter 5, emphasize the importance of the incorporation and subsequent use of Static *and* Dynamic LCC in the value proposition for monolithic and fractionated spacecraft. It is evident from the analysis in Chapter 4, specifically in Figure 4-27, that Static LCC is an inappropriate LCC metric for quantifying the LCC implications of nuances in spacecraft design (*e.g.,* use of shared resources) when spacecraft have stochastic lifecycles (due to the risks resulting from lifecycle uncertainties). This observation therefore serves as motivation for the use of both Static *and* Dynamic LCC as complements in the value proposition, rather than treat them as competing LCC paradigms. Static LCC is useful for assessing monolithic and fractionated spacecraft value propositions from the perspective of the best-case scenario lifecycle for a spacecraft, as Static LCC ignores the adverse LCC implications of risks resulting from lifecycle uncertainties. Additionally, the Static LCC is beneficial for instilling confidence in the Dynamic LCC and hence dynamic lifecycle simulation model within the SET, if a logical path of reasoning can demonstrate differences in (or transition between) Static and Dynamic LCC for a given spacecraft (see Figure 4-27 and associated discussion). However, in considering more realistic stochastic spacecraft lifecycles, Dynamic LCC becomes unequivocally more appropriate than Static LCC because it fully accounts for the adverse LCC implications of risks resulting from lifecycle uncertainties (see Section 3.3.3).

5.2. Response to the Research Questions

Based on the breadth and depth of the analysis provided in Chapter 1 and subsequent discussion in Section 5.1, the most succinct manner to address the research questions is to formulate responses to each research question and with respect to each metric of the value proposition. Section 5.2.1, 5.2.1, and 5.2.3 are therefore devoted to addressing the first, second, and third research question respectively. Section 5.2.4 then provides a condensed summary of the responses to all the research questions.

The responses to the research questions provided hereafter are qualitative to keep the responses representative, meaningful, and tractable. These qualitative responses therefore elicit the aggregate responses to the research questions based on *all the data* presented in the analysis in Chapter 4. Employing this approach for responding to the research questions is far more constructive than providing a (very long) laundry list of quantitative responses to each respective research question based on *all the data* presented in the analysis in Chapter 4. Therefore, even though it may appear that the responses to the research questions are qualitative, they are in fact not because quantitative instantiations these responses proliferate throughout the analysis in Chapter 4. To find a quantitative instantiation corresponding to a given response in Section 5.2, simply find data in the analysis in Chapter 4 captured by that response, which is not difficult given the long list of trends explicitly called out in Chapter 4. To this end, Table 5-2 through Table 5-5 are very helpful as they cite specific case studies containing quantitative instantiations (results) corresponding to each of the respective responses to the research questions provided in Section 5.2.

Please note that a table (*i.e.,* Table 5-2*,* Table 5-3*, and* Table 5-4) is provided at the end of the response to each respective research question that summarizes the differences in monolithic and fractionated spacecraft value propositions based on that research question. In this summary, with respect to each of the 11 metrics in the value proposition, the value propositions for monolithic and fractionated spacecraft relative to one another are cited as being stronger, equal, and/or weaker; here stronger, equal, and weaker respectively mean better, equivalent, and/or worse. In addition, in the summary the specific context (case study results) used to form the conclusion as to the relative difference between monolithic and fractionated value propositions is provided for reference purposes. This is also done to reinforce that the conclusions made with regard to monolithic and fractionated spacecraft value propositions are only valid for the case studies and results presented herein, and therefore may not be applicable in other contexts. Table 5-1 will be of use in finding the specific results corresponding to the research question responses elicited hereafter as it provides a listing of the figures found in Section 4.3 through 4.7 containing the respective results for each of the three case studies

Table 5-1. Case study results guide.

Case Study					
1		2		3	
Figure 4-	Page No.	Figure 4-	Page No.	Figure 4-	Page No.
19	108	21	112	23	114
20	109	22	113	26	119
24	115	25	118	30	126
27	121	29	124	33	129
28	123	35	133	34	131
31	127	36	134		
32	127				

5.2.1. Research Question 1: Spacecraft Architectures

How do the value propositions for monolithic and fractionated spacecraft compare across alternative spacecraft architectures (designs)?

System-Level Value Proposition Metrics

1. Mission Lifetime: Given the hypothetical situation posited in Section 4.2.4. **(a)** Regardless of the payload Ground Resolution, fractionated spacecraft that share resources have an equal or longer Mission Lifetime capability than a comparable monolith. And in terms of comparable (*i.e.,* identical use of shared resources) fractionated spacecraft, two and three-module fractionated spacecraft have an identical Mission Lifetime capability relative to that of a comparable monolith. **(b)** Amongst the fractionated spacecraft considered, two and three-module spacecraft that share the Comm_CS_C&DH and ADS_GNS resources, but no Power (*i.e.,* the Payload Module produces 100% of its power) consistently have the largest Mission Lifetime capability (per unit LCC) relative to a comparable monolith, regardless of payload performance (*i.e.,* Ground Resolution). **(c)** Two and three-module spacecraft that share all resources, that is, Comm_CS_C&DH, ADS_GNS, and Power have similar, but always lesser, Mission Lifetime capabilities (per unit LCC) as compared to two and three-module spacecraft sharing only the Comm_CS_C&DH and ADS_GNS resource. **(d)** For fractionated spacecraft that share all resources, as the amount of Power sharing increases (*i.e.,* the amount of power produced by the Payload Module decreases), the Mission Lifetime capability of fractionated spacecraft relative to that of a comparable monolith increases.
2. Static LCC: **(a)** On the basis of Static LCC, monolithic spacecraft are always less expensive than comparable two, three, and four-module fractionated spacecraft. **(b)** Among the fractionated spacecraft considered, for comparable two, three, and four-module fractionated spacecraft, the Static LCC of these fractionated spacecraft, from highest to lowest, will always correspond to the four, three, and two-module fractionated spacecraft respectively. **(c)** For a given class of fractionated spacecraft (*e.g.,* two-module spacecraft), the spacecraft that share the Comm_CS_C&DH and ADS_GNS resource, but no Power will always have the smallest Static LCC relative to that of a comparable monolith. **(d)** Additionally, for fractionated spacecraft that share all resources, that is, Comm_CS_C&DH, ADS_GNS, and Power, as the inter-module separation distance increases, their respective Static LCC increases proportionally.
3. Median Dynamic LCC: **(a)** The Median Dynamic LCC for three and four-module fractionated spacecraft is always appreciably larger than that of a comparable monolith, whereas in contrast, two-module fractionated spacecraft can have larger, equal, or smaller Median Dynamic LCC values than a comparable monolith. **(b)** For four-module fractionated spacecraft, the spacecraft that are always the most Median Dynamic LCC-competitive to a comparable monolith are those spacecraft that do not employ shared resources. **(c)** For three-module fractionated spacecraft, the spacecraft that are always the most Median Dynamic LCC competitive to a comparable monolith are those spacecraft that only share the Comm_CS_C&DH and/or ADS_GNS resource, but no power. **(d)** There are specific contexts (*e.g.,* a case study and its respective SET inputs) that provide quantitative instantiations of a monolithic spacecraft having a larger *and* smaller Median Dynamic LCC than a comparable two-module fractionated spacecraft. **(e)** Across all case studies and fractionated spacecraft considered, two-module fractionated spacecraft that share only the Comm_CS_C&DH and/or ADS_GNS resource are always the most Median Dynamic LCC-competitive to a comparable monolith. **(f)** For all fractionated spacecraft considered, as the inter-module separation distances decreases, their Median Dynamic LCC disparity relative to a comparable monolith *stays the same* and *decreases* if the fractionated spacecraft *does not* and *does employ* the Power shared resource, respectively.

4. Payload Performance: **(a)** All monolithic and fractionated spacecraft are capable of the same payload performance (*i.e.,* Ground Resolution).
5. Mass: **(a)** Relative to a comparable monolith, two, three, and four-module fractionated spacecraft are always more massive. **(b)** For each respective class of fractionated spacecraft (*e.g.,* two-module spacecraft) and regardless of the inter-module separation distance, the least massive spacecraft relative to the Mass of a comparable monolith is always the spacecraft that does not employ shared resources.
6. Propellant Usage: **(a)** Since Propellant Usage is a linear transformation of Mass (see Figure 4-19), the response with regard to (5) Mass, is identical for (6) Propellant Usage.

Payload Module-Level Value Proposition Metrics

7. Mission Lifetime: Given the hypothetical situation posited in Section 4.2.4. **(a)** At all Ground Resolutions (*i.e.,* payload performances) considered, Payload Modules have an equal or longer Mission Lifetime capability than a comparable monolith. **(b)** For a given Payload Module and its respective reliance on shared resources, the relationship between Mission Lifetime capability to that of a comparable monolith and Ground Resolution is nonlinear, with a Ground Resolution of 5 m enabling the yielding the longest Mission Lifetime capability. **(c)** For Payload Modules that rely only the Comm_CS_C&DH and ADS_GNS shared resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 30, and 0.5 m respectively. **(d)** And for Payload Modules that rely on the Comm_CS_C&DH, ADS_GNS, and Power shared resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 0.5, and 30 m respectively.
8. Median Dynamic LCC: **(a)** The Payload Module Median Dynamic LCC will always be less than that of a comparable monolith. **(b)** The Payload Module Dynamic LCC is driven by the aggregate number of replenishments (not mass) of the Payload Module. Therefore, as the number of replenishments increases for the Payload Module, the Median Dynamic Payload Module LCC increases. Subsequently, as a direct result of this, Payload Modules corresponding to fractionated spacecraft that yield similar or identical aggregate numbers of replenishments *and* distribution of replenishments across their respective modules will end up having a similar Median Dynamic LCC values. **(c)** For the Payload Module Median Dynamic LCC, there is not a positive correlation between the shared resources the Payload Module relies on and the context (*e.g.,* a case study and its respective SET inputs) in which the Payload Module operates.
9. Payload Performance: **(a)** All monolithic spacecraft and Payload Modules are capable of the same payload performance (*i.e.,* Ground Resolution).
10. Mass: **(a)** For a given context (*e.g.,* a case study and its respective SET inputs), the Payload Module Mass remains nearly constant, regardless of the capacity in which the Payload Module relies on shared resources. **(b)** In relation to a comparable monolith, the Payload Module Mass is always less.
11. Propellant Usage: **(a)** In a fractionated spacecraft, the relationship between the System Propellant Usage and Mass is simply a scaled aggregation of the linear relationships between the Propellant Usage and Mass of each respective module in the spacecraft. Therefore, the results in Figure 4-19 apply by analogy to the Payload Module. Subsequently, the Payload Module Propellant Usage is a linear transformation of Payload Module Mass and therefore the response with regard to (10) Mass, is identical for (11) Propellant Usage.

Based on the responses to the first research question provided in Section 5.2.1, Table 5-2 summarizes the differences in the monolithic and fractionated value propositions. Thereafter, a succinct response to the first research question is given.

Table 5-2. Research Question 1: summary of responses with regard to the value proposition.

			Monolithic Spacecraft	Fractionated Spacecraft	
			Relative Strength		
		Value Proposition Metric	Stronger/Equal/Weaker		Context (Results)
System	1	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	2	Static LCC	Stronger	Weaker	Case Study 1
	3	Median Dynamic LCC	Stronger/Equal/Weaker	Weaker/Equal/Stronger	Case Study 1
	4	Payload Performance	Equal	Equal	Case Study 1
	5	Mass	Stronger	Weaker	Case Study 1
	6	Propellant Usage	Stronger	Weaker	Case Study 1
Payload Module	7	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	8	Median Dynamic LCC	Weaker	Stronger	Case Study 3
	9	Payload Performance	Equal	Equal	Case Study 3
	10	Mass	Weaker	Stronger	Case Study 3
	11	Propellant Usage	Weaker	Stronger	Case Study 3

Fractionated spacecraft can have a **stronger or equal** value proposition with regard to the specified metric

Fractionated spacecraft can have a **weaker** value proposition with regard to the specified metric

How do the value propositions for monolithic and fractionated spacecraft compare across alternative spacecraft architectures (designs)? Relative to the monolithic and fractionated spacecraft architectures investigated in Case Study 1, 2, and 3 as well as the metrics of the value proposition quantified with regard to this research question, fractionated spacecraft…

- *can be stronger or equal* than monolithic spacecraft with respect to 8 out of 11 (or 73% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be stronger or equal* than a comparable monolith:
  - System: Mission Lifetime, Median Dynamic LCC, and Payload Performance
  - Payload Module: Mission Lifetime, Median Dynamic LCC, Payload Performance, Mass, and Propellant Usage
- *can be weaker* than monolithic spacecraft with respect to 4 out of 12 (or 33% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be weaker* than a comparable monolith:
  - System: Static LCC, Median Dynamic LCC, Mass, and Propellant Usage
  - Payload Module: none

5.2.2. Research Question 2: Payload Performance

How do the value propositions for monolithic and fractionated spacecraft compare relative to changing payload requirements (*i.e.,* Ground Resolution)?

System-Level Value Proposition Metrics

1. Mission Lifetime: Given the hypothetical situation posited in Section 4.2.4. **(a)** At all Ground Resolutions (*i.e.,* payload performances) considered, fractionated spacecraft have an equal or longer Mission Lifetime capability than a comparable monolith. **(b)** For a given two or three-module fractionated spacecraft and its respective use of shared resources, the relationship between Mission Lifetime capability to that of a comparable monolith and Ground Resolution is nonlinear, with a Ground Resolution of 5 m providing the longest Mission Lifetime capability. **(c)** For fractionated spacecraft that share only the Comm_CS_C&DH and ADS_GNS resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 30, and 0.5 m respectively. **(d)** And for fractionated spacecraft that share the Comm_CS_C&DH, ADS_GNS, and Power resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 0.5, and 30 m respectively.
2. Static LCC: Response cannot be formulated based on the results presented in Chapter 4.
3. Median Dynamic LCC: **(a)** On the basis of Ground Resolution (*i.e.,* payload performance), fractionated spacecraft can have Median Dynamic LCC values that are larger, equivalent, and smaller than that of a comparable monolithic spacecraft. **(b)** For a given fractionated spacecraft, the relationship between Ground Resolution and disparity in that spacecraft's respective Median Dynamic LCC relative to that of a comparable monolith is nonlinear and, additionally, this nonlinear relationship differs across all spacecraft architectures considered. **(c)** Independent of the number of modules in a fractionated spacecraft and Ground Resolution, fractionated spacecraft the share only the Comm_CS_C&DH and ADS_GNS resource are consistently the most Median Dynamic LCC-competitive spacecraft with respect to a comparable monolith. **(d)** At high Ground Resolutions (*i.e.,* 0.5 and 1 m), two-module fractionated spacecraft that share only the Comm_CS_C&DH and ADS_GNS resource will be less expensive than a comparable monolithic spacecraft.
4. Payload Performance: **(a)** All monolithic and fractionated spacecraft are capable of the same payload performance (*i.e.,* Ground Resolution).
5. Mass: **(a)** Regardless of Ground Resolution, fractionated spacecraft are always more massive than a comparable monolith. **(b)** For a given fractionated spacecraft, there is a positive correlation between Ground Resolution and disparity in that spacecraft's respective Mass relative to that of a comparable monolith. **(c)** Independent of the number of modules, for fractionated spacecraft that share only the Comm_CS_C&DH and ADS_GNS resource, their respective *relative* Mass disparity to a comparable monolith, from smallest to largest, corresponds to a Ground Resolution of 30, 0.5, 1, and 5 m respectively. **(d)** And independent of the number of modules, for fractionated spacecraft that share Comm_CS_C&DH, ADS_GNS, and Power resource, their respective *relative* Mass disparity to a comparable monolith, from smallest to largest, corresponds to a Ground Resolution of 30, 5, 1, and 0.5 m respectively.
6. Propellant Usage: **(a)** Since Propellant Usage is a linear transformation of Mass (see Figure 4-19), the response with regard to (5) Mass, is identical for (6) Propellant Usage.

