PROJECT LIFE-CYCLE AND IMPLEMENTATION FOR A CLASS OF SMALL SATELLITES

By

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To my family and friends for all their support

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I would like to thank my family for their continued support throughout my life. Without them, I wouldn’t have had the privilege to study in the United States.

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LIST OF ABBREVIATIONS

2D	Two Dimensional
3D	Three Dimensional
ACSA	Average Cross-Sectional Area
AI&T	Assembly Integration and Test
AR	Acceptance Review
BLOB	Binary Large OBject
CAD	Computer-Aided Design
Cal Poly	California State Polytechnic University
Caltech	California Institute of Technology
CDH	Command and Data Handling
CDR	Critical Design Review
CHDK	Canon Hack Development Kit
CMG	Control Moment Gyroscopes
CNSA	Chinese National Space Administration
CONOPS	Concept of Operations
COPUOUS	Committee on the Peaceful Uses of Outer Space
COTS	Commercial-off-the-shelf
CRR	Commissioning Result Review
CSLI	CubeSat Launch Initiative
DAMAGE	Debris Analysis and Monitoring Architecture to the GEO Environment
DCS	Debris Categorization System
DIR	Design Implementation Review

DITL	Day-In-The-Life
DoD	Department of Defense
DR	Disposal Review
ECSS	European Corporation for Space Standardization
EDU	Engineering Development Unit
ELaNa	Educational Launch of Nanosatellites
ELR	End of Life Review
EPS	Electrical Power System
ESA	European Space Agency
FCC	Federal Communications Commission
FMECA	Failure Modes, Effects, and Criticality Analysis
FRR	Flight Readiness Review
FTA	Fault Tree Analysis
Gage R&R	Gage Repeatability and Reproducibility
GEO	Geostationary Orbit
GEVS	General Environmental Verification Standard
GNC	Guidance, Navigation, and Control
GPS	Global Positioning System
HIL	Hardware-In-the-Loop
HVI	Hypervelocity Impact
IAA	International Academy of Astronautics
IADC	Inter-Agency Space Debris Coordination Committee
INCOSE	International Council on Systems Engineering

ISAS	Institute of Space and Astronautical Science
ISRO	Indian Space Research Organisation
JAXA	Japan Aerospace Exploration Agency
JERG	JAXA Engineering Requirement and Guideline
JMR	JAXA Management Requirements
JSpOC	Joint Space Operations Center
KARI	Korea Aerospace Research Institute
L_C	Characteristic Length
LEO	Low Earth Orbit
LRR	Launch Readiness Review
MBSE	Model Based System Engineering
MCR	Mission Concept Review
MCR	Mission Close-out Review
MDR	Mission Definition Review
MEO	Medium Earth Orbit
MOC	Mission Operations Center
NAL	National Aerospace Laboratory of Japan
NASA	National Aeronautics and Science Administration
NASDA	National Space Development Agency of Japan
NLAS	Nano –Satellite Launch Adapter System
NOAA	National Oceanic and Atmospheric Administration
NPOESS	National Polar-orbiting Operational Environmental Satellite System

NPR	NASA Procedural Requirements
OPAL	Orbiting Picosat Automated Launcher
ORR	Operational Readiness Review
P-POD	Poly-Picosatellite Orbital Deployer
PBS	Product Breakdown Structure
PDR	Preliminary Design Review
PFR	Post-Flight Review
PLAR	Post-Launch Assessment Review
PQR	Post-Qualification Review
PRR	Preliminary Requirements Review
QR	Qualification Review
RF	Radio Frequency
RGB	Red-Green-Blue
ROSCOSMOS	Russian Federal Space Agency
SATCAT	Satellite Catalog
SDR	System Definition Review
SE	Systems Engineering
SOCIT	Satellite Orbital Debris Categorization Impact Test
SR	Safety Review
SRR	System Requirements Review
SSN	Space Surveillance Network
SSTL	Surrey Satellite Technology Limited
SSWG	Space Systems Working Group

STEM	Science, Technology, Engineering, and Math
STK	System Tool Kit
SWaP	Size, Weight, and Power
SysML	Systems Modeling Language
TLE	Two Line Elements
TRL	Technology Readiness Level
TTC	Telemetry, Tracking, and Command
UF	University of Florida
UHF	Ultra High Frequency
UN	United Nations
USSTRATCOM	United States Strategic Command
V&V	Verification and Validation
VHF	Very High Frequency
VLF	Very Low Frequency
WBS	Work Breakdown Structure

Abstract of Dissertation Presented to the Graduate School
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PROJECT LIFE-CYCLE AND IMPLEMENTATION FOR A CLASS OF SMALL SATELLITES

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With advancements in miniaturization technologies, novel and innovative approaches to space and planetary explorations are being realized. An outcome of these innovations is a new class of small satellites referred to as CubeSats. CubeSats are popular within the space community (including new space entrants) due to their smaller form factor, lower costs, and faster development times as compared to traditional monolithic satellites. Currently, most CubeSat missions are developed in an ad-hoc manner due to the lack of structured procedures and protocol since suitable project life-cycle processes do not exist. Existing project life-cycles developed by NASA and other space/government agencies were developed specifically for traditional monolithic satellite missions and are not suitable for CubeSat class missions. Thus, there is a need to reimagine a project life-cycle for CubeSat class satellites.

This dissertation develops a comprehensive project life-cycle (inception, design, development, and operation/retirement) for a class small satellites that are launched from containers (i.e., containerized satellites). The “Containerized Satellite Mission Life-Cycle” leverages appropriate aspects of various existing project life-cycles and

engineering activities performed by the space/government agencies and the small satellite community. The project life-cycle has six phases, Pre-Phase I through Phase V, where Pre-Phase I is a systems engineering training activity catering to new space entrants and/or academic institutions. Phase I identifies the mission concept and a preliminary design is developed. Phase II matures the design into detailed design and Phase III addresses component/subsystem development, integration, and testing. Phase IV addresses the system level assembly and integration, environmental testing, and launch preparation. Phase V addresses the post launch operations up to and including retirement and disposal. Reviews are used to transition between the phases. The efficacy of the project life is assessed through two applications, one is an actual small satellite mission known as SwampSat II and the other is a non-mission project known as DebriSat. Through these applications it was shown that the Containerized Satellite Mission Life-Cycle is a structured process that is adaptable and flexible, and can be applied to containerized satellite missions as well as non-satellite missions.

CHAPTER 1
BACKGROUND

History of Satellites and Satellite Catalog

Since Sputnik-I, the first successful spacecraft launched by the Soviet Union in 1957 [1], thousands of spacecraft have been launched to space by numerous countries. These spacecraft are launched for various reasons such as, technology demonstration, human and non-human exploration, scientific experiments, remote sensing, communication, Earth and space weather, human capacity development, and many more. A satellite catalog (SATCAT) from Celestrak [2], shows a list of cataloged objects in space with various information (i.e., catalog numbers, object names, launch date, altitudes, periods, etc.). As of March 10, 2018, there are a total of 43,234 numbers shown in SATCAT, where the numbers are sequentially cataloged based on the launched date. Utilizing the date information (launch and decay) from SATCAT, Figure 1-1 was generated to showcase the growth of space objects, decayed objects (i.e., atmospheric reentry), and in-orbit objects since 1957. 56% of the objects have decayed (24,302) and the remaining 44% (18,932) of the objects are currently in orbit. The number of in-orbit objects have continued to grow as the years have progressed. This trend may be a safety concern for future space missions due to the orbit congestion. In this document, the terms spacecraft and satellite are used interchangeably.

The SATCAT does not limit the cataloged objects to just spacecraft. All of the cataloged objects are categorized and listed as either spacecraft, rocket body, or debris. A distribution of all of the objects in space is shown in Figure 1-2, where two third of the counts are debris (66%) and the other third is either spacecraft (20%) or rocket bodies

(14%). This distribution is concerning for the space community since majority are debris and may pose as threats to the current and future space missions.

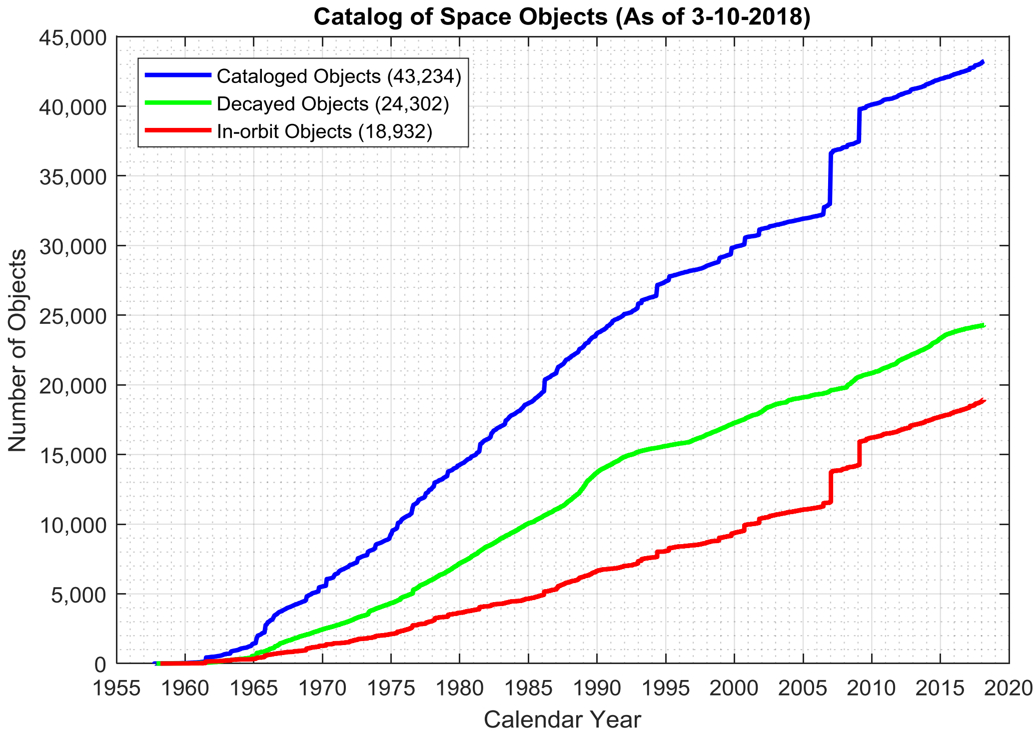


Figure 1-1. History of cataloged objects in space since 1957.

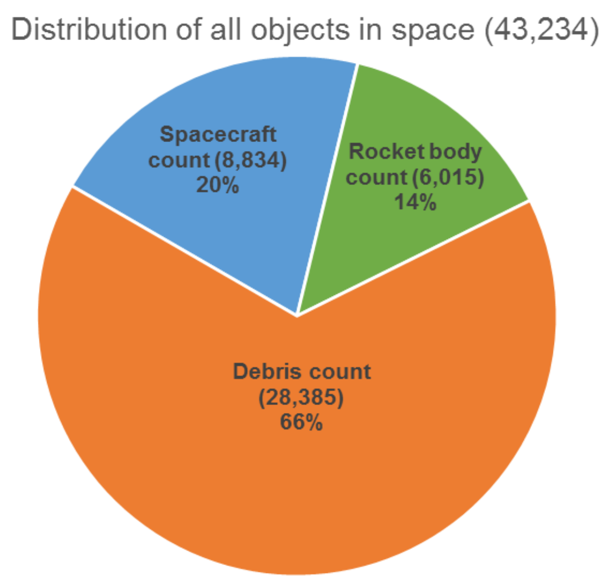


Figure 1-2. Distribution of cataloged objects in space since 1957.

All objects launched into space are released at a particular altitude. Altitude is calculated from the Earth's sea levels and typically 100 km represents the boundary separating Earth's atmosphere and outer space [3] [4]. Figure 1-3A represents perigee altitude for all objects launched into space and Figure 1-3B shows where majority of the objects are launched (less than 40,000 km). In Figure 1-3B, there are three distinct regions where the objects are launched: low Earth orbit (LEO, less than 2,000 km), medium Earth orbit (MEO, around 20,000 km) and geostationary orbit (GEO, within 200 km of 25,786 km). To note, some of the altitudes for the cataloged objects are not listed in SATCAT data (e.g., space probes such as PIONEER 4, MAVEN, VOYAGER-2, and spacecraft such as AEROCUBE 4, CINEMA, etc.).

Utilizing the SATCAT data and eliminating objects with apogee and perigee altitudes greater than 2,000 km, a total number of objects in LEO were determined. A total of 36,825 objects were cataloged in LEO, which is over 85% of the total number of objects. An examination of the distribution of these 36,825 objects shows that 18% of objects are spacecraft, 12% of objects are rocket bodies and the remaining 70% of objects are debris. This is consistent with the distribution of all cataloged objects (shown in Figure 1-2). As of March 10, 2018, a total of 13,422 cataloged objects are currently in LEO. Figure 1-4A shows the distribution of these 13,422 objects; 21% are spacecraft, 6% are rocket bodies, and 73% of objects are debris fragments. This distribution is also consistent with the distribution of the cataloged objects (shown in Figure 1-2) where it shows that majority of the cataloged objects are debris and is concerning for the space community. Figure 1-4B shows the altitudes at which these 13,422 objects are currently at, where majority of the objects are in the 500 km to 1,000 km region.

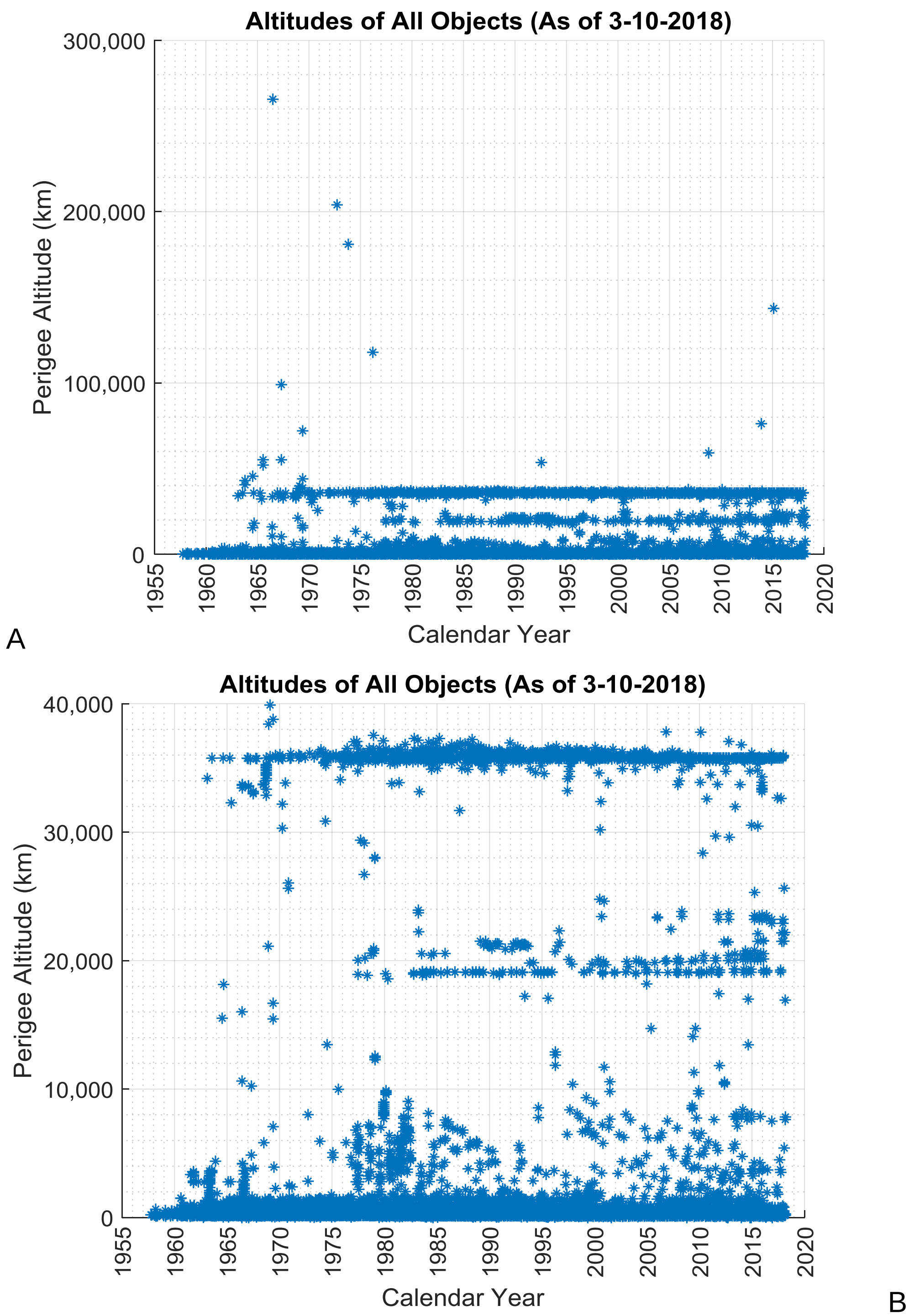
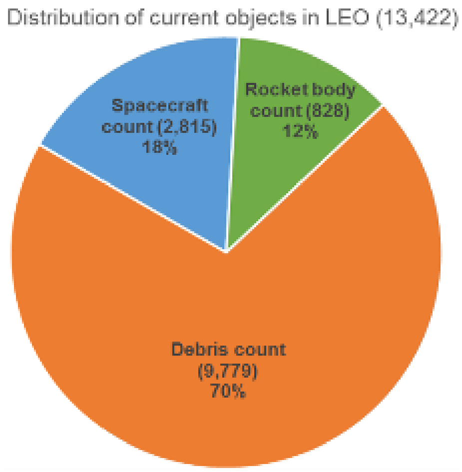
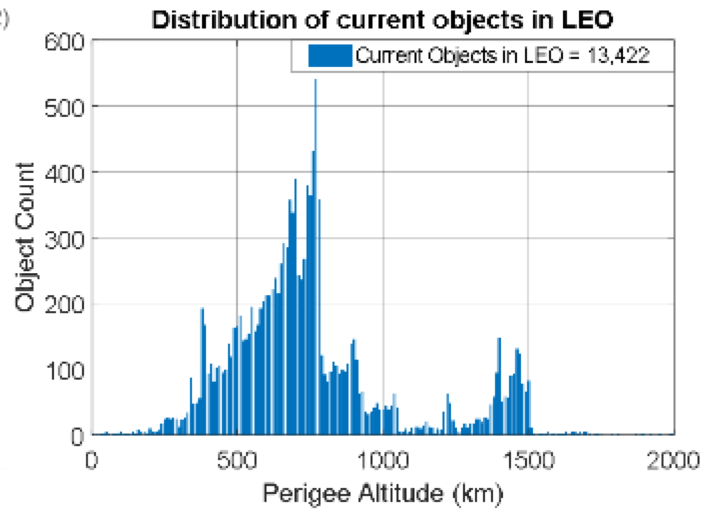


Figure 1-3. Altitudes of cataloged objects (as of 3-10-2018); A) all objects and B) zoomed in.



A



B

Figure 1-4. Distribution of current objects in LEO (as of 3-10-2018).

Why are more objects launched to LEO rather than MEO and GEO? One reason is that LEO is the closest to Earth and the launch vehicles (rockets) require less fuel to carry payloads to LEO, which results in lower launch costs. Another reason LEO is popular is for the maturation of the technologies (i.e., hardware and software) in relevant environments. The maturation of the technologies advances the Technology Readiness Levels (TRLs). TRLs were first introduced at National Aeronautics and Science Administration (NASA) by Sadin, et al. in 1989 [5] and has since been expanded. TRLs are systematic metrics used to assess the maturity of particular technology [6] and these metrics are implemented depending on the technology (i.e., hardware and/or software) being developed. TRL is organized into nine levels, where TRL 1 is the lowest maturation level (technology's basic principle) and TRL 9 is the highest maturation level (on-orbit validation of technology).

LEO is suitable to mature new space technologies due to lower launch and spacecraft costs. Due to its shorter development time, reduced mass, and reduced cost (launch and development), small spacecraft have been popular alternatives compared to traditional monolithic spacecraft where the technologies have to have been matured prior to incorporation. In order to test new technologies in the space environment, the technology developers aim for a cost-effective spacecraft. As a result, innovative spacecraft (specifically small satellites) have been developed as a platform to test the new technologies in LEO.

Small Satellites

The definition of “small satellites” has been an on-going discussion among the space communities and there is no universally accepted definition. Table 1-1 shows some examples of the ambiguity in the satellite classifications definitions as per organization. For example, the National Aeronautics and Space Administration (NASA) defines small satellites as satellites less than 180 kg [7], whereas the European Space Agency (ESA) defines small satellites as 350 kg to 700 kg [8], the Japan Aerospace Exploration Agency (JAXA) defines it as less than 100 kg [9] [10], Surrey Satellite Technology Limited (SSTL) defines small satellites as 500 kg to 1,000 kg satellites [11] [12] [13], and the International Academy of Astronautics (IAA) defines small satellites as all satellites less than 1,000 kg [14]. SSTL has been a pioneer in low-cost small satellites and their definition of small satellites has been more widely accepted. However, further studies, IAA in particular, to universally define the term has been on-going. Some study groups are trying to rename the term to “lean” satellites. For the purpose of this research, the definitions by IAA will be utilized; small satellites are all satellites less than 1,000 kg and classifications of small satellites as shown in Table 1-1.

Small satellites are not a recent development, they have been launched since the beginning of space exploration. Specifically, the first spacecraft were all small satellites; Sputnik-1 (83.6 kg [1]), Sputnik-2 (508.3 kg [15]) and Explorer-1 (13.97 kg [16]) where all fit the definition of small satellites. During the early years of the space age, the technology and the capability of launch vehicles limited the size of the payload. With the improvements in technologies, the capabilities of launch vehicles increased. In addition, as the number of advanced technologies increased, the number of technologies integrated in a single spacecraft increased. This increase in number of integrated technologies lead to higher power consumption which results in the need for more power generation. Which ultimately resulted in the larger “traditional” monolithic spacecraft of the 20th century. Thousands of spacecraft have been launched with various mission objectives, however, designs, developments, and launches of these larger spacecraft are time consuming and have extremely high costs due to the complexity of the spacecraft. In addition, operations during the mission lifetime (typically over five years for these larger spacecraft) add to higher cost. These costs were prohibitive and as such most spacecraft were either owned by government or by large corporations (particularly communication companies). The lower cost of small spacecraft have opened up a new paradigm of space utilization with academia and non-government owned spacecraft becoming ever popular

Yet in the mid-1970s Dr. Sir Martin Sweeting and his colleagues, at the University of Surrey, decided to develop a small satellite utilizing only standard consumer technology, also known as commercial-off-the-shelf (COTS) components [17] [18]. Sweeting and his team built UoSAT-1 and successfully launched it in 1981 with the help of NASA. The 72 kg satellite was cheaper, lighter, and took less time to build compared to traditional satellites. Following the success of UoSAT-1, the second microsatellite, UoSAT-2, was also developed by Sweeting and his team and was launched by NASA in 1984. Both microsatellites were launched to LEO from Vandenberg Air Force base in the United States. Dr. Sweeting founded SSTL in 1985 after the successes of the microsatellites to remain at the forefront of small satellite innovation. These two microsatellites demonstrated the potential for small satellites and utilization of COTS components in small satellite platforms. Furthermore, the success of these two microsatellites by SSTL revolutionized the small satellite market and opened doors for others. Specifically, the popularity of small satellites in academia grew for the educational hands-on experience while at lower costs.

Table 1-1. Definitions of small satellites.

Organization	Classification	Mass Range
NASA	Small	< 180 kg
	Mini	100 kg – 180 kg
	Micro	10 kg – 100 kg
	Nano (CubeSat)	1 kg – 10 kg
	Femto and Pico	< 1 kg
ESA	Small	350 kg – 700 kg
	Mini	80 kg – 350 kg
	Micro	50 kg – 80 kg
JAXA	Ultra Small	< 100 kg
	Micro-Nano	1 kg – 50 kg
	Nano-Pico	1 kg
SSTL	Small	500 kg – 1,000 kg
	Mini	100 kg – 500 kg
	Micro	10 kg – 100 kg
	Nano	1 kg – 10 kg
IAA	Mini	< 1000 kg
	Micro	< 100 kg
	Nano	< 10 kg
	Pico	< 1 kg

CubeSats

The small satellites developed in university-level engineering programs during the 1980s and 1990s were all nano- and microsatellite classes (1 kg to 100 kg). However, the lack of funding and launch opportunities made it very difficult to launch these small satellites [3] [17]. CubeSats, were introduced by Professors Robert "Bob" Twiggs at Stanford University and Jordi Puig-Suari at California State Polytechnic University (Cal Poly) in late 1999 [19]. The concepts of CubeSats originated from Orbiting Picosat Automated Launcher (OPAL), a 23 kg microsatellite developed by students at Stanford University and The Aerospace Corporation, to demonstrate deploying pico-satellites on-orbit via larger satellite [20]. OPAL was a significant achievement in small satellites by demonstrating the concept of pico-satellites and innovative on-orbit deployment system. The goal for Dr. Twiggs and Dr. Puig-Suari was to develop a standard set of dimensions for the pico-satellite class structure that can easily interface to an orbital deployer. As a result, the CubeSat form factor was defined and Poly-Picosatellite Orbital Deployer (P-POD) was developed [21]. A 1U standard CubeSat is a 10 x 10 x 10 cm cube with a mass of 1.33kg or less [22]. The CubeSats launched to date are 1U, 1.5U, 2U, 3U, and 6U, however, in recent years, new specifications has increased the size up to 12U and 27U (shown in Figure 1-5) [23].

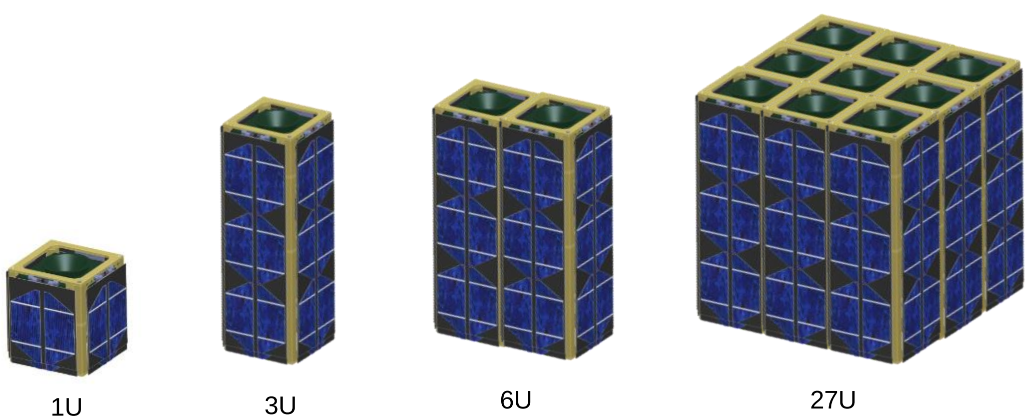


Figure 1-5. Different CubeSat form factors.

With introductions of CubeSats and P-PODs and on-orbit demonstration of OPAL, the launch opportunities for university-built small satellites increased. CubeSats are typically launched as a secondary payloads and are deployed into orbit through the use of deployment containers (deployment containers interface to the launch vehicle). While size, weight, and power (SWaP) constraints challenge CubeSat designers and developers with innovative designs, interface to the launch vehicle and ejection into orbit are not part of design considerations. Various programs such as NASA's CubeSat Launch Initiative (CSLI) [24] and Educational Launch of Nanosatellites (ELaNa) [25] provide "piggyback" rides with very little cost or no cost which attribute to the increase in the popularity of CubeSats.

Containerized Satellites

As more innovative CubeSat class satellites are being developed, various containers (e.g., P-POD [21], X-POD [26], ISIPOD [27], etc.) have also been developed to deliver these satellites into orbit. Moreover, these containers interface one or more satellites to the launch vehicle and prevent any harm to the launch vehicle and to others in the same container. Due to the development of these containers the number of "containerized" satellites launched into orbit (specifically LEO) have increased. Referring to satellites that are delivered to orbit via deployment containers as "containerized satellites", Figure 1-6 was generated to show the history of containerized satellites launched since 2002 (i.e., first CubeSat launch). Each block in the figure represents a single launch and the colors represents the countries of the launch providers. In addition, the launches are shown in chronological order (left to right). In this document, containerized satellites are defined as follows:

"A containerized satellite is any satellite that is enclosed in a container that interfaces the satellite to the launch vehicle. Such a container (e.g., P-POD [21], X-POD [26], ISIPOD [27], etc.) may contain one or more satellites and is designed to prevent harm to the launch vehicle (and other satellites) as well as deploy the containerized satellite(s) into orbit”.

Since 2002, 373 containerized satellites have been launched (as of July 23, 2015) and over 75% of them have come in between years 2013 and 2015. In the earlier years, most containerized satellites were launched outside of the United States, however, since 2009, the majority are launched from the United States. The data displayed in Figure 1-6 was generated through various sources and including individual websites [28] [29] [30] [31]. This figure differs from those of Janson [32] and Swartwout [33] since this considered all containerized satellites with masses less than 30 kg (includes those that experienced launch failure). In addition, less than 1U size (i.e., femtosatellites with masses less than 0.1 kg or “satellites-on-a-chip”) was not considered. A list of the containerized satellites are shown in Appendix A.

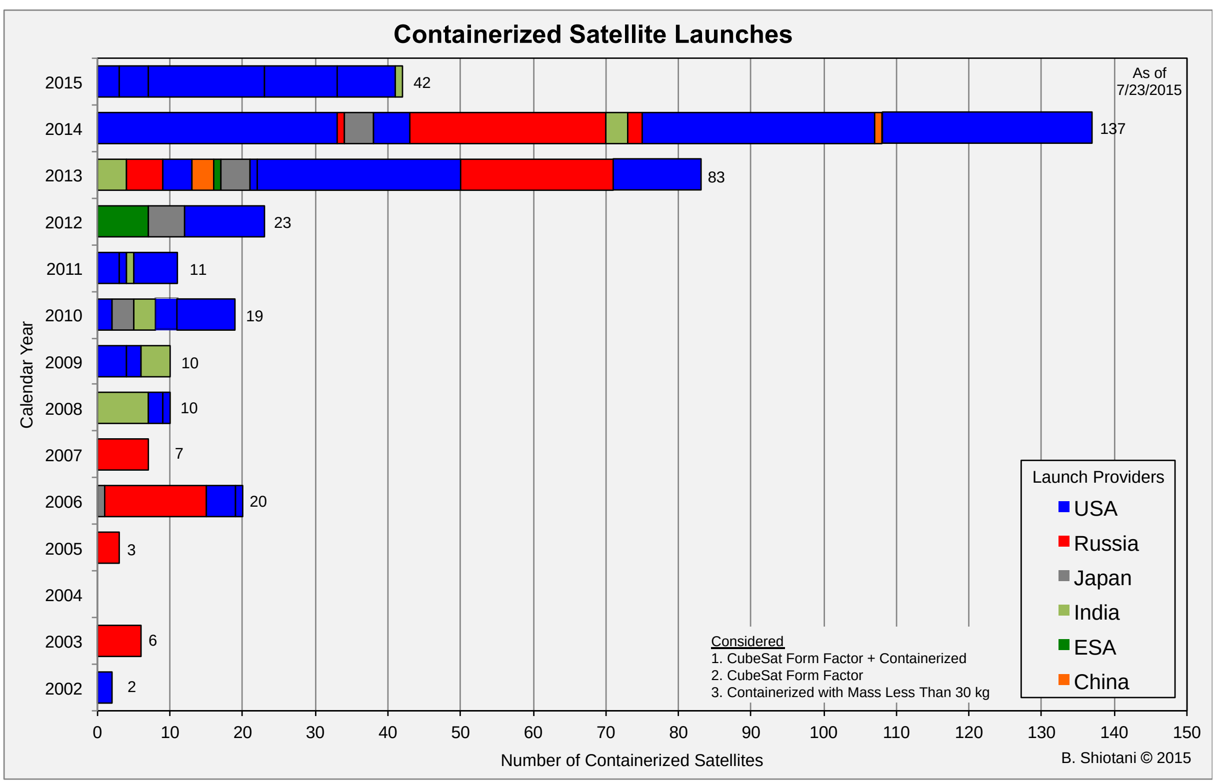


Figure 1-6. History of containerized satellite launches.

With advancements of the deployment containers, launch opportunities increased and the CubeSat class satellites were launched in swarms (i.e., multiple spacecraft in a single launch). Due to the increase in the number of CubeSats launched into space and their smaller form factors, there was a perception that these CubeSats were contributing to the debris population. However, only less than 400 CubeSats have been launched since 2002, compared to over 7,000 non-CubeSat class satellites launched since 1957. In this document, containerized satellite and CubeSat class satellites are used interchangeably.

Orbital Debris and Space Situational Awareness

Orbital debris is any non-operational object in orbit and is classified as either natural or man-made objects [34]. Natural objects include meteoroids and asteroids, and man-made objects are objects launched into orbit. In 1971, Kessler [35] explained the lack of information regarding the natural objects would present considerable danger to spacecraft. In his paper, he presented estimates to model the collision frequency in the asteroid belt. Then in 1977, Kessler and Cour-Palais [36] explained that as the number of artificial satellites in Earth orbit increases, the probability of collision between satellites increases at an even faster rate, which in turn would produce more fragments and increase probability of further collisions, known as the “Kessler syndrome.” Following this, in 1978, Kessler and Burton [37] developed a model that described the environment resulting from orbiting satellites. In 1987, Johnson and McKnight [38] published a book entirely on artificial space debris, which most describe as the first book ever published on artificial (i.e., man-made) debris. After thirty years since the first man-made object was launched to space, the community’s focus started to shift from natural orbital debris to man-made orbital debris.

For the purpose of this research, only man-made space debris is considered and is precisely defined as follows. “Space debris is all man-made objects, including fragments and elements thereof, in Earth orbit or re-entering the atmosphere, that are non-functional [39].” These man-made objects include retired satellites, upper rocket stages, and break-ups from past on-orbit collisions. A 2008 study by NASA showed that 48% of on-orbit objects are due to satellite breakups [40]. Furthermore in the same study, the primary causes of satellite breakups are propulsion-related events and deliberate (mission) actions, but one in five breakups causes are unknown [40].

In 1995, NASA was the first space agency to issue a set of orbital debris mitigation guidelines which the U.S. National Science and Technology Council distributed among agencies [41]. Two years later, the U. S. government adopted NASA guidelines and developed its own orbital debris mitigation standard practices [42]. In 1999, the United Nations (UN) Committee on the Peaceful Uses of Outer Space (COPUOS) published a technical report on space debris that discussed the measurement, modeling, and mitigation strategies for space debris [43]. After a multi-year effort, the Inter-Agency Space Debris Coordination Committee (IADC) developed a set of guidelines in 2002 [44]. The IADC is an international organization where members from space agencies exchange information and research activities regarding space debris. The IADC members include government space agencies from, Britain (BNSC), Canada (CSA), China (CNSA), Europe (ESA), France (CNES), Germany (DLR), India (ISRO), Italy (ASI), Japan (JAXA), Russia (ROSCOSMOS), South Korea (KARI), Ukraine (NSAU), and the United States (NASA) [45]. After five years, in 2007, the UN

COPUOUS adopted the IADC guidelines and developed their own space debris mitigation guidelines [46].

There are three main research areas of debris: measurement, modeling, and mitigation. The details of each are discussed in the following sections.

Measurements

Ground and space-based measurements are taken between low Earth orbit (LEO, less than 2,000 km altitude) and geostationary orbit (GEO, within 200 km of 35,786 km). The ground measurements utilize radar and optical instruments and the space measurements are performed through impact and optical detectors. Ground instruments are capable of measuring up to few mm in LEO and 10 cm in GEO, however, currently only 10 cm or larger objects are actively tracked in LEO through the United States, Space Surveillance Network (SSN) [47] and other space agencies [48] [49] [50] [51]. Space-based in-situ instruments are capable of measuring sub-millimeter debris by impact detectors [52]. An impact detector called the Space Debris Sensor has been launched and installed on the international space station in January 2018 and has been collecting in-situ measurements [53].

Modeling

To compensate for high cost of debris measuring instruments, space agencies around the world have developed space debris environmental models. Space debris environmental models provide distribution, movement and flux, and physical characteristics of objects in space. These models use data from historical records of satellite characteristics, launch activities, orbit collisions and breakups, and measurements (ground- and space-based). Furthermore, these models are developed to characterize the current and future debris environment. Specifically, the short term models typically consider up to 10 years and the long term models consider the environment for more than 10 years. Many debris environmental models (short term and long term) have been developed by various space agencies [43] [54] [55] [56] [57] [58] [59] and the results are used and shared in IADC working groups [60] [61]. Table 1-2 shows some of the models developed by different space agencies.

Table 1-2. Debris environmental models.

Space Agency (Country)	Short Term Model	Long Term Model
NASA (USA)	ORDEM DAS MEM	LEGEND EVOLVE
ESA (Europe)	MASTER	DELTA
ASI (Italy)		SDM
ISRO (India)		KSCPROP
JAXA (Japan)		LEODEEM
UKSA (United Kingdom)	IDES	DAMAGE
ROSCOSMOS (Russia)	SPDA	

With the assistance of these models, various collision risk assessments among operational spacecraft and space debris have been conducted. In 1994, Rossi, et al., presented a collision analysis of debris and stated that an exponential growth is expected in altitude regions between 700 and 1,000 km and between 1,400 and 1,500 km [62]. In 2006, Klinkrad [63] published a book on space debris modeling and risk analysis largely focused on European activities. In Klinkrad's book, he describes conjunction prediction and collision avoidance is possible through careful analyses of two line elements (TLEs). TLEs represent orbital information for each cataloged objects in space. Recently, there have been more studies of CubeSat collision and conjunction risk assessments: in 2011, Oltrogge and Leveque assessed orbital lifetime of CubeSats [64], in 2013 Springmann et al., conducted investigation to the on-orbit conjunction between CubeSats [65], and in 2014 Lewis et al., conducted an assessment of CubeSat collisions utilizing Debris Analysis and Monitoring Architecture to the GEO Environment (DAMAGE, developed by University of Southampton, U.K.) [66]. The increase in the studies for CubeSats is largely attribute to the shift in paradigm and as more CubeSat class satellites are being developed, the space community is becoming more aware that these CubeSat class satellites are increasing the on-orbit population. Currently, TLEs are cataloged and maintained by US Strategic Command (USSTRATCOM) and its Joint Space Operations Center (JSpOC) tracks and identifies all artificial objects in Earth orbit. JSpOC notifies spacecraft owners with proximity predictions and distributes alerts in case of close approaches [67]. The close approach notifications from JSpOC are one of the methods seen in debris mitigations.

Mitigation

The space debris mitigation guidelines adopted by the UNCOPUOS in 2007 outlines the space debris mitigation strategies during the entire life-cycle phases for spacecraft and launch vehicles. These life-cycle phases include, mission planning, design, manufacture, verification, and operation (launch, mission, and disposal) phases. There are seven guidelines listed in the UNCOPUOUS document and these guidelines can be broadly categorized into two stages: pre-launch stage and post-launch stage. The pre-launch stage include mission planning, design, manufacturing, and verification phases, while the post-launch stage include the operation phase. For each guideline, the mitigation strategies can be implemented in both the pre-launch and the post-launch stages. The seven debris mitigation guidelines are:

1. Limit debris released during normal operations
2. Minimize the potential for break-ups during operational phases
3. Limit the probability of accidental collision in orbit
4. Avoid intentional destruction and other harmful activities
5. Minimize potential post-mission break-ups resulting from stored energy

6. Limit the long-term presence of spacecraft and launch vehicle orbital stages in the low-Earth orbit (LEO) region after the end of their mission
7. Limit the long-term interference of spacecraft and launch vehicle orbital stages with the geosynchronous Earth orbit (GEO) region after end of their mission

The guidelines #1 through #4 recommends spacecraft and launch vehicle designers to implement strategies during the pre-launch stage, specifically in the mission planning and in the design phases. For example, in early years of the space age, both launch vehicle and spacecraft designers intentionally released mission-related objects (e.g., nozzle covers, lens caps, deployment mechanisms, etc.) into orbit. To limit intentionally released mission-related objects, the designers shall not incorporate such objects in their designs. If the objects are required to be released, those objects can be designed with a tether or similar so that it does not get released into orbit. Another example is for the designers to select orbits (altitude and inclination) that have less probability of accidental collision during the mission planning phase. If the orbit cannot be selected, the spacecraft and launch vehicle designers can incorporate systems capable of orbit maneuvers to change orbits to avoid accidental collisions.

Guidelines #5 through #7 are aimed at post-launch stage of the mission. Guideline #5 recommends the spacecraft and launch vehicle operators to deplete any stored energy post-mission to reduce potential break-ups resulting from stored energy. For example, the pressurized tanks can be de-pressurized post-mission so that the stored energy in the tanks are reduced. There are two ways to dispose of space objects, de-orbit and enter Earth's atmosphere, or maneuver to different altitudes. For objects that de-orbit, the guideline encourages objects to be removed from orbit in a controlled fashion as well as to not pose risk to people or property if objects survive reentry. For objects that are disposed by changing orbits, the guideline recommends spacecraft to maneuver above the GEO region to avoid interference with other spacecraft in the GEO region.

Challenges and Issues

While debris research and activities have been an on-going worldwide participation, there are several noticeable challenges. First, for debris measurements, only objects 10 cm and greater are actively tracked in LEO. This is a growing concern since smaller fragments still pose threat to manned and unmanned space missions. The exact numbers of objects less than 10 cm are unknown and as more femto-class satellites are being developed and launched to LEO, the problem is compounded. For example, the KickSat mission of May 2014 [68] was to deploy 128 chip-sized (~5 cm square) femto-class satellites, however, the deployment failed and none of the chip-sized satellites were deployed into orbit.

Another challenge is regarding the debris mitigation guidelines. The UNCOPUOS mitigation guidelines are only recommended and are not bounded by law. There are countries that have their own space laws and enforce them to space users. Moreover, some countries enforce space laws that are not shown in the UNCOPUOS debris mitigation guidelines. For example, NASA programs and projects must conduct formal orbital debris assessment to satisfy the “25-Year-Rule” where the spacecraft and upper stage in LEO must be disposed (i.e., de-orbit) within 25 years after completion of mission [42] [69].

Another challenge is uncertainties in the debris environmental models. There are no clear studies to determine uncertainty in these models. To overcome this, various ground impact tests have been conducted to characterize the on-orbit breakup models [70]. One of the well-known is the Satellite Orbital Debris Characterization Impact Test

(SOCIT) series performed in early 1990s [71] [72]. The fourth SOCIT series targeted a 1970s defunct Navy satellite and subjected it to a hypervelocity impact test. In addition, seven micro-satellite impact tests were conducted through collaboration between Kyushu University in Japan and NASA (completed in 2007) [73]. Data from SOCIT series and the Kyushu University-NASA tests have been used in the current satellite breakup models. Another ground impact test, known as the DebriSat project, was conducted in April 2014 to update the current satellite breakup model [74].

There were two major on-orbit collision events where the current breakup models were used to compare with the actual observed debris fragments. The events were the 2007 Fengyun 1C missile test [75] and the 2009 accidental collision of Iridium 33 and Cosmos 2251 [76]. From these catastrophic events, over 5,700 objects have been cataloged in the SATCAT to date, however, these cataloged are those that are actively tracked (i.e., 10 cm and greater). After these catastrophic events, NASA utilized its current breakup model and compared to the SATCAT data. NASA's model predictions matched well for breakups of old satellite (i.e., Cosmos 2251), however, there were noticeable differences for the modern satellites (i.e., Fengyun 1C and Iridium 33). Figure 1-7 shows the comparisons between the NASA model predictions and the SATCAT data; Figure 1-7A shows the comparison between NASA model and Cosmos 2251 fragments [77], Figure 1-7B compares NASA model predictions and the Iridium 33 fragments [77], and Figure 1-7C compares NASA model predictions and Fengyun 1C fragments [78].

The discrepancies in the NASA's breakup model predictions were due to the fact that the model used data from older satellites while Iridium 33 and Fengyun 1C were developed using newer materials and process techniques. Based on these inaccuracies in the predictions, NASA and the Department of Defense (DoD) decided to update the current satellite breakup models. In order to update the model, a representative LEO satellite using modern materials and components referred to as DebriSat test article was developed and was subjected to a hypervelocity impact in April 2014 [74] [79]. The fragments collected from the DebriSat project are analyzed and used in updating the current breakup models. Details of the DebriSat project are explained in the later chapter.

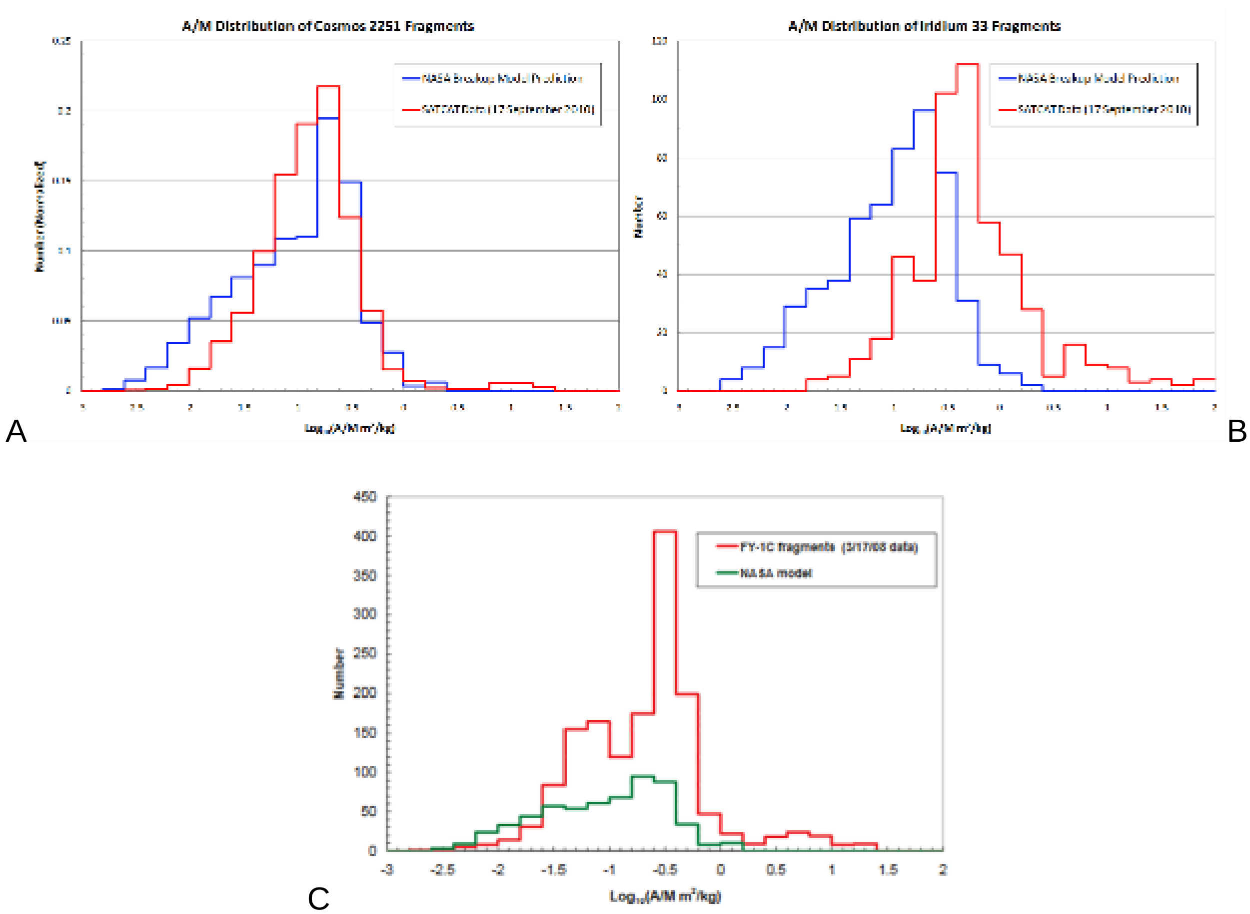
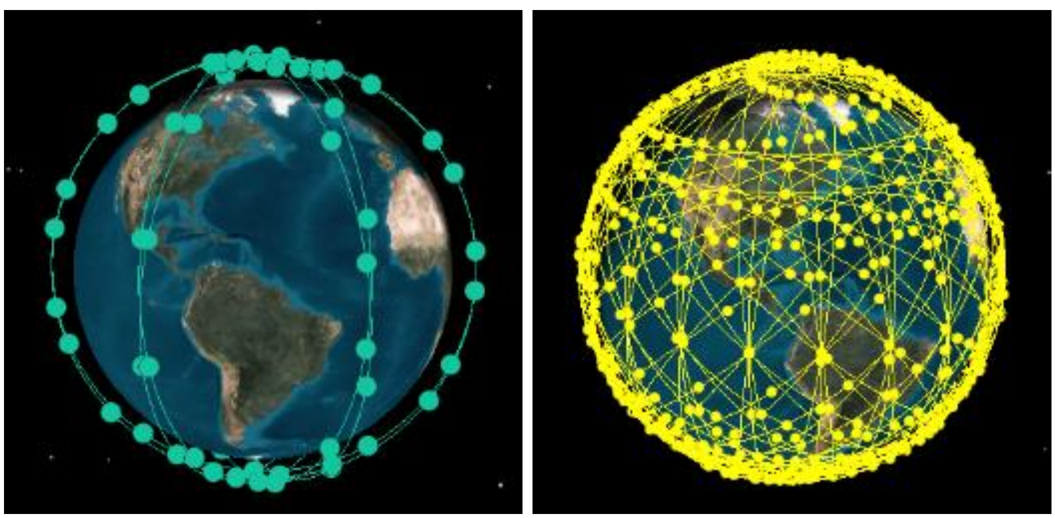


Figure 1-7. Area-to-mass distributions of the NASA breakup model prediction and A) Cosmos 2251 [77], B) Iridium 33 [77], and C) Fengyun 1C [78].

Another growing concern is regarding the mega constellations in low Earth orbits that industries such as OneWeb and SpaceX are planning in the near future [80] [81]. Current satellite constellations such as Iridium [82] utilize less than 100 satellites in their constellation, however, the large constellations plan to utilize thousands of small satellites. Figure 1-8 shows the example of the two constellations. With such large satellite constellations, the orbit population increases by the thousands and the space environment becomes more congested. Failure of satellites in the large constellations can be catastrophic as they instantly become debris and add to the debris population. The growth in the debris population becomes harmful to current and future manned and unmanned space missions.



A B

Figure 1-8. Satellite constellations: A) traditional satellites (e.g., Iridium) and B) future satellites (e.g., OneWeb SpaceX, PlanetLabs).

As opportunities for space increase with the emergence of small satellites, the responsibilities as space users and spacecraft developers must also increase. There are instances where dummy loads (also known as balance mass) are launched into orbit to adjust the spacecraft’s center of mass for launch. These dummy masses are essentially debris placed in orbit. Spacecraft developers should avoid these practices and must ensure that functional spacecraft are placed in orbit.

Motivation

The emergence of a new class of small satellites, specifically the CubeSat class satellites, has led to rapid increase in the number of small satellites launched into space. These new class of small satellites are appealing to both amateurs (mostly academic) and professionals (industry and government) due to their smaller form factor, shorter development time, and reduced costs (launch and development). However, these first CubeSats developed by the small satellite community have high failures. Swartwout maintains a CubeSat database [83] and showed that over 50% of the first CubeSat missions developed by 193 organizations, both amateurs and professionals, do not achieve mission objective(s). 125 out of 193 organizations are academia and over 60% of their first CubeSat mission fail. The high failures may be attributed to the lack of experience and lack of structured procedures for these new class of small satellites. Without fundamental procedures and protocols in place, there is a tendency to proceed in an ad-hoc manner. Spacecraft and for that matter any system developed in such ad-hoc manner are less reliable and will result in greater chance of failure.

The existing project life-cycles developed by NASA and other government space agencies are specifically developed for larger monolithic satellites and in the current form are not suitable for CubeSat class satellites. For example, the larger monolithic satellites could have several payloads to satisfy numerous mission objectives, thus, require more time and cost. On the other hand, due to its smaller form factor and size, weight, and power (SWaP) constraints, CubeSats typically do not have multiple payloads and thus require shorter development cost and time. While the CubeSat class satellites do not require the level of detail that larger monolithic satellites may need, the CubeSat class satellites still require a structured process. The structured process for the traditional spacecraft are too rigid and not flexible to be adapted for CubeSat class satellites. One notable example of this is the National Polar-orbiting Operational Environmental Satellite System (NPOESS).

In the 1990’s, the National Polar-orbiting Operational Environmental Satellite System (NPOESS) program was created as a joint project for NASA, DoD, and the National Oceanic and Atmospheric Administration (NOAA) to integrate their weather satellite systems [84] [85]. The objective of the NPOESS program was to plan, develop, and operate a polar-orbiting remote-sensing spacecraft while reducing the cost and risk with the integration of three agencies. However, due to significant cost overruns and schedule delays the NPOESS program was terminated in 2010. The significant cost overruns and schedule delays may be attributed to the lack of flexibility of the processes. When three government agencies implement their respective project life-cycles, it becomes extremely time consuming, which results in increase costs. As CubeSat class satellites have evolved out of the paradigm, the structured process in developing them shall also evolve. Therefore, there is a need to reimagine a project life-cycle process that is tailored for the CubeSat class satellites.

One recent effort in developing a project life-cycle for CubeSats was by NASA’s CubeSat Launch Initiative (CSLI). In October 2017, NASA’s CSLI released a document known as the “CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers” that is aimed at first-time CubeSat developers [86]. The document is for CubeSat developers who are working specifically with NASA’s CSLI. The CSLI opportunities are only available to NASA centers, U.S. non-profit organizations, and accredited U.S. educational organizations. This document may be suitable for first time CubeSat developers that could qualify for the CSLI opportunities and may not for those who are intermediate CubeSat developers and are outside of the U.S. Although the document could be used by experienced and intermediate CubeSat developers as references, the exact process cannot be implemented. Currently, there is no project life-cycle for those with prior knowledge of systems engineering principles and CubeSat experience.

Another effort in developing a project life-cycle for CubeSats is by the International Council on Systems Engineering (INCOSE). INCOSE has a space systems working group (SSWG) that is currently developing a reference model for a CubeSat using a model based system engineering (MBSE) systems modeling language (SysML) [87]. The goal of the CubeSat reference model is to provide the CubeSat projects with a formal project life-cycle, from concept to retirement. Specifically, the CubeSat reference model follows the INCOSE project life-cycle and is aimed to promote the use of MBSE. While the SSWG's CubeSat reference model follows a structured process, it requires the use of MBSE SysML modeling tools. The MBSE SysML modeling tools are not readily available and requires extensive training of the modeling tool that many CubeSat developers may not have access to. The CubeSat reference model is bounded by the use of SysML and not flexible to be adapted by CubeSat developers. The project life-cycle should be adaptable and should not require specific tools to implement.

This document presents a project life-cycle that provides the fundamental procedures and protocols for a complete life-cycle for small satellites (specifically CubeSat class satellites). The project life-cycle is developed based on the understanding of systems engineering principles and existing project life-cycles. Furthermore, the project life-cycle is aimed at intermediate CubeSat developers and to provide a thorough end-to-end process to reduce the tendency for ad-hoc development. The project life-cycle is flexible (does not require specific tools to implement) and is adaptable to many systems. This document also presents the implementations of the life-cycle to various space systems.

The dissertation is organized as follows. Chapter 1 provides the background and history of satellites, and introduces small satellites. Chapter 1 also discusses the motivation for the research. Chapter 2 explores the existing project life-cycles and associated engineering activities. Chapter 3 describes the project life-cycle for the containerized satellite satellites. Chapter 4 provides implementation examples of the project life-cycle. Chapter 5 presents the conclusion and future work.

CHAPTER 2
PROJECT LIFE-CYCLES AND ENGINEERING ACTIVITIES

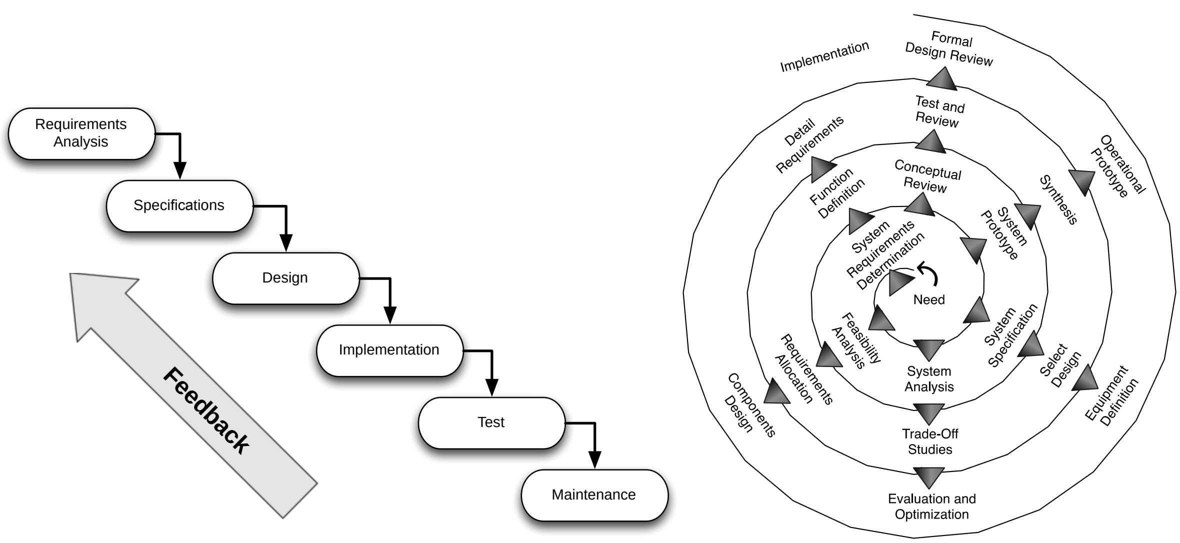
Systems Engineering and Systems Engineering Process Models

A system is a construct or collection of different elements that together produce results not obtainable by elements alone [88]. A subsystem is a lower level (i.e., component) of a larger system. Some subsystems and systems are classified as system of systems. Systems engineering is an interdisciplinary approach that's meant to enable the realization of successful systems [89]. One of the main characteristics of systems engineering is applying a systematic process/approach to engineering problems throughout the entire system. Such systematic processes/approaches have similar principles and objectives, however, the implementation varies depending on the nature of the system being developed. Systems engineering differs from classical engineering where classical engineering focuses on development of products and its performance and systems engineering is a process focused on the system as a whole and provides robust solutions throughout the entire process [90].