Payload Module-Level Value Proposition Metrics

7. Mission Lifetime: Given the hypothetical situation posited in Section 4.2.4. **(a)** At all Ground Resolutions (*i.e.,* payload performances) considered, Payload Modules have an equal or longer Mission Lifetime capability than a comparable monolith. **(b)** For a given Payload Module and its respective reliance on shared resources, the relationship between Mission Lifetime capability to that of a comparable monolith and Ground Resolution is nonlinear, with a Ground Resolution of 5 m enabling the yielding the longest Mission Lifetime capability. **(c)** For Payload Modules that rely only the Comm_CS_C&DH and ADS_GNS shared resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 30, and 0.5 m respectively. **(d)** And for Payload Modules that rely on the Comm_CS_C&DH, ADS_GNS, and Power shared resource, their respective Mission Lifetime capability relative to that of a comparable monolith, from longest to shortest, corresponds to a Ground Resolution of 5, 1, 0.5, and 30 m respectively.
8. Median Dynamic LCC: Response cannot be formulated based on the results presented in Chapter 4.
9. Payload Performance: Response cannot be formulated based on the results presented in Chapter 4.
10. Mass: Response cannot be formulated based on the results presented in Chapter 4.
11. Propellant Usage: Response cannot be formulated based on the results presented in Chapter 4.

Based on the responses to the second research question provided in Section 5.2.2, Table 5-3 summarizes the differences in the monolithic and fractionated value propositions. Thereafter, a succinct response to the second research question is given.

Table 5-3. Research Question 2: summary of responses with regard to the value proposition.

			Monolithic Spacecraft	Fractionated Spacecraft	
			Relative Strength		
		Value Proposition Metric	Stronger/Equal/Weaker		Context (Results)
System	1	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	2	Static LCC	Response cannot be formulated based on the results presented in Chapter 4.		
	3	Median Dynamic LCC	Stronger/Equal/Weaker	Weaker/Equal/Stronger	Case Study 2
	4	Payload Performance	Equal	Equal	Case Study 2
	5	Mass	Stronger	Weaker	Case Study 2
	6	Propellant Usage	Stronger	Weaker	Case Study 2
Payload Module	7	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	8	Median Dynamic LCC	Response cannot be formulated based on the results presented in Chapter 4.		
	9	Payload Performance	Response cannot be formulated based on the results presented in Chapter 4.		
	10	Mass	Response cannot be formulated based on the results presented in Chapter 4.		
	11	Propellant Usage	Response cannot be formulated based on the results presented in Chapter 4.		

Fractionated spacecraft can have a **stronger or equal** value proposition with regard to the specified metric

Fractionated spacecraft can have a **weaker** value proposition with regard to the specified metric

How do the value propositions for monolithic and fractionated spacecraft compare relative to changing payload requirements (*i.e.,* Ground Resolution)? Relative to the monolithic and fractionated spacecraft architectures investigated in Case Study 2 and the metrics of the value proposition quantified with regard to this research question, fractionated spacecraft…

- *can be stronger or equal* than monolithic spacecraft with respect to 4 out of 6 (or 67% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be stronger or equal* than a comparable monolith:
  - System: Mission Lifetime, Median Dynamic LCC, and Payload Performance
  - Payload Module: Mission Lifetime
- *can be weaker* than monolithic spacecraft with respect to 3 out of 6 (or 50% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be weaker* than a comparable monolith:
  - System: Median Dynamic LCC, Mass, and Propellant Usage
  - Payload Module: none

5.2.3. Research Question 3: Lifecycle Uncertainties

How do the value propositions for monolithic and fractionated spacecraft compare relative to risks resulting from spacecraft lifecycle uncertainties (*e.g.,* on-orbit failure)?

System-Level Value Proposition Metrics

1. Mission Lifetime: Response cannot be formulated based on the results presented in Chapter 4.
2. Static LCC: Response cannot be formulated based on the results presented in Chapter 4.
3. Median Dynamic LCC: **(a)** On the basis of any of the nine combinations of Mission Lifetime and PoIM values considered in Case Study 3, fractionated spacecraft can have Median Dynamic LCC values that are larger, equivalent, and smaller than that of a comparable monolithic spacecraft. **(b)** Across all fractionated spacecraft considered, with regard to the Median Dynamic LCC-competiveness of fractionated spacecraft relative to a comparable monolith, there is not a consistent optimum Mission Lifetime and PoIM, that is, not all fractionated spacecraft have their smallest Median Dynamic LCC with regard to same combination of Mission Lifetime and PoIM. **(c)** Aside from a lack of optimum Mission Lifetime and PoIM, a trend does exist given all nine combinations of Mission Lifetime and PoIM parings. Independent of the number of modules, fractionated spacecraft that share only Comm_CS_C&DH and ADS_GNS resource are consistently the most Median Dynamic LCC-competitive spacecraft relative to a comparable monolith. These specific types of fractionated spacecraft thus provide the most assurance across potential combinations of Mission Lifetimes and PoIM in terms of minimizing Median Dynamic LCC for fractionated spacecraft. **(d)** As the use of shared resources increases in fractionated spacecraft, across all Mission Lifetime and PoIM values, the Median Dynamic LCC-competiveness of fractionated spacecraft relative to a comparable monolith decreases. **(e)** There are certain combinations of Mission Lifetime and PoIM that demonstrate two-module fractionated spacecraft as being less expensive than a comparable monolith.
4. Payload Performance: **(a)** All monolithic and fractionated spacecraft are capable of the same payload performance (*i.e.,* Ground Resolution).
5. Mass: **(a)** Regardless of Mission Lifetime and PoIM, fractionated spacecraft are always more massive than a comparable monolith. **(b)** Mass is dependent on Mission Lifetime and independent of PoIM. **(c)** For a given fractionated spacecraft, as the Mission Lifetime increases, the Mass of the spacecraft relative to a comparable monolith increases. **(c)** For a given Mission Lifetime and class of fractionated spacecraft (*e.g.* two-module spacecraft), as the use of shared resources in a fractionated spacecraft

increases, the Mass disparity increases between that fractionated spacecraft and a comparable monolithic spacecraft, in favor of the monolith. **(d)** And with regard to comparable two and three-module fractionated spacecraft, as the Mission Lifetime increases, the two-module spacecraft will always have a lower Mass disparity relative to a comparable monolith than that of three-module fractionated spacecraft.

6. Propellant Usage: **(a)** Since Propellant Usage is a linear transformation of Mass (see Figure 4-19) the response with regard to (5) Mass, is identical for (6) Propellant Usage.

Payload Module-Level Value Proposition Metrics

7. Mission Lifetime: Response cannot be formulated based on the results presented in Chapter 4.
8. Median Dynamic LCC: **(a)** The Payload Module Median Dynamic LCC will always be less than that of a comparable monolith. **(b)** The Payload Module Dynamic LCC is driven by the aggregate number of replenishments (not mass) of the Payload Module. Therefore, as the number of replenishments increases for the Payload Module, the Median Dynamic Payload Module LCC increases. Subsequently, as a direct result of this, Payload Modules corresponding to fractionated spacecraft that yield similar or identical aggregate numbers of replenishments *and* distribution of replenishments across their respective modules will end up having a similar Median Dynamic LCC values. **(c)** For the Payload Module Median Dynamic LCC, there is not a positive correlation between the shared resources the Payload Module relies on and the context (*e.g.,* a case study and its respective SET inputs) in which the Payload Module operates.
9. Payload Performance: **(a)** All monolithic spacecraft and Payload Modules are capable of the same payload performance (*i.e.,* Ground Resolution).
10. Mass: **(a)** For a given context (*e.g.,* a case study and its respective SET inputs), the Payload Module Mass remains nearly constant, regardless of the capacity in which the Payload Module relies on shared resources. **(b)** In relation to a comparable monolith, the Payload Module Mass is always less.
11. Propellant Usage: **(a)** In a fractionated spacecraft, the relationship between the System Propellant Usage and Mass is simply a scaled aggregation of the linear relationships between the Propellant Usage and Mass of each respective module in the spacecraft. Therefore, the results in Figure 4-19 apply by analogy to the Payload Module. Subsequently, the Payload Module Propellant Usage is a linear transformation of Payload Module Mass and therefore the response with regard to (10) Mass, is identical for (11) Propellant Usage.

Based on the responses to the third research question provided in Section 5.2.3, Table 5-4 summarizes the differences in the monolithic and fractionated value propositions. Thereafter, a succinct response to the third research question is given.

Table 5-4. Research Question 3: summary of responses with regard to the value proposition.

		Value Proposition Metric	Monolithic Spacecraft	Fractionated Spacecraft	Context (Results)
			Relative Strength		
			Stronger/Equal/Weaker		
System	1	Mission Lifetime	Response cannot be formulated based on the results presented in Chapter 4.		
	2	Static LCC	Response cannot be formulated based on the results presented in Chapter 4.		
	3	Median Dynamic LCC	Stronger/Equal/Weaker	Weaker/Equal/Stronger	Case Study 3
	4	Payload Performance	Equal	Equal	Case Study 3
	5	Mass	Stronger	Weaker	Case Study 3
	6	Propellant Usage	Stronger	Weaker	Case Study 3
Payload Module	7	Mission Lifetime	Response cannot be formulated based on the results presented in Chapter 4.		
	8	Median Dynamic LCC	Weaker	Stronger	Case Study 3
	9	Payload Performance	Equal	Equal	Case Study 3
	10	Mass	Weaker	Stronger	Case Study 3
	11	Propellant Usage	Weaker	Stronger	Case Study 3

Fractionated spacecraft can have a **stronger or equal** value proposition with regard to the specified metric

Fractionated spacecraft can have a **weaker** value proposition with regard to the specified metric

How do the value propositions for monolithic and fractionated spacecraft compare relative to risks resulting from spacecraft lifecycle uncertainties (*e.g.*, on-orbit failure)? Relative to the monolithic and fractionated spacecraft architectures investigated in Case Study 3 and the metrics of the value proposition quantified with regard to this research question, fractionated spacecraft…

- *can be stronger or equal* than monolithic spacecraft with respect to 6 out of 8 (or 75% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be stronger or equal* than a comparable monolith:
  - System: Median Dynamic LCC and Payload Performance
  - Payload Module: Median Dynamic LCC, Payload Performance, Mass, and Propellant Usage
- *can be weaker* than monolithic spacecraft with respect to 3 out of 8 (or 38% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be weaker* than a comparable monolith:
  - System: Median Dynamic LCC, Mass, and Propellant Usage
  - Payload Module: none

5.2.4. Summary of Responses to the Research Questions

Table 5-5 summarizes the respective response to each research question in Section 5.2.1, 5.2.2, and 5.2.3. Following this is a succinct response to all three research questions.

Table 5-5. Research Questions: summary of responses with regard to the value proposition.

			Monolithic Spacecraft	Fractionated Spacecraft	
			Relative Strength		
		Value Proposition Metric	Stronger/Equal/Weaker		Context (Results)
System	1	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	2	Static LCC	Stronger	Weaker	Case Study 1
	3	Median Dynamic LCC	Stronger/Equal/Weaker	Weaker/Equal/Stronger	Case Study 1, 2, & 3
	4	Payload Performance	Equal	Equal	Case Study 1, 2, & 3
	5	Mass	Stronger	Weaker	Case Study 1, 2, & 3
	6	Propellant Usage	Stronger	Weaker	Case Study 1, 2, & 3
Payload Module	7	Mission Lifetime	Weaker/Equal	Stronger/Equal	Case Study 2 (Section 4.2.4/4.7)
	8	Median Dynamic LCC	Weaker	Stronger	Case Study 3
	9	Payload Performance	Equal	Equal	Case Study 3
	10	Mass	Weaker	Stronger	Case Study 3
	11	Propellant Usage	Weaker	Stronger	Case Study 3

Fractionated spacecraft can have a **stronger or equal** value proposition with regard to the specified metric

Fractionated spacecraft can have a **weaker** value proposition with regard to the specified metric

Given the specific context of the three case studies considered herein, Table 5-5 thereby demonstrates that based on the value proposition, fractionated spacecraft…

- *can be stronger or equal* than monolithic spacecraft with respect to 8 out of 11 (or 73% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be stronger or equal* than a comparable monolith:
  - System: Mission Lifetime, Median Dynamic LCC, and Payload Performance
  - Payload Module: Mission Lifetime, Median Dynamic LCC, Payload Performance, Mass, and Propellant Usage
- *can be weaker* than monolithic spacecraft with respect to 4 out of 12 (or 33% of the) value proposition metrics. The following are the value proposition metrics for which fractionated spacecraft *can be weaker* than a comparable monolith:
  - System: Static LCC, Median Dynamic LCC, Mass, and Propellant Usage
  - Payload Module: none

The synthesis of monolithic and fractionated spacecraft value propositions with regard to each research question provided in Section 5.2.1, 5.2.2, and 5.2.3 was made to be as objective (unbiased) as possible to maintain a fair comparison between monolithic and fractionated spacecraft. As such, the summarized responses to all three research questions, as is provided in Section 5.2.4 and Table 5-5, intentionally account for the value proposition *of every* fractionated and monolithic spacecraft considered in the three case studies. Subsequently, these responses do not preclude (hide) the fact that fractionated spacecraft do not always provide the strongest value proposition. Therefore, as a point of interest to fractionated spacecraft advocates, Appendix E adopts a perspective in comparing monolithic and fractionated spacecraft value propositions that answers the suppliant question: within a given context, what is the absolute best fractionated spacecraft can do in terms of each respective value proposition metric relative to a comparable monolith? Appendix E subsequently only considers the most competitive fractionated spacecraft relative to a comparable monolithic spacecraft, thereby ignoring the adverse implications of fractionation purposefully cited in Section 5.2 to maintain a holistic understanding of the implications of fractionation. Most succinctly put Appendix E supplies quantitative data, from which, fractionated spacecraft advocates may derive appreciable value.