There are several different system engineering process models, where each system engineering process models shows different implementation approaches [91]. For this research five systems engineering process models were examined; waterfall [92] (Figure 2-1A), spiral [93] (Figure 2-1B), “Vee,” [94] (Figure 2-2A) “W,” [95] (Figure 2-2B) and NASA's systems engineering engine [88] (Figure 2-3).

Winston W. Royce first introduced the waterfall model in 1970 specifically for software development [92]. The process flows downward through each phase and transitions between phases are done once defined goals are accomplished. If the developed product does not meet requirements, the process starts over with improvements using feedbacks. The waterfall model is best suited for small projects and programs, however, it is typically used with well-defined requirements. Also, changes throughout the process are difficult to adopt.

The spiral model was first introduced by Barry Boehm in 1986 as an adaptation of the waterfall model. The spiral model was intended to introduce risk-driven approach for software development where the risks are identified and assessed at each phase of the project. For each identified risk, a detailed analysis is conducted and appropriate risk mitigation strategies are implemented to reduce the risks. The risk assessments are conducted and mitigation strategies are implemented each phase, thus, this model is complex, time consuming, and cost intensive to implement.

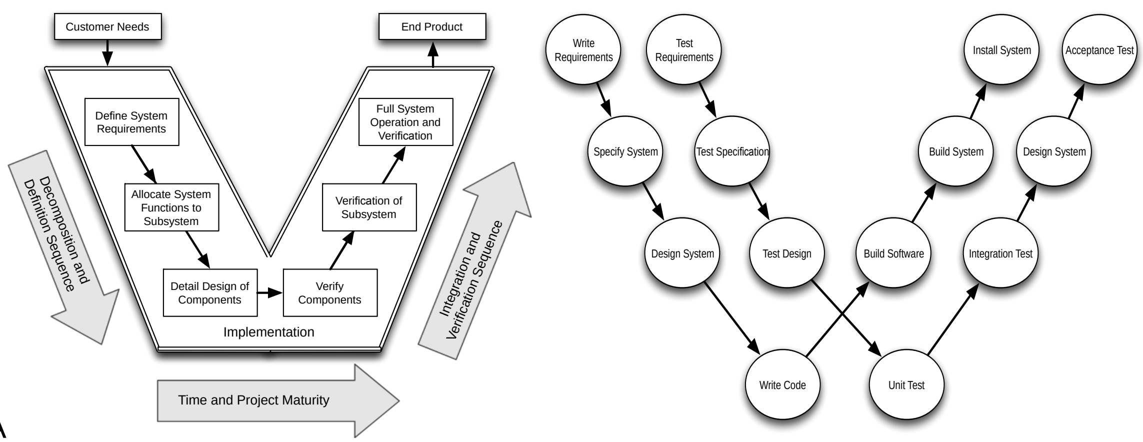


A B

Figure 2-1. Systems engineering process models: A) Waterfall process model and B) Spiral process model.

The “Vee” model provides guidance for planning and realization of projects/programs. The model starts with user needs and ends with a validated end product. The left side of the “Vee” shows decomposition and definition activities from higher level to lower levels (i.e., system to subsystem to component levels). The right side shows integration and verification activities as it moves from lower to higher levels. The transitions between phases are typically done through reviews and project maturity/progression is depicted as model is implemented. However, this model is not adaptable to change. If additional customer needs are added, the model will not be able to account for it, thus, the process needs to start over.

The “W” model is an implementation of multiple “Vee” models and the multiple “Vee” models are implemented in parallel. For example, testing can occur in parallel with program management and progress in parallel. The “W” model is suited for complex systems, however, as the number of “Vee” models increase, the complexity rises and can be difficult to manage.



A B

Figure 2-2. More systems engineering process models: A) “Vee” process model and B) “W” process model.

The NASA systems engineering (SE) engine is specifically designed for complex space systems. The “engine” adds to the “Vee” model by adding optimization and control. There are 17 process activities for system design, realization, and management. The NASA engine is applied throughout the life-cycle phases for NASA projects and shown in Figure 2-3. Table 2-1 summarizes the pros and cons for each systems engineering process models.

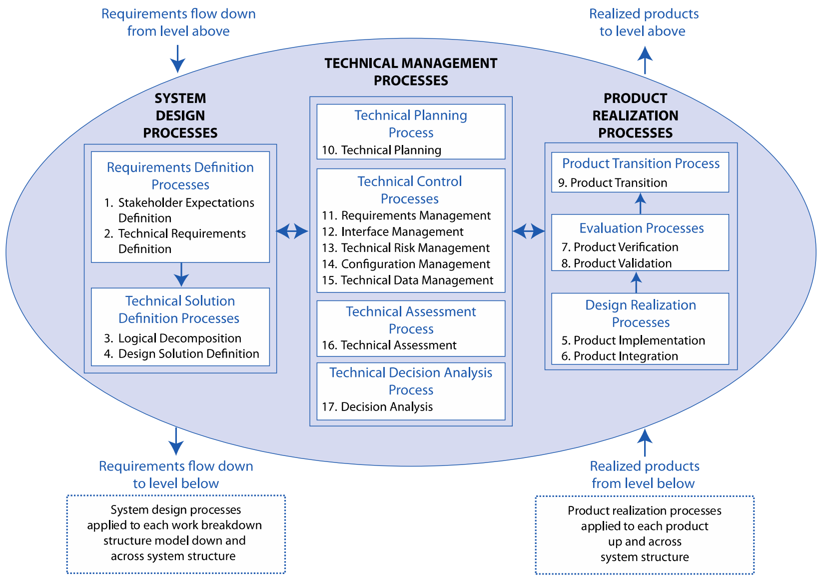


Figure 2-3. NASA systems engineering engine [88].

Table 2-1. Summary of systems engineering process models.

Model	Pros	Cons
Waterfall	Simple and easy to use Clearly defined phases Easy scheduling Suitable for small projects	Does not allow for revisions Difficult to accommodate changes in requirements Not suitable for complex systems Integration and testing only done at the end of the process, thus, difficult to identify any bottlenecks early
Spiral	Risk evaluation Suitable for complex systems Adaptive to changes Incorporates feedback	Costly to implement Requires risk analysis Not suitable for small projects Process is complex
"Vee"	Highly disciplined model Simple and easy to use Easy to manage schedule Reviews and deliverables at each phase	Not suitable for complex system High risk and uncertainty Nota adaptable to change Once in testing phase, difficult to go back and change functionality
"W"	Highly disciplined model Implement other process in parallel Suitable for complex system Each stages are well defined	Difficult to manage High risk due to complexity
NASA SE Engine	Thorough process Suitable for complex systems Resembles both technical development and management	Very complex Time consuming High cost to implement

Project Life-Cycles

In general, the systems engineering process models are implemented throughout a life-cycle process that are typically divided into phases or stages; Needs, Requirements, Design, Development, Verification and Validation (V&V), Utilization (i.e., operations and maintenance) and Support, and Retirement, as shown in Figure 2-4 [89] [90]. First the stakeholders identify the needs, then requirements are established based on the stakeholder needs and constraints. Conceptual designs are developed based on the requirements. Conceptual designs are matured to detailed designs during the Design phase. Prototypes and final designs are developed during Development. During the V&V phase, verification activities such as metrology, assembly integration, and tests (AI&T) are performed. Once V&V is completed, the project will move to Utilization and Support, where the system/product are put into use and operation. Any repairs, replacements, and failures are addressed during this phase. Once the system/product reaches its end of life, the Retirement phase is executed to dispose of the system/product. Transition between phases is typically done through reviews and approval from stakeholders. The life-cycle process is divided into phases to allow the development team to assess their progress, estimate system and project performance, and plan the next. In addition, the division allows stakeholders and decision makers to assess management and technical progress. Like the system process models, many life-cycle processes are developed and implemented by different organizations for their projects. The following sections details selected project life-cycles from various organizations.

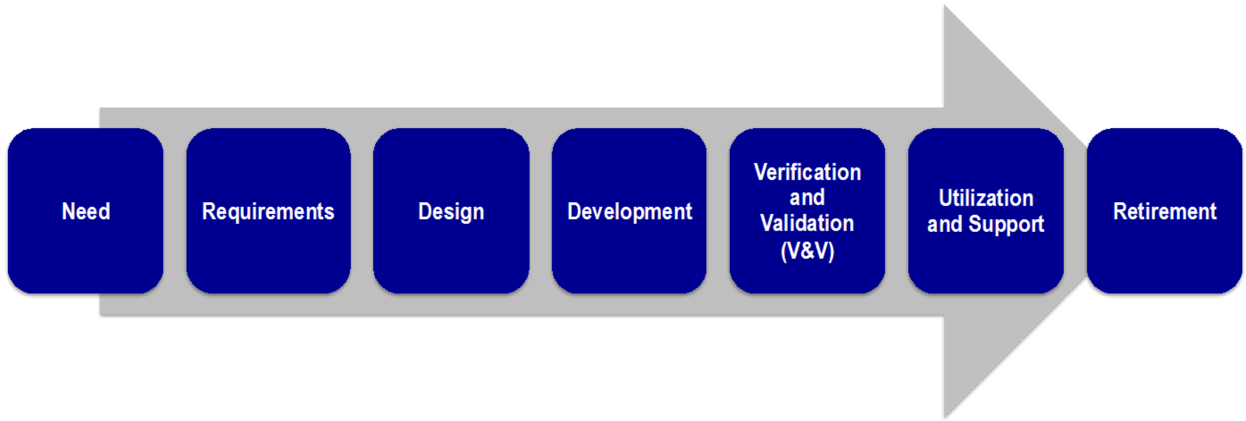


Figure 2-4. General project life-cycle.

INCOSE Project Life-Cycle

The International Council on Systems Engineering (INCOSE) project life-cycle is divided into seven stages; Exploratory Research, Concept, Development, Production, Utilization, Support, and Retirement [89] (Figure 2-5). During Exploratory Research, stakeholders' needs are identified and ideas and technology solutions are explored. The stakeholders' needs are refined and feasible concepts are explored in the Concept stage. System requirements are refined, system is built, and verification and validation of the system are conducted in Development. Production stage produces systems and once systems are produced, they are inspected and verified. After the system is verified, it is put into use during the Utilization and in the Support stage, sustainment of the system are conducted. Finally, in the Retirement stage, the system is stored, archived, or disposed of. For the INCOSE project life-cycle, the transition between the stages are through decision gates, where the decision options are same for all decision gates (shown as triangles in Figure 2-5). The decision options are:

- Proceed with next stage
- Proceed and respond to action items
- Continue this stage
- Return to preceding stage

- Put a hold on project activity
- Terminate project

The INCOSE life-cycle is unique since it allows for returning to preceding stages. Most life-cycles do not allow for returns, such as the Department of Defense (DoD) life-cycle and the project life-cycles by space agencies.

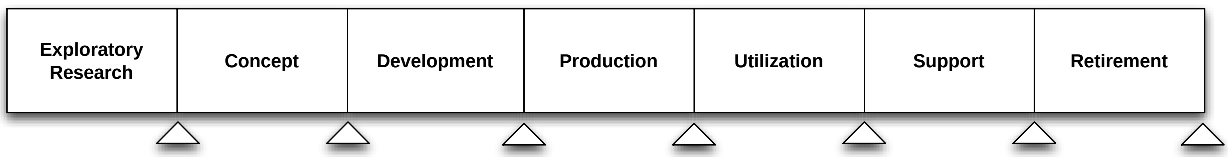


Figure 2-5. INCOSE project life-cycle.

DoD Life-Cycle

The Department of Defense (DoD) life-cycle is divided into five phases; Material Solution Analysis, Technology Development, Engineering and Manufacturing Development, Production and Deployment, and Operations and Support (Figure 2-6) [96]. Material Solution Analysis phase assess potential material solutions and development strategies to meet requirements. During this phase, the metrics are defined to assess desired performances. Technology risks are reduced and decisions on which technologies to be integrated are identified. During the Engineering and Manufacturing Development phase, detailed integrated designs are developed with producibility and operational supportability in mind. The purpose of the Production and Deployment phase is to finalize product support and maintenance plans while initial production commences. Upon successful evaluations and tests of the initial production, the full rate production and deployment takes place. During the Operations and Support the systems are put into use and effectiveness of the systems are assed to ensure requirements are satisfied. Maintenance and repairs are conducted during this phase as well as demilitarization/disposal. The DoD utilizes this life-cycle and adapts accordingly to specific programs. In the operation of the Defense Acquisition System instructions [97], there are four varied models of the life-cycle; hardware intensive, software intensive, combination of hardware and software, and accelerated acquisition. The combination and accelerated models are hybrid models where some of the life-cycle phases are merged/overlapped. The triangles shown in Figure 2-6 represents milestone decisions and the diamond shape represents decision points. In addition, reviews are conducted to assess the progress and influence decisions by the program management.

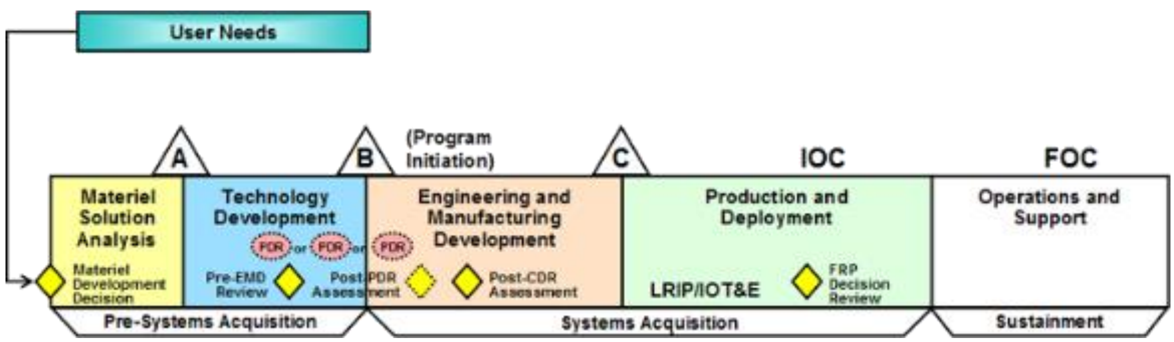


Figure 2-6. DoD project life-cycle [96].

Project Life-Cycles for Space Agencies

Space agencies around the world have adopted and implemented their own mission life-cycle definitions. For example, NASA's project life-cycle is divided into seven phases: Pre-Phase A to Phase F [88]. The European Space Agency (ESA) has a project life-cycle similar to NASA's, which includes seven phases, Phase 0 to Phase F [98]. The Japan Aerospace eXploration Agency (JAXA) also has a similar project life-cycle which includes five phases; Phase 0 to Phase 4 [99]. Other space agencies around the world, such as the Russian Federal Space Agency (ROSCOSMOS), Chinese National Space Administration (CNSA), Indian Space Research Organization

(ISRO), and Korea Aerospace Research Institute (KARI), project life-cycles are not publically available. Therefore, in this research, the life-cycles of NASA, ESA, and JAXA space agencies are presented.

NASA Project Life-Cycle

NASA space missions are typically developed through seven phases of its project life-cycle, Pre-Phase A to Phase F, as shown in Figure 2-7 [88]. The phases are separated by control/progression gates, which are typically reviews and upon completion of the reviews, the phase transitions. These first three phases are categorized as program formulation and the remaining four phases are categorized as program implementation. The program formulation phases initiate the planning of a new project and perform analyses required to formulate the project. The program implementation phases execute the project to ensure the goals and objects of the project are satisfied. The Pre-Phase A produces various mission concepts, draft of system-level requirements, and evaluates the possible missions. After a mission concept review (MCR), the project moves to Phase A, where the concept and technology development plans result in final mission concept and system-level requirements with preliminary concept of operations (CONOPS) are also developed. Upon completion of the mission definition review (MDR), the project transitions to Phase B, which is preliminary design and technology completion where the designs are further matured and the CONOPS are finalized. In addition, preliminary software development, such as simulation and analysis are conducted. Successful completion of the preliminary design review (PDR) leads to Phase C, where the detailed design of the system is finalized, fabrication begins, and early flight software is developed. Once the critical design review (CDR) is completed, the hardware procurement and software coding commences. In Phase D, assembly, integration, verification, and validation (including environmental testing) are performed while satisfying system requirements. As part of this phase, the systems are launched into orbit and commissioned. A flight readiness review (FRR) validates if the system is operation-ready and can be delivered to the launch provider. After successful launch, Phase E involves the day-to-day activities to conduct the mission and to monitor and maintain the system performance as designed and expected. A post-launch assessment review (PLAR) assesses the system to validate mission objectives and operations. During PLAR, it is possible that the project may decide on extending the mission. Upon completion of PLAR, project will move on to Phase F, where system decommissioning disposal plan is implemented to determine the final closeout of the mission. A disposal review (DR) is typically conducted to determine how the de-commissioning will be implemented and executed. At each phase, the NASA systems engineering engine (see Figure 2-3) is implemented.

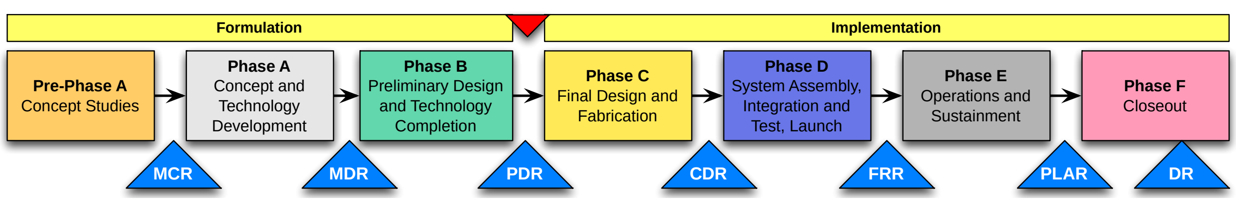


Figure 2-7. NASA project life-cycle.

ESA Project Life-Cycle

The European Space Agency (ESA) has similar project life-cycle to NASA; ESA’s project life-cycle include seven phases, Phase 0 to Phase F, shown in Figure 2-8 [98]. ESA calls Phases 0, A, and B as “preparatory phases” and are mainly focused on i) elaboration of system functional and technical requirements, ii) identification of all activities and tasks required to develop the space and ground segments, and iii) initial assessments of technical and programmatic risks. Phases C and D are referred to as the development phase where all activities are performed to develop the space and ground systems and their products. Phase E comprises of all activities required to operate, utilize, and maintain the deliverable products. Phase F is where the disposal of all space and ground systems occurs. As shown in Figure 2-8, upon successful reviews, the project transitions to the succeeding phase. Table 2-2 summarizes each review during ESA's life-cycle, where from the preliminary requirements review (PRR) to the preliminary design review (PDR) follows a "top-down" approach and from the critical design review (CDR) to the acceptance review (AR) follows a "bottom-up" approach. This approach is a typical systems engineering "Vee" model where the "top-down" sequence starts from the top-level customers and suppliers and continues down to the lowest level suppliers and the "bottom-up" sequence where the lowest level suppliers and continues up to the top level customer and suppliers [98].

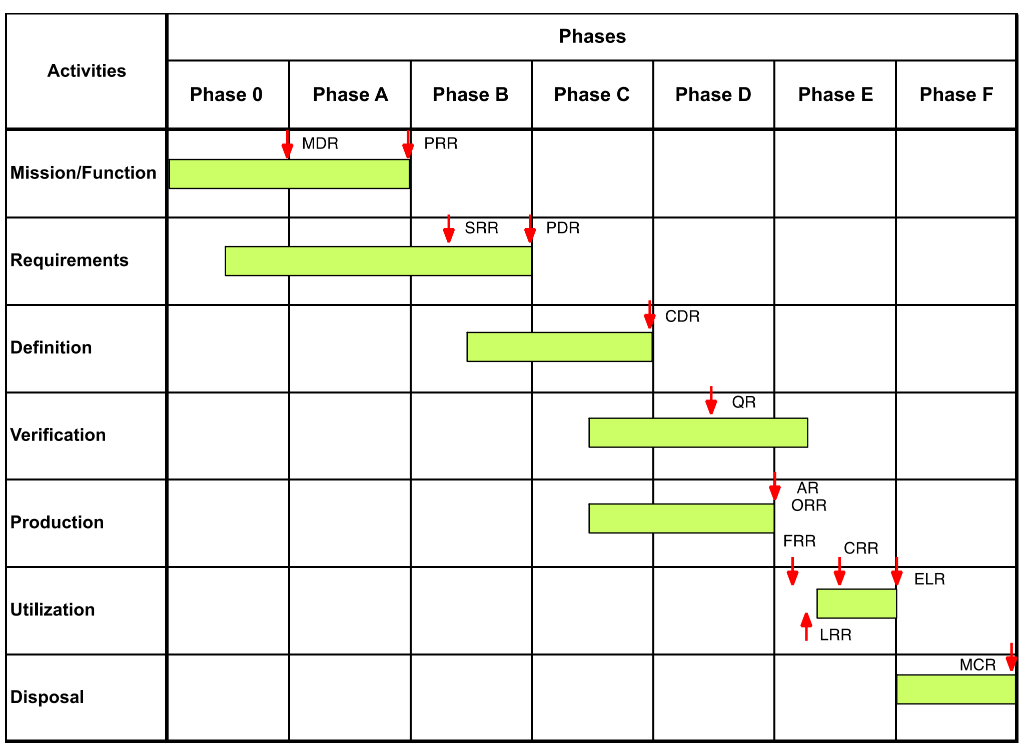


Figure 2-8. ESA project life-cycle.

Table 2-2. ESA review descriptions.

Review	Objectives
Mission Definition Review (MDR)	Confirm mission requirements and preliminary programmatic assessment
Preliminary Requirements Review (PRR)	Confirmation of preliminary requirements, technical and programmatic feasibility
System Requirements Review (SRR)	Assessment of preliminary design definition
Preliminary Design Review (PDR)	Verification of preliminary design against project and system requirements
Critical Design Review (CDR)	Assess readiness of development testing and pre-qualification testing
Qualification Review (QR)	Verify all qualification tests are completed and satisfy requirements
Acceptance Review (AR)	Verify all acceptance tests are completed and satisfy requirements
Operational Readiness Review (ORR)	Assure inter-operability between space and ground segment, prior or after delivery
Flight Readiness Review (FRR)	Certify total launch configuration and supporting systems are ready
Launch Readiness Review (LRR)	Declare readiness for launch of all flight and ground systems
Commissioning Result Review (CRR)	Verify all components of the system are performing to parameters
End of Life Review (ELR)	Verify completion of mission and operations. Configure for disposal
Mission Close-out Review (MCR)	Ensure all mission disposal activities are completed

JAXA Life-Cycle

The Japan Aerospace eXploration Agency (JAXA) was established by merging the Institute of Space and Astronautical Science (ISAS), the National Space Development Agency of Japan (NASDA), and the National Aerospace Laboratory of Japan (NAL) in 2003. For JAXA, there are several life-cycles shown; i) one that shows five phases, Phase 0 through Phase 3 and Operations phase [99], ii) one that follows a standard systems engineering life-cycle, concept study, concept development, project formulation, preliminary design, final design, production and testing, launch operations, and operations [100], and iii) one that follows a life-cycle similar to NASA's, Pre-Phase

A through Phase F [101]. In this paper, JAXA's life-cycle with five phases are used for discussions and is shown in Figure 2-9. The five phases are: Phase 0 is where the conceptual studies are performed and conceptual designs and project plan decisions are developed, Phase 1 and Phase 2 are referred to as the preliminary design and critical design phases where the design commences and matures, Phase 3 is when the designs are manufactured and tested for verification, and the Operations phase that includes the launch and mission operations. Similar to NASA's and ESA's project life-cycle, JAXA utilizes reviews to assess the project progress and proceed to the succeeding phases upon successful reviews. The reviews in JAXA's project life-cycle are; Mission Definition Review (MDR), System Requirement Review (SRR), System Definition Review (SDR), Preliminary Design Review (PDR), Critical Design Review (CDR), Post-Qualification Review (PQR), Launch Readiness Review (LRR), and Post-Flight Review (PFR). In addition to the reviews, JAXA conducts safety review (SR) at the end of each phase and a post-Phase 3 safety review if necessary. The safety reviews are conducted to confirm safety requirements established according to each hazard identified and its compliance with requirements.

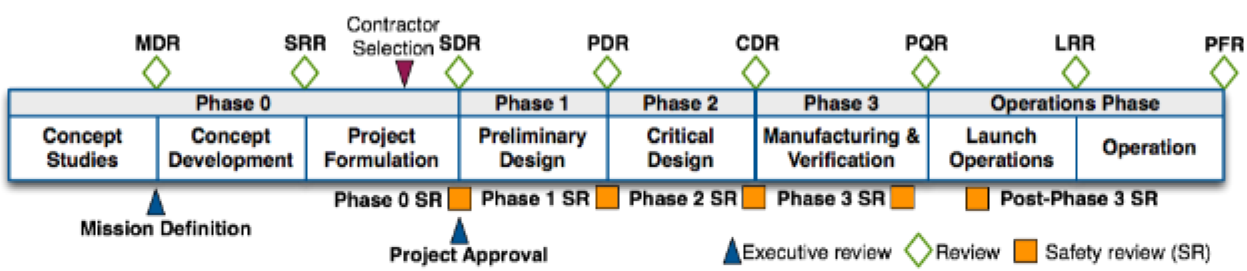


Figure 2-9. JAXA project life-cycle.

Project Life-Cycle Discussion

Based on the various project life-cycles, it is evident that the three space agencies utilize reviews to transition between phases/stages. Some have reviews throughout the phases/stages, however, most critical decisions are made during phase transition reviews. One evident fact is that these life-cycle processes are extremely time consuming. For example, NASA utilizes the systems engineering engine (17 activities) in each of the seven life-cycle phases. Iterations within the engine results in more than 120 process activities in the project life-cycle, which clearly shows the amount of time that is required to develop space systems (on top of excessive addition of functionalities in one spacecraft).

In addition to their own project life-cycle, all three space agencies have their own classifications of smaller spacecraft/payloads; NASA utilizes four risk levels for the payloads, Class A to Class D [102], and ESA [103] and JAXA [101] have three mission classifications (Large-size mission, or L-class; Medium-size mission, or M-class; and Small-size mission; or S-class). Table 2-3 summarizes small spacecraft/payload missions from each agency. As shown in Table 2-3, the cost and duration are significantly higher than what CubeSat class satellites may require.

Another key consistency in the project life-cycles by NASA, ESA, and JAXA is that their entire life-cycle follows a typical systems engineering process, where the need and goals are first identified, the requirements are developed and the system is designed and developed based on the requirements. The developed system is then verified and upon verification, the system is put into operation. During operation, the system is validated and maintained to satisfy the mission objectives. After mission objectives are accomplished, the system is decommissioned and disposed. Essentially, the life-cycle starts with a top-level need and it gets decomposed into component levels through design then these components gets recomposed into subsystem and system level through development, verification, and validation. Based on the life-cycles by NASA, ESA, and JAXA, it is critical that a project life-cycle must follow an end-to-end systems engineering process. Many space systems must go through rigorous processes due to its extreme cost and risk, however, these project life-cycle processes are not entirely necessary for containerized satellites. In addition to the project life-cycles, it is critical to identify the engineering activities that are needed throughout the project life-cycle. The engineering activities conducted by NASA, ESA, and JAXA are presented in the next section.

Table 2-3. Summary of small spacecraft/payloads for space agencies.

Name	Mass	Start Date	Launch Date	Years##	Cost
NASA – Class D missions [102]					
SNOE [104]	120 kg	Spring 1994	Feb 1998	4	4.3 million (USD)
TERRIERS [105]	125 kg	Spring 1994	May 1999	5	4.3 million (USD)
ESA – S-Class missions [103]					
CHEOPS [106]	58 kg	Oct 2012	June 2019**	7	50 million (Euro)
SMILE [107]	TBD	Sept 2015	Dec 2021**	6	92 million (Euro)
JAXA – S-Class missions [101]					
INDEX [108]	60 kg	1999	Aug 2005	6	4 million (USD)

- Represents design and development time from project start to launch
** - Projected launch dates

Engineering Activities

It is critical to perform various engineering activities throughout the project life-cycle to ensure higher mission success for space missions (also known as mission assurance). Similar to the project life-cycle, the space agencies and organizations have adopted varying definitions of mission assurance. NASA defines mission assurance as “Providing increased confidence that applicable requirements, processes, and standards for the mission are being fulfilled [109].” The U.S. DoD defines mission assurance as “A process to protect or ensure the continued function and resilience of capabilities and assets – including personnel, equipment, facilities, networks, information and information systems, infrastructure, and supply chains – critical to the performance of DoD mission essential functions in any operating environment or conditions [110].” ESA defines space product assurance as “To ensure that space products accomplish their defined mission objectives in a safe, available and reliable way [111].” JAXA defines mission assurance as “An operation action performed throughout the development and operation of spacecraft in order to ensure the mission success [112].” In this section the engineering activities performed throughout the project life-cycle by the space agencies (NASA, ESA, and JAXA) are presented. In addition, a survey was disseminated to the small satellite community (including academia, industry, and government agencies) to understand the engineering activities performed throughout their project life-cycle [113] and the results are presented.