6. Conclusion

The conclusion consists of four sections that provide closure to the research effort, reflection to encapsulate the lessons learned, and motivation for future fractionated spacecraft assessments. Section 6.1 revisits the unique research contributions discussed in Section 2.4 whereas Section 6.2 revisits the objectives of the research methodology discussed in Chapter 1. Section 6.3 then offers some important reflections based on this research effort with regard to (1) the Spacecraft Evaluation Tool, (2) the value proposition, (3) the role of shared resources, (4) further exploration of the value proposition, and (5) previous assessments of fractionation. Lastly, Section 6.4 provides closure to the assessment of monolithic and fractionated spacecraft performed through this research effort by eliciting a succinct response to what is commonly referred to as "the case for fractionation."

6.1. Research Contributions Revisited

The research contributions given in Section 2.4 are based on the limitations of previous assessments of fractionated spacecraft. These limitations are repeated below for convenience.

Limitations

1. Narrow scope of the value proposition
    a. Low fidelity models
        i. Parametric models or design-by-analogy
    b. Small number of fractionated spacecraft architectures investigated
2. Lack of dynamic lifecycle considerations
3. Minimal focus on cardinal measures of benefit and/or value

Based on these limitations, the research contributions were formulated to ensure this research effort and subsequent outcomes provide a meaningful contribution of knowledge pertaining to monolithic and fractionated spacecraft value propositions. These research contributions are repeated below for convenience.

Research Contributions

1. Provide a high fidelity, bottom-up, dynamic quantitative assessment of monolithic and fractionated spacecraft value propositions.
2. Enable an understanding of the monolithic and fractionated spacecraft value propositions using cardinal, "traditional" measures of effectiveness (MoE)
3. Provide the ability to explore monolithic and fractionated spacecraft value propositions in both breadth and depth.

These three unique research contributions are met through this research effort and its respective outcomes. The first research contribution is met through the development (Chapter 3) and subsequent application of the Spacecraft Evaluation Tool (SET) (Chapter 4). The second research contribution is met through the analysis and synthesis of the monolithic and fractionated spacecraft value propositions (Chapter 4 and 5 respectively), which are composed of quantitative, "traditional" cardinal measures of effectiveness. And the third research contribution is met through the capability to explore a wide range of value propositions using the SET (Chapter 3 and 4) and, additionally, by the breadth and depth of the monolithic and fractionated spacecraft value propositions presented in the analysis, and subsequently synthesized (Chapter 4 and 5 respectively).

6.1.1. Implications of achieving the Second Research Contribution

Many previous assessments of fractionation, especially in academia, have almost exclusively used (or at least relied heavily on) the metric of utility to quantify and compare the benefits of monolithic and fractionated spacecraft[23]. The utility metric is indeed useful in its ability to quantify the aggregate benefit of a spacecraft based on multiple attributes (*i.e.,* measures of effectiveness); however, it has some significant shortcomings, most notably, that it is an ordinal measure (see Section 2.3.3, specifically the third limitation and associated discussion). Subsequently, utility tends not to resonate well with beneficiary stakeholders who are used to assessing/evaluating systems (*e.g.,* monolithic and fractionated spacecraft) in the context of "traditional" cardinal measures of effectiveness (MoE) (*e.g.,* LCC and mass of a system). The prevalence of employing utility for assessments of fractionated spacecraft has undoubtedly been due to the desire to quantify multiple measures of benefit for a given spacecraft in a single value (metric). However utility may also have been ushered to the forefront of many (but not all, please note) previous assessments of fractionated spacecraft because of an inability to demonstrate that fractionated spacecraft are competitive with comparable monolithic spacecraft based on cardinal MoE. Subsequently a common outcome of many previous assessments of fractionation is, based on utility, fractionated spacecraft most often (if not always) provide more benefit than comparable monolithic spacecraft. However, in terms of cardinal MoE (*e.g.,* LCC, mass), fractionated spacecraft are never more desirable than comparable monolithic spacecraft.

The value propositions for monolithic and fractionated spacecraft generated from the SET purposefully do not incorporate utility due to its often-poor resonance with beneficiary stakeholders as well as only being able to discern a ranked ordering from utility values. Additionally, utility is not incorporated in the value propositions in this research to emphasize that the value propositions for fractionated spacecraft can be stronger (*i.e.,* better) than that of comparable monolithic spacecraft based on cardinal MoE; this is demonstrated in the analysis and synthesis (Chapter 4 and 5 respectively) and Appendix E. Specifically, in Chapter 4 and 5 and Appendix E, it is shown that of the 11 "traditional" cardinal MoE (*i.e.,* metrics) composing the value proposition considered in this research, up to 9 of 11 (or 82%) of the metrics (LCC being one of the eight) show fractionated spacecraft as being more desirable than a comparable monolithic spacecraft. These results have significant implications because they quantitatively substantiate that monolithic spacecraft are *not always* the best spacecraft architecture, in terms of the value proposition metrics considered in this research. This conclusion therefore not only provides a specific instantiation of attaining of the second research contribution, but also an instantiation that demonstrates that "traditional" cardinal MoE, including LCC, can be used to show that fractionated spacecraft are more desirable than comparable monolithic spacecraft (for pointing-intensive, RSM space missions). However, recognize that achieving the second research contribution does not diminish the worth of previous assessments of fractionation that employ utility, because utility is far more encapsulating of fractionated spacecraft benefits than a single metric (MoE) could ever hope to be. It is, however, hoped that the respective outcomes of this research, through which "traditional" cardinal MoE (including LCC) show fractionated spacecraft as being more desirable than comparable monolithic spacecraft, will *reenergize* investigations of fractionation on the basis of cardinal MoE. Our collective understanding of the implications of fractionated spacecraft for the current spacecraft paradigm could greatly benefit through future assessments of fractionation specifically corroborating (and/or refuting) the results from this research investigation.

[23] Using the word *utility* instead of *multiple attribute utility*, which is actually the correct terminology given that utility is nearly always derived from more than one attribute, is done for purposes of simplicity. Additionally, the using utility instead of multiple attribute utility does not detract from the message being conveyed.

Based on the discussion in Section 6.1.1 thus far, the suppliant question yet to be addressed is, why has this research effort proven fruitful in the area of demonstrating that fractionated spacecraft are competitive to comparable monolithic spacecraft based on cardinal MoE, specifically LCC? Without intimate knowledge of the models employed in other assessments of fractionation, no conclusive response to this question can be formulated; however, a few insights as to the response can be offered. As compared to the models developed and employed in other assessments of fractionation, the SET may (or may not) differ significantly in both its breadth and depth as a monolithic and fractionated spacecraft model. And, in addition, the SET has an expansive and flexible launch vehicle analysis and selection model (see Section 3.2.1) that may (or may) not be present in the fractionated spacecraft models employed in previous assessments. Therefore, in terms of a response to the fruitful nature of this research effort in demonstrating the benefits of fractionation based on "traditional" cardinal MoE, it is the working hypothesis that this is the cause of the following four attributes of the SET. (1) The SET quantifies of Static *and* Dynamic LCC and the Dynamic LCC, as is discussed in Section 3.3.3, accounts for potential LCC savings provided by fractionation relative to a comparable monolith. (2) The SET is readily able to examine an infinite number of fractionated spacecraft architectures, thereby having the potential to capture a broad scope of fractionated spacecraft value propositions. (3) The SET fidelity enables the accurate discrimination between the launch vehicle usage based on nuances in monolithic and fractionated spacecraft designs (see Section 3.2.1 and 3.3.2) not possible with low fidelity, predominantly (or fully) parametric spacecraft models. (4) And the SET launch vehicle analysis and selection model (see Section 3.2.1 and 3.3.2) allows for the selection of up 22 candidate launch vehicles for spacecraft deployments and replenishments. Additionally, the launch vehicle analysis and selection model fully accounts for payload fairing size and mass constraints as well as the important relationship between launch site latitudes and launch payload mass. In particular, the fidelity of the SET and the launch vehicle selection model are believed to be the key to discerning nuances between monolithic and fractionated spacecraft architectures, which enabled this research effort to demonstrate that fractionated spacecraft can have stronger value propositions than comparable monolithic spacecraft.

6.2. Research Methodology Objectives Revisited

The research methodology serves as the foundation for the development and progression of this research. Subsequently, there were objectives set forth specifically regarding the research methodology, which ensured that each stage of the research development and progression was a demonstration of scholarly achievement. The four the research methodology objectives are:

The research methodology must...

1. Formulate comprehensive, appropriate, and quantitative responses to the research questions (see Section 2.5).
2. Engender the unique contributions of this research (see Section 2.4).
3. Produce meaningful results reliably and repeatedly.
4. Be readily extendable to demonstrate broad applicability of the methodology to problems (*i*.e., questions) that were beyond the original scope of this research.

These four research methodology objectives are met through this research effort and its respective outcomes. The first and second research methodology objectives are met through the synthesis (Chapter 5) and achieving the research contributions (see Section 6.1), respectively. The third research methodology objective is met implicitly through the development and application of the SET (Chapter 3 and 4 respectively), because the inherent construction and V&V of the SET enabled results to be reliably and repeatedly generated for the analysis (Chapter 4). And the fourth research methodology objective is met

based on the discussion pertaining to the analysis (Chapter 4), through which questions appreciably different from the three research questions were answered – these questions are in fact the trends elicited in the analysis, which are simply reordered questions.

6.3. Reflections

Section 6.3 offers several reflections pertaining to the progression of this research over the past two years as well as the subsequent outcomes of the research documented herein. These reflections provide not only closure for this research effort but also serve as a catalyst for motivating future fractionated spacecraft assessments. The four five reflections pertain to (1) the Spacecraft Evaluation Tool, (2) the value proposition, (3) the role of shared resources, (4) further exploration of the value proposition, and (5) previous assessments of fractionation.

6.3.1. The Spacecraft Evaluation Tool

The Spacecraft Evaluation Tool (SET) development was guided directly by the research contributions, this thereby requiring that the SET be a high fidelity, almost exclusively non-parametric, automated spacecraft modeling tool. As such, it was necessary to devote a significant amount of time (the majority of the time allotted for this research effort) for the SET development and verification & validation. Subsequently, in both form and function, the SET achieved its intended objectives. In terms of form, the SET has a simple graphical user interface (GUI) where the inputs are easily specified and outputs are directly reported to an easy-to-read GUI (note, this is not shown in Chapter 3). And in terms of function, the SET is a high fidelity, predominantly non-parametric spacecraft modeling tool that reliably and repeatedly quantifies monolithic and fractionated spacecraft value propositions. Subsequently, the SET was found be capable of formulating appropriate responses to the research questions, the physical instantiation of this being the data produced by the SET and presented in the analysis in Chapter 4. However, in retrospect of the SET development and ensuing application, there are two important insights worth mentioning that, although seemingly obvious, were reinforced continuously throughout the development and application of the SET. These two insights therefore may be of benefit to future quantitative assessments of monolithic and fractionated spacecraft. These insights complement the insights provided in Section 3.6, specifically with regard to the SET limitations and its respective implications for the research contributions.

The first insight with respect to the SET has to do with (spacecraft) modeling scope (capability). Admittedly, the SET development went headfirst into the realm of high fidelity spacecraft modeling in order to achieve the first research contribution, that is, the creation of a high fidelity, predominantly non-parametric spacecraft model. Consequently, this resulted in a long development time for the SET, in particular because of the need to maintain a uniform level of depth across all constituents in the SET – a necessary requirement of any spacecraft model. Therefore, despite the SET performing better than expected in generating results/responses for the research questions, in hindsight the SET development and subsequent generation of results could have been more efficient (*i.e.,* less time consuming). Specifically, this could have been achieved by explicitly defining the scope of the SET from the start given the fidelity required to respond to the research questions. The scope of a given (spacecraft) model serves as an invaluable source of guidance for the model development as it controls the fidelity (consistency) of the model, which in turn, dictates how long it will take to develop the model and subsequently generate results from it. Therefore, the scope of a spacecraft model should be defined *before* the spacecraft model development commences in order to maximize the efficiency (*i.e.,* time spent) for the model development and its subsequent application; this being in contrast to a more ad hoc approach for a model's respective development based on knowing that the model must be, for example, "high fidelity." Ultimately, having an

ill-defined or nonexistent model scope will lead to inefficiencies in developing a model and generating results/outputs from it. Additionally, a lack of model scope can create the potential for the respective capabilities of a spacecraft model to be (appreciably) beyond what is required to adequately and appropriately address the questions of interest (*e.g.,* research questions).

The second insight with regard to the SET pertains to biases, or lack thereof, of the model developer and subsequently the comparison of monolithic and fractionated spacecraft value propositions. For a given spacecraft model, decisions with regard to every constituent of the model need to be made (the higher the model fidelity, the more decisions to be made). Subsequently, the implications of these decisions on the outputs/results of the model can be (and often are) significant given the intricacies and dependencies of a spacecraft model's respective constituents. This therefore enumerates the importance of model developer biases, as the decisions made about the model constituents, given these biases, will absolutely dictate whether fractionation is unfairly penalized, overly favored, or balanced relative to a comparable monolith. And this in turn has profound implications for the relative value propositions of monolithic and fractionated spacecraft. It is therefore necessary to understand, articulate, and ultimately maintain, as best as possible, the biases (or lack thereof) when developing a spacecraft modeling tool so that the model outcomes can be understood relative to that particular perspective (*i.e.,* bias) adopted.