Engineering Activities – Space Agencies

NASA, ESA, and JAXA have developed and utilized guidelines and standards for their space missions. NASA utilizes the NASA Procedural Requirements (NPR) and NASA systems engineering handbook, ESA utilizes the European Corporation for Space Standardization (ECSS), and JAXA utilizes JAXA Management Requirements (JMR) and JAXA Engineering Requirement and Guideline (JERG) documents throughout their project life-cycles. In these documents, each space agency lists the different engineering activities and product artifacts required for each phase of the project life-cycle. While each space agency implements its own project life-cycle, many of the engineering activities performed throughout their life-cycles are similar. To demonstrate this, Table 2-4 summarizes the various engineering activities performed by each space agency throughout their project life-cycles. The engineering activities are organized into five phase: concept, preliminary design, critical design, verification and validation, and operations and disposal. For each phase, the corresponding phases from each project life-cycles are also shown.

The concept phase activities include defining mission objectives, performing feasibility studies on mission concepts, and defining success criteria. High level requirements are developed with constraints and initial cost estimates and schedule are determined. The concept phase activities are focused on project formulation; how the mission goals will be realized into systems with inputs from the stakeholders. After the concept phase, the preliminary design phase activities include the initial design of the system that includes the internal and external interfaces as well as risk, reliability and safety assessments. Majority of the technical plans for the later stages are developed: assembly and integration plans, verification and validation (V&V) plans, operations plans, and disposal plans. Once the reviews are successfully completed, the designs are matured into the final design of the systems and the life-cycle transitions to the critical design phase. Prototypes and engineering development units (EDUs) of critical systems are developed and tested during this phase before the critical design review. The plans are further matured while the launch site and launch preparations commence. In the verification and validation (V&V) phase, the systems are integrated and assembled with various V&V tests performed. For the tests, each space agency has their own testing guidelines/standards: NASA utilizes the General Environmental Verification Standard (GEVS) [114], ESA utilizes the ECSS Space engineering: testing [115], and JAXA utilizes the spacecraft testing standard [116]. In all three testing guidelines, the systems go through rigorous V&V tests including tests in on-orbit like environments. After successfully completing the V&V phase, the system is launched into space and on-orbit operations are conducted. Data obtained throughout the mission operations are analyzed and upon completion of the mission operations, the disposal plan is executed in the disposal phase. Additional data are analyzed, processed, and shared with stakeholders

Table 2-4. Summary of engineering activities for space agencies.

Phases	NASA: Pre-Phase A & Phase A	ESA: Phase 0 & Phase A	JAXA: Phase 0
Concept phase	Define mission objective(s) Identify mission concept(s) Perform feasibility studies on mission concept(s) Define success criteria and performances Develop mission requirements Identify constraints	Develop requirements verification matrix Top level work breakdown structure Develop cost and schedule Identify risks Identify personnel and responsibilities Develop plans for succeeding stages Reviews	
Phase	NASA: Phase B	ESA: Phase B	JAXA: Phase 1
Preliminary design phase	Preliminary design of systems Trade studies Identify internal and external interfaces Design documents Update cost and schedule Update risks Reliability assessment Safety assessment	Budgets with margins Mature plans Assembly and integration plans Verification and validation (V&V) plans Operation plans Disposal plans Project management plans Refine WBS Reviews	
Phase	NASA: Phase C	ESA: Phase C	JAXA: Phase 2
Critical design phase	Detailed design of systems Finalize interfaces Development and testing of critical items Launch site and launch preparations Update cost and schedule Update risks Update reliability assessment	Update safety assessment Mature plans Assembly and integration plans V&V plans Operations plans Disposal plans Project management plans Reviews	

Table 2-4. Continued.

Phase	NASA: Phase D	ESA: Phase D	JAXA: Phase 3
Verification and validation phase	Integration and assembly V&V tests (hardware and software) Component testing Subsystem testing System testing Environmental testing Update risks	Iterate design (if necessary) Launch preparations Mature plans Operations plans Disposal plans Document lessons learned Reviews	
Phase	NASA: Phase E and F	ESA: Phase E and F	JAXA: Operations Phase
Operations and disposal phase	Launch On-orbit operations Mission verification and validation	Finalize and implement disposal plan Data analyses Document lessons learned Reviews	

The engineering activities shown in Table 2-4 are applicable to many types of space missions, robotic (non-human space flight) missions to human space flight missions. The top-level product breakdown structure (PBS) of the space mission is divided into three systems: space system, ground system, and launch system (shown in Figure 2-10). Based on the type of space mission, the elements of the PBS as well as the work breakdown structure (WBS) differs; the WBS defines relationships between all project elements (including the PBS) necessary to successfully complete the space mission.

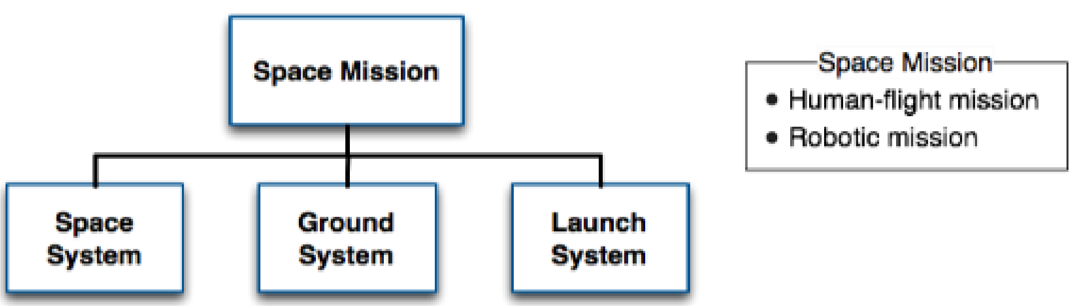


Figure 2-10. Product breakdown structure (PBS) of space missions.

The containerized satellites are non-human space flight (i.e., robotic) missions and not all engineering activities shown in Table 2-4 are applicable. Additionally, the containerized satellites are typically launched as secondary payload, thus, have no control of the launch system. Therefore, the engineering activities and the project life-cycles from the space agencies need to be tailored for containerized satellites. In addition to the tailoring of the engineering activities, the engineering activities performed by the small satellite community must be understood. In order to understand the engineering activities performed by the small satellite community, a survey was developed and disseminated [113].

Engineering Activities – Small Satellite Community

In order to identify engineering activities performed by the small satellite community throughout their project life-cycle, a survey was designed and disseminated to the community; i.e., to academia, industry, space, and government agencies (both domestic and international). To prevent redundant responses, the survey inquired regarding containerized satellites and aimed for a single response per organization (e.g., a small team, group, or could be as large as an entire division or company). The survey was organized into five sections: i) past and planned launches, ii) “25-Year-Rule”, iii) engineering activities, iv) commercial-off-the-shelf (COTS) and in-house components, and v) voluntary questions.

Section 1 questions inquired about basic spacecraft characteristics (i.e., size, mass, status, etc.) of past and planned launches. Section 2 inquired about the “25-Year-Rule” and tasks performed to satisfy the guideline. The “25-Year-Rule” is a debris mitigation guideline that states that spacecraft and upper stages in LEO must be disposed within 25 years after completion of the mission [42]. However, this guideline is not enforced internationally, therefore, enforcement of the guideline is left to each nation and its governing agency(ies). Section 3 inquired about the participant's various engineering activities throughout the spacecraft mission life-cycle. Section 4 inquired about the usage of COTS and in-house built components in the development of the spacecraft. Section 5 were optional and inquired about the mission objectives, organization affiliation, and project cost of missions. The survey was disseminated to the small satellite community through mailing lists (e.g., CubeSat, AMSAT, and working groups of INCOSE and IAA) and personal contacts starting January 2015 and continued for three months. The survey results were published in Reference [113]. The survey's flow chart and the questions are detailed in Appendix B.

Survey Results

There were a total of 200 survey links opened, however, not all participants responded. In addition, the number of responses decreased as the respondents progressed through the survey; 121 responses for Section 1, 104 responses for Section 2, 95 responses for Section 3, 88 responses for Section 4, and 73 responses for Section 5. The survey was anonymous and was designed to only ask non-proprietary information, unless the participants decided to disclose their information inside the survey. The average duration of the survey was 35 minutes, however, there were several responses that were over 4 hours, which indicated that some responses were not answered in one sitting.

Section 3 of the survey asked the participants to identify and briefly describe their engineering activities performed throughout their project life-cycle. The engineering activities were categorized into eight activities: 1) simulations and analysis, 2) reliability analyses, 3) requirements verification and traceability, 4) documentation control and management, 5) hardware verification and validation, 6) software verification and validation, 7) internal and/or external review, and 8) others. Figure 2-11 shows the various engineering activities the survey respondents performed, where the percentages of respondents performing each engineering activity are also shown. For example, 64% of the respondents stated that they performed simulation and analysis and 63% of the respondents said they performed hardware verification and validation (V&V) for their spacecraft. The simulations and analyses, hardware V&V, and software V&V were the most common engineering activities performed by the survey respondents, while performing reliability analyses were the least common activity. Table 2-5 shows these engineering activities and selected responses from the survey participants.

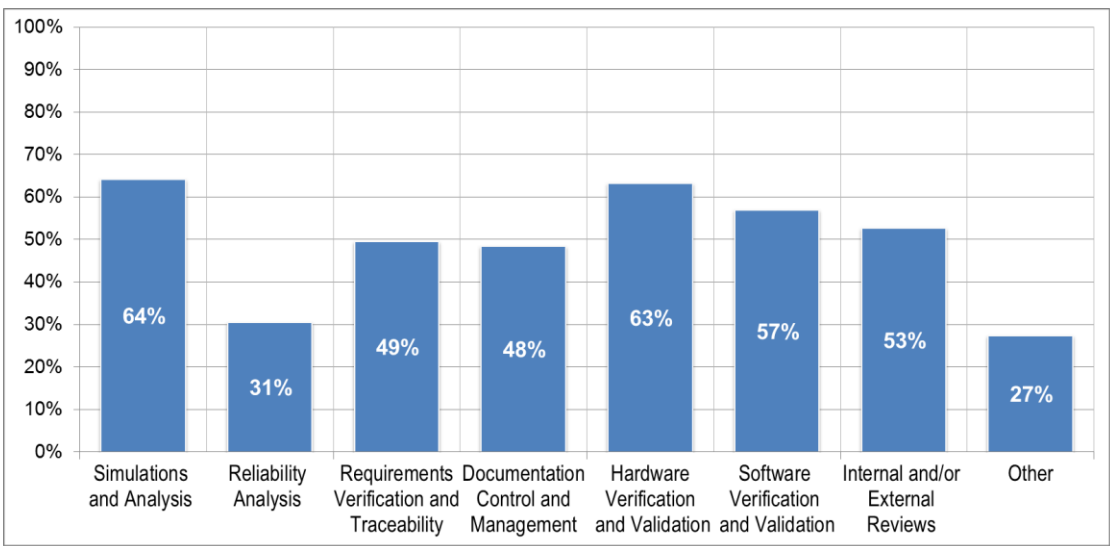


Figure 2-11. Engineering activities performed by the small satellite community.

Table 2-5. Engineering activities performed by the small satellite community.

Engineering activity	Selected response from the survey question 3
Simulations and Analyses	Orbital simulations (NASA DAS, STELA, STK) Analysis of optimal orbits based on requirements (mission, attitude, and comms) Structural and thermal analyses (FEA, THERMICA) Antenna modeling and system simulation

Table 2-5. Continued.

Reliability Analysis	Fault Tree Analysis (FTA) Anechoic chamber verification Using component data Duration testing
Requirements Verification and Traceability	Part of systems engineering process Utilizing software (e.g., RMTrak) Subject to review board (requirements document/review) AES9100 Quality Assurance Working groups to develop ICD and spacecraft target specification Product assurance plan Achieve the requirement and correction
Documentation Control and Management	ESA standards Versioning and server system (e.g., SVN, Git) Subject to review board and change board to manage changes AES9100 quality assurance Documentation and configuration management plan Custom procedure
Hardware Verification and Validation (including environmental tests)	Based on manufacturer data Environmental test facilities Vibration, thermal vacuum, thermal bake-out, radiation, shock Verification plan procedures Functional testing Test to qualification level on EM and acceptance level on FM Hardware and software-in-the-loop
Software Verification and Validation	End-to-end test and with hardware Utilizing software simulator (e.g., ptran, marc, solid-works, Nastran) Hardware and software-in-the-loop Emulator and real time but limited use, build it test it, code/recode test NEVER allow last minute small changes
Internal (peer) and/or External (subject matter expert) Reviews	Multiple design reviews (internal and external) Professors and working groups External reviews (French and ESA experts) PDR, CDR, MTR, FRR Sometimes non-software expert found bugs than software team NASA GSFC team review
Others	None Launch and in-orbit experience None – Regulate LVs and not payloads yet None – we are government regulator, thus, mission success is not regulated

The simulations and analyses include orbital simulations performed to determine the best suited orbits and the system behavior in those orbits. Respondents also stated that they perform structural and thermal analyses of the system as well as simulations of the antenna performance. For reliability analyses, the survey respondents stated they perform fault tree analysis to identify root causes of failures as well as tests, such as anechoic chamber verification and duration test. For requirements verification and traceability, respondents stated they utilize quality assurance standard AES9100 as well as product assurance plan. Other respondents stated that they utilize reviews and working groups as well as software for requirements verification and traceability. For documentation control and management, respondents stated that they utilize a software and server system as well as document configuration management plan. Some respondents stated they have custom procedure within their own for documentation control. The majority of the respondents that performed hardware verification and validation tests listed different environmental tests they performed: vibration tests, thermal vacuum and thermal bake-out tests, radiation, and shock tests. In the software verification and validation, some respondents stated that they perform the verification and validation with hardware and not software alone. Some respondents stated they utilize software simulators and emulators to perform software verification and validation. For the reviews, respondents stated they hold internal (peers) and external (subject matter experts) reviews and multiple reviews throughout the project life-cycle. In the "Other" category, some respondents stated that they performed no engineering activities and some stated that they regulate launch vehicles and not spacecraft.

The survey results were further analyzed in an effort to identify, if any, relationships between the operational status(es) of the containerized satellite(s) launched by each respondent and their engineering activities and/or their affiliation. First, the survey results were sorted to retain the responses that included the operational status of the launched containerized satellites. Next, these results were further sorted into responses that included respondent's affiliation and engineering activities. As a result of this sorting, 28 responses were identified; 18 of them were from academia, four were government respondents, four respondents from industry, and two were private respondents. Inspecting the status and the respondent's affiliation, each affiliation had similar proportion of containerized satellites that were operational, non-operational, and deorbited, thus, there were no clear affiliation relationships that could be identified. In this study, non-operational spacecraft referred to those that are not functional and do not communicate. Following this, the engineering activities were examined along with the affiliations. All but one of the 28 responses had performed hardware verification and validation, while only eight had performed reliability analyses. The other engineering activities add had more than 20 responses each: internal/external reviews and software verification and validation had 23 responses each, requirements verification and traceability had 22 responses, and simulation and analysis and document control and management had 21 respondents. The responses showed that there is a lack of reliability analyses performed by the small satellite community, thus, there is a need to perform more reliability analyses. This result is shown in Figure 2-12.

Further examination of the survey results showed that the respondents that have performed all the listed engineering activities (i.e., seven activities excluding "Other") in the survey have at least one operational containerized satellite currently in orbit. Another observation seen from the survey results is that as more engineering activities are performed, there are more operational containerized satellites regardless of affiliation. However, similar results were seen for non-operational satellites. This states that the engineering activities are performed at random and proves that there is a lack of structured process. The status of containerized satellites based on engineering activities is shown in Figure 2-13.

Affiliation	Operational Status and Count				Mission Assurance Activities and Count								
	O	N	D	U	SIM	REL	REQ	DOC	H/W V&V	S/W V&V	REV	Other	Count
Industry	>3				x		x	x	x	x	x		6
Industry	>3		>3		x		x	x	x	x	x		6
Private	>3	2	1		x	x	x	x	x	x		x	6
Academia	3	1	1		x		x	x	x	x	x		6
Academia	3						x	x	x	x	x	x	5
Academia	3				x	x	x	x	x	x	x		7
Academia	2	1			x		x	x	x	x	x		6
Academia	2		1		x	x	x	x	x	x	x		7
Industry	2				x		x	x	x	x	x		6
Academia	1				x				x	x			3
Academia	1	1			x	x		x	x	x	x		6
Academia	1		1		x		x	x	x	x	x		6
Government	1								x	x			2
Private	1						x		x	x	x		4
Academia	1	1			x		x	x	x	x	x		6
Academia	1	3			x	x	x	x	x		x		6
Academia	1	>3			x	x	x		x	x	x		6
Government	1	3	3		x	x	x	x	x	x	x		7
Industry	1	3			x	x	x	x	x	x	x		7
Academia		1	1									x	0
Academia		1					x	x	x				3
Academia		1			x				x		x		3
Government				1	x		x	x	x	x	x		6
Academia		1							x		x		2
Academia		1					x	x	x	x	x		5
Academia		1			x		x	x	x	x	x		6
Academia			1		x		x	x	x	x	x		6
Government		1	>3		x		x	x	x	x	x		6

O: Operational; N: Non-operational; D: Deorbited; U: Unknown
SIM: Simulation & analysis; REL: Reliability analyses; REQ: Requirements verification; DOC: Documentation
H/W V&V: Hardware verification and validation; S/W V&V: Software verification and validation; REV: Reviews

Figure 2-12. Analysis of survey responses.

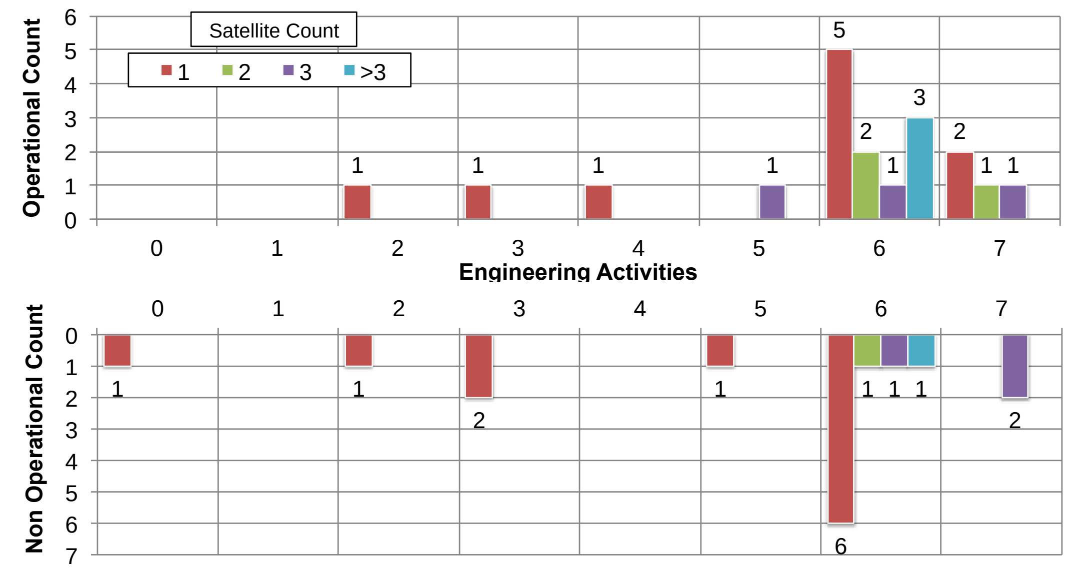


Figure 2-13. Status of containerized satellites based on engineering activities.

Limitations of the Study

As the survey results were being examined, several limitations were noted. One of the major limitations that was observed was the lack of survey responses. While the survey collected valuable responses, the number of responses was insufficient to identify clear relationships to the operational status of the containerized satellites. Other limitations were the anonymity and the lack of detailed responses. The survey was designed such that non-proprietary information was asked, however, by doing so the results lacked details in some of the responses. In addition, due to the anonymity, when there were questions regarding some of the responses, there was no way to get in contact with the respondents unless their contact information was provided.

Engineering Activities Discussion

The engineering activities performed by the space agencies throughout their project life-cycle are applicable for any space missions, robotic and non-robotic missions. In addition, the engineering activities by the space agencies include activities to support the launch systems (e.g., launch vehicles). The containerized satellites are all robotic missions and do require activities to support launch systems, therefore, the engineering activities from the space agencies must be tailored for these containerized satellites.

A survey was designed and disseminated to the small satellite community (i.e., academia, industry, space and government agencies, both domestic and international) to inquire about their containerized satellites. One of the questions in the survey inquired about the engineering activities performed by the respondent throughout their project life-cycle. The use of hardware V&V, software V&V, and internal/external reviews were the most common engineering activities performed by the survey respondents, while performing reliability analyses were the least common activity. The survey showed that as more engineering activities are performed, there is a higher probability of mission success. However, even when engineering activities are performed, there are times that missions fail. Furthermore, the survey did not ask the respondents when in the project life-cycle these specific engineering activities were performed and based on the survey responses, it was evident that majority of the containerized satellite developers were not following a structured process. Specific engineering activities must be performed at specific times during the project life-cycle but without a structured process the engineering activities appear to be performed at random times during the project life-cycle. The results from the survey emphasized the need for a structured process for the containerized satellite developers.

CHAPTER 3
PROJECT LIFE-CYCLE FOR A CLASS OF SMALL SATELLITES

While conventional space missions may require the development of the space, ground, and launch systems, typical containerized satellite missions do not require the development of all three systems. The containerized satellites are typically launched as secondary payloads and launched out of containers, thus, do not require the development nor perform activities for the launch systems. Additionally, containerized satellite missions are all robotic space missions that do not require special space systems necessary for human-flight missions.

The size, weight, and power (SWaP) constraints imposed by the CubeSat specifications have limited their technical performances and design spaces. The SWaP constraints are driven by the various containers that deliver these satellites into orbit, specifically, these containers constrain the size and weight of the satellites which influence the power of the satellites. For example, Tyvak's Nanosatellite Launch Adapter System (NLAS) can accommodate up to a 6U form factor (340.8.0 mm x 246.3 mm x 120.0 mm and 14 kg) [117] , therefore, the satellite designs are limited to be within the 6U form factor. These containers interface one or more satellites to the launch vehicle and prevent any harm to the launch vehicle and to others in the same container. Due to the advancements of these containers, the number of "containerized" satellites launched into orbit (specifically LEO) have increased. These factors, among others, significantly influence the overall design and development of these containerized satellites and hence motivate the need to reimagine the mission life-cycle of these class of satellites. A containerized satellite is defined as follows:

“A containerized satellite is any satellite that is enclosed in a container that interfaces the satellite to the launch vehicle. Such a container (e.g., P-POD [21], X-POD [26], ISIPOD [27], etc.) may contain one or more satellites and is designed to prevent harm to the launch vehicle (and other satellites) as well as deploy the containerized satellite(s) into orbit”.

These containerized satellite missions do not require the development nor perform activities to support the launch systems, but these missions typically have to develop space and ground systems. In general, the ground system is only developed once, while the space system is required to be developed each mission. Both space and ground systems gets decomposed and recomposed throughout the development. The systems are decomposed into subsystem and component levels during concept, preliminary design, and critical design. The components are then recomposed and integrated into subassemblies/subsystems and ultimately into the system during verification and validation (see Figure 3-1). Once the systems complete verification and validation, they are put into operations until retirement and disposal.

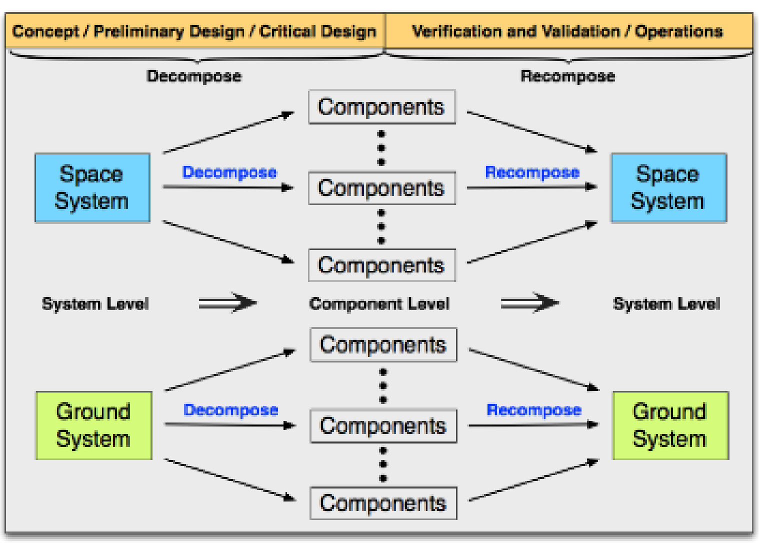


Figure 3-1. System decomposition and recomposition throughout the development.

Based on the review of the existing project life-cycles and engineering activities in Chapter 2, seven key engineering activities were identified that are critical during the decomposition and recomposition of the systems. These engineering activities are organized and mapped to life-cycle phases shown in Figure 3-2. The simulations and analyses such as orbit, attitude, power, battery, communication, structural, and thermal are performed at the concept phase and through preliminary and critical designs to identify appropriate components for designs to satisfy requirements. The reliability analyses are performed during preliminary and critical designs to identify high risk items and the results from the analyses are used to determine risk mitigation strategies. The requirements verification and traceability activities begin in the concept phase when the top-level requirements are defined. The requirements are refined and finalized as the project proceeds and the requirements are satisfied and verified prior to launch and operations. The requirements are typically listed in a requirements verification matrix, where the requirements along with its verification methods (e.g., analysis, inspection, design, and test) are identified. Documentation control and management begins early in the life-cycle when everything from mission objectives to concept studies are documented and archived. Requirements documents, design documents, interface control documents, verification and validation documents, and review documents are developed as the life-cycle progresses. The hardware and software verification and validation (V&V) activities begin in preliminary design when prototypes are developed and continue through as the prototypes mature into engineering development units and flight units. The hardware and software are verified during V&V phase and are validated during on-orbit operations. Reviews, typically with external subject matter experts and stakeholders, are conducted between each phase and internal (peer) reviews are conducted throughout the phases. The external reviews are critical events conducted to examine if the project can transition to the proceeding phase based on the feedback from the subject matter experts and the stakeholders.

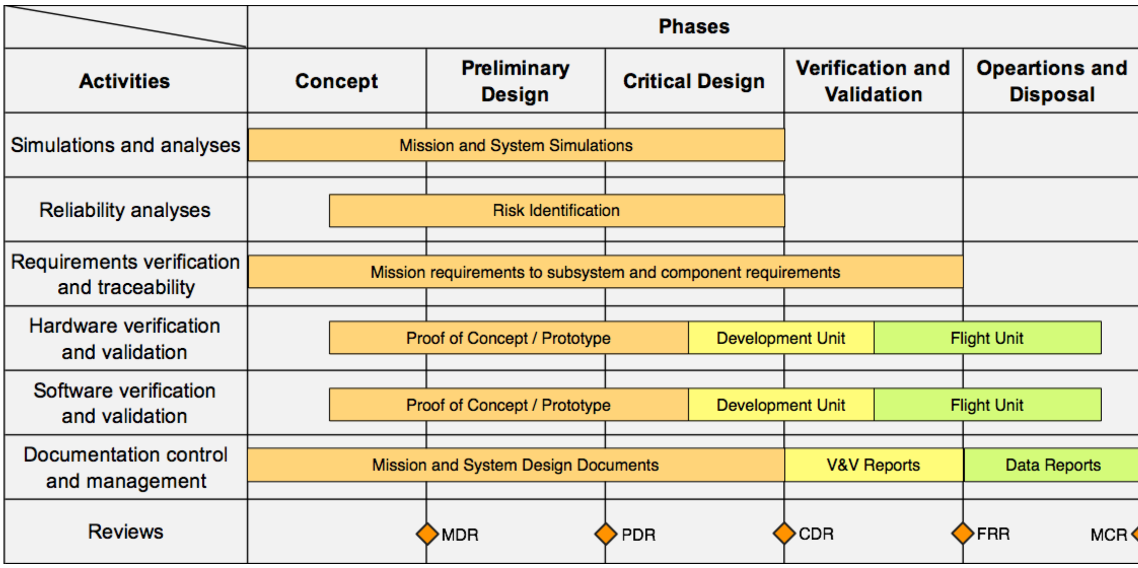


Figure 3-2. Engineering activities mapped to life-cycle phases.

With the review of the existing project life-cycles and the seven key engineering activities, previous research efforts, and the lessons learned from them, a systems engineering based comprehensive mission life-cycle for containerized satellites was developed. The mission life-cycle, here in referred to as the Containerized Satellite Mission Life-Cycle, is presented in the following sections.

The Containerized Satellite Mission Life-Cycle

The Containerized Satellite Mission Life-Cycle is a comprehensive mission life-cycle for containerized satellites and is developed particularly to cater towards the paradigm change in the inception, design, development, operation, and retirement of the containerized satellites. The Containerized Satellite Mission Life-Cycle differs from existing project life-cycles by providing the framework for fundamental procedures and protocols best suited for containerized satellites while streamlining the process to reduce cost and time. It is developed based on (i) a thorough review of existing project life-cycles of various space agencies and (ii) engineering activities performed by these space agencies and the small satellite community.