In response to the second insight, throughout the SET development, decisions made with regard to the modeling of monolithic and fractionated spacecraft were as unbiased as possible, subsequently forming an appreciably equal basis for the comparison of monolithic and fractionated spacecraft. As such, during the SET development, for example, the hardware associated with all respective subsystems was kept the same for monolithic and fractionated spacecraft. In addition, inter-module dependencies were accounted for as a factor in dictating the number of lifecycle replenishments for fractionated spacecraft, rather than assuming that fractionated spacecraft fail as many times as a comparable monolith. However, in contrast, in the hypothetical case in which the SET was biased towards fractionated spacecraft, the SET development could have easily employed, for example, higher efficiency solar cells on fractionated spacecraft than on monolithic spacecraft, thereby unfairly penalizing monolithic spacecraft in terms of the power generation subsystem mass. Additionally, the SET could have not accounted for the inter-module dependencies in fractionated spacecraft for determining their respective number of replenishments, thereby "leveling the playing field" with comparable monoliths in this regard. The point of illustrating how these biases could have been manifested in the SET is to emphasize two conclusions. First, there is no correct bias to have when modeling monolithic and fractionated spacecraft, as each bias provides a unique perspective with regard to monolithic and fractionated spacecraft value propositions and is accurate in its own right. And second, model biases can have significant implications for monolithic and fractionated spacecraft value propositions, and therefore if biases exist they must be strictly observed during a model development and subsequently accounted for in the discussion of outputs from the model.

6.3.2. The Value Proposition

With regard to forming monolithic and fractionated spacecraft value propositions, as was enumerated at the end of Section 6.3.1, the SET was developed to be as unbiased (neutral) toward monolithic and fractionated spacecraft as possible to maintain a fair assessment. Subsequently, the value propositions generated from the SET are impartial towards monolithic and fractionated spacecraft. The unbiased nature of the value propositions for monolithic and fractionated spacecraft is the subject of the first point of reflection with regard to the value proposition. As alluded to in Section 6.3.1, the value proposition can readily be manipulated by a spacecraft model developer to unfairly advantage either monolithic or fractionated spacecraft. It is therefore important to reemphasize the unbiased nature of the monolithic and fractionated spacecraft value propositions provided and discussed in the analysis and synthesis (Chapter 4 and 5 respectively). These value propositions therefore provide a necessary common foundation for accurately understanding the comparative merits and limitations of monolithic and fractionated spacecraft.

In terms of the value proposition LCC metrics, an important observation was elicited in Section 3.3.3 and Section 5.1.2 (and in Appendix B) stating that both Static *and* Dynamic LCC should be used as complements in the value proposition. This observation is instantiated in the analysis (Chapter 4) through which the Static LCC fails to capture the majority of the benefits and costs associated with monolithic and fractionated spacecraft given their naturally stochastic lifecycles. Therefore, given that many previous assessments of fractionation only quantify the Static LCC, the respective outcomes of this research provide a meaningful contribution for understanding monolithic and fractionated spacecraft value propositions with regard to Dynamic LCC (relative to Static LCC).

In Section 2.1.10 it was stated/hypothesized that if parametric cost models are used to estimate the cost of a spacecraft during the early stages of its respective design (*e.g.,* conceptual), it may be inappropriate to quantify the uncertainty in a cost estimate for the spacecraft due to uncertainty in the parametric cost model. The discussion in Section 2.1.10 pertaining to this hypothesis highlights that parametric cost model uncertainty fails to quantify the critical uncertainty associated with forming cost estimates of a spacecraft during the early stages of its design. Subsequently based on this hypothesis, the relevant question is, how appropriate is it to quantify the cost uncertainty associated with a spacecraft cost estimate if the design of the spacecraft is hardly certain? There is no correct response to this question, however it should be recognized that if uncertainty in parametric cost models is used to quantify uncertainty in cost estimates of spacecraft made during the early stages of its design (*e.g.,* conceptual design phase), the cost uncertainty associated with the estimate represents an absolute lower bound.

And the last point of reflection with regard to the value propositions for monolithic and fractionated spacecraft concerns itself with launch vehicles. As is enumerated in Section 5.1.1 and through the analysis in Chapter 4, launch vehicles can yield Dynamic LCC savings for fractionated spacecraft, so much so that fractionated spacecraft can sometimes have a lesser Dynamic LCC than a comparable monolith. Specifically, this is because fractionated spacecraft are sometimes able to use multiple, smaller and less expensive launch vehicles as compared the launch vehicle employed by a comparable monolith. Subsequently, this can have the effect of appreciably reducing the launch and thus replenishment costs (which drive the Dynamic LCC) of fractionated spacecraft relative to that of a comparable monolith. Ultimately, this is *the reason* as to why certain fractionated spacecraft can be Static and Dynamic LCC-competitive to comparable monoliths, despite their higher masses. Therefore, based on this reasoning, as fractionated spacecraft have access to smaller, more reliable, and less expensive launch vehicles, this will further increase their LCC-competitiveness relative to a comparable monolithic spacecraft.

6.3.3. The Role of Shared Resources

Employing shared resources in a fractionated spacecraft changes the mass and size distribution across its respective modules. The outcomes of this research have shown that this change in mass and size distribution has both advantages and disadvantages for fractionated spacecraft. The advantages of sharing resources can include a potential reduction in LCC and an increase in mission lifetime capability of fractionated spacecraft relative to that of a comparable monolithic spacecraft. And the disadvantages of sharing resources can include a consistently larger mass and often LCC of fractionated spacecraft relative to that of a comparable monolithic spacecraft. The outcomes of this research therefore provide quantitative instantiations supporting the conclusion that the optimal use of shared resources in a fractionated spacecraft, in terms of minimizing and maximizing LCC and mission lifetime capability respectively, is to share the Comm_CS_C&DH and/or ADS_GNS subsystem resources (but not Power generation and storage). In the results produced for this research, it was discovered that sharing the Power generation and storage resource incurs too large a mass and size (and hence LCC) penalty as compared to having fractionated spacecraft modules produce and store all of their own power. Therefore, one noteworthy insight gained through this research with regard to shared resources is that shared resources are beneficial for fractionated spacecraft in terms of their respective value propositions, but employing shared resources reaches a point of diminishing returns. This insight thereby emphasizes the significance of the manner in which shared resources are modeled/incorporated in fractionated spacecraft.

6.3.4. Further Exploration of the Value Proposition

One of the trends observed in fractionated spacecraft architectures was that as the number of modules increases, the number of inter-module dependencies increases and this ultimately has the effect of increasing the aggregate number and cost of replenishments. Therefore, given the significance of the number and cost of replenishments on the Dynamic LCC of a spacecraft, it would be of benefit to investigate fractionated spacecraft with a lesser number of inter-module dependencies. Specifically, this could be accomplished by increasing the system-wide redundancy in a fractionated spacecraft such that all Infrastructure Modules no longer share *any* resources; hence, for a given three and four-module fractionated spacecraft architecture considered in Case Study 1, 2, and 3, all Infrastructure Modules will be identical to Module 1[24]. The working hypothesis with regard to this suggestion is that fractionated spacecraft with fully redundant and identical Infrastructure Modules will have a lesser Dynamic LCC than a fractionated spacecraft will non-redundant Infrastructure Modules as are depicted in Figure 4-3 through Figure 4-7[25]. If this hypothesis is true then by the introduction of these new three and four-module fractionated spacecraft architectures having fully redundant Infrastructure Modules, the value proposition competiveness, specifically with respect to Dynamic LCC, for the three and four-module fractionated spacecraft classes may increase appreciably.

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[24] Note that since two-module fractionated spacecraft only have one Infrastructure Module, this suggestion is not applicable to two-module fractionated spacecraft.

[25] Preliminary tests have quantified this hypothesis as to an increase in system-wide redundancy yielding a lesser Dynamic LCC despite a higher system mass. Specifically, SET simulations were run to test this hypothesis for Spacecraft Architectures 9 and 10 (see Section 4.1.3). The Median Dynamic LCC corresponding to Spacecraft Architecture 9 and 10 as they are in Case Study 1 (Modules 1 and 3 are not identical) was compared to their respective Median Dynamic LCC if Modules 1 and 3 are identical (*i.e.,* the hypothesis suggestion). This comparison showed that the hypothesis is correct for Spacecraft Architecture 9 and 10 because when system-wide redundancy increases (*i.e.,* Module 1 and 3 are identical), the Median Dynamic LCC for Spacecraft Architecture 9 and 10 decreases by 115.94 and 114.03 (FY2008$M) respectively. Therefore, the results of this test are encouraging in terms of reducing the Median Dynamic LCC disparity even further between three and four-module fractionated spacecraft and a comparable monolith by having redundant infrastructure modules.

For the fractionated spacecraft considered in the analysis in Chapter 4, regardless of the number of modules, the Power shared resource (when employed) had a dominating effect on a fractionated spacecraft, especially for high resolution spacecraft with large (approximately 1 km) inter-module separation distances. However, this dominating effect is entirely dependent on the Wireless Power Distribution (WPD) system employed in fractionated spacecraft and its respective hardware (see Section 2.1.5 and 4.1.3). Therefore, one interesting and fruitful investigation would be to explore the implications of a new WPD system for fractionated spacecraft to employ. The WPD system modeled in the SET was a Laser Power Beaming WPD system but three other options exist: (1) radio and microwave power transmission, (2) electric conduction, and (3) concentrated, reflected sunlight (see Section 2.1.5). Before the SET development, a quantitative investigation of WPD systems for fractionated spacecraft took place to assess the four WPD system options. Subsequently, laser power beaming was selected as the "best" WPD system option, and the only other feasible WPD system was determined to be a concentrated, reflected sunlight WPD system[26]. (Radio and Microwave Power Beaming can wirelessly stream enough power but the ratio of power transmitted-to-received scales with the inverse of separation distance squared; and the proposed power source required for electric conduction is far too significant to be incorporated in fractionated spacecraft). Therefore, to change the value proposition for fractionated spacecraft, for better or worse, a concentrated, reflected sunlight WPD system can be used when employing the Power shared resource rather than a laser power beaming WPD system. Subsequently, using a different WPD system will have significant implications for the relative differences between comparable monolithic and fractionated spacecraft value propositions, not enumerated in the analysis in Chapter 4.

And the last thought with regard to expanding the value proposition, is that the fractionated spacecraft architectures considered in the analysis in Chapter 4 could easily have been transformed into optical space interferometers by having RSM payloads on every module. A fractionated spacecraft interferometer combines the electromagnetic waves captured by each RSM payload on its respective modules, and subsequently forms a single image of an object in space. The advantage of using interferometers is manifested in the images they create as they can have significantly more resolution than an image created by a monolithic spacecraft with a single RSM payload (Makins, 2002; Steel, 1967). Given the physics of optical interferometry, fractionated spacecraft could prove to be beneficial in two ways. First, the RSM payloads on each module in a fractionated spacecraft can be smaller than the RSM payload on a comparable monolith to achieve the same image resolution as the monolith. And second, the RSM payloads on each module in a fractionated spacecraft can be kept the same size as the RSM payload on the monolith and thereby achieve a far greater image resolution than can be attained by the monolith. Therefore, with respect to the value proposition and specifically payload performance, the insights gained from an investigation into the implications of fractionation for optical space interferometers would be rewarding.

6.3.5. Previous Assessments of Fractionation

Each previous assessment of fractionation (see Section 2.3) offers a unique and valuable contribution of knowledge towards understanding fractionated spacecraft. Therefore, identifying, discussing, and citing the limitations of these previous assessments was done only out of the need to develop a rational and structured path to ensure that this research effort did not progress in vain but rather made new and meaningful contributions to our shared knowledge of fractionated spacecraft. Therefore, the insights gained from this research effort and all previous assessments of fractionation should be appropriately treated as complements that collectively achieve the common goal of understanding fractionated spacecraft.

[26] Alternatively, the lack of options beyond the concentrated, reflected sunlight WPD system could serve as a source of motivation for the research and development of a new WPD system for fractionated spacecraft power sharing.

6.4. Is there a "Case for Fractionation"?

The "case for fractionation" is of interest to advocates and proponents of fractionated spacecraft alike. The case for fractionation is, in essence, a particular context through which it can be demonstrated that fractionated spacecraft are in some capacity "better," "equal," or "worse" than (comparable) monolithic spacecraft. Subsequently, one pertinent question with regard to making a "case for fractionation" is whether or not there is a unifying (*i.e.,* all encompassing and consistent) case for fractionation.

As can be ascertained from the assessment of monolithic and fractionated spacecraft value propositions performed through this research, there is a plentiful supply of cases for fractionation that demonstrate fractionated spacecraft are being "better", "equal", or "worse" than monolithic spacecraft. Therefore, the consistent conclusion formulated through this research effort with regard to a unifying case for fractionation is that there is no unifying case. Additionally, based on the respective outcomes of this research, it is likely that as the scope (context) in which monolithic and fractionated spacecraft are assessed broadens, the probability of attaining a unifying case for fractionated will decrease.

Therefore with regard to the "case for fractionation", based on this research effort, it is concluded that rather than a unifying case for fractionation existing, there is in fact a copious supply of cases for fractionation that exist, which collectively justify *and* refute the suitability of fractionated spacecraft in the current spacecraft paradigm. This conclusion thereby emphasizes an invaluable lesson learned through the progression of this research, namely, the significance of the perspectives subjectively chosen to compare monolithic and fractionated spacecraft.

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Appendix A Implementation Challenges for Fractionation

The negative hypotheses about fractionation (see Section 2.2.1) can be thought of as implementation challenges for fractionated spacecraft. The implementation challenges can be categorized as belonging to one of four classifications: (1) technological, (2) managerial, (3) economic, and (4) political. In addition, the implementation challenges can be organized relative to various phases of a spacecraft development cycle: (5) Research and Development; (6) IA&T and Production; (7) Launch and Operations; (8) Market; (9) Performance and Perceived Value Proposition; and (10) Acquisition. Each of these ten organizational domains is enumerated below.

Classifications

1. Technological: technology development for fractionated spacecraft.
2. Managerial: top-down program planning and current program/management paradigms for (fractionated) spacecraft development.
3. Economic: the nature of fractionated spacecraft benefits and costs as well as spacecraft developers' cost-benefit perspective of (fractionated) spacecraft.
4. Political: the current nature of (fractionated) spacecraft acquisition and the implications of this for resource allocation for (fractionated) spacecraft development.

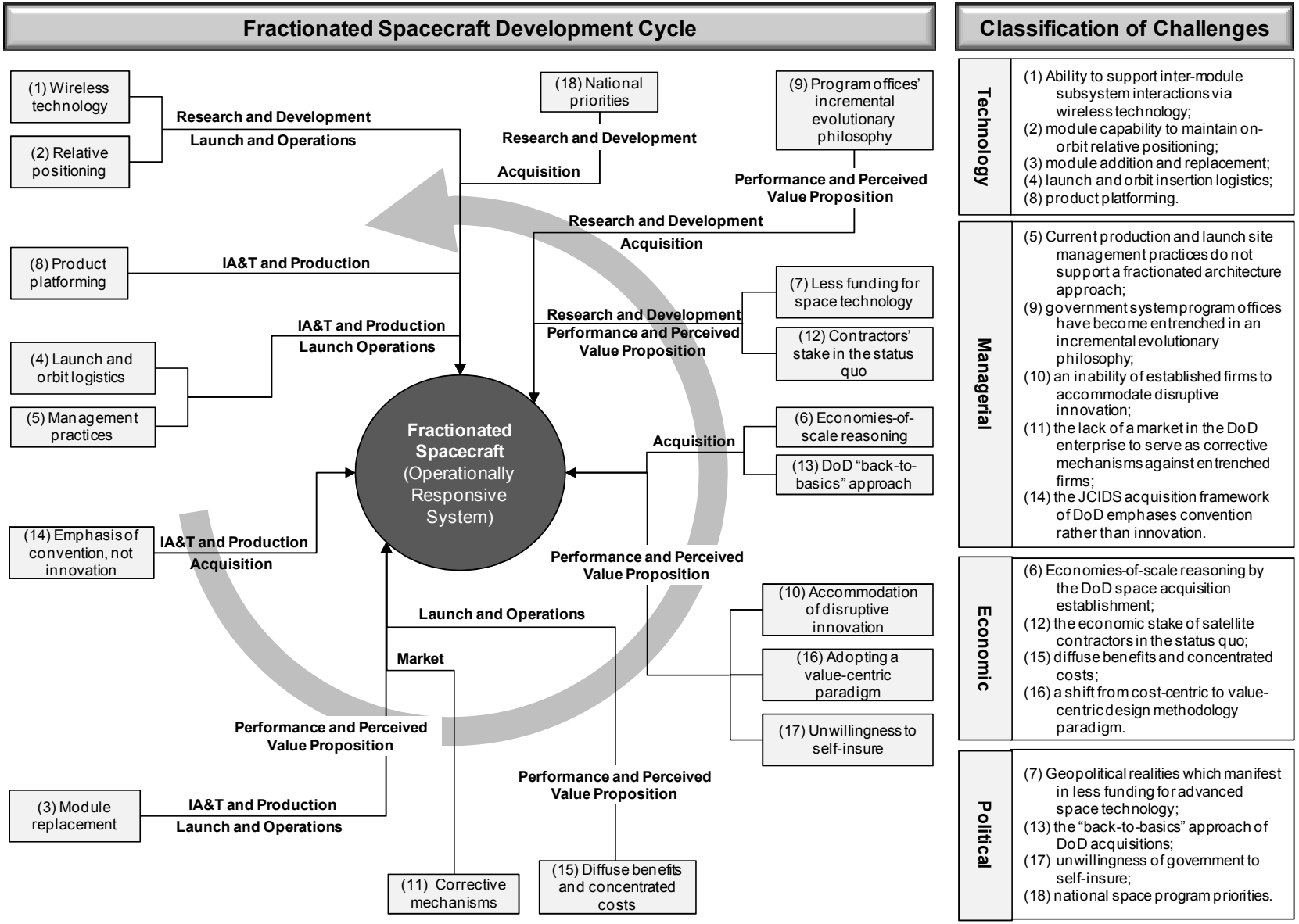
Spacecraft Development Cycle Phases

5. Research and Development: activities regarding the research of key technologies for fractionated spacecraft and their subsequent development for testing and manufacture.
6. IA&T and Production: activities regarding the IA&T of a fractionated spacecraft and, concurrently with this, the full-scale production of a flight-ready fractionated spacecraft.
7. Launch and Operations: activities regarding the deployment (*i.e.,* launch) of a fractionated spacecraft and its subsequent operation for the effective operational mission lifetime.
8. Market: activities regarding the dependency of the market (*i.e.,* customers for the services provided by a fractionated spacecraft).
9. Performance and Perceived Value Proposition: activities regarding the performance (*i.e.,* value delivery) of fractionated spacecraft and the manner in which the market perceives this value.
10. Acquisition: activities regarding the procurement of fractionated spacecraft in terms of developing and funding a program for fractionated spacecraft.

Organizing the implementation challenges along these ten dimensions provides the necessary context for understanding how to allocate resources to overcome each of these challenges (barriers-to-entry) for fractionated spacecraft – this may be of particular importance to future fractionated spacecraft developers. For example, if an implementation challenge is classified as a technological and it is part of the Research and Development (R&D) phase of the spacecraft development cycle, it narrows the scope for understanding the origin of the challenge and subsequently how it can be overcome. In this example, the challenge could thus be overcome by allocating more resources in the R&D phase, specifically for technology development.

In Figure A-1 numerous implementation challenges for fractionated spacecraft are presented, which are derived from the negative hypotheses about fractionated spacecraft (see Section 2.2.1). Using Figure A-1, one can identify a challenge's respective classification as well as where it influences the spacecraft development cycle. Figure A-1 therefore provides a more transparent view of how to respond to the negative hypotheses about fractionated spacecraft.

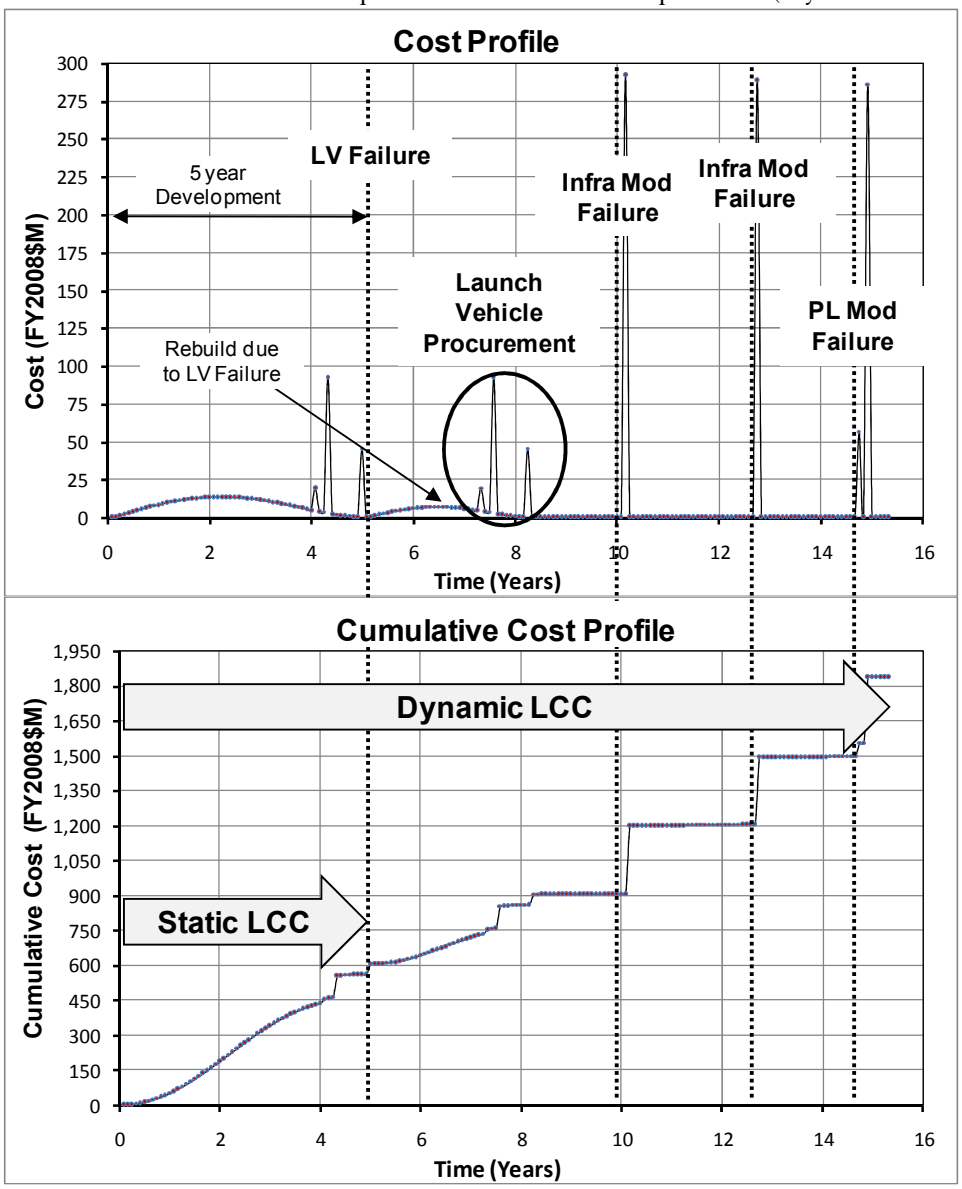
Figure A-1. Implementation challenges for fractionated spacecraft.



Appendix B Static and Dynamic Lifecycle Cost

Figure B-1 provides a representative example of the cost and cumulative cost profile for a two-module fractionated spacecraft corresponding to one MCA trial. The cost profiles in Figure B-1 represent a case in which spacecraft have a very severe (harsh) lifecycle, that is, during the lifecycle there is a high probability that risks resulting from lifecycle uncertainties will occur at any given time. This severe lifecycle, although unrealistic, is used for the two-module fractionated spacecraft under consideration in order to increase the frequency of on-orbit and launch vehicle failures. Subsequently, this serves the purpose of better demonstrating potential variations in the cost and cumulative cost profile for a (fractionated) spacecraft as well as more clearly illustrating the difference between Static and Dynamic LCC.

Figure B-1. Cost and cumulative cost profile for a fractionated spacecraft (7-year mission lifetime).



The following paragraph describes the reasoning as to the form of the cost profiles seen in Figure B-1. **(1)** In Figure B-1, the time period from 0-5 years is the design, development, manufacture, and initial deployment of the spacecraft. For this time period, a NASA-standard Beta curve is used for the time-dispersion of costs. **(2)** Then at 5 years when the spacecraft is being launched for the first time, the launch vehicle fails and therefore the spacecraft must be rebuilt, which takes 3.25 years (this is one of the SET inputs, see Section 3.2 'Build Time Learning Factor'). **(3)** Then at 8.25 years, the spacecraft is launched again, this time successfully. **(4)** The spacecraft continues to operate in space until about 10 years on the time axis (hence the spacecraft has only been in space for 1.75 out of the 7 years intended) when the fractionated spacecraft's Infrastructure Modules fails. The Payload Module also fails when the Infrastructure Module fails since, for the fractionated spacecraft under consideration, the Payload Module is dependent on the Infrastructure Module due to its reliance on shared resources. **(5)** The rebuild time of 3.25 years following this failure is suppressed Figure B-1 but the costs of having to build and launch a spacecraft again are fully accounted for - this is why there is a big spike in the cost profile. **(6)** Then after the 3.25 years it takes to rebuild the spacecraft pass, the spacecraft operates without failure for until about 12.75 years on the time axis (hence being in space for a total 4.5 out of the 7 years intended) before the Infrastructure Module failing and subsequently the Payload Module fail again. **(7)** After the spacecraft is rebuilt once again over a period of 3.25 years (hence another spike is apparent in the cost profile), it is the spacecraft is launched again. **(8)** The spacecraft is operated in space for another 2 years (hence the spacecraft has been in space for a total of 6.5 out of the 7 years intended) until the Payload Module fails. **(9)** The Payload Module is rebuilt in 3.25 years and then launched only to see the Infrastructure Module fail directly thereafter. **(10)** And ultimately, after the spacecraft is rebuilt a fourth time (which takes another 3.25 years), it carries out the remaining 0.5 years of the mission without failure.

The vital lesson to be drawn from the cost and cumulative cost profile presented in Figure B-1 is that the *Static* LCC accounts for two items. First, the cost incurred up to the first five years, and second, the cost of operation for the rest of the 7-year mission lifetime (this is equal to the sum of the cost as is shown by the zero-slope line at the bottom of the cost profile plot). Therefore, the Static LCC does not capture any of the costs associated with the stochastic nature of a spacecraft's respective lifecycle (*e.g.,* on-orbit and launch vehicle failure). In contrast, the Dynamic LCC grows through time, accounting for not only the cost elements quantified by the Static LCC but also all on-orbit and launch vehicle failures and their respective costs. In this light, Dynamic LCC is a more appropriate measure of LCC than Static LCC is for spacecraft with stochastic lifecycles[27]. To accentuate the difference between Static and Dynamic LCC, consider the cost profile in Figure B-1. Here the Static and Dynamic LCC is about 600 $M and 1,840 $M, respectively - therefore the Static LCC is less than the Dynamic LCC by 1,240 $M or 67 %, a significant difference.

Understandably, the Dynamic LCC and its respective appropriateness as a quantitative estimate of LCC depends on the stochastic spacecraft lifecycle model employed to capture lifecycle uncertainties (see Appendix C). Therefore, it is not the intent herein to suggest that the Static LCC is less accurate or statistically meaningful than the dynamic LCC as they are both accurate relative to the perspective of a spacecraft lifecycle they each adopt. Rather, the suggestion is that the Static *and* Dynamic LCC should be used together as complements – thus providing a more holistic understanding of the LCC of a spacecraft. As such, both the Static and Dynamic LCC of monolithic and fractionated spacecraft are computed and analyzed via the SET and subsequently used for comparisons.

[27] In comparing static to Dynamic LCC one should note that Dynamic LCC is *more appropriate* than Static LCC due to Dynamic LCC accounting for lifecycle uncertainties. This, however, does not insinuate differences in accuracy between Static and Dynamic LCC because they are each accurate relative to the perspective of a spacecraft's lifecycle that they adopt.

A notable question with regard to the LCC profiles shown in Figure B-1 is whether it is worth replacing a module or set of modules if they fail close to the end of the mission lifetime. Alternatively stated, is it more LCC-effective to not replace a module or set of modules that fail near the end of the mission lifetime? Figure B-1 provides a concrete example of the dilemma enumerated by this question, as there are two failures within roughly 0.5 years of completing the intended 7 year mission lifetime. Given this situation, one might decide that a 6.5 mission lifetime is acceptable and thus they do not need to replace the modules, and in doing so, reduce the LCC appreciably by not having to rebuild the spacecraft a fourth time.