The Containerized Satellite Mission Life-Cycle is organized into six phases, Pre-Phase I through Phase V, and is shown in Figure 3-3. An appreciation for systems engineering practices and procedures is a significant requirement for carrying out a containerized satellite mission, thus Pre-Phase I is a systems engineering training activity geared primarily for new entrants and/or academic institutions. Phase I identifies the mission concept and develops a preliminary design. Phase II matures the design into a detailed design and Phase III addresses component/subsystem development, integration, and testing. Phase IV addresses the system level assembly, environmental testing, and launch preparation. Phase V addresses the post launch operations up to and including retirement and disposal.

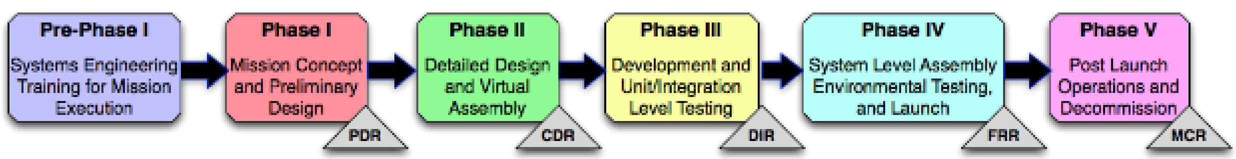


Figure 3-3. The Containerized Satellite Mission Life-Cycle.

Pre-Phase I – Systems Engineering Training for Mission Execution

For containerized satellite missions executed by new space entrants or academic institutions, the team may be comprised primarily of individuals with limited or no systems engineering knowledge. For such teams, the Pre-Phase I activities focus on imparting systems engineering knowledge through training. The systems engineering training is implemented to teach the team about the life-cycle of a spacecraft mission from concept to grave (i.e., concept, design, fabrication, verification and validation testing, operations, and decommissioning). The systems engineering training can be offered as a comprehensive course, short course, or an extensive seminar/workshop. During this phase, the team may recruit from academia and/or industry professors, scientists, and engineers who have an established expertise in one or more of the following areas:

- Payload expertise
- Communications engineering – hardware and protocol
- Electrical and electronics engineering
- Power systems engineering – harnessing, storage and distribution
- Computer science and engineering
- Mechanical design and optimization engineering
- Navigation and control engineering
- Propulsion engineering
- Systems engineering

The training identifies the need, develops requirements, designs and develops the systems based on the requirements and constraints, performs verification and validation on the systems, launches and operates the systems, retires the systems, and post-process the flight data. As part of the training, the following activities are performed:

Phase I – Mission concept and preliminary design
1. Identify a feasible mission for a containerized satellite
2. Develop a mission concept of operations
3. An overview understanding of containerized satellite subsystems
4. Requirements gathering process for a containerized satellite mission
5. Develop work breakdown structure (WBS)
6. Allocating and deriving mission requirements
7. Trade study of components and subsystems available as COTS products
8. Mission concept and preliminary design report and review

Phase II – Detailed design and virtual assembly
9. Detailed design process for each subsystem
10. Detailed design report and review

Phase III – Development and unit/integration level testing
11. Acquisition and development of components and subsystems

12. Adequate test planning for each component and subsystem
13. Assembly, integration, and testing of components and subsystems
14. Assembly, integration and testing report

Phase IV – System level assembly, environmental testing, and launch

15. System level verification and validation tests, including environmental tests
16. Verification and validation test report and flight readiness review
17. Delivery and launch vehicle integration

Phase V – Post launch operations and decommission

18. Perform on-orbit operations to validate mission objectives
19. Implement decommissioning process
20. Post-process data and document lessons learned

The above activities follows Phase I through Phase V of the Containerized Satellite Mission Life-Cycle, specifically activities 1 through 8 are performed in Phase I, activities 9 and 10 are performed in Phase II, activities 11 through 14 are performed in Phase III, activities 15 through 17 are performed in Phase IV, and activities 18 through 20 are performed in Phase V. The details of each activity are discussed in the following subsections.

The benefit of the systems engineering training for academic institutions and new space entrants is for the team members to learn and understand the importance of the end-to-end process since the team members perform different tasks and sub-tasks to realize the space and ground systems. This is known as a conjunctive group task, where the team members interact with one another and influence one another to produce a product [118] [119]. In a conjunctive group task, the performances of the group members are improved compared to a group member working individually due to what is referred to as the “Kohler effect”. The Kohler effect occurs when the group members exert more effort to avoid being the inferior group member in the conjunctive group task [119].

One method to implement the systems engineering training is to utilize SABRE-I. SABRE-I was a 3U high-altitude balloon mission to emulate an end-to-end space mission on a CubeSat by following a systems engineering process [120] [121]. SABRE-I began as a CubeSat-derived activity to provide an end-to-end hands-on experience throughout the project life-cycle. SABRE-I more closely represented a CubeSat class system compared to other educational satellite platforms such as CanSats through the use of CubeSat standards like the PC/104 interface, 1U PCB dimensions, and modularized subsystems for power distribution and command and data handling. SABRE-I was designed and developed using almost entirely COTS components from hobbyist electronics stores to reduce the overall cost of the system. In addition, the other educational satellite platforms provide the “assembly integration and utilization” options (i.e., later phases of the life-cycle) while SABRE-I provides the entire life-cycle experience including the earlier phases of the life-cycle (i.e., concept and design) due to available COTS components and reduced cost of the system. SABRE-I has more leniency for failure due to its reduced cost, thus, allows the new space entrants to gain more experiences if failure occurs rather than during the development of the containerized satellites.

SABRE-I was successfully launched on a tethered high-altitude balloon in the middle of 2015 and since been used in a wide range of educational platforms from capstone design courses to science, technology, engineering, and math (STEM) outreach projects. The pictures of SABRE-I are shown in Figure 3-4.

In addition to providing an end-to-end life-cycle experience, this training program allows the transfer of knowledge from one generation of personnel to the next. Teams executing containerized satellite missions, particularly those in academia, experience significant attrition rates and team members need to be constantly replaced. By receiving formalized training in the methods of systems engineering and hands-on experiences, the team is not overly reliant on individuals. Similar to a traditional space mission, a team well versed in the methods of systems engineering is equipped to address accountability, traceability and repeatability. It is important to mention here that for containerized satellite missions executed by experienced teams and corporate space institutions, this phase need not be exclusively adopted since systems engineering practice and training are typically an integral part of the work culture for these teams.

Figure 3-4. Images of SABRE-I. Courtesy of author.

Phase I – Mission Concept and Preliminary Design

During this phase, the team will delve into the design of the space system and the ground system, specifically of the payload, the satellite bus, and the ground station by following the outline shown in Figure 3-5 [122] [123]. As shown in Figure 3-5, the mission definition is the starting point of the design process. A well-articulated mission definition leads to identification of specific primary and secondary mission objectives. With the mission objectives determined, a mission success criteria is developed which assesses the mission success based on how well the mission objectives were achieved and fulfilled. Although mission objectives are the primary influence for defining the requirements and developing the mission concept of operations (CONOPS), external drivers which include financial cost, schedule, constraints, and lessons learned from previous missions and/or parallel systems design and development, have significant influence. Specific mission requirements, both functional and performance, which are described in technical terminology, are identified and associated with the mission objectives.

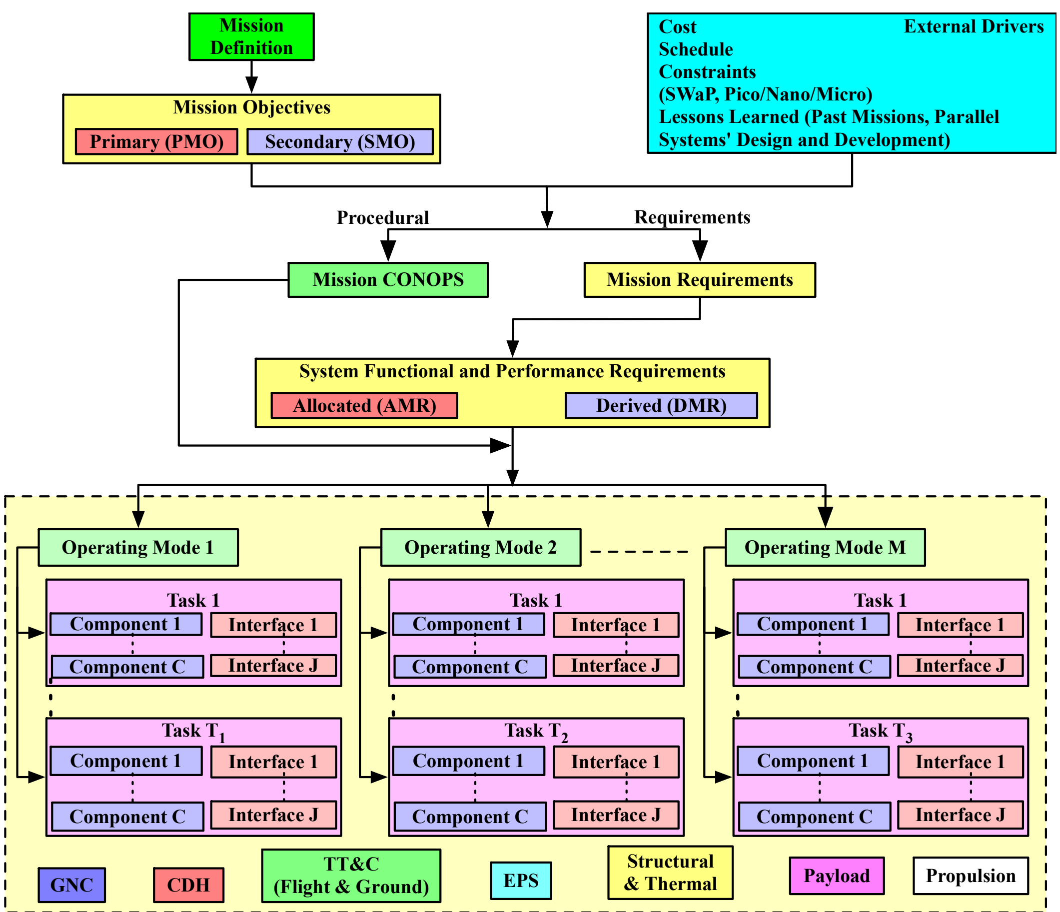
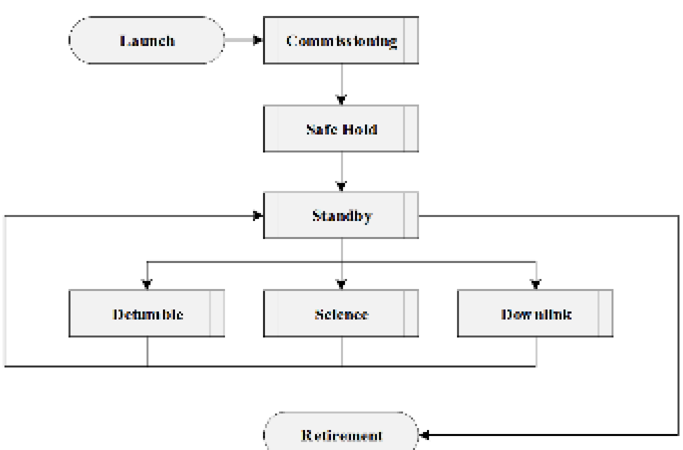


Figure 3-5. Mission and requirements flowdown [122] [123].

The mission CONOPS is designed to describe the procedure for validating mission objectives on orbit. The containerized satellites are generally launched as secondary payloads, thus, require a non-interference with the primary payload during launch and for a period of time after ejection from the container. The wait period will depend on the primary payload and the launch vehicle. When the containerized satellites are ejected from their containers, they typically are rotating at some rate, thus, they require a detumble mode to stabilize the spacecraft prior to on-orbit operations. During on-orbit operations, payload data and spacecraft bus health data are downlinked to the ground station and commands are uplinked from the ground station to the spacecraft. Once on-orbit operations are concluded, the containerized satellites are retired. An example of a containerized satellite's CONOPS that includes a wait period after ejection from the deployment container, commission, detumble, on-orbit operations, and retirement are shown in Figure 3-6. With the mission CONOPS identified, orbital simulations using tools such as MATLAB and AGI's System Tool Kit (STK) are performed to refine the mission CONOPS and mission requirements.



Operational Mode	Description
Launch	Launch and eject from deployer. Wait 45 min.
Commissioning	Power up spacecraft. Perform initial checks. Deploy housekeeping antennas. Detumble the spacecraft.
Safe Hold	Minimum operation mode. Beacon and check health.
Standby	Beacon and check health. Maintain attitude.
Detumble	Beacon and check health. Detumble the spacecraft.
Science	Deploy payload if not deployed. Take measurements. Beacon and check health.
Downlink	Downlink payload data to the ground. Beacon and check health.
Retirement	Retire the spacecraft.

Figure 3-6. An example mission CONOPS.

The mission CONOPS and system requirements are decomposed into components, interfaces, and tasks, the basic building blocks of the various subsystems shown in the dotted box of Figure 3-5. Each component is associated with one or more interfaces and the tasks, which involve one or more components and their interfaces, are grouped together to form operating modes as per the mission CONOPS. The subsystems blocks, expanded in Figure 3-7, show the categories and components of each subsystem for space and ground systems. The subsystems for the space system (i.e., containerized satellite) are payload, command and data handling (CDH), electrical power system (EPS), guidance, navigation, and control (GNC), propulsion, telemetry, tracking, and command (TTC), and structural and thermal. The subsystems for the ground systems include radio frequency (RF) ground stations and optical ground stations depending on the mission. The subsystems and the components are selected based on the mission CONOPS and the mission requirements, thus, not all subsystem and components shown in Figure 3-7 are used. The expanded space and ground subsystems blocks are also known as the product breakdown structure (PBS) of the containerized satellite mission.

Based on the PBS and the decomposition, a detailed work breakdown structure (WBS) is developed to identify the tasks and the activities needed to develop and operate the two systems. An example of a WBS is shown in Figure 3-8, where project management, verification and validation, and mission operations activities are shown in addition to the PBS. It is important to note that each component in the WBS are numbered to help in organizing and numbering the requirements. With the detailed WBS developed, tasks and responsibilities are distributed and assigned to team members. It is highly recommended to use a file-share system like a server system to share and use common documentations and files throughout the team for consistency.

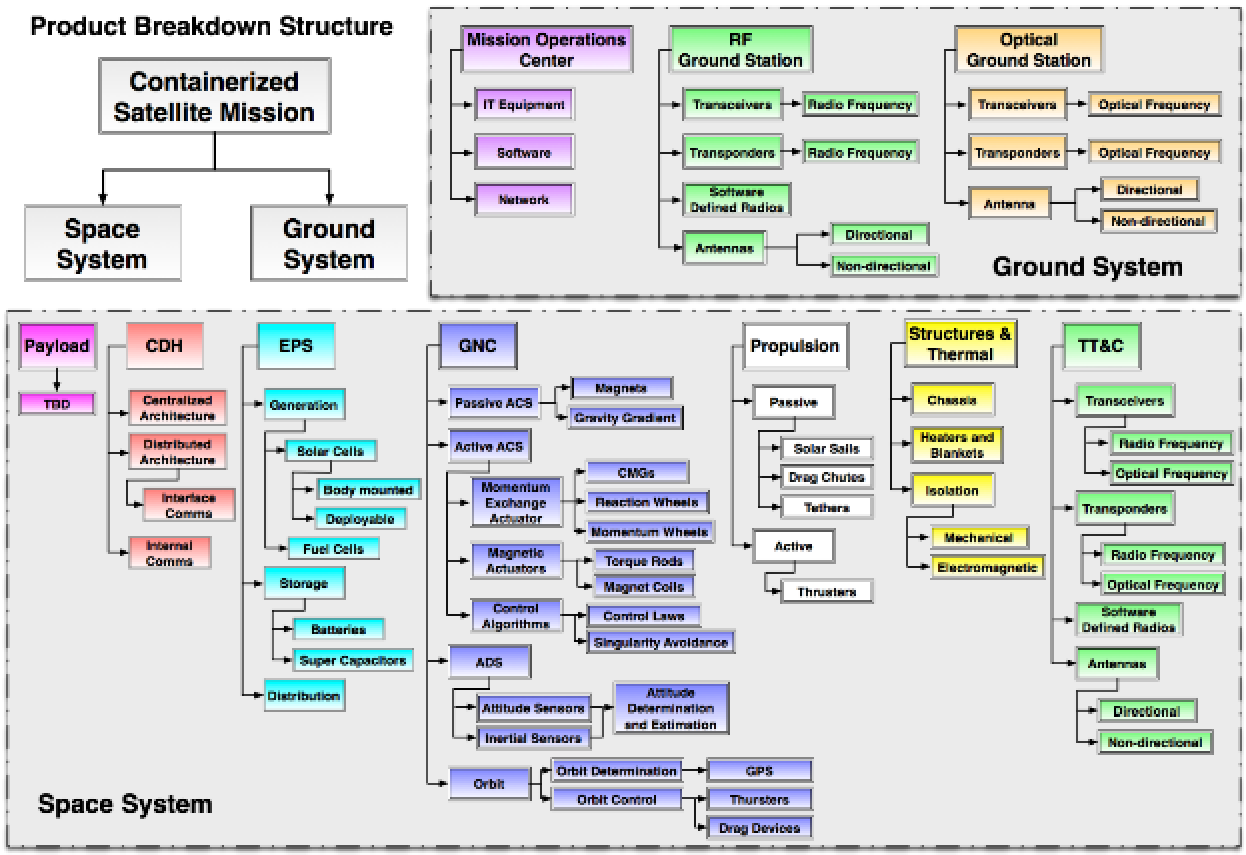


Figure 3-7. Subsystem level flowdown.

Based on decomposition and the detailed WBS, the system level mission requirements are decomposed and functionalities are allocated into subsystem level and component level requirements. All the requirements are listed in a requirements verification matrix for traceability and verification. The requirements verification matrix shows how the requirements are verified. An example requirements verification matrix is shown in Figure 3-9 where the matrix includes the requirement number (based on the WBS), requirement description, verification method, verification artifact, and its status. The verification methods include:

- A – analysis and simulation
- O – observation and inspection
- D – design
- T – test and measurement
- R – reference and datasheet

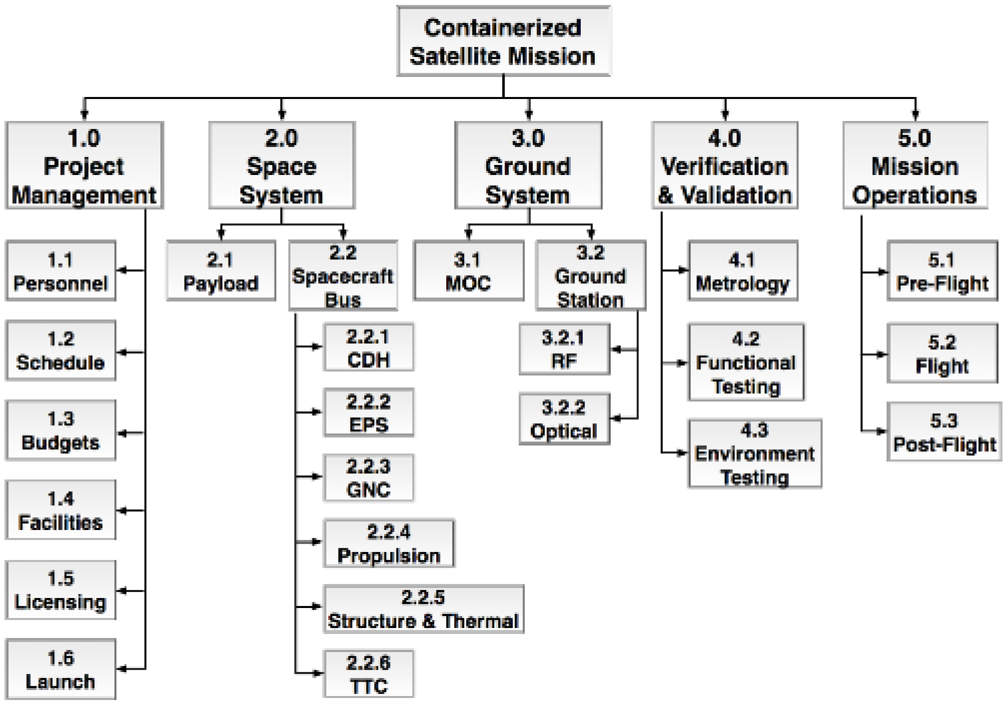


Figure 3-8. An example of a work breakdown structure.

Number	Requirement	Verification Method					Verification Artifact	Status
		A	O	D	T	R		
2.2.2	**EPS Subsystem**							
2.2.2.1	The EPS subsystem shall include a rechargeable battery to provide sufficient power to all peripherals on SABRE-I			X		X	Design and Reference Document	
2.2.2.2	The EPS subsystem shall include a power distribution module that will deliver power to all peripherals on SABRE-I			X		X	Design and Reference Document	
2.2.2.3	The EPS subsystem shall include a power generation module to charge batteries			X		X	Design and Reference Document	

Figure 3-9. An example requirements verification matrix.

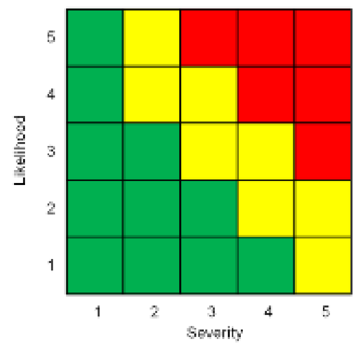
With subsystem and component requirements established, the designs for each subsystem commences with the definition of the preliminary system budgets (i.e., power, mass, volume, link, and telemetry) that shows the design limits. A trade study to identify COTS systems, which may include subsystem boards, sensors, actuators, etc., is a significant part of the preliminary design for each subsystem. Weighted design matrices are used to perform the trade study of the components. The design matrices are developed by listing the characteristics and performance parameters from the requirements. Each parameter is ranked and weights are added based on the ranking. Each component's performance is scored and a value is determined by multiplying the weighting factor and the score for each. By summing each characteristic value, an overall score for each component is determined and used to identify the best-suited selection. An example design matrix is shown in Figure 3-10.

Housekeeping Transceiver			CPUT VUTRX			ISIS UHF/VHF			AstroDev Li-1		
Parameter	Weight	Units	Mag.	Score	Value	Mag.	Score	Value	Mag.	Score	Value
Max Transmit Power	0.25	Watts (W)	2	2	0.50	0.5	1	0.25	4	3	0.75
Max Power Draw	0.25	Watts (W)	10	1	0.25	4	3	0.75	10	1	0.25
Receive Sensitivity	0.25	dBm	-117	3	0.75	-100	1	0.25	-104.7	2	0.50
Flight Heritage	0.15	Missions	2	2	0.30	1	1	0.15	> 3	3	0.45
Extra Features	0.10	Level	High	3	0.30	Low	1	0.10	Medium	2	0.20
		Overall Value:			2.10			1.50			2.15

Figure 3-10. Design matrix example.

In addition to the trade studies, each subsystem should perform risk assessment using tools such as failure modes, effects, and criticality analysis (FMECA) and/or fault tree analysis (FTA) to identify potential failures. For each failure mode in the FMECA, the failure causes are identified and for each failure cause the likelihood and severity are determined and used to determine the criticality from the risk matrix. Failure detection and preventative actions are also listed for each failure cause. The FMECA is typically documented in a tabular format. The FTA complements the FMECA by starting with a top-level failure effect and traces the failure to potential lowest level causes, creating a fault tree. The fault tree evaluates the combination of root failures that could lead to the top event. The fault tree is constructed using logic gates to represent the events and consequences and describe the logical relationship between events. Reliability analyses using those two methods, FMECA and FTA, were conducted for a containerized satellite called SwampSat and detailed in [124]. An example of a FMECA with a risk matrix is shown in Figure 3-11 and an example of a FTA is shown in Figure 3-12.

Failure Mode	Failure Cause	S	L	C	Failure Detection	Preventative Action
No UHF Downlink	UHF antenna deployment failure	5	1	5	Sparse UHF downlink	Rigorous testing on ground prior to launch
	UHF/VHF transceiver failure	5	1	5	No UHF downlink	Rigorous testing on ground prior to launch
	Connection failure	5	2	10	No UHF downlink	Rigorous testing on ground prior to launch. Multiple inspections on connections
	EMI or Radiation	5	2	10	Sparse UHF downlink	Conformal coating on transceiver board. Orbit simulations and analysis of SAA
No S-band Downlink	Ground station equipment failure	5	3	15	No S-band downlink	Extensive ground testing prior to launch
No response to Uplink	VHF Receiver failure	5	1	10	No response to Uplink	Rigorous testing on ground prior to launch.



High Risk	Significant Impact
Moderate Risk	Some Impact
Low Risk	Minimum Impact

Figure 3-11. Example FMECA and risk matrix.

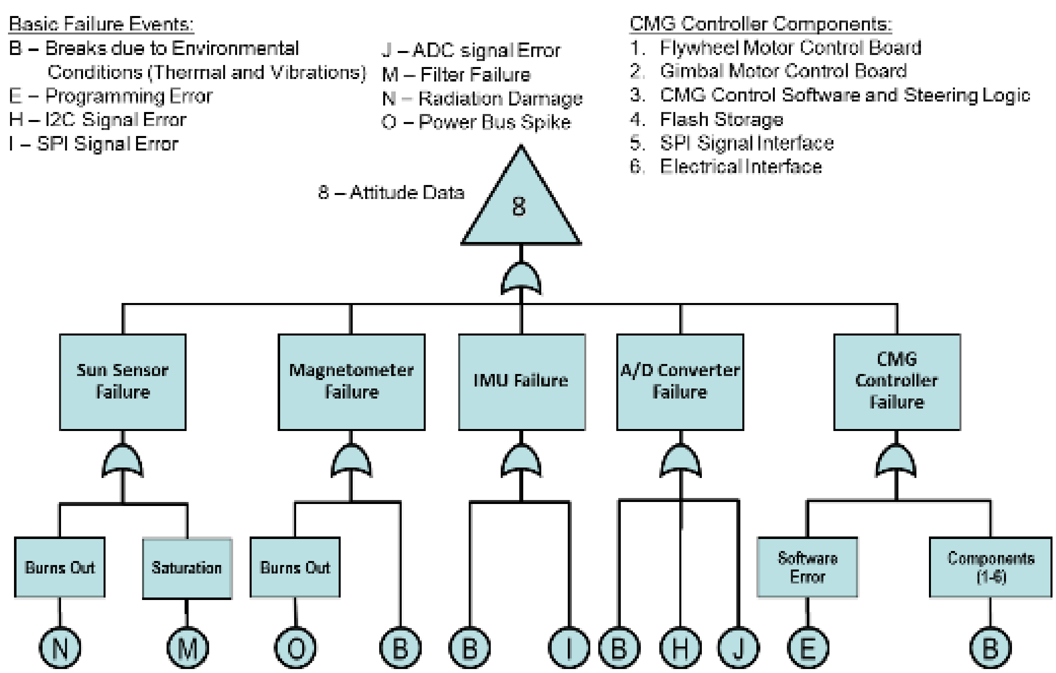


Figure 3-12. Example FTA.

With the preliminary designs completed for each subsystem, the system budgets (i.e., power, mass, volume, link, and telemetry) are all updated. The system budgets should include contingencies where higher contingencies are allocated at this phase and as the design matures, the contingencies are reduced. For example, a mass budget at this phase should have a 20% contingency and as the design matures into detailed design, a 10% contingency is applied.

Also in this phase, each subsystem must develop verification and validation test plans that describe how each component will be tested. The V&V test plans are developed in steps: metrology, component level testing, subassembly level testing, subsystem level testing, and system level testing (see Figure 3-13). In metrology, the acquired components are inspected and their sizes and masses are measured to ensure the measurements are within specifications. At component level, functions for each component are tested and verified. Once the functions for the components are verified, the components are integrated into subassemblies. Those subassemblies are then verified through testing to ensure functionality. Upon verification, the subassemblies are then integrated into subsystems and are verified through testing to ensure that the tasks specified in the operating modes are accomplished. The subsystems are then assembled into a system and subjected to tests for verification. Environmental tests (i.e., vibration, thermal-vacuum, shock, and/or radiation tests) are conducted at system level to verify functionality in relevant operating environment. While the environmental tests may be conducted at lower system levels, generally they are conducted at system level to reduce stress and fatigue on components. In addition to the environmental tests, a day-in-the-life (DITL) test is conducted at system level to verify that the system is capable to perform all the tasks to satisfy the mission and its objectives. Once verified, the system is ready to be put into operation. An example V&V test plan developed during this phase is shown in Figure 3-14. The V&V test plans also help identify necessary testing apparatus to conduct the tests. These testing apparatus are either developed in-house or outsourced depending on the team.

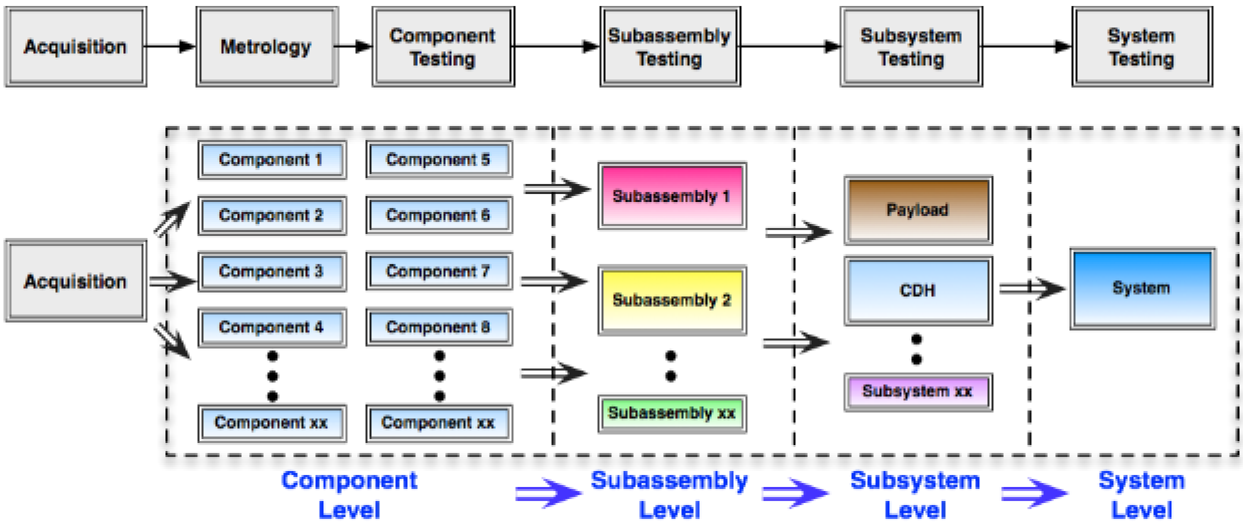


Figure 3-13. Verification and validation test methodology.

Verification Test	Housekeeping Radio	S-band Radio
Acquisition	Acquire hardware and develop test beds	
Metrology	Inspect acquired hardware	
Component testing	Functionality test with frequency counter	
Subassembly testing	Test with hand-held	Test with SDR
Subsystem testing	Integrate with EPS and CDH and test with ground station	
System testing	Integrate with spacecraft and test with ground station	

Figure 3-14. An example of the V&V test plan during Phase I.

To highlight the details of the V&V methodology shown in Figure 3-13, consider its application to the acquisition of a transceiver board that utilizes ultra high frequency (UHF, 420 – 450 MHz) for downlink and very high frequency (VHF, 130 – 150 MHz) for uplink. Once the transceiver is acquired and unpackaged, visual inspection is conducted on the transceiver to ensure no visible damages. Furthermore, sizes and mass are measured to ensure they meet specifications (acquisition and metrology). After metrology, the transceiver is powered up and different functions such as downlink and uplink are configured and setup. After the setup, 50 ohm loads are connected to the receive and transmit connections on the transceiver and the downlink and uplink functions are tested and verified (component level test). After verification, other components such as UHF and VHF antennas are assembled with the transceiver and tested with another transceiver (subassembly level). Once verified, the assembly is integrated with other subassemblies like the flight computer board from CDH and battery board from EPS and tested (subsystem level). After verification, the assemblies are integrated into a single system (system level) and verified through testing to ensure all components are functional. After system verification, the system is then put into use.

For each subsystem, a subsystem design document is required to archive the design drivers and the analyses performed during the preliminary design. In addition, the trade studies, the risk assessments, the system budgets, and the V&V plans are also documented in the design documents. The preliminary designs for each subsystems are detailed in the following subsections.

Payload

At this phase, if the containerized satellite mission is a technology demonstrator or experiments, the concept technology (i.e., sensors, actuators, instrumentation, etc.) is designed, analyzed and in certain scenarios, prototyped as well [125] [126]. Generally, the payload has higher risks than the spacecraft bus, thus, it is critical to design, develop, and test the proof of concepts as early in the life-cycle as possible. For example, consider a miniaturized control moment gyroscope (CMG) assembly capable of three-axis control is the payload for a containerized satellite mission (technology demonstrator). The CMG assembly consists of a flywheel, a gimbal, and electronics to control the two and the CMG assembly generates a torque. The torque generated by a single CMG lies in a plane, therefore, multiple CMGs in appropriate configurations are required to produce torque in a 3D space. During this phase, a prototype of the CMG assembly is designed and developed based on mission requirements and simulation results, where the simulations determine the feasible CMG designs and configurations. Trade studies are conducted on the CMG components to identify the best-suited components for development.

CDH

The command and data handing (CDH) subsystem is the on-board computer system and its associated software that collect and process component telemetries within the spacecraft while receiving and sending data through the telemetry, tracking, and command (TTC) subsystem. The typical parameters/specifications of the on-board computer are processing capabilities, timing accuracies, data storage, and interfaces. These parameters depend on what the mission CONOPS and requirements are. Using the parameters and specifications, trade studies are conducted to identify best-suited options for the on-board computer. A software development plan is needed to determine the software development language and process. The software development process typically begins with the drivers (i.e., component-level) and based on the drivers, different functions are developed. The functions are arranged and organized into tasks and ultimately into operating modes. Also during this phase, a preliminary telemetry and data budget is developed with the TTC team that describes the communication protocol and data packaging formats. Each component associated with one or more interfaces and tasks are identified for each operating mode, thus, the telemetry required to monitor these tasks and components are identified. It is recommended to compile a list of telemetries from each component and use the list to determine the preliminary telemetry and data budget.

EPS

The electrical power system (EPS) has three functions: power generation, energy storage, and power distribution. Solar panels are commonly used to generate power for the containerized satellites. In the preliminary design, a power budget is required to ensure that sufficient power is available during mission operations. The power budget is organized into operating modes and shows power consumption and power generation with contingencies. In the preliminary design phase, a 20% contingency is sufficient. The power consumptions are determined based on selected subsystem components and the power generation is determined via simulations. Both the power consumption and power generation are typically calculated per orbit and shown in Watt-hours. The power consumptions for each component is duty cycled per orbit, specifically, if the duty cycle is 1, then the component will be turned on and consuming power all the time, and if duty cycle is 0.5, then the component will only be turned on and consuming power half of the orbit. Perform simulations in tools such as MATLAB or AGI’s System Tool Kit (STK) to determine the power generation with different solar panel configurations. For example, simulate the power generation of all body mounted panels, single-deployable solar panels, and double-deployable solar panels on summer and winter solstices (i.e., longest daylight and shortest daylight) while assuming a constant solar cell efficiency, constant altitude, and constant inclination. From the simulation results, the minimum, the average, and the maximum power generations for the three solar panel configurations are determined. Use the minimum power generations and compare how many power negative modes exist. The minimum power generation represents the worst-case scenarios. An example of a power budget is shown in Figure 3-15, where majority of the mission is in power positive modes. With the power budget updated, select the best-suited solar panel configuration and perform trade studies to identify the available COTS solar panels.

Power Budget	Commission	Detumble	Safe-Hold	Standby	Science	Downlink
Subtotal (Whr)	2.0	3.3	2.0	3.5	4.5	10.5
20% Contingency (Whr)	0.4	0.7	0.4	0.7	0.9	2.1
Power consumption (Whr)	2.4	3.9	2.4	4.2	5.4	12.6
Power generation (Whr)	5.3	5.3	5.3	5.3	5.3	5.3
Margin (Whr)	2.9	1.4	2.9	1.1	-0.1	-7.3

Figure 3-15. An example of a power budget with 20% contingency.

Generally there are two options for storing electrical energy: battery or super capacitors. The containerized satellites typically use batteries over super capacitors. Trade studies are performed to identify the best-suited option based on the EPS requirements such as storage capacity as well as size and mass. Larger storage capacity typically is larger in size and heavier in mass, therefore, requirements must be clearly defined. Similarly for power distribution, trades studies are performed to identify power distribution units that provide the necessary voltage busses. It is highly recommended that all three EPS components (power generation, energy storage, and power distribution) are selected from the same vendor due to compatibility. For example, if the solar panel voltage is not higher than the energy storage voltage, the current does not flow and the solar panels will not be able to charge the energy storage devices. Furthermore, if the energy storage voltage differs from the power distribution unit voltage, the power distribution unit is inefficient or in some cases unable to distribute voltages to the spacecraft components. Thus, it is critical to examine the compatibility between the three EPS components during the design.

GNC

The guidance, navigation, and control (GNC) subsystem consists of attitude and orbit determination and control. The attitude determination and control is needed to orient the spacecraft in orbit and the orbit determination and control is needed for orbit maintenance and/or to maneuver between orbits. Both utilize sensors and actuators for the determination and control. Attitude control can be achieved passively or actively where passive attitude control uses gravity gradient or permanent magnets and active control uses momentum exchange devices (control moment gyroscopes, reaction wheel, or momentum wheel) or a reaction control system that uses thrusters. When momentum exchange devices are used on the spacecraft, stored angular momentum saturates the wheels, thus, external torques are needed to dump the momentum. The external torques are typically implemented by magnetic actuators (magnet torque rods or magnet coils) and thrusters to dump momentum. For active attitude control, attitude determination using sensors such as Sun sensors, star and Earth cameras, and magnetometer are required. Orbit control is achieved by using thrusters or drag sails/chutes and orbit determination is typically achieved by global positioning system (GPS).

Typical GNC requirements include attitude and orbit knowledge, pointing accuracy, and orbit maintenance. Attitude and/or orbit simulations (see Figure 3-16) are conducted to determine the technical parameters required to meet GNC requirements. For example, an angular stabilization mode (i.e., detumble mode) once ejected from the container is simulated to determine the necessary attitude actuator/sensor combinations using the setup in Figure 3-16. By selecting an altitude and inclination (e.g., 450 km and 45°), external disturbances (aerodynamic drag, gravity, magnetic field, and solar radiation) are calculated via tools such as AGI’s STK or MATLAB. For the spacecraft dynamics, assume a CubeSat form factor and its maximum allowable mass (e.g., 3U CubeSat with 4 kg) with evenly distributed moments of inertia. Select a control algorithm (e.g., B-dot proportional control), initial angular rates (e.g., 5 degrees/sec per axis) and final angular rates (e.g., 0.5 degrees/sec per axis). Perform the simulation and determine the actuator and sensor performances and time required to detumble to the final angular rates. The control algorithm, initial angular rates, final angular rates are varied to examine the performances and the time required to detumble. Based on the technical parameters and requirements, component trade studies are conducted to identify the best-suited options.

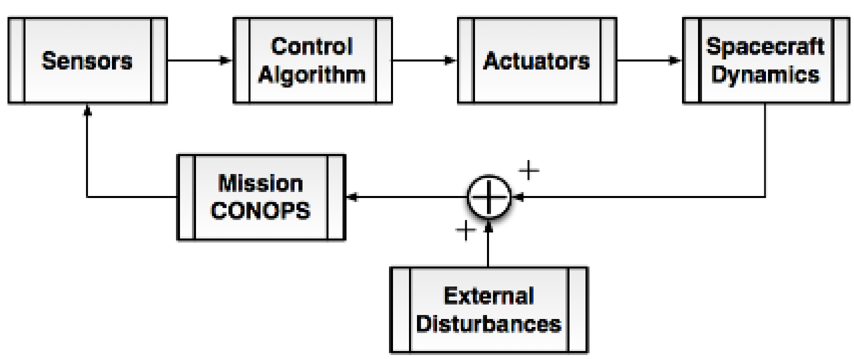


Figure 3-16. Attitude/orbit simulation example.

Propulsion

The propulsion subsystem is required for orbit maneuvering, orbit maintenance, or attitude control and the functions of the propulsion subsystem are determined based on the CONOPS and the requirements. Based on the functions, determine and quantify parameters such as delta-V, thrust, and total impulse to achieve the functions. Orbital simulations are performed to identify and quantify the required parameters. To note, these simulations are typically conducted with the GNC subsystem. A propellant budget is created based on different maneuvers and for each maneuvers, the delta-V, impulse, and propellant usage are shown. At this phase, the propellant budget also includes a 20% contingency. Perform trade studies of the components that satisfy the parameters and determine the best-suited components for the propulsion subsystem. Propulsion subsystem typically requires propellant, tank, thrusters, and control electronics that takes up a lot of size and mass that many containerized satellites cannot accommodate. In addition, many containerized satellite missions do not require orbit maneuvering (changing orbits). Therefore, many containerized satellites typically do not have propulsions systems.

Structures and Thermal

The structures and thermal subsystem is organized into the spacecraft structure and the thermal control system. The structure is the backbone of the spacecraft and provides the supporting framework to house the payload and the spacecraft bus during launch and on-orbit operations. The thermal controls and manages the temperature to ensure the operability of the spacecraft and its components. The structure and thermal ensure that the spacecraft survive the launch and on-orbit environments. Majority of the structure requirements are derived (or directly) by the containers, specifically, the stowed dimensions, mass, center of gravity, structure material, and interfaces. Based on the requirements, trade studies are conducted to identify the best-suited options for the structure. During this phase, it is highly recommended to begin developing a computer-aided design (CAD) model of the containerized satellite. The CAD model is used to determine the outer dimensions, mass, center of gravity, and moments of inertia. In addition to the CAD model, a mass budget is required to list all the masses for each subsystem including contingency to stay within the total mass requirement. An example mass budget is shown in Figure 3-17 where masses for subsystems and contingency (denoted as “Others” in figure) are shown.

Subsystem	Components	Mass (kg)	Percentage
Payload	Technology	1.20	30.00
CDH	On-board computer	0.10	2.50
EPS	Power distribution unit, battery, solar panels	1.00	25.00
GNC	Attitude sensors and actuators	0.20	5.00
Structures & Thermal	Structure, fasteners, cables	0.40	10.00
TT&C	Transceiver, antennas	0.20	5.00
Others	Miscellaneous (epoxy, Kapton, etc.)	0.40	10.00
	Subtotal (kg)	3.50	
	Allowable Mass (kg)	4.00	
	Margin (%)	12.50	

Figure 3-17. An example mass budget with contingency.

There are two methods for thermal protection, passive and active, where passive protection include insulations and coatings and active protection include electrical heaters/coolers and heat pipes/radiators. The thermal design is driven by the thermal environment, thus, orbital simulations using tools like AGI’s STK or C&R Technologies’ Thermal Desktop are performed to determine the orbit temperatures. For example, examine the maximum and minimum external temperatures that the spacecraft will experience at a certain altitude and inclination on each external face of the containerized satellite. From the external temperatures, internal temperature is determined via heat transfer. The external and internal temperature limits are used to check the operability and survivability of the components and to assess if thermal control is required in the design.

TTC

The telemetry, tracking, and control (TTC) subsystem is required for communication between the containerized satellite and the ground station. TTC typically uses radio frequency (RF) communication between the satellite and the ground station. The containerized satellite downlink payload data and satellite's health telemetry and the ground station uplinks commands to the satellite. The TTC design parameters such as data rate, frequency, power, and modulation are based on the CONOPS and the requirements. A link budget is required to ensure that the communications between the containerized satellite and the ground station are possible. The link budget for containerized satellites are developed using tools such as AGI's STK (RF and optical) or AMSAT Link Budget Excel developed by Jan King [127] (RF only). Parameters such as transmit power levels, antenna options, and data modulations are varied in the tools to examine different configurations and link margins. A healthy link margin is required to show that the satellite and the ground station are able to communicate. An example of RF link margins with different hardware configurations using AMSAT Link Budget Excel is shown in Figure 3-18. Based on the link budgets, trade study of the COTS components are conducted to determine the best-suited selections.

Spacecraft Transmit Power	Monopole	Dipole	Turnstile	Patch
0.5 W	-2.54 dB	4.18 dB	4.03 dB	7.93 dB
1.0 W	0.47 dB	7.19 dB	7.04 dB	10.94 dB
2.0 W	3.48 dB	10.21 dB	10.05 dB	13.95 dB
3.0 W	5.24 dB	11.97 dB	11.81 dB	15.72 dB

Good Link	Marginal Link	No Link
> 6 dB	> 0 dB and < 6 dB	< 0 dB

Figure 3-18. An example of a RF link margin for different configurations.

Ground System

The ground system is the base for mission operations that monitors, tracks, and commands the containerized satellites. The ground system is divided into the ground station (RF or optical) and mission operations center (MOC). The ground station communicates with the satellite's TTC subsystem using RF or optical depending on the mission. The MOC is the command center that monitors and supports the operations. Most containerized satellites use RF communications and many use amateur frequencies since many the amateur frequencies do not require specific licenses and are free to use. The design of the ground station and MOC is driven by the CONOPS, requirements, the TTC subsystem, and the link budget. Also, the cost in building and erecting a ground station becomes a factor. An example of a RF ground station is shown in Figure 3-19, where an antenna system, a converter system to modify RF signals, a radio to receive and transmit signals, a modem to demodulate and modulate signals, a computer to receive and transmit packets, and an antenna controller system to control the antenna system are shown.

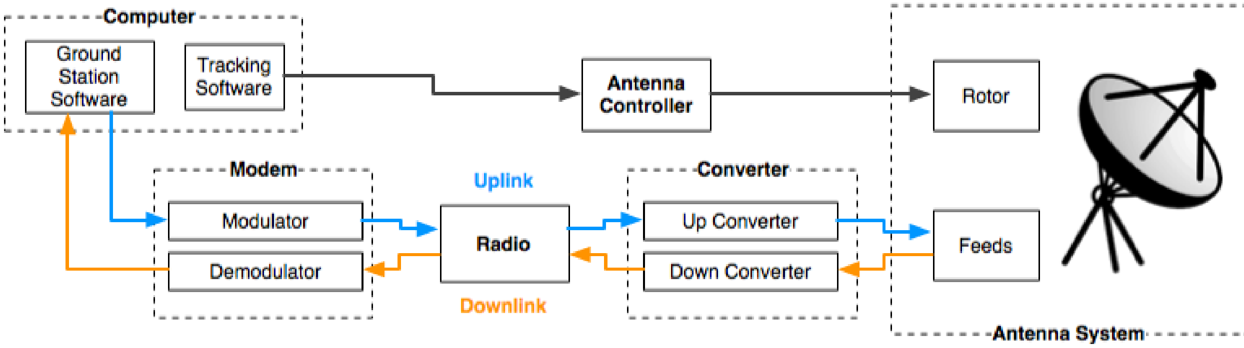


Figure 3-19. An example of a RF ground station.

Interfaces

One of the critical activities during the design is the identification of interfaces. With the preliminary designs for each subsystem, the interfaces must be identified and examined to ensure compatibility. An interface diagram such as N2 diagram that shows the mechanical and electrical interfaces between each component is required. An example of a N2 diagram is shown in Figure 3-20. In addition, most containerized satellite electronic boards utilize the PC/104 header connectors, thus, by combining all the connector pinouts and creating a master PC/104 header pinout ensures no interference between the electronics boards. Furthermore, it is highly recommended to begin developing a CAD model of the containerized satellite in this phase. The external and internal interfaces of the satellite are identified by using the CAD model.

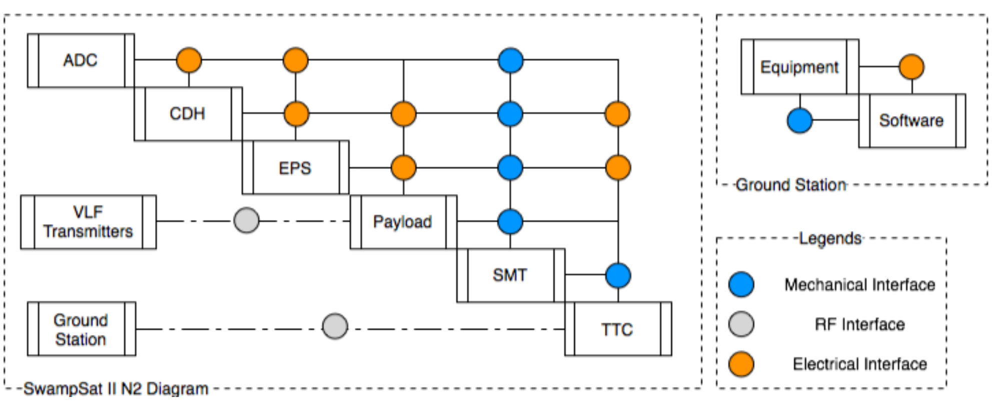


Figure 3-20. Example of a N2 diagram.

Project Management

In addition to the design of the space and ground systems, project management tasks are needed. These tasks are typically organized by the project managers and include the following: identify roles, responsibilities, tasks, while organizing team personnel, develop schedule that shows the entire life-cycle, develop cost budget, identify required facilities to development and operations, determine required licenses, and secure launch opportunities. For the team personnel, it is critical to use consistent document formatting and terminology. In addition, a file share system like a server file system is recommended to be used within the team. A schedule is required and recommended to be shown in life-cycle phases. The schedule is required to be maintained throughout the project life-cycle and typically developed in a form of a Gantt chart. Allocate contingencies in the schedule to provide sufficient time for development and testing. The cost budget needs to reflect the material (shipping costs too) and non-recurring engineering (research, development, and test) cost for at least one unit and should also include launch and operations cost. Contingencies in the cost budget are also required and at this phase higher contingencies are applied. As the mission progresses, lower contingencies are applied since the designs are matured and the costs are more accurately estimated. The facilities for development and operations are also required to be identified. For example, if the team needs to build and erect a ground station, the program management must include the costs in the budget. Licensing is another task that is required for containerized satellites and depends on the containerized satellite's country of origin. For example, containerized satellites developed in the United States may require two licenses, one from the Federal

Communication Commission (FCC) for RF communications and one another license from the National Oceanic and Atmospheric Administration (NOAA) for remote sensing. Securing launch opportunity is also a required task for the project management. Particularly for academic institutions, there are launch opportunities at very little or almost no cost through programs like NASA's CubeSat Launch Initiative [24] and ESA's Fly Your Satellite [128].

Preliminary Design Review

Feedback mechanisms are significant in project maturation, thus, a preliminary design review (PDR) with external reviewers is critical for the team. Therefore, to advance to the next phase (i.e., exit criteria), the containerized satellite team must successfully complete a PDR. The PDR shall be conducted with external subject matter experts and funding stakeholders. In addition to the reviews by external reviewers, it is highly recommended to have regular internal peer reviews to obtain feedbacks. The new entrant teams should be mandated to seek at least two successful PDRs – an internal PDR with peers and an external PDR with subject matter experts and funding stakeholders – for advancing to the next phase. Successful PDR leads to increased confidence in the design and allows the team to advance to the next phase.

The mission concept and preliminary design are presented in a form of a preliminary design report and in the PDR. The following are presented in the preliminary design report and in the PDR:

- Statement of mission goal
- Objectives of proposed mission - primary and secondary
- Background study to show feasibility and uniqueness
- Preliminary mission CONOPS
- Work breakdown structure (WBS)
- Team organization with roles and responsibilities

- Mission requirements flowdown and requirements verification matrix
- System budgets with contingencies/overhead
    - Power budget
    - Mass budget
    - Size budget
    - Link budget
    - Telemetry budget
    - Mission schedule/timeline
    - Mission cost
- Preliminary design documents for each subsystem
    - Payload
    - Command and data handling (CDH)
    - Electrical power system (EPS)
    - Guidance, navigation, and control (GNC)
    - Propulsion
    - Telemetry, tracking, and command (TTC)
    - Structural and thermal
    - Ground systems (e.g., RF ground station, optical ground station, etc.)
- Interfaces and interface control document
    - Internal and external interfaces
    - Preliminary CAD layout/model of mission payload and satellite bus
- Verification and validation plans
    - Details how verification and validation tests are going to be performed
- Reliability analyses document
    - Identify risks and its mitigation strategies
- Regulatory licensing status
    - Identify required licenses
- Launch status
    - Identify launch opportunities

The activities presented in this phase are typically associated with the first three phases of a NASA project life-cycle, Pre-Phase A through Phase B. The mission concepts and objectives are identified, CONOPS and mission requirements are developed, requirements are decomposed into subsystem and component level requirements (i.e., requirements flowdown), and a preliminary design is developed. For the containerized satellites, the mission and design spaces are finite due to the size, weight, and power (SWaP) constraints imposed by the containers. Moreover, the containerized satellite missions are all non-human space flight (robotic) missions that do not require development of human safety plans and the containerized satellites do not develop launch systems (e.g., launch vehicles). The engineering activities presented in this phase are tailored specifically for containerized satellites.

Phase II – Detailed Design and Virtual Assembly

During this phase, the team further matures the designs and ultimately decide on a detailed design of the containerized satellite and ground system. The results from the reliability analyses using techniques such as FMECA and FTA from Phase I are used to refine designs (e.g., adding redundancies) and update verification and validation test plans to reduce potential risks. In addition, feedbacks and action items from the PDR are addressed in this phase. Based on the preliminary designs, the CONOPS and the requirements are refined and technical performance requirements are added. If the containerized satellite is manifested for a particular launch, there are additional requirements from the launch provider. These requirements are typically environmental tests (vibrations, shock, and thermal-vacuum) to survive launch and inhibits to ensure no interference with the other payloads on the launch vehicle. The containerized satellites are generally launched as secondary payloads, therefore, non-interference is required during launch (and after ejection from container in some cases). The inhibits are typically switches that are connected to the EPS to ensure no power is provided to the satellite when the switch is engaged. With the updates to the requirements, the requirements verification matrix is also updated.

The systems budgets are updated with lower contingencies compared to the budgets from preliminary design; if a 20% contingency was applied during preliminary design, apply a 10% contingency in this phase. The system budgets must be within the design limits and the contingencies/margins are reduced as the mission progresses.

In this phase, the V&V test plans for each level (component level, subassembly level, subsystem level, and system level) are updated and developed in detail. With the detailed design for each subsystem, the functions for each component are identified and listed. For each function, a functionality/verification test plan is developed where the test plan identifies how the functions are tested. For example, consider a COTS transceiver board has several functions including transmit and receive RF signals, measure on-board temperature, and measure current consumption. For each function, detailed test plans are developed to verify its component-level functionalities (see Figure 3-21). Next at subassembly level, a hand-held transceiver is used to communicate (downlink and uplink) with the COTS transceiver and subassembly level test plans are developed for the communication tests (see Figure 3-22). At subsystem level, the subassemblies from CDH and EPS are integrated with the TTC subassembly to test the downlink and uplink with the ground stations (see Figure 3-23). At system level, the subsystems are integrated into a system and each operating modes in the mission CONOPS are tested. Detailed test plans for each operating mode are developed (see Figure 3-24). In addition to the operating mode tests, environmental tests are required at system level. The environmental tests and its loads are varied depending on the launch provider, however, the test plans are still required. After the environmental tests, a day-in-the-life (DITL) test is also conducted at system level where all of the operating modes are tested along with the ground station. With the detailed V&V test plans, the necessary testing apparatus and equipment are all identified and decisions are made for each testing apparatus if they are either developed in-house or outsourced.

Component	Functions/Tasks	Functionality/Verification Test
UHF/VHF Transceiver board	Power Up	Connect power from power supply. Monitor current draw on power supply. Command and read device IDs.
	Transmit at UHF	Connect frequency counter. Turn transmitter ON, transmit UHF, and measure the transmit frequency on frequency counter
	Receive at VHF	Connect 50 ohm load. Turn receiver ON, transmit VHF, command and measure the received signal strength
	Measure temperature	Command temperature sensor and measure temperature. Apply heat on the temperature sensor and measure temperature.
	Measure current draw	Turn receiver ON, command and measure current draw. Turn transmitter ON, transmit UHF, command and measure current draw.

Figure 3-21. An example component level V&V test plan.

Subassembly	Components	Functions/Tasks	Functionality/Verification Test
TTC_SA_1	UHF/VHF transceiver board	Downlink UHF packets and uplink VHF packets	Connect UHF and VHF antennas to transmit and receive connections. Connect UHF/VHF hand-held radio to computer. Downlink UHF packets from transceiver, receive UHF packets on hand-held radio. Check received packets. Uplink VHF packets from hand-held radio to transceiver. Check receive signal strength on transceiver. Check received packets
	UHF antenna		
	VHF antenna		
	UHF/VHF hand-held radio		
TTC_SA_2	S-band transmitter board	Transmit and receive S-band packets	Connect S-band antenna to transmitter. Downlink S-band packets from transmitter and receive S-band packets on the SDR. Check received packets
	S-band transmitter antenna		
	Software defined radio		

Figure 3-22. An example subassembly level V&V test plan.

Subsystem	Subassembly	Components	Functions/Tasks	Functionality/Verification Test
CDH EPS TTC	CDH_SA_1	SSII Flight computer	• Downlink CDH and EPS telemetry using UHF and S-band. • Uplink commands to downlink telemetry and to switch frequencies.	• Turn ON EPS. CDH to acquire temperature telemetry from CDH and EPS. Store them in the EEPROM. • Uplink VHF to downlink telemetry via UHF. CDH read from EEPROM and sends to UHF transmitter. • Downlink UHF packets and receive UHF packets on ground station. Check received packets. • Uplink VHF to downlink telemetry via S-band. CDH read from EEPROM and sends packets to S-band transmitter. • Downlink S-band packets from transmitter and receive S-band packets on the ground station. Check received packets.
		EEPROM		
		Temperature sensor		
	EPS_SA_1	40 Whr Battery		
		Power distribution board		
		Temperature sensor		
	TTC_SA_1	UHF/VHF transceiver board		
		UHF antenna		
		VHF antenna		
		UHF/VHF ground station		
	TTC_SA_2	S-band transmitter board		
		S-band transmitter antenna		
		S-band ground station		

Figure 3-23. An example subsystem level V&V test plan.