In response to this dilemma, the SET assumes that such a situation, as is shown in Figure B-1 can occur; subsequently there is no "logic gate" that determines whether a spacecraft/module should be rebuilt based on proximity to the end of the mission lifetime. Based on the PoIM value and the risks created from launch, technical, environmental, and operational lifecycle uncertainties; the objective of the SET is to enumerate all possible dynamic lifecycle scenarios a spacecraft may experience via thousands of MCA trials. In doing so, the SET thereby provides a holistic spectrum of potential Dynamic LCC values for a spacecraft – some of which that may be instantiations of the dilemma mentioned previously and shown in Figure B-1, but also many that correspond to dynamic lifecycles in which none of the modules fail appreciably close to the end of the mission lifetime.

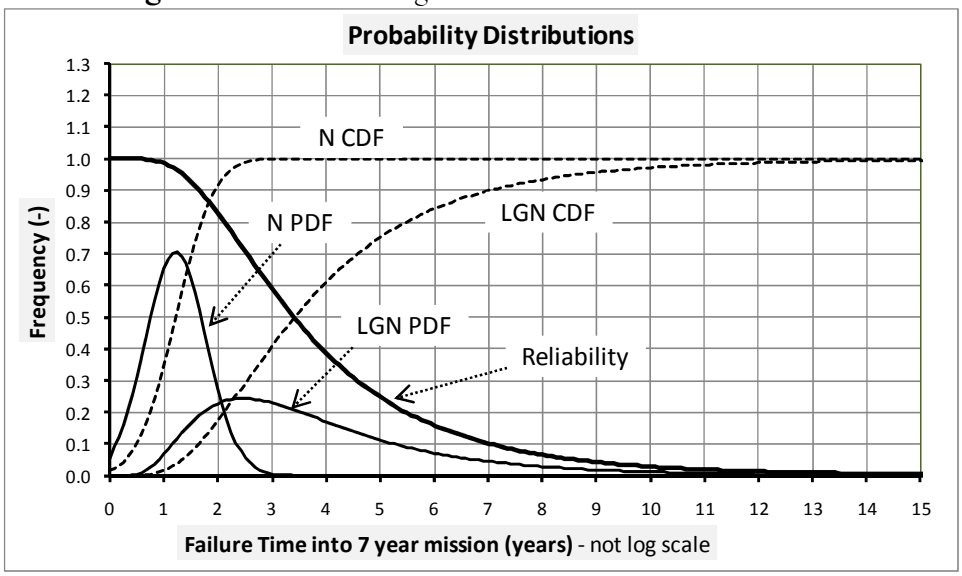
In terms of the value proposition for a given spacecraft, it is most important to appropriately quantify the measure of central tendency with regard to Dynamic LCC based on the MCA trials employed *for a given* (1) mission lifetime, (2) PoIM, and (3) lifecycle uncertainties. As such, incorporating a logic gate in the SET that would prematurely end mission lifetimes, based on reducing LCC by not choosing to replace modules that fail near the end of the mission, would provide an inappropriate measure of Dynamic LCC central tendency. This is because the logic gate enables the mission lifetime to vary for each MCA and consequently the Dynamic LCC values corresponding to each MCA can no longer be compared or aggregated based on a uniform mission lifetime – something of value to beneficiary stakeholders. Ultimately, incorporating a logic gate in the SET that ends missions prematurely diminishes the value of specifying a specific mission lifetime in the first place for which to compare the value propositions for spacecraft. Based on this reasoning, the SET chooses not ignore the dilemma's such as those illustrated in Figure B-1 in which appreciable LCC savings can be had by not rebuilding a spacecraft when it fails close to end of the mission lifetime. Subsequently, the SET provides the most unbiased, comprehensive understanding of potential Dynamic LCC values for monolithic and fractionated spacecraft *for a given mission lifetime*.

Appendix C The SET and Lifecycle Uncertainties

For the SET development, there were eight potential lifecycle uncertainties considered, but ultimately in the SET only four of eight were incorporated into the stochastic modeling of spacecraft and subsequent quantification of Dynamic LCC (see Section 2.2.2). The four lifecycle uncertainties considered are launch, technical, environmental, and operational. The combined implication of the technical, environmental, and operational uncertainties is an aggregate risk of on-orbit failure, and the implication of launch uncertainties is a risk of launch vehicle failure. The risk of on-orbit and launch vehicle failure due to technical, environmental, operational, and launch lifecycle uncertainties is not modeled as a constant across all MCA trials or even within one MCA trial. Therefore, the risks randomly occur and thereby more appropriately represent the true nature of a stochastic spacecraft lifecycles. As such, each of the risks resulting from the four lifecycle uncertainties considered has a unique probability of occurrence at a given point in time in the simulation of a spacecraft's lifecycle, as is embodied one MCA trial. This probability of occurrence is determined from a statistically meaningful, probability density function (pdf) representing the population of all occurrences of that risk relative to the spacecraft under consideration.

Risk of on-orbit failures due to technical, environmental, operational lifecycle uncertainties during a spacecraft's respective lifecycle are incorporated and subsequently modeled in the SET using a lognormal distribution pdf. From this lognormal pdf, a random number generator is used to select failure times for a given spacecraft/module such that the resulting distribution of failure times over a spacecraft's respective lifecycle forms the lognormal distribution. The form of the lognormal distribution can be readily changed by the SET inputs, specifically by the PoIM and mission lifetime. The PoIM and mission lifetime dictate both the mean and variance of the lognormal distribution (this is implicit in the SET model) and as such each PoIM and mission lifetime will create a different lognormal distribution. And as the PoIM and mission lifetime change in value, the lognormal distribution changes to reflect the change in PoIM and mission lifetime value via the on-orbit failure times during a given spacecraft's lifecycle (if any). Therefore, through the PoIM and mission lifetime, one has the freedom to control/change the risk of on-orbit failures (*i.e.,* occurrences) as desired. A representative lognormal distribution is given in Figure C-1 (also included is the lognormal cdf, normal pdf and cdf, and reliability distributions corresponding to the lognormal pdf).

Figure C-1. Notional lognormal distribution of failure times.



Due to the time-expensive nature of quantifying the risk of on-orbit failure due technical, environmental, operational lifecycle uncertainties from the ground up (*i.e.,* using a conditional probability based method), a lognormal distribution corresponding to a given PoIM and mission lifetime was selected to form the basis of determining the failure times for spacecraft and modules[28]. For the conceptual design of (fractionated) spacecraft, lognormal distributions are appropriate and subsequently used in industry because the distributions represent the pdf of a risk (*e.g.,* on-orbit failure) in the case in which the risk is a multiplicative aggregate of numerous, assumed independent and randomly occurring risks/failures. Although the assumption made with a lognormal distribution that the individual risks contributing to the aggregate risk are independent does not hold true for actual failures of spacecraft, this assumption can be logical reasoned for the *conceptual* modeling of spacecraft. Therefore, this assumption is made for the purposes of this research because without making this assumption, modeling the technical, environmental, operational lifecycle uncertainties for fractionated spacecraft would not have been within the scope of this work.

In contrast to on-orbit failures, launch vehicle (LV) failures are modeled as a Bernoulli Trial Sequence (BTS) using each LVs respective failure rate, which is known from the launch vehicle database in the SET (see Section 3.2.1). For a given launch vehicle, failures are assumed independent, and although this is not entirely accurate, it is an assumption that has to be made if intimate knowledge about launch vehicle failures is unknown. Therefore, for a given launch vehicle (or set of launch vehicles), launch vehicle failures can be modeled as the "flip of a coin (or set of coins)" which is (are) weighted according to the launch history of that (set of) launch vehicle(s). For a given launch vehicle (or set of launch vehicles), a random number generator is used to determine when a launch vehicle fails and succeeds based on its respective BTS distribution.

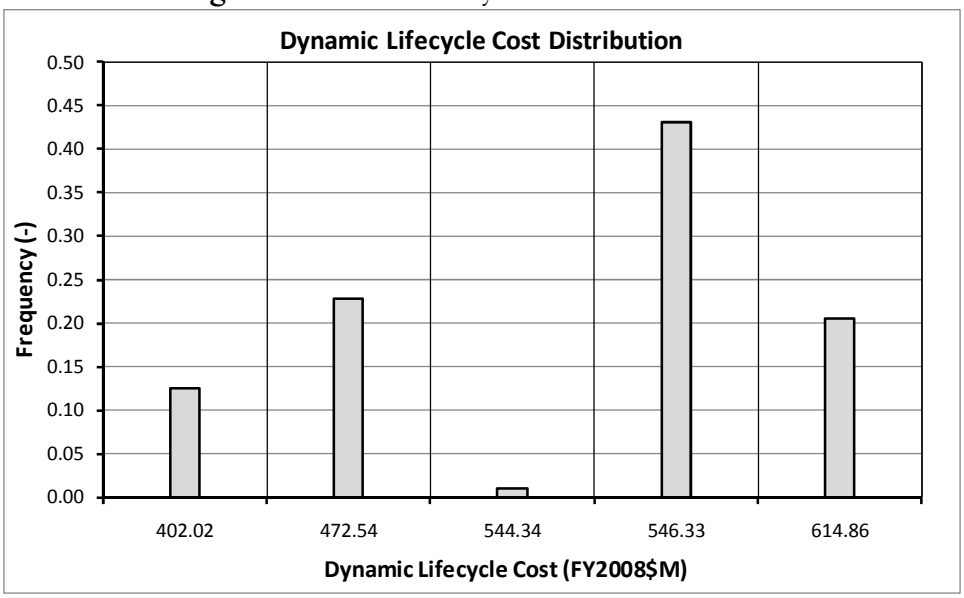
The objective of briefly introducing the methods for operationalizing the lifecycle uncertainties in the SET in Appendix C is to facilitate an understanding of *one option* for modeling the random nature of the risks resulting from launch, technical, environmental, operational lifecycle uncertainties. The risks for a given lifecycle uncertainty, as modeled in the SET, do not pose the same threat at all times during a spacecraft's lifecycle, and as such, more appropriately mimic the actual behavior risks for real spacecraft. And additionally, one should note that the randomness of the occurrence of risks due to lifecycle uncertainties as modeled in the SET is not "blind" but rather rooted in a statistically meaningful sense by using random number generators concurrently with statistical meaningful continuous and discrete distributions. Lastly, given the uncertain nature of how launch, technical, environmental, and operational uncertainties affect spacecraft, the risks that these uncertainties impose on a spacecraft are purposefully kept as input variables to the SET so that as a further understanding of the influence of these uncertainties is gained, it can be readily and accurately reflected in the SET.

[28] An alternative approach for modeling spacecraft reliability is to employ a Kaplan-Meier (KM) estimator for calculating reliability as a function of mission lifetime (Castet & Saleh, 2009), albeit this is not equivalent in functionality to the approach employed herein. The KM estimator relies on data for historical spacecraft failure rates.

Appendix D LCC Distributions and MCA Implications

Due to the manner in which the Dynamic LCC is quantified for a given spacecraft in each MCA trial (see Appendix B and C), the Dynamic LCC value is not constant across all MCA trials. Hence, the Dynamic LCC across *n* MCA trials will create a population of Dynamic LCC values that can be represented as a statistical distribution – this is the subject of Section 2.1.10. The Dynamic LCC distribution raises two questions. First, given the distribution of Dynamic LCC values, what should be my measure of central tendency and confidence in that measure of central tendency? And second, how many MCA trials should be used to ensure that I have appropriately captured the Dynamic LCC distribution? To facilitate this discussion, consider the Dynamic LCC distributions generated from the SET in Figure 4-31, Figure 4-32, and Figure D-1. These three Dynamic LCC distributions are multimodal distributions (MMDs) and representative of the Dynamic LCC MMDs generated by the SET for monolithic and fractionated spacecraft.

Figure D-1. Notional Dynamic LCC distribution.



The first question, given the distribution of Dynamic LCC values, what should be my measure of central tendency and confidence in that measure of central tendency, is addressed in full in Section 2.1.10.

The second question, how many MCA trials should be used to ensure that I have appropriately captured the Dynamic LCC distribution, seeks to understand how well the Dynamic LCC distribution produced by the SET MCA (see Figure 4-31, Figure 4-32, and Figure D-1 for examples) characterizes all possible Dynamic LCC values for the spacecraft. The response to this question is important, as it is desirable to use the most appropriate/representative Dynamic LCC distribution for spacecraft to inform the value proposition. The solution to the dilemma enumerated by this question is to determine the minimum number of MCA trials needed in the assessment of a spacecraft's lifecycle to appropriately represent the Dynamic LCC distribution. Intuitively, it is desirable to employ the minimum number of trials in a MCA since as the number of MCA trials used increases, the computational time increases. Thus, a sensitivity study was conducted to determine this minimum number of MCA trials. The study specifically considered three different spacecraft architectures: a monolith, a two-module fractionated spacecraft, and a three-module

fractionated spacecraft. For each spacecraft architecture, dynamic lifecycle assessments were performed using four different MCA trial numbers: 20,000; 2,500; 250; and 10. Then for each for each spacecraft architecture, the Dynamic LCC MMD corresponding to each of the four MCA trial values used was plotted. Given the four plots corresponding to the three spacecraft architectures corresponding to the four respective number of MCA trials considered, it was possible to form an approximate estimate as to the minimum number of MCA trials needed to appropriately characterize the Dynamic LCC distribution for a spacecraft. Figure D-2, Figure D-3, and Figure D-4 present the results from this study for the monolithic, two-module fractionated, and three-module fractionated spacecraft respectively.

Figure D-2. Dynamic LCC distributions: monolithic spacecraft.

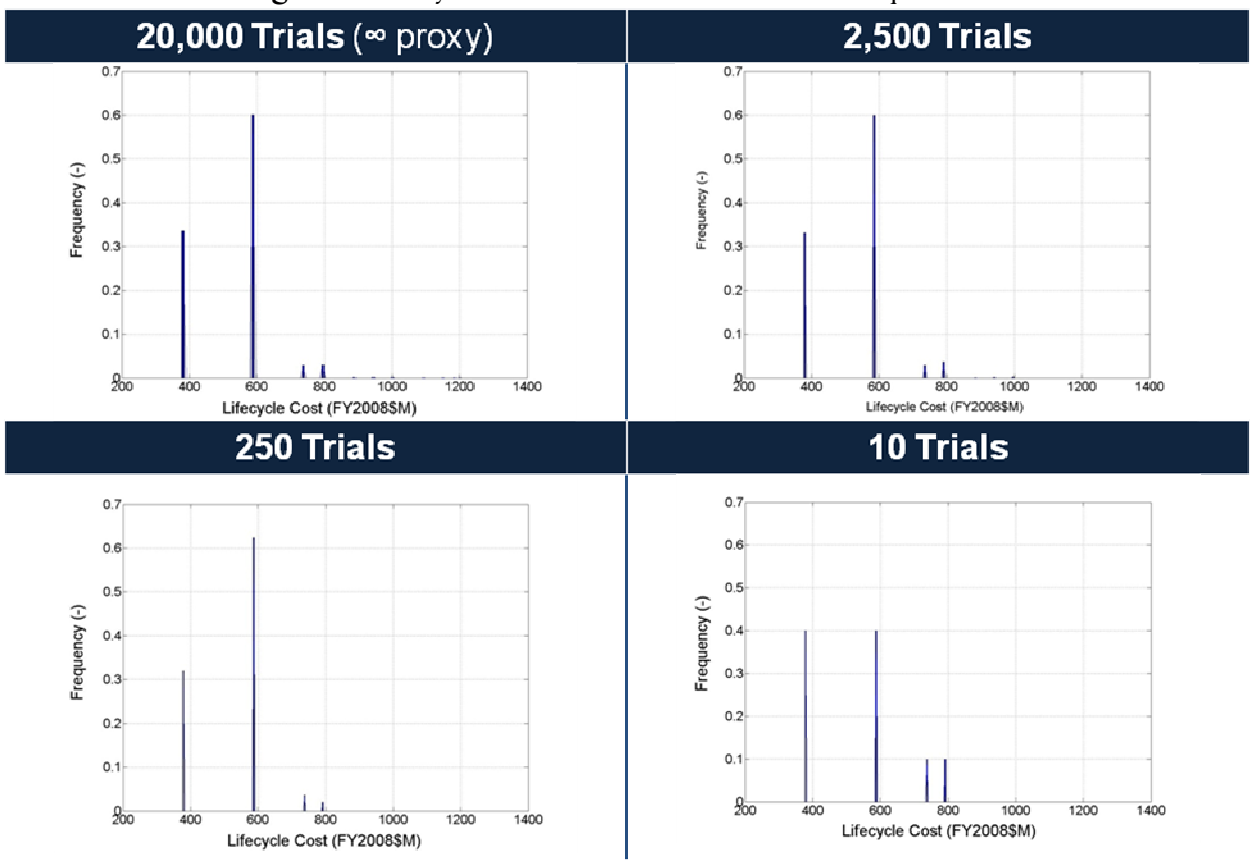


Figure D-3. Dynamic LCC distributions: two-module fractionated spacecraft.

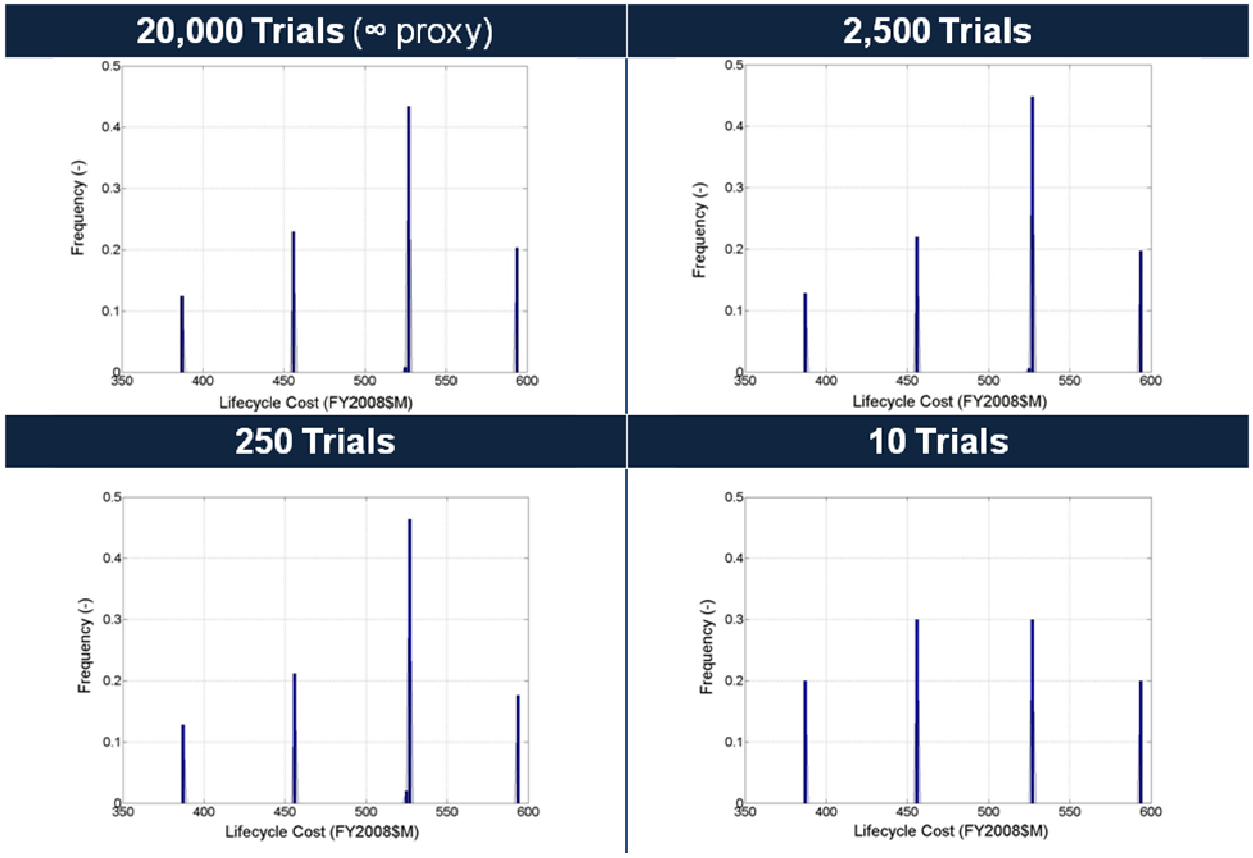
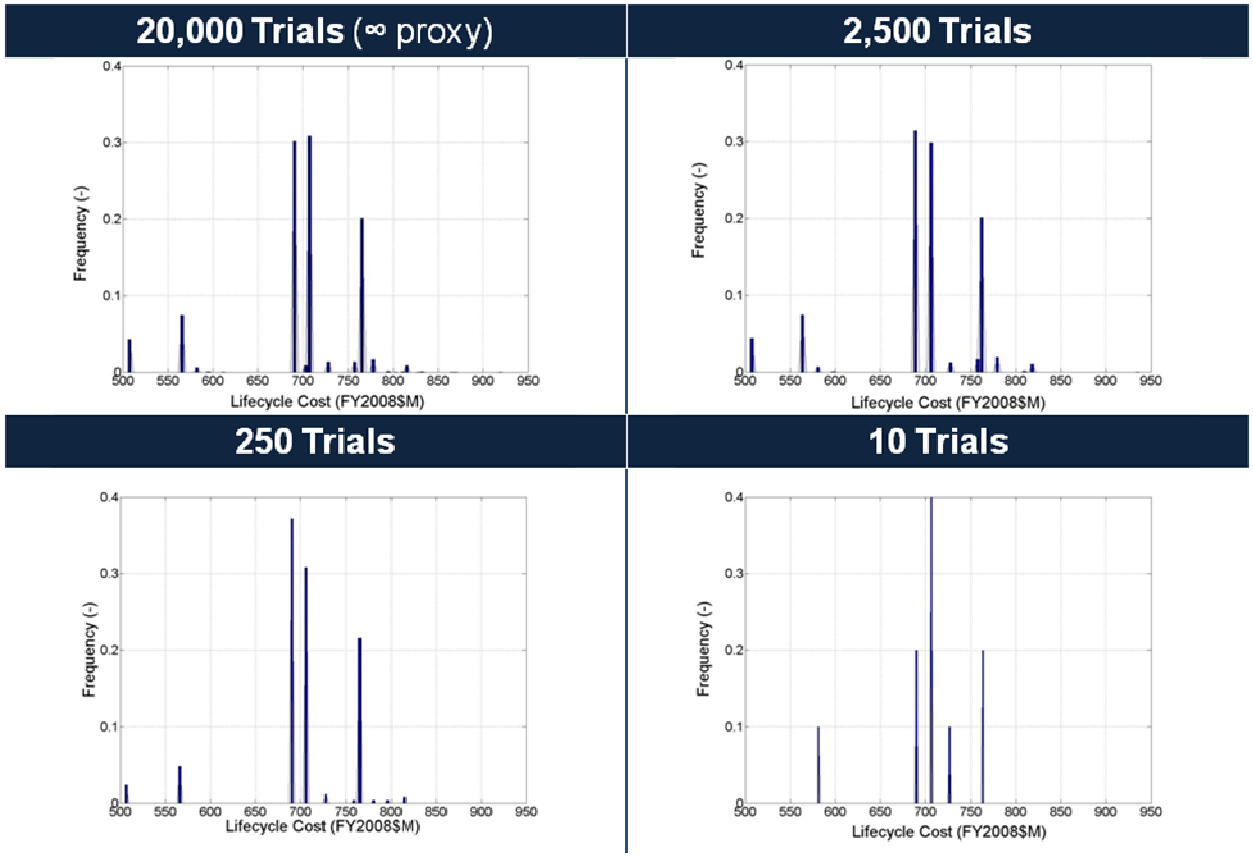


Figure D-4. Dynamic LCC distributions: three-module fractionated spacecraft.



A value of 20,000 MCA trials was used as a proxy for infinity to form the most appropriate/representative Dynamic LCC distribution. Comparing Figure D-2, Figure D-3, and Figure D-4 suggests that at least 2,500 MCA trials are needed to ensure that the Dynamic LCC distribution for a given spacecraft architecture is accurately captured. One should note that the Dynamic LCC distributions corresponding to 250 and 10 MCA trials in Figure D-2, Figure D-3, and Figure D-4 are fairly different (inaccurate) as compared to the Dynamic LCC distribution corresponding to 20,000 MCA trials. Therefore, the Dynamic LCC distribution corresponding to 2,500 MCA trials provides a nice balance between ensuring that the distribution is appropriate/representative (*i.e.,* accurate) and that the time required to run an SET simulation is not more than it has to be. Specifically, this conclusion is evident from the close correlation between the 20,000 and 2,500 MCA trial-produced Dynamic LCC distributions in Figure D-2, Figure D-3, and Figure D-4 and the fact that it takes on the order of $1/6^{th}$ the time to assess a spacecraft using 2,500 MCA trials as compared to using 20,000 MCA trials.

Based on the implications of the MCA trial sensitivity study presented in Appendix D, all results generated for the three case studies in Chapter 4 were obtained using 2,500 MCA trials. It is recognized that although 2,500 MCA trials may not be the true minimum required to appropriately reflect the 20,000 MCA trial-produced Dynamic LCC distributions (as the sensitivity study is rather course), it is sufficiently close.

Appendix E Optimistic Perspective of Fractionated Spacecraft

The synthesis of monolithic and fractionated spacecraft value propositions with regard to each research question provided in Section 5.2.1, 5.2.2, and 5.2.3 was made to be as objective (unbiased) as possible to maintain a fair comparison between monolithic and fractionated spacecraft. As such, the summarized responses to all three research questions, as is provided in Section 5.2.4 and Table 5-5, intentionally account for the value proposition *of every* fractionated and monolithic spacecraft considered in the three case studies. Subsequently, these responses do not preclude (hide) the fact that fractionated spacecraft do not always provide the strongest value proposition. In doing this, within the limits of this research, the responses research questions are equally fair to monolithic and fractionated spacecraft.

With this said, given the wealth of information provided in the analysis and synthesis in Chapter 4 and 5 respectively, the value proposition comparison between monolithic and fractionated spacecraft can be made in considering only the fractionated spacecraft with the strongest value propositions. This approach to quantifying and comparing the value propositions of monolithic and fractionated spacecraft therefore ignores the broader (and adverse) implications of fractionation, as are encapsulated in the responses the research questions in Section 5.2.1, 5.2.2, 5.2.3, and 5.2.4 because it compares *each dimension* of the value proposition based on the "best" fractionated spacecraft against the comparable monolith. However, this does provide value to both advocates and proponents of fractionation as it answers the suppliant question: within a given context, what is the absolute best fractionated spacecraft can do in terms of each respective value proposition metric relative to a comparable monolith?

The response to this question is quantified in Table E-1 and Table E-2. Specifically, Table E-1 and Table E-2 report the value proposition for the fractionated spacecraft within a given class (*e.g.,* two-module spacecraft) that is the most competitive (*i.e.,* "best") relative to the comparable monolith, based on each respective metric in the value proposition and for all three case studies. Table E-1 and Table E-2 are analogous to Table 5-2 through Table 5-5 in that they compare the value propositions of monolithic and fractionated spacecraft, however, they differ in that Table E-1 and Table E-2 adopt the most optimistic perspective of the fractionated spacecraft paradigm.

Table E-1. The most competitive fractionated spacecraft value propositions for Case Study 1 and 2.