System	Subsystem	Functions/Tasks	Functionality/Verification Test
SwampSat II	Payload CDH EPS GNC SMT TTC	Power-up and detumble	Disengage flight switches and turn ON EPS. Boot up CDH and perform health check. Detumble spacecraft.
		Deploy UHF/VHF antennas and solar panels	Measure angular rates. Command deployment of UHF/VHF antennas. Measure angular rates and verify deployment. Command solar panel deployments. Measure angular rates, solar panel voltage and currents to verify deployments
		Point solar arrays to the Sun	Activate Sun pointing mode. Measure solar panel voltage and currents, temperature, Sun sensors to verify Sun pointing
		Deploy payload antenna	Command payload antenna deployment. Measure angular rates, capture images every 30 sec, monitor motor increments. Capture images after deployment to verify deployment
		Receive and record VLF signals	Sample VLF signals from A/D and store in the EEPROM
		Downlink VLF signals	Compress and package VLF signals into packets. Downlink S-band packets from transmitter to ground station

Figure 3-24. An example system level V&V test plan.

The level of testing that is required depends on the technology readiness level (TRL), where the TRL evaluates the maturity level of a particular technology [6]. Lower TRL technologies require a more rigorous testing compared to higher TRL technologies since lower TRL technologies have higher risks. To increase the TRL and to reduce risks, the technologies and components require validation in relevant operating environments. Repeated tests must be performed on lower TRL technologies to increase the TRL, therefore, all the tests must be repeatable. Typically, the prototypes and engineering development units (EDUs) are subjected to qualification level tests, which are more rigorous than acceptance level tests. When the prototypes and EDUs pass the qualification level tests, the flight unit is assembled and subjected to acceptance level tests that are less rigorous than the qualification tests. The qualification level and acceptance level tests are common for environmental tests. In the environmental tests, a pre-test, during-test, and post-test functionalities tests are required to ensure that the system can survive the environments.

In addition to the detailed V&V test plan for each subsystem, an operations plan that describes how the CONOPS are executed is required. The operations plan includes schedule and personnel that will execute the mission CONOPS from the ground. This operations plan is required and tested during the day-in-the-life (DITL) tests.

A rule of thumb during the design, particularly for in-house designed components, is to consider the manufacturability as well as assembly and integration aspects (later phases in the life-cycle). Basically, avoid designs that are impossible to manufacture and/or cannot be assembled and integrated. To verify the assembly and integration procedure, a virtual assembly is required. By developing a virtual assembly of the containerized satellite, the assembly and integration procedures are verified.

For each subsystem, the subsystem design document is updated to archive the design traceability and the detailed design. Along with the detailed designs, the updated risks, the updated system budgets, and the detailed V&V plans are also documented. The detailed designs for each subsystem are discussed in the following subsections.

Payload

In this phase, a prototype of the payload is required to be developed and tested to verify its functionality. Generally, the payload has higher risks than the rest of the satellite bus, therefore, it is critical for the payload design to be matured and further tested. A detailed V&V test plan is required that shows the testing plans at each level to mature the technology readiness level (TRL). One difference for the payload compared to other subsystems is to perform rigorous testing at component level where the payload is subjected to on-orbit and launch environment tests. With the detailed V&V test plan, the testing apparatus required for the V&V tests are identified. For the CMG assembly example, a prototype developed in Phase I must be characterized to ensure appropriate torque in 3D space is generated. In order to characterize the torque, a torque sensing platform like GATorSense [129] [130] is required. If the prototype requires design changes, the design is refined and prototyped. The updated prototype is tested to ensure the performance meets the requirements. Once the performance of the prototype is characterized and verified, a detailed V&V test plan that includes rigorous testing of the prototype in on-orbit and launch environments is developed. With the detailed V&V test plan, necessary testing apparatus such as thermal-vacuum chamber and vibration tables are identified.

CDH

The CDH design must include the final selection of the on-board computer based on the requirements. It is recommended to acquire a development board of the processor selected during PDR to verify the selection. Typically, the processors have development boards that can be acquired, thus, acquisition and verification of the selection is critical. Also with the development boards, software development can begin. While the COTS components typically come with their own software, the mission software must be developed in-house. The CDH team must select a software language to be used for the software development. It is recommended to use tools such as Git and SVN for software development since these tools have version control for software. In addition, the CDH team must develop a detailed software flow chart that follows the operating modes where each operating mode is further broken down into specific tasks and functions (see Figure 3-25). In addition to the operating modes, it is highly recommended to have two additional operating modes: reset and transmitter off. The reset mode and the transmitter off modes are uplinked from the ground station for the containerized satellite to reset and to turn the transmitter off, respectively. The reset mode is executed when there is a malfunction during mission operations and the transmitter off mode is executed when the regulatory entities (e.g., Federal Communications Commission in the United States) request no RF transmission from the containerized satellites. The detailed software flow chart with all the operating modes and its function is established and used in developing the software.

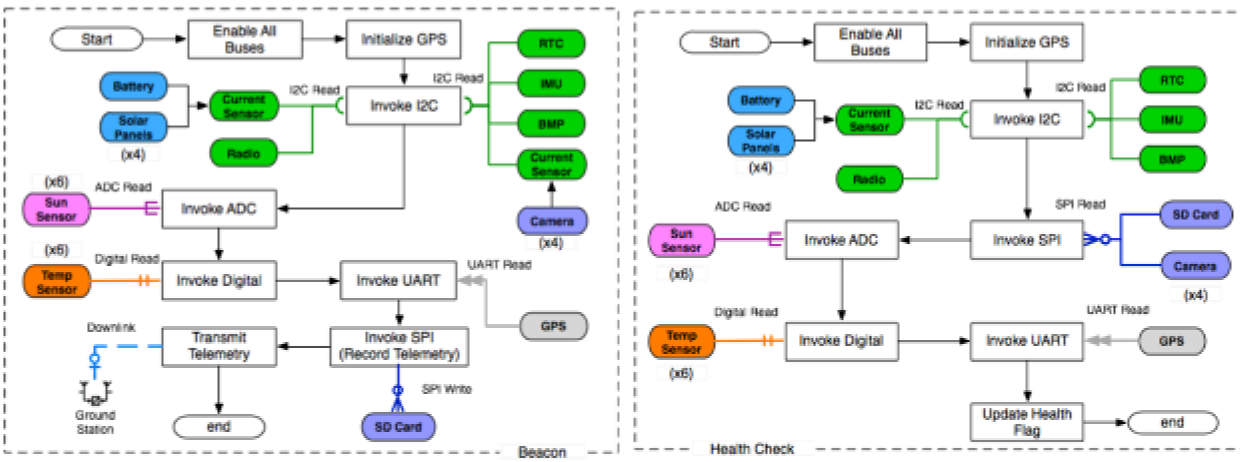


Figure 3-25. An example of a detailed software flowchart.

The list of telemetries are organized based on each operating modes; the downlink and uplink telemetries and data formats are decided for each operating mode. In general, the data storage formats follow the telemetry formats so that no processing is required prior to data downlink. Once uplink is received, the CDH will parse the data and execute associated commands.

With the components selected for the detailed design, the risks are updated and the V&V plans for each component and software are updated. The CDH will include both hardware and software, therefore, the detailed V&V test plans must include both hardware and software.

EPS

For the detailed design of the EPS, select the power generation, energy storage, and power distribution. Based on the updated designs (including other subsystems), update the power budget where the power consumptions are updated based on the selected components and the power generations are updated from simulations. The power generation simulations performed in Phase I are updated according to operating modes and updated designs. For example, after ejection from the container and up to detumble, the deployable solar panels are stowed and after detumble is completed, the solar panels are deployed. Based on this, the power generation varies depending on the operating mode and is reflected in the power budgets. In Phase I, the contingency in the power budget was 20%, however, in the detailed design, the contingency needs to be lower at 10% since the designs are more matured. An updated power budget example is shown in Figure 3-26 with 10% contingency.

Power Budget	Commission	Detumble	Safe-Hold	Standby	Science	Downlink
Subtotal (Whr)	2.8	4.4	2.4	3.9	4.4	13.4
10% Contingency (Whr)	0.3	0.4	0.2	0.4	0.4	1.3
Power consumption (Whr)	3.1	4.9	2.6	4.3	4.8	14.7
Power generation (Whr)	2.7	2.7	2.7	7	7	7
Margin (Whr)	-0.4	-2.2	0.1	2.7	2.2	-7.7

Figure 3-26. An example of a power budget with 10% contingency.

Using the power budget, the energy storage is simulated during the mission CONOPS to ensure that the energy storage capacity is sufficient. For example, simulate the ejection from the container, commission, and detumble assuming no solar panel deployment for power generation. In most cases, the energy storage capacity will not be full after ejection from the container, therefore, assume a 50% capacity when ejected from the container and simulate. The results will show if the energy storage capacity is sufficient for the mission or not.

As previously mentioned, inhibits are required for the containerized satellites to ensure no power is provided to the spacecraft during launch and some period after ejection from the deployment container. These inhibits are typically on the EPS systems and are controlled via switches on the satellite. An inhibit diagram, shown in Figure 3-27, is required to show that the inhibits are included in the design and in some cases, there are multiple inhibits.

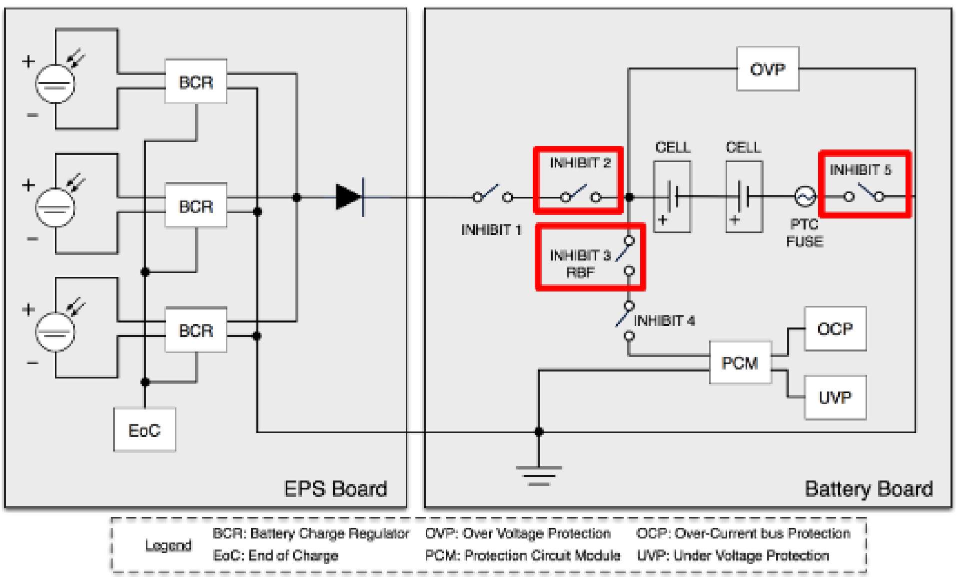


Figure 3-27. An example inhibits diagram.

The design selection for the power distribution is based on the trade studies, however, as stated in Phase I, it is highly recommend to use the power generation, energy storage, and power distribution from the same vendor for to its compatibilities. With the EPS components selected, the risks are updated and detailed V&V plans are developed. The detailed V&V plans for the EPS must include test plans for power generation, energy storage, and power distribution. In most cases, the energy storage devices require regular monitoring to ensure certain energy levels are maintained. Thus, this regular monitoring and maintenance of the energy levels must be included in the V&V plans.

GNC

For the GNC subsystem, the components selected in Phase I must be verified via updated simulations. Specifically, the spacecraft dynamics (including mass and moments of inertia) are updated from Phase I. Therefore, the simulations are updated and executed to ensure that the selected components satisfy the mission CONOPS and the requirements. Once the updated simulations are conducted and the components performances are verified, the risks are updated and detailed V&V plans for each component are developed. The V&V test plans for the GNC must include test plans for the sensors and actuators and in some cases, specific testing apparatus may be required (e.g., Helmholtz cage to isolate the Earth’s magnetic fields, air bearing tables to provide friction-less environment, and others).

Propulsion

If the containerized satellite includes the propulsion system, the detailed design includes the selection of the propulsion components (propellant, tank, thrusters and control electronics). Similar to the GNC subsystem, the simulations are updated and executed to verify that the selected propulsion components satisfy the mission CONOPS and requirements. The propellant budget is also updated and the contingency is lowered to 10% since the designs are matured. With the components selected, the risks are updated and the detailed V&V plans are developed. The detailed V&V test plans for the propulsion must include plans for all propulsion components (actuators and sensors). Similar to the GNC subsystem, specific testing apparatus may be required for the propulsion components as well (e.g., thermal vacuum chamber).

Structures and Thermal

During this phase the designs for the structure and thermal are selected. The selected structure in Phase I is simulated using finite element analysis software for vibrations and compression loads to emulate launch. In general, the loads from NASA's General Environmental Verification Standard (GEVS) are used, however, the loads differ depending on the launch vehicle. Based on the simulation results, design updates are made as necessary. During this phase, a virtual assembly (i.e., CAD model) of the spacecraft is required that includes all the components of the spacecraft. The virtual assembly is great way to perform a fit check to ensure that (i) all the components fit within the satellites, (ii) no interference within the satellite, and (iii) the satellite will fit inside the deployment container. With the virtual assembly, the outer dimensions, the mass, the center of gravity, and the moments of inertia are determined. Furthermore, with the detailed design, the mass budget is updated (see Figure 3-28). The mass budget in this phase is a better estimate compared to the mass budget during preliminary design (see Figure 3-17). While it is a better estimate, the contingencies such as cables, epoxies, Kapton tape, etc. must still be included in the mass budget.

For thermal, detailed thermal simulations are performed using the updated CAD model. Similar to the thermal simulations in Phase I, the external and internal temperatures of the spacecraft are determined. With the detailed CAD model and thermal simulations, the maximum and the minimum temperatures for each component are determined and used to compare to the operating temperatures of the components.

The results are assessed to examine if thermal control is required for the components. If thermal control is required, passive or active thermal protection and control are implemented. Trade studies are developed to select the best suited options for thermal control if necessary.

Subsystem	Components	Mass (kg)	Percentage
Payload	Technology	1.30	33.68
CDH	On-board Computer	0.10	2.59
EPS	Power distribution unit, battery, solar panels	0.93	24.09
GNC	Attitude Sensors and actuators	0.40	10.36
Structures & Thermal	Structure, fasteners, cables	0.50	12.95
TT&C	Transceiver, antennas	0.33	8.55
Others	Miscellaneous (epoxy, Kapton, etc.)	0.30	7.77
	Subtotal	3.86	
	Allowable Mass	4.00	
	Margin (%)	3.50	

Figure 3-28. An example of updated mass budget.

With the detailed designs, the risks are updated and detailed V&V plans are developed. The V&V plans for the structure and thermal must include environmental tests; typical environmental tests include vibration, shock, and thermal-vacuum tests. The environmental test parameters are dependent on the launch providers.

TTC

For TTC, the communication components and their antennas are selected for communication with the ground stations. The component selections are dependent on the ground station equipment, thus, the link budgets are updated and used to identify the best suited components. An example of a RF link budget is shown in Figure 3-29 where there are three link margins, i) VHF uplink, ii) UHF downlink, and iii) S-band downlink, show healthy link margins. With the selected components and the updated link budgets, the telemetry packet formats for both downlink and uplink are finalized.

Containerized satellites typically use AX.25 protocols for amateur VHF and UHF bands, where the AX.25 is a communication protocol that is common to amateur radio operators around the world [131]. The AX.25 packet includes a frame header (16 bytes) with designation address and source address, a data field (256 bytes), and a checksum (2 bytes). When using the AX.25 protocol, 256 bytes is the maximum amount of data per transmission. It is highly recommended to downlink satellite's health via beacons at least every 30 seconds. If the beacons can be downlinked more often, it would be recommended since many of the containerized satellites are launched in swarms (i.e., multiple containerized satellites in a single launch), therefore, there is a higher chance of finding your satellite within the swarm with a higher beacon rate.

With the components selected, link budget updated, and packet formats determined, the risks are also updated. In addition to the risk, detailed V&V plans for the TTC are also developed. The V&V plans for the TTC must include ground station in order to perform communication tests.

Parameter	VHF Uplink	UHF Downlink	S-Band Downlink
Frequency	145.800 MHz	437.450 MHz	2300.000 MHz
Data Rate	1200 bps	9600 bps	1 Mbps
Modulation Method	AFSK	GMSK	QPSK
Transmitter Output Power	100 W	2 W	1 W
Total Transmitter Line Losses	2.94 dB	0.1 dB	0.1 dB
Transmitter Antenna Gain	12.8 dBi	2.2 dBi	6.0 dBi
Calculated Transmitter EIRP	29.9 dBW	5.1 dBW	6.0 dBW
Total Radio Link Losses	142.0 dB	151.3 dB	170.0 dB
Receiver Antenna Gain	2.2 dBi	16.1 dBi	38.9 dBi
Total Receiver Line Losses	0.00 dB	2.3 dB	0.4 dB
Receiver LNA Temperature	75 K	60 K	60 K
Receiver LNA Gain	0.0 dB	20.0 dB	20.0 dB
System Noise Temperature	821 K	257 K	174 K
Calculated Eb/No	58.7 dB	32.3 dB	21.5 dB
Eb/No Threshold	24.2 dB	10.6 dB	10.6 dB
Link Margin	+34.5 dB	+21.7 dB	+10.9 dB

Figure 3-29. An example RF link budget.

Ground System

During this phase, the ground system designs are finalized, specifically, the ground stations and the MOC are determined. The selection is based on the requirements as well as the TTC design. An example of an UHF/VHF ground station setup is shown in Figure 3-30, where it shows the UHF/VHF antennas with pre-amplifiers that are connected to the rotor and on a mast, a radio that receives and transmit signals to the antennas, terminal node controllers (modems) that modulate and demodulate signals, and a computer equipped with ground software to control the rotors to track the containerized satellites. It is very important to include a proper grounding of the equipment and the antenna for the ground stations to reduce lightening damages.

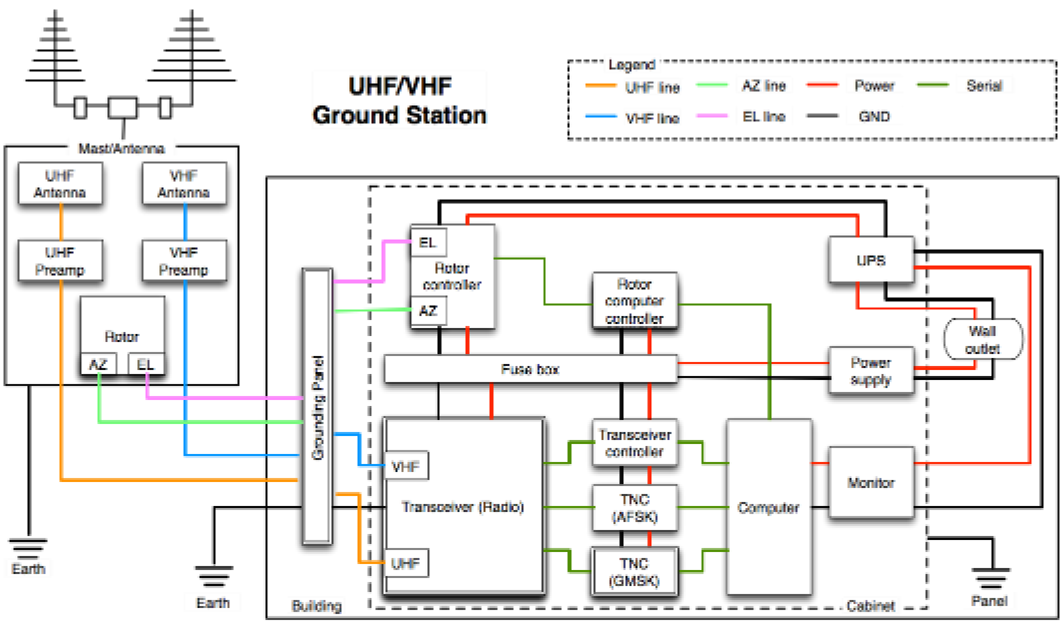


Figure 3-30. An example of an UHF/VHF ground station setup.

With the updated ground system design, the risks are updated and detailed V&V plans are developed with the TTC subsystem. For RF ground stations, it is highly recommended to perform communication tests with satellites in orbit to verify the equipment prior to verification with the containerized satellite. In addition to the V&V

plan, an operations plan that describes how, who, and when the mission operations are conducted are detailed. The operations plan must also include scenarios when there are unforeseen events and how these events are handled.

Interfaces

With the detailed design of the subsystems, the interfaces are identified and the interface diagrams are updated. An interface diagram, such as a detailed N2 diagram, is necessary to show the mechanical and electrical interfaces between each component (see Figure 3-31). In addition to the interface diagram, the PC/104 header pinout is updated to show that there are no interferences between the electronic boards. In addition to the interface diagrams and updated PC/104 header pinouts, a virtual assembly (i.e., CAD model) of the containerized satellite is developed using the detailed designs and used to help identify the external and internal interfaces. Furthermore, developing a cabling/wiring diagram is highly recommended. A cabling/wiring diagram shows all the connections and interfaces between each component and helps identify the necessary cable harnesses (connectors and lengths).

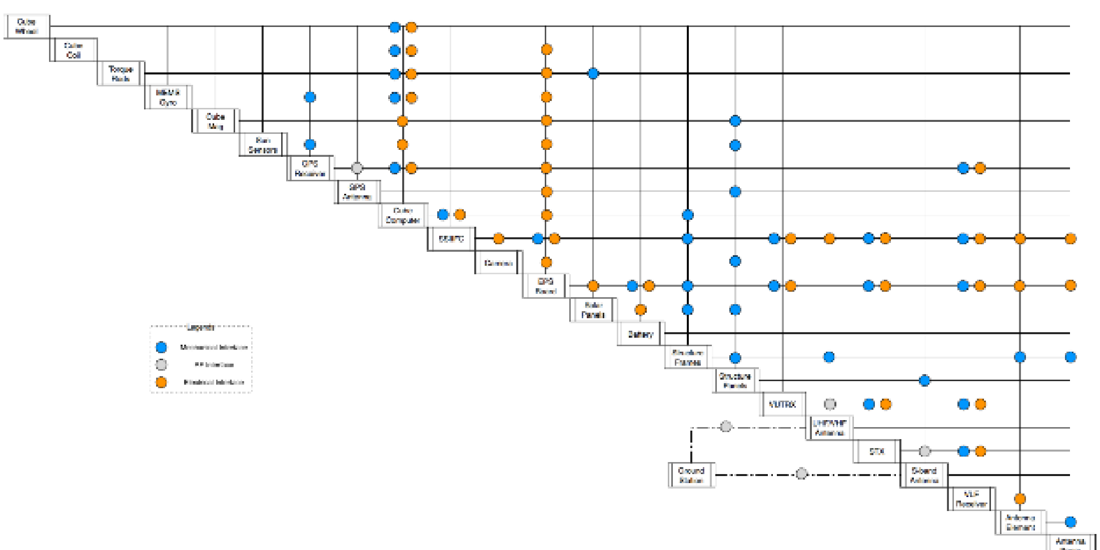


Figure 3-31. Example of a detailed N2 diagram.

The CAD model is a great way to perform virtual assembly and integration of the spacecraft. The virtual assembly and integration in computer-aided software is critical to ensure that all components interface together without any interferences, size and mass are within the requirements, and helps team members visually assemble and integrate prior to the physical assembly and integration. All the interfaces, diagrams, and CAD models are documented in a form of an interface control document.

Project Management

During this phase, the project managers maintain and organize the personnel and team members, update the project schedule, update cost budget, update facilities, work on licensing, and secure launch opportunities. The project managers must frequently communicate with the team members to ensure the project progresses. The schedule developed in Phase I is updated to reflect the status of the project. The schedule also includes contingencies to account for unforeseen events. During scheduling, it is recommended to identify the long lead items because most COTS components have 3-6 months lead time and must be accounted for. The cost budget is also updated with the detailed design and the budget must include non-material costs. Similar to the schedule, the cost budget also needs to include contingencies. Procurement of facilities is needed such that the later phases of the project life-cycles can be implemented. The facilities also include any ground station(s) that are required to operate/communicate with the containerized satellites. The RF licensing and frequency allocation typically requires dedicated ground station for uplink, therefore, procurement of the ground station is needed. The licensing depends on the country of origin and may require details of the launch, therefore, securing launch opportunity and manifestation may be required.

Critical Design Review

To advance to the next phase (i.e., exit criteria), the team must have a successful critical design review (CDR) with external reviewers. The CDR shall be conducted with external subject matter experts and funding stakeholders to obtain feedbacks. If the containerized satellite is manifested on a launch, the CDR shall include the launch provider as well to assess the project maturity. Successful CDR leads to increased confidence in the mission and allows the team to advance to the next phase.

The detailed designs are typically presented to reviewers in critical design review and a detailed design report. The teams will be required to provide the following design aspects for the critical design review:

- Action items from PDR
- Detailed mission CONOPS
- Refined work breakdown structure (WBS)
- Updated requirements and requirements verification matrix
- System architecture and overview
- Updated system budgets with contingencies/overhead
    - Power budget
    - Mass budget
    - Link budget
    - Telemetry budget
    - Mission schedule/timeline
    - Mission cost
- Detailed design documents for each subsystem
    - Payload
    - Command and data handling (CDH)
    - Electrical power system (EPS)
    - Guidance, navigation, and control (GNC)
    - Propulsion
    - Structural and thermal
    - Telemetry, tracking, and command (TTC)
    - Ground system (e.g., RF ground station, optical ground station, etc.)
- Refined interfaces and updated interface control document

- Internal and external interfaces
  - Detailed N2 diagram
  - Wiring diagram

- Virtual assembly and integration using CAD software
- Updated verification and validation plans including detailed test procedures
- Launch vehicle integration plan
- Mission operations plan
- Updated reliability analyses document (i.e., risk assessment)
- Updated regulatory licensing status
- Update on launch opportunity

It is important to note that the reviewers may differ from the reviewers from PDR, therefore, the team must present the action items from PDR and how they were addressed during this phase. As part of the CDR, the team would showcase the detailed mission CONOPS and requirements gathering and a means for verification (requirements verification matrix). A detailed design document for each subsystem to archive traceability and maturation from preliminary designs are needed. A review of the software design should also be conducted as part of the critical design review. Based on the detailed designs, the interfaces are all updated and archived in the interface control document. Updated V&V plans are presented including detailed test procedures for each subsystem. Also during this phase, launch vehicle integration plan and a mission operations plans are developed. The launch vehicle integration plan lists procedures up to the containerized satellite's integration to the deployment container at the integration site. The procedures typically include battery charging and health check. The mission operations plan include details of how the operations are going to be conducted, including personnel and schedules. It is important to note that after integration to the launch vehicle, there will be a wait period (anywhere from 1-6 months) that the launch vehicle waits until its launch. This period will vary on the launch vehicle and the launch opportunity, however, the team may need to include provisions in the

CONOPS prior to starting mission operations. For example, after ejection from the deployment container, wait 24 hours to charge the batteries prior to starting mission operations. Updated risks are also presented with mitigation strategies and the statuses of the regulatory licensing and launch opportunities are also presented in the CDR. The team should be mandated to seek at least one critical design review with external reviewers and highly recommended to conduct multiple internal (peer) reviews throughout this phase. Upon successful CDR, the team proceeds to the next phase and begin acquisition.

Phase III – Development and Unit/Integration Level Testing

Upon successful completion of Phase II, the team will delve into the acquisition and development of the various components and subsystems of the containerized satellite mission. The development phase will advance the design identified in Phase II and the goal of this phase is to develop a flat-satellite or even an engineering development unit (EDU) of the containerized satellite and the ground system. It is critical to get to this phase since there are unforeseen challenges when working with hardware and software. Developing a flatsat and/or EDU and verifying its functionality identify any design flaws and design iterations are made as necessary. In addition to the space and ground system(s), necessary testing apparatus are required to be developed in this phase.

The development phase begins with the acquisition of components. Some COTS components have longer lead times (up to 6 months) and must be accounted for in the project schedule. Once the components are acquired, metrology must be performed; inspect the components for visible damages and measure the sizes and masses. The sizes and masses are measured and compared to the specifications (these measurements are critical for items with tight tolerances). Take pictures during metrology for documentations. Once metrology is completed, component level verification is performed where functionalities of each component are verified. Once the components are verified, the components are integrated into subassemblies. These subassemblies are then verified through testing to ensure functionality. Upon verification, the subassemblies are then integrated into subsystem level and verified through testing. The virtual assembly and integration procedures from Phase II are used during the physical integration of components into subassemblies and subsystems. The verification process follows and implements the V&V plans developed in Phase I and Phase II (see Figure 3-13 and Figure 3-14). It is very important to note that the V&V tests must be conducted at least in pairs so that each action is checked and verified.

One of the challenges and risks during the development phase is to ensure that the payload is further developed compared to the satellite bus. The satellite bus components supports the payload, thus, if the payload is not fully developed, the spacecraft bus cannot be developed. To mitigate this, it is highly recommended to design, develop, and test the payload prior to this phase such that a functional prototype of the payload is available during this phase.