				Case Study 1					Case Study 2 (Part 1)				
				Separation Distance		Lifecycle Parameters			Ground Resolution		Lifecycle Parameters		
		Value Proposition Metric	Value Proposition Units	VP: More (+), Equal (/), Less (-)	Arch ID	Res. (m)	Mission Lifetime (years)	PoIM (%)	VP: More (+), Equal (/), Less (-)	Arch ID	Res. (m)	Mission Lifetime (years)	PoIM (%)
Two Module Fractionated Spacecraft	*System*	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5	(/) 7	~	all	7.00	1.5
		Static LCC	FY2008$M	(+) 11.04	A4	0.5	7.00	1.5	(+) 10.28	A4	1.0	7.00	1.5
		Median Dynamic LCC	FY2008$M	(-) 90.73	A4	0.5	7.00	1.5	(-) 94.35	A4	1.0	7.00	1.5
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5	(/) 0.5, 1, 5 , 30	~	all	7.00	1.5
		Mass	kg	(+) 139.38	A2	0.5	7.00	1.5	(+) 153.23	A4	5.0	7.00	1.5
		Propellant Usage	kg	(+) 84.06	A2	0.5	7.00	1.5	(+) 91.34	A4	5.0	7.00	1.5
	Payload Module	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5	(/) 7	~	all	7.00	1.5
		Median Dynamic LCC	FY2008$M	(-) 449.80	A7	0.5	7.00	1.5	(-) 441.83	A7	0.5	7.00	1.5
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5	(/) 0.5, 1, 5 , 30	~	all	7.00	1.5
		Mass	kg	(-) 264.47	A7	0.5	7.00	1.5	(-) 217.18	A7	0.5	7.00	1.5
		Propellant Usage	kg	(-) 161.00	A7	0.5	7.00	1.5	(-) 130.36	A7	0.5	7.00	1.5
Three Module Fractionated Spacecraft	*System*	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5	(/) 7	~	all	7.00	1.5
		Static LCC	FY2008$M	(+) 145.82	A10	0.5	7.00	1.5	(+) 144.49	A10	1.0	7.00	1.5
		Median Dynamic LCC	FY2008$M	(+) 168.95	A10	0.5	7.00	1.5	(+) 167.26	A10	1.0	7.00	1.5
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5	(/) 0.5, 1, 5 , 30	~	all	7.00	1.5
		Mass	kg	(+) 279.36	A8	0.5	7.00	1.5	(+) 314.18	A10	5.0	7.00	1.5
		Propellant Usage	kg	(+) 168.44	A8	0.5	7.00	1.5	(+) 188.05	A10	5.0	7.00	1.5
	Payload Module	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5	(/) 7	~	all	7.00	1.5
		Median Dynamic LCC	FY2008$M	(-) 435.91	A10	0.5	7.00	1.5	(-) 439.02	A10	1.0	7.00	1.5
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5	(/) 0.5, 1, 5 , 30	~	all	7.00	1.5
		Mass	kg	(-) 264.47	A13	0.5	7.00	1.5	(-) 217.18	A13	0.5	7.00	1.5
		Propellant Usage	kg	(-) 161.00	A13	0.5	7.00	1.5	(-) 130.36	A13	0.5	7.00	1.5
Four Module Fractionated Spacecraft	*System*	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5	Not Applicable				
		Static LCC	FY2008$M	(+) 295.61	A22	0.5	7.00	1.5					
		Median Dynamic LCC	FY2008$M	(+) 402.00	A20	0.5	7.00	1.5					
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5					
		Mass	kg	(+) 419.07	A20	0.5	7.00	1.5					
		Propellant Usage	kg	(+) 252.69	A20	0.5	7.00	1.5					
	Payload Module	Mission Lifetime	year	(/) 7	~	0.5	7.00	1.5					
		Median Dynamic LCC	FY2008$M	(-) 436.67	A22	0.5	7.00	1.5					
		Payload Performance	m	(/) 0.5	~	0.5	7.00	1.5					
		Mass	kg	(-) 264.47	A25	0.5	7.00	1.5					
		Propellant Usage	kg	(-) 161.00	A25	0.5	7.00	1.5					

* System and Payload Masses are equivalent for all four ground resolutions investigated. The System and Payload Masses corresponding to the 0.5, 1, 5, and 30 meter Resolution is 1,668.74; 962.91; 404.37; and 195.56 kg respectively.

** System and Payload Propellant Usage are equivalent for all four ground resolutions investigated. The System and Payload Propellant Usage corresponding to the 0.5, 1, 5, and 30 meter resolution is 986.72; 569.34; 239.56; and 117.30 kg respectively.

Table E-2. The most competitive fractionated spacecraft value propositions for Case Study 2 and 3.

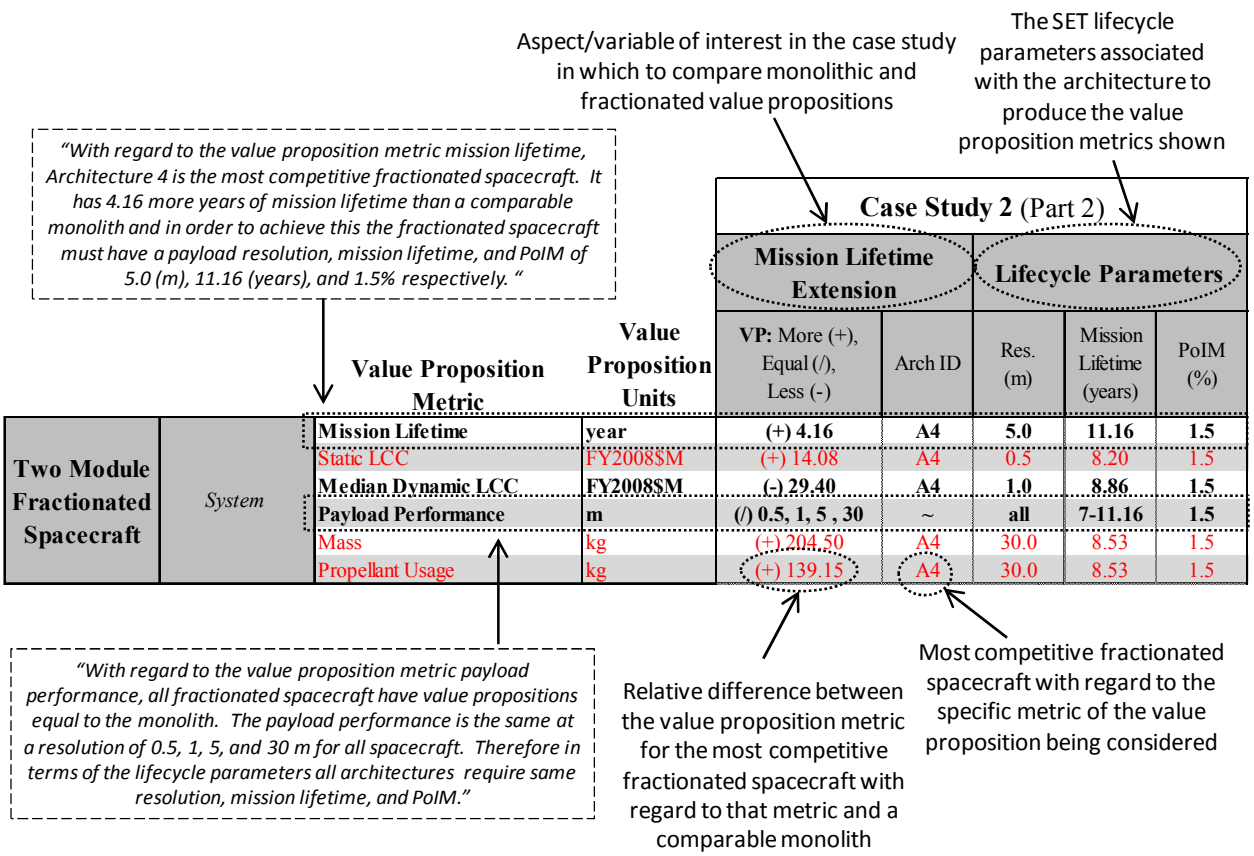
				Case Study 2 (Part 2)					**Case Study 3**				
				Mission Lifetime Extension		**Lifecycle Parameters**			**Mission Lifetime and PoIM**		**Lifecycle Parameters**		
		Value Proposition Metric	**Value Proposition Units**	**VP:** More (+), Equal (/), Less (-)	Arch ID	Res. (m)	Mission Lifetime (years)	PoIM (%)	**VP:** More (+), Equal (/), Less (-)	Arch ID	Res. (m)	Mission Lifetime (years)	PoIM (%)
Two Module Fractionated Spacecraft	*System*	**Mission Lifetime**	**year**	**(+) 4.16**	**A4**	**5.0**	**11.16**	**1.5**	**(/) 5, 7, 9**	**~**	**0.5**	**all**	**all**
		Static LCC	FY2008$M	(+) 14.08	A4	0.5	8.20	1.5	(+) 10.77	A4	0.5	5.00	all
		Median Dynamic LCC	**FY2008$M**	**(-) 29.40**	**A4**	**1.0**	**8.86**	**1.5**	**(-) 147.46**	**A4**	**0.5**	**9.00**	**5.0**
		Payload Performance	**m**	**(/) 0.5, 1, 5 , 30**	**~**	**all**	**7-11.16**	**1.5**	**(/) 0.5**	**~**	**0.5**	**all**	**all**
		Mass	kg	(+) 204.50	A4	30.0	8.53	1.5	(+) 138.25	A4	0.5	5.00	all
		Propellant Usage	kg	(+) 139.15	A4	30.0	8.53	1.5	(+) 74.50	A4	0.5	5.00	all
	Payload Module	**Mission Lifetime**	**year**	**(+) 4.16**	**A4**	**5.0**	**11.16**	**1.5**	**(/) 5, 7, 9**	**~**	**0.5**	**all**	**all**
		Median Dynamic LCC	**FY2008$M**	**(-) 376.84**	**A7**	**1.0**	**9.30**	**1.5**	**(-) 533.63**	**A7**	**0.5**	**9.00**	**5.0**
		Payload Performance	**m**	**(/) 0.5, 1, 5 , 30**	**~**	**all**	**7-11.16**	**1.5**	**(/) 0.5**	**~**	**0.5**	**all**	**all**
		Mass	**kg**	**(/) 1,668.74***	**~**	**all**	**7-11.16**	**1.5**	**(-) 263.88**	**A7**	**0.5**	**9.00**	**all**
		Propellant Usage	**kg**	**(/) 986.72****	**~**	**all**	**7-11.16**	**1.5**	**(-) 173.19**	**A7**	**0.5**	**9.00**	**all**
Three Module Fractionated Spacecraft	*System*	**Mission Lifetime**	**year**	**(+) 4.16**	**A10**	**5.0**	**11.16**	**1.5**	**(/) 5, 7, 9**	**~**	**0.5**	**all**	**all**
		Static LCC	FY2008$M	(+) 149.04	A10	0.5	8.20	1.5	(+) 145.29	A10	0.5	5.00	all
		Median Dynamic LCC	FY2008$M	(+) 272.33	A10	0.5	8.20	1.5	(+) 114.17	A10	0.5	9.00	0.0
		Payload Performance	**m**	**(/) 0.5, 1, 5 , 30**	**~**	**all**	**7-11.16**	**1.5**	**(/) 0.5**	**~**	**0.5**	**all**	**all**
		Mass	kg	(+) 385.12	A10	30.0	8.53	1.5	(+) 277.19	A10	0.5	5.00	all
		Propellant Usage	kg	(+) 247.13	A26	30.0	7.00	1.5	(+) 149.89	A10	0.5	5.00	all
	Payload Module	**Mission Lifetime**	**year**	**(+) 4.16**	**A10**	**5.0**	**11.16**	**1.5**	**(/) 5, 7, 9**	**~**	**0.5**	**all**	**all**
		Median Dynamic LCC	**FY2008$M**	**(-) 376.84**	**A10**	**1.0**	**8.86**	**1.5**	**(-) 525.50**	**A10**	**0.5**	**9.00**	**5.0**
		Payload Performance	**m**	**(/) 0.5, 1, 5 , 30**	**~**	**all**	**7-11.16**	**1.5**	**(/) 0.5**	**~**	**0.5**	**all**	**all**
		Mass	**kg**	**(/) 1,668.74***	**~**	**all**	**7-11.16**	**1.5**	**(-) 263.88**	**A13**	**0.5**	**9.00**	**all**
		Propellant Usage	**kg**	**(/) 986.72****	**~**	**all**	**7-11.16**	**1.5**	**(-) 173.19**	**A13**	**0.5**	**9.00**	**all**
Four Module Fractionated Spacecraft	*System*	**Mission Lifetime**	**year**	**Not Applicable**									
		Static LCC	FY2008$M										
		Median Dynamic LCC	FY2008$M										
		Payload Performance	**m**										
		Mass	kg										
		Propellant Usage	kg										
	Payload Module	**Mission Lifetime**	**year**										
		Median Dynamic LCC	**FY2008$M**										
		Payload Performance	**m**										
		Mass	**kg**										
		Propellant Usage	**kg**										

* System and Payload Masses are equivalent for all four ground resolutions investigated. The System and Payload Masses corresponding to the 0.5, 1, 5, and 30 meter resolution is 1,668.74; 962.91; 404.37; and 195.56 kg respectively.

** System and Payload Propellant Usage are equivalent for all four ground resolutions investigated. The System and Payload Propellant Usage corresponding to the 0.5, 1, 5, and 30 meter resolution is 986.72; 569.34; 239.56; and 117.30 kg respectively.

A few notes with regard to Table E-1 and Table E-2. First, some of the data used to populate these tables is not included in the analysis in Chapter 4 for reasons of keeping the data presentation and subsequent analysis in Chapter 4 tractable. Second, in Table E-1 and Table E-2, if a metric in a value proposition is stronger for the fractionated spacecraft than the monolith, that metric's respective value is in BOLD font. If this is not the case, meaning that the value proposition for the fractionated spacecraft is weaker than the monolith with respect to some metric, then that metric's respective value are listed in RED/LIGHT font. Third, the lifecycle parameters listed next to a given spacecraft architecture are the three SET inputs that correspond to the value proposition metric for that specific spacecraft architecture. The remaining SET inputs required to produce the value proposition metric listed next to the spacecraft architecture were held constant in the case studies; thus, these are not given in Table E-1 and Table E-2 but can be found in Table 4-2 and Table 4-3. And fourth, interpreting Table E-1 and Table E-2 is explained in Figure E-1.

Figure E-1. Reading the information in Table E-1 and Table E-2.



Based on the comparison the value proposition for monolithic and fractionated spacecraft given in Table E-1 and Table E-2, it is clear that regardless of the context (*i.e.,* case study) fractionated spacecraft never have stronger value propositions with regard to System Static LCC, System Mass, and System Propellant Usage.

Two-module fractionated spacecraft provide the highest number of value proposition metrics in which the fractionated spacecraft is stronger (better) than a comparable monolith.

And in terms of two, three, and four-module fractionated spacecraft, the most frequently occurring spacecraft architecture is that which employs only the Comm_CS_C&DH and ADS_GNS shared resource

(but no Power) (*e.g.,* Arch 4). In this sense, these spacecraft architectures are the most competitive fractionated spacecraft across all three case studies. This therefore enumerates that the "best" fractionated spacecraft, among those considered, in terms of the value proposition metrics of Mission Lifetime, Static LCC, and Median Dynamic LCC are those that employ the Comm_CS_C&DH and ADS_GNS shared resources only.

With regard to System Mass and Propellant Usage, the "best" fractionated spacecraft are those that employ no shared resources, thereby not incurring the mass penalties associated with the hardware for modules that are shared resource sources and recipients.

And with regard to all five Payload Module metrics in the value proposition, fractionated spacecraft that minimize the Payload Module mass (and size) are consistently the "best." Intuitively, these fractionated spacecraft happen to employ all the shared resources and their respective Payload Module produces and stores 0% of the power it requires, thereby yielding the smallest, and least massive Payload Module possible.

The specific reasoning as to why each fractionated spacecraft architecture in Table E-1 and Table E-2 is the most competitive with respect to a given value proposition metric, can be directly ascertained from the discussion in analysis in Chapter 4, and will therefore not be enumerated herein.