The development of each subsystem will fall into one of the following three categories:

- Development based around a COTS product
- Development based on a custom design
- Development based on a hybrid approach

For a subsystem developed around COTS products, the mechanical and electrical interfaces must be first identified. Based on those interfaces, testing apparatus are setup. It is highly recommended to acquire COTS products that have been validated in orbit (i.e., flight heritage), since it will have a higher TRL. Once the COTS products are acquired, metrology and functional testing must be performed. In some instances, the COTS products have a set of functionality tests programmed (i.e., health check), therefore, during component level testing, the set of functionality tests are used. For COTS products with low TRL and no flight heritage, functionality must be verified in orbit-like environments and TRL must be matured. For COTS products with flight heritage, the TRL does not need to be matured, however, the COTS product must be characterized before its integration with other components during component level testing. The COTS products are typically equipped with function-level software, thus, software development is typically not required to verify functionality of the component(s).

For a subsystem based on custom design, typically there are multiple revisions of the design, thus, sufficient time and testing must be allotted before the final product is realized. For the custom designs, it is crucial to repeat the development and testing cycle such that any design iterations and improvements can be made before the final design. Additionally, these custom designs will begin with a low TRL, therefore, it is necessary to perform multiple tests in relevant environments to mature the TRL. A subsystem based on custom design can optionally consider testing the entire subsystem as opposed to testing its individual components. However, for a component, the component level testing and characterization must be considered. Testing a custom design may require a specific testing apparatus, thus, the team and group responsible must plan during the design phase and develop the apparatus during this phase.

Last but not the least, a subsystem designed to accommodate a hybrid process, which will include both COTS products and custom designs will also require testing. Since the custom design begins with a low TRL, it must be verified in orbit-like environments to advance the TRL. Additionally, when the custom designs are integrated with COTS product with higher TRL, the assembled product becomes a new product and requires TRL maturation. The custom designs and the COTS products must be validated individually prior to integrating into an assembled product. The hybrid subsystem also requires verification and validation tests to mature the TRL. This hybrid process that includes both COTS products and custom designs are typical for payloads.

The CMG assembly presented as examples in previous phases followed this hybrid process where it utilized both COTS components and custom designs. Specifically, the flywheel motor, the gimbal motor, and the bearing were COTS components and the structural housing and the electronics board were custom designed. A prototype of the CMG assembly was first developed and tested and a flight version of the CMG assembly was developed and tested. For both assemblies, four single CMGs were developed and tested individually (component level) and upon verification, they were assembled into the CMG assembly. The CMG assembly was tested to ensure functionality (subassembly level). Upon verification, the CMG assembly was integrated with other subsystems and tested (subsystem level). The CMG assembly example is shown in Figure 3-32, where 1.0 represents the prototype and 2.0 represents the flight versions.

At this phase, it is highly recommended to perform hardware-in-the-loop (HIL) verification of the subassemblies and subsystems, as shown in Figure 3-33. The HIL tests emulate operational conditions and allows for rigorous testing of software and hardware at different levels. The HIL tests are conducted with various combinations of physical hardware/software and emulated components to verify functionality and operability. The mission CONOPS, spacecraft dynamics, and orbit disturbances are simulated. The results of the simulation are used to update the flight software. After metrology and component level verification tests are completed on the hardware, the components are assembled and integrated as subassemblies which replace the emulated systems in the HIL test. These subassemblies are then subjected to HIL testing for verification. Verified subassemblies are then integrated into subsystems and subjected to the HIL testing for verification.

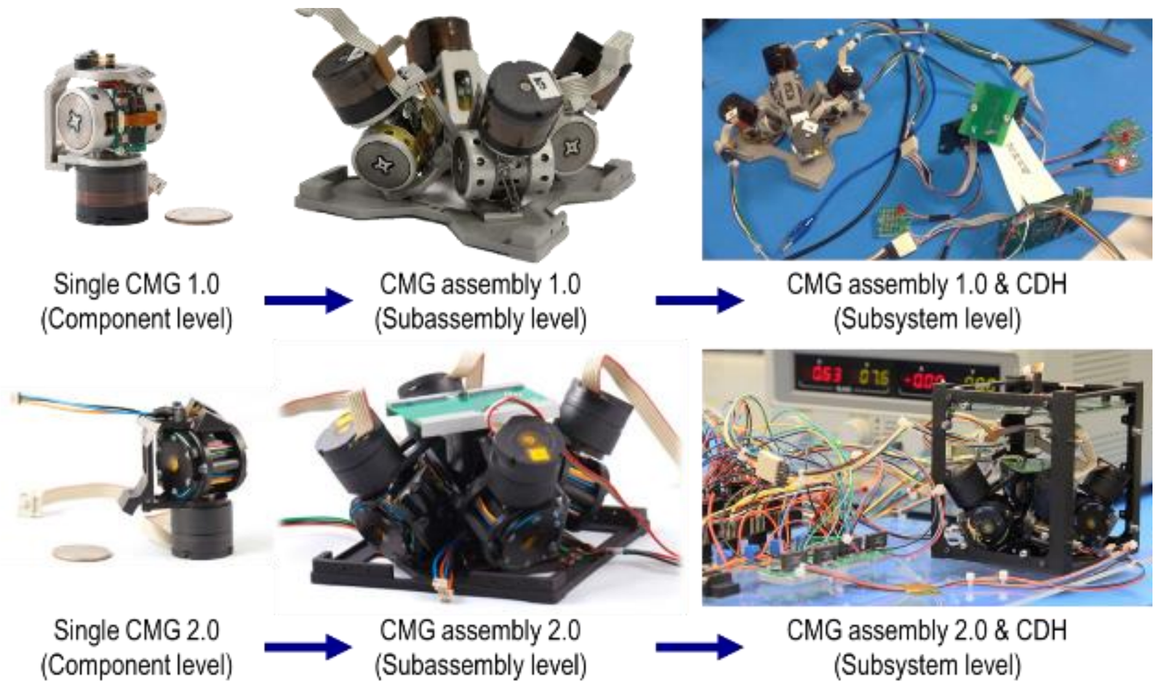


Figure 3-32. CMG assembly example. Courtesy of author.

As part of Phase III, a majority of the software development is also expected to be complete. Software for verifying functionality of components must be completed, however, the mission software may not be fully developed. The mission software must be completed in Phase IV, prior to performing a day-in-the-life (DITL) test. Accordingly, each team will demonstrate the successful development, testing, and integration of their subsystem by preparing verification and validation documents (including software) for each subsystem that includes the following documents:

- Verification and validation testing document for each subsystem that includes:
  - Unit/component level testing report
  - Subassembly level testing report
  - Subsystem level testing report
- Software verification report

The V&V testing documents detail how each test was conducted for each level and the results from each test. The software verification report is similar to the V&V testing documents and details how the software was verified starting at driver level and through function level and task level.

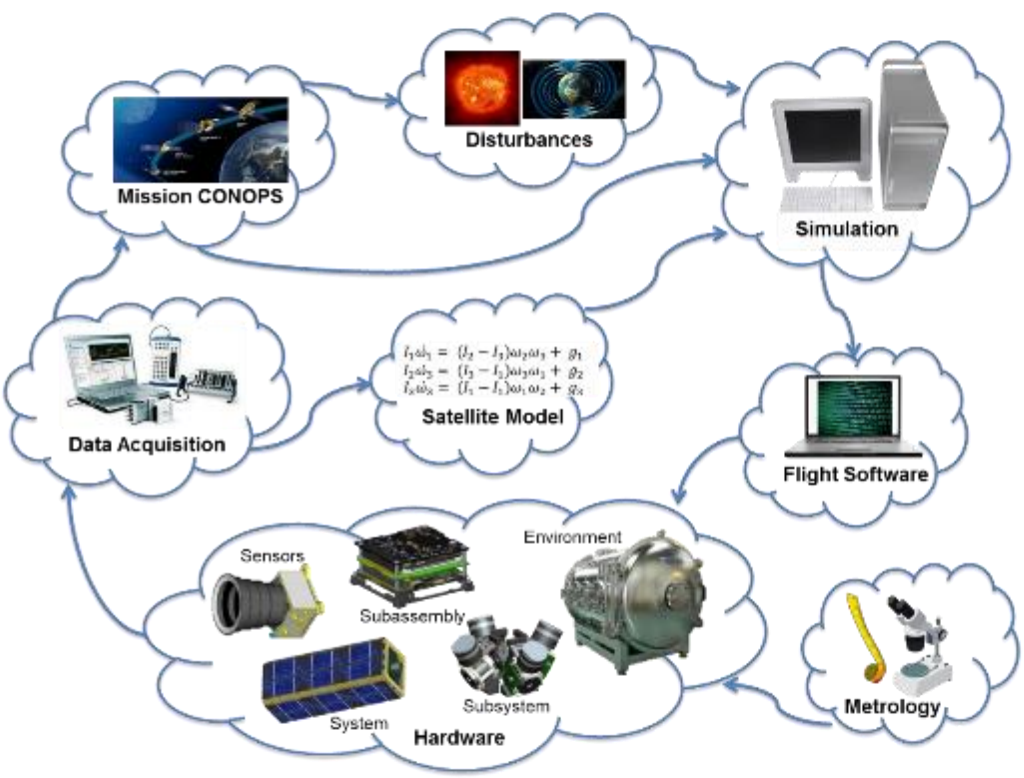


Figure 3-33. Hardware-in-the-loop (HIL) testing.

Design Implementation Review

At this phase, the team shall be mandated to seek a design implementation review (DIR) with external reviewers as the exit criteria from this phase. Successful DIR will advance the project to the next phase. The following are presented In the DIR:

- Action items from CDR
- Mission overview and system architecture
- Updated mission cost and schedule
- Verification and validation for each subsystem including software
    - Unit/component level
    - Subassembly level
    - Subsystem level
- Updated V&V plan for system level testing
- Updated launch vehicle integration plan
- Updated mission operations plan
- Updated regulatory licensing status
- Update on launch opportunity

As part of the DIR, the team presents the action items from CDR and how they were addressed during this phase. As previously mentioned, the reviewers may differ from previous reviews, therefore, it is critical to present the action items from CDR. The mission overview and system architecture along with updated mission cost and schedule are presented to the reviewers. At this phase, the team should be mandated to present at least a flatsat but an engineering development unit (EDU) is desired. The flatsat and/or EDU's assembly and integration procedures along with the verification tests are captured in the V&V documents. Although a clean room facility would be highly recommended for assembling the flatsat and/or EDU, a relatively clean environment, free of dust and static fields may be considered as an effective alternate (unless there are moving components in the satellite design). If the EDU is developed at this phase, it should be mandated to pass a fit check with a deployment container such as a poly-picosatellite orbital deployer (P-POD). The system level V&V plan is presented in detail along with updated launch vehicle integration plan and mission operations plan. Updates to regulatory licensing status and launch opportunities are also presented in the DIR. With successful DIR, the team advances and proceeds with the system assembly, integration, and testing.

Phase IV –System Level Assembly, Environmental Testing, and Launch

As part of Phase IV, the final phase of development, the team will assemble and integrate the various subsystems into space and ground systems. In this phase, an engineering development unit (EDU) of the containerized satellite is assembled and tested prior to integrating the flight unit of the containerized satellite. The system assembly follows the process from Phase III and will primarily address the mechanical and electrical integration of the various components. The EDU is subjected to qualification level tests to ensure that the design satisfies the requirements and once the EDU successfully completes the qualification level tests, the flight unit is assembled. The team must assemble the flight unit in a clean room facility. During the assembly and integration (EDU or flight unit), the team is required to ensure the functionality at each step of the integration. Additionally, it is recommended to measure the size and mass of each assembly and of the system once it is fully assembled. These size and mass measurements of the flight unit are typically required by the launch provider for launch.

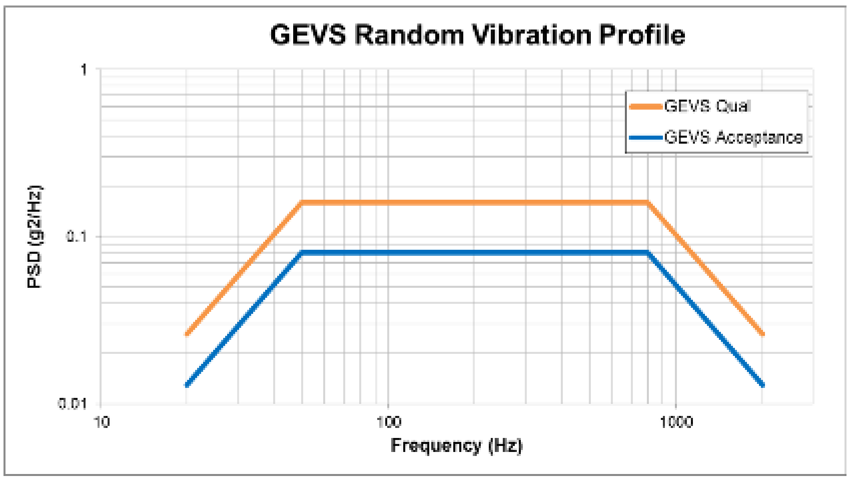
As part of the final phase of development, the teams will implement the system level test plan (including a reporting section), that includes functionality tests to ensure that all components are function after the assembly and integration. The same functionality tests are conducted pre and post the environmental tests to ensure the components survived the environmental tests. It is critical to perform the same functionality tests pre and post environmental tests to identify any failures to the components based on the environmental tests. Depending on the launch provider, the environment tests may vary, however, these are typical environmental tests that assembled containerized satellites are subjected to:

- Vibration testing to qualify for a potential launch vehicle and launch environment
- Thermo-vacuum testing to qualify for the space environment
- Shock testing to qualify for a potential launch vehicle and launch environment
- Radiation testing to qualify for the space environment

It is important to note that the vibration and shock tests must be conducted with the containerized satellite integrated into the deployment container. These tests are performed to qualify the containerized satellite for the launch environment, thus, the containerized satellite must be integrated in the deployment container. Typically, random vibration tests are conducted on three-axes, one random vibration test per axis. For thermal-vacuum testing, a temperature profile that follows an orbit-like environment is applied once the containerized satellite is in vacuum. Similar for radiation tests, the gamma rays are emitted towards the containerized satellite once in vacuum.

As previously mentioned, the details of the environmental tests depend on the launch provider, but in general, there are two levels of tests: qualification and acceptance. Qualification level tests are more rigorous and are tested on EDUs and acceptance level tests are less rigorous than qualification tests and are for flight units. As an example, the GEVS random vibration profiles for qualification and acceptance levels are shown in Figure 3-34, where the qualification levels (14.1 G_{rms}) are more rigorous than the acceptance levels (10.0 G_{rms}).

Once successful environmental tests are conducted, the team will perform a day-in-the-life (DITL) test that tests the entire mission CONOPS on the ground. The DITL test uses the space and ground systems for the test, therefore, the software for both space and ground systems must be completed. The DITL test is performed to refine the mission operations plans and to ensure that the space system (containerized satellite) and the ground system can communicate. The DITL test is critical for the mission and is required to be successfully completed.



	Qual	Acceptance
Frequency	PSD	PSD
(Hz)	(g^2/Hz)	(g^2/Hz)
20	0.026	0.013
50	0.16	0.08
800	0.16	0.08
2000	0.026	0.013
G_{rms}	14.1	10.0

Figure 3-34. GEVS random vibration profile [114].

Flight Readiness Review

With the successful completion of the system level tests, the team addresses the fulfillment of each requirement identified in the requirements document and in the requirements verification matrix. The team should be mandated to seek a flight readiness review (FRR) with external reviewers. As part of the FRR, the external review team would evaluate all the system level functional and performance test reports. The review team will also be presented with environmental testing reports, which will include the above qualifications. It is important to note that the FRR is conducted with the launch provider, therefore, the containerized satellite must be manifested on a launch.

The following are presented in the FRR:

- Action items from DIR
- Updated mission cost and schedule
- System level verification and validation test report
- Updated requirements verification matrix
- Updated launch vehicle integration plan
- Updated mission operations plan
- Updated regulatory licensing status

In the FRR, the action items from the DIR and how the action items were addressed are presented. Updated mission costs and schedule are presented. The system level test results, including environmental tests and functionality tests, are presented to show that the containerized satellite is ready for launch. The updated requirements verification matrix show that all the requirements have been verified and provides confidence to the reviewers and launch provider that the team is ready for launch and operations. The launch vehicle integration and operations plans are updated and finalized. The regulatory licensing status is also presented to show the launch providers that the licenses have been acquired. In most cases, the launch provider will require a regulatory license prior to launching the satellite, therefore, the regulatory license must be obtained at this time.

A successful FRR means that the containerized satellite is ready to be launched. After a successful FRR, the team packages the containerized satellite and is required to hand over and deliver the containerized satellite to the launch provider or a launch facilitator for its journey into an orbit. In some instances, the teams are allowed to perform functionality checks at the launch vehicle integration site after delivery, therefore, the launch vehicle integration plan is implemented and a final checkout before integration is conducted. After integration to the launch vehicle, the launch vehicle awaits the launch opportunity.

Phase V – Post Launch Operations

The final phase of the containerized satellite mission involves the post launch operation. As part of the early phases (Phase II, Phase III, and Phase IV) a detailed mission operations plan was developed and the plan is implemented in this phase. This operations plan and the ground system would be emulated and tested for functionality in Phase III and Phase IV during V&V tests. The RF ground station facilities will be tested for successful operation by communicating with existing satellites of the same range of frequency. It is important to note that when RF communications are used between the containerized satellite and the ground station, only radio-licensed operators may uplink commands to the satellite.

As part of the mission operations, the team will implement a data collection, storage, and distribution scheme, where the payload data along with satellite health data are gathered, stored, and downlinked to the ground. Commands from the ground station are uplinked to the containerized satellite if necessary. If there are malfunctions during mission operations, a flatsat or an EDU developed during Phase III is used to troubleshoot. Typically, a reset command is uplinked to the ground station if malfunctions occur.

Mission Closeout Review

After the mission and mission objectives are completed, the collected data are analyzed and shared with the stakeholders. Additionally, a final report is created to document the collected data as well as the lessons learned. The lessons learned are important and should be obtained from the team members and used in future missions.

A mission closeout review (MCR) with the stakeholders and/or external team are conducted and a decision to continue the mission or decommission the containerized satellite is made. When the mission is decommissioned, the containerized satellite is retired and typically deorbits into Earth's atmosphere.

Summary

The Containerized Satellite Mission Life-Cycle presented a structured end-to-end process for small satellites, specifically for containerized satellites. The mission life-cycle provides the fundamental procedures and protocols to limit ad-hoc development for small satellites. The first phase, Pre-Phase I, provides systems engineering training to teach the team the complete life-cycle. This first phase is unique and for those with experiences with systems engineering, this phase does not need to be exclusively adopted. On the other hand, academic institutions and new space entrants with limited systems engineering experiences are required to implement this phase. The next phase, Phase I, identifies the mission concept and the preliminary design commences. During this phase, it is critical to clearly identify the mission objective(s) and requirements since the designs are based on them. In other words, the mission objectives and system level requirements are decomposed and allocated into lower level requirements (i.e., requirements flowdown), therefore, a clear and well-written mission objectives and requirements are necessary. Based on the requirements, a preliminary design is developed. The preliminary design is then matured into detailed design and a virtual assembly of the design is performed in Phase II. The virtual assembly visually aids in the assembly, integration, and testing of hardware in the proceeding phases. In the next phase, Phase III, components are developed, integrated, and tested. The key activity during this phase is to develop at least a flatsat or an engineering development unit to ensure the designs satisfy the requirements and to make design provisions if necessary. There are unforeseen challenges during the assembly and verification of hardware and software, thus, it is critical to perform assembly, integration, and testing to develop a flatsat or an engineering development unit. The next phase, Phase IV, develops and assembles the flight unit and subjects the flight unit to environmental tests to test functionality in operational conditions. After successful environmental tests, a day-in-the-life test is conducted. The day-in-the-life test is critical for mission operations. Once the flight unit is verified, the flight unit is launched and enters the operations. Mission operations are executed to achieve mission objective(s) in Phase V. Once mission objective(s) are achieved, the spacecraft is decommissioned and retired. Reviews with external subject matter experts and stakeholders are utilized to transition between the phases and internal (peer) reviews are performed throughout the phases.

Figure 3-35 summarizes the different levels of the space and ground systems over the Containerized Satellite Mission Life-Cycle, where the life-cycle begins at system level, decompose into subsystem and component level for design (Phase I and Phase II), then components are developed and integrated into subassembly and subsystem level (Phase III), and ultimately integrated as a system (Phase IV). Once the space and ground systems are verified, they are put into operations.

This research effort began with performing reliability analyses on SwampSat, a 1U CubeSat designed and developed at the University of Florida (shown in Figure 3-36). SwampSat's mission was to validate on-orbit a compact, three-axis attitude actuator using four control moment gyroscopes (CMGs) in an assembly. SwampSat successfully launched in November 2013 through NASA's ElaNa IV program. Two reliability analysis techniques were utilized, a failure modes, effects, and criticality analysis (FMECA) and fault tree analysis (FTA) to identify potential failure modes for SwampSat. Based on the analyses, various risks were identified and risk mitigation strategies were developed and implemented [124] [132]. One of the main risk mitigation strategies were to perform rigorous testing in different environments; starting from component level to subassemblies and subsystem level and at system level. These rigorous tests were developed and implemented for SwampSat.

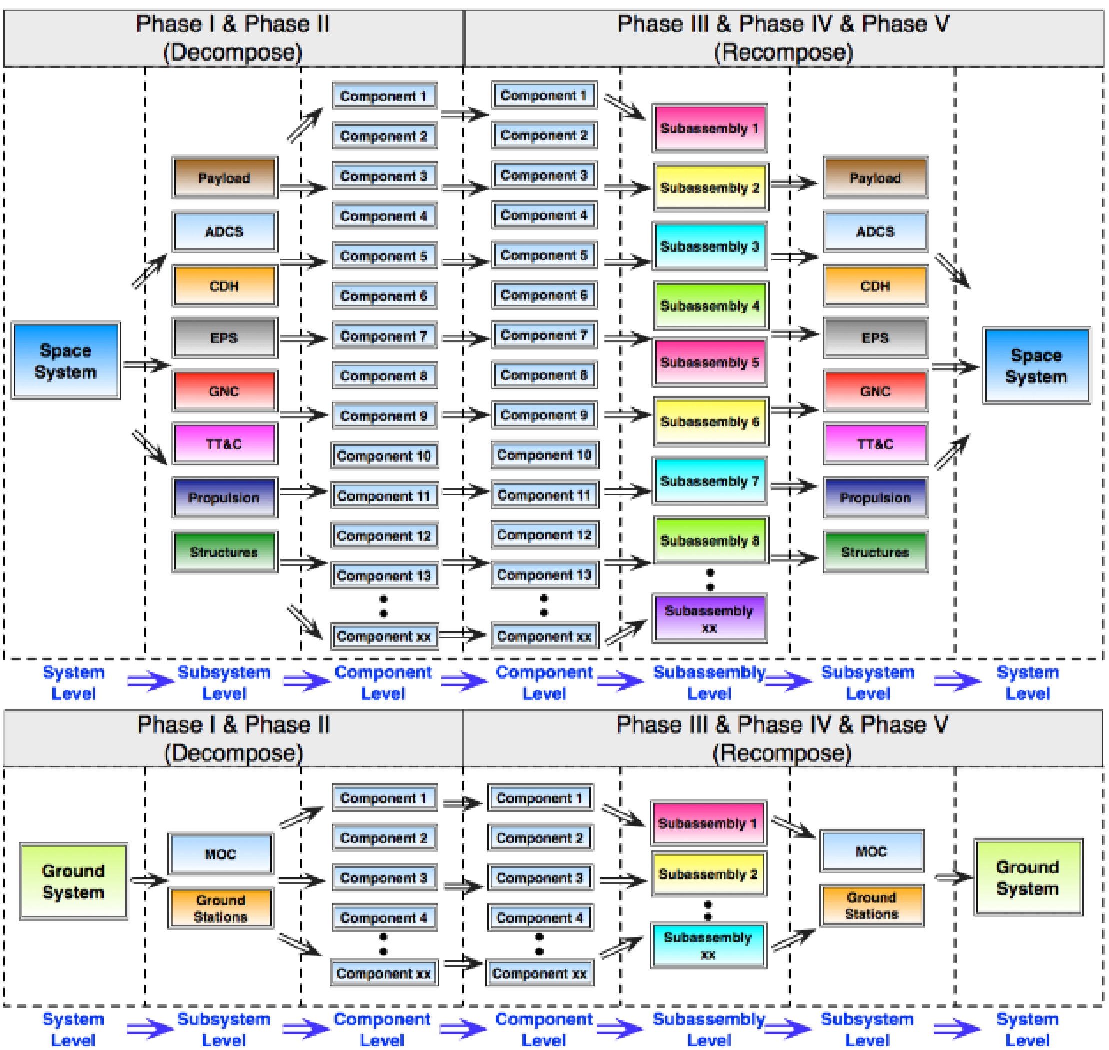


Figure 3-35. Different levels of system over the project life-cycle.

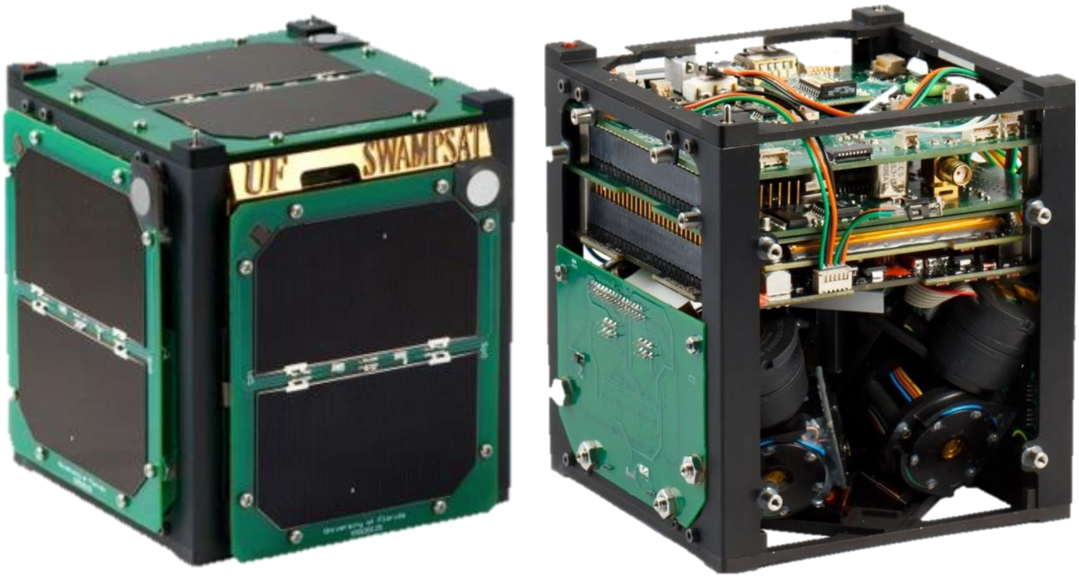


Figure 3-36. Pictures of SwampSat. Courtesy of Kunal Patankar.

While rigorous tests and risk mitigation strategies were developed and implemented for SwampSat, many of the engineering activities during SwampSat development were performed in an ad-hoc manner where no structured process was followed. During the development of SwampSat, it was realized that a structured systems engineering process was required for development of containerized satellites. By utilizing a systematic approach, a more robust system capable of adapting to potential failures can be developed. The Containerized Satellite Mission Life-Cycle was developed based on the experiences of SwampSat and other small satellite projects.

CHAPTER 4
PROJECT LIFE-CYCLE IMPLEMENTATION

The Containerized Satellite Mission Life-Cycle presented in Chapter 3 showed the end-to-end process for small satellites, specifically for satellites that are launched from containers (i.e., containerized satellites). The Containerized Satellite Mission Life-Cycle is flexible and adaptable and as such it was applied and implemented for two missions, one to an actual mission known as SwampSat II and the other to a non-mission project known as DebriSat. The details are presented in this chapter.

SwampSat II

The objective of the SwampSat II mission is to characterize naturally-occurring, intentionally transmitted, and lightning-generated narrowband very low frequency (VLF) signals and their propagation through the lower ionosphere to better understand the natural and controlled loss of energetic radiation belt particles. Theoretical predictions of the VLF radio wave power that is injected into the magnetosphere by ground-based sources, such as man-made transmitters, remain an essential component of understanding the natural and controlled loss of energetic radiation belt particles. Significant discrepancies exist between the theoretical predictions and experimental observations at low Earth orbit (LEO) for VLF waves [133] [134]. These discrepancies greatly impact previous theoretical calculations that analyzed the relative importance of physical mechanisms for radiation belt decay. Significant model improvements may be necessary to determine where the VLF power propagates.

The VLF frequencies are between 3 – 30 kHz and requires an antenna-receiver system capable of sampling frequencies at those range. In order to complete the mission, SwampSat II will deploy a payload consisting of a 16 meter square-loop antenna on a 3U CubeSat (see Figure 4-1). Deploying large-scale loop antennas introduces the risk of storage and entanglement, not seen in other large deployable missions, but offers a higher quality VLF signals compared to other antenna designs.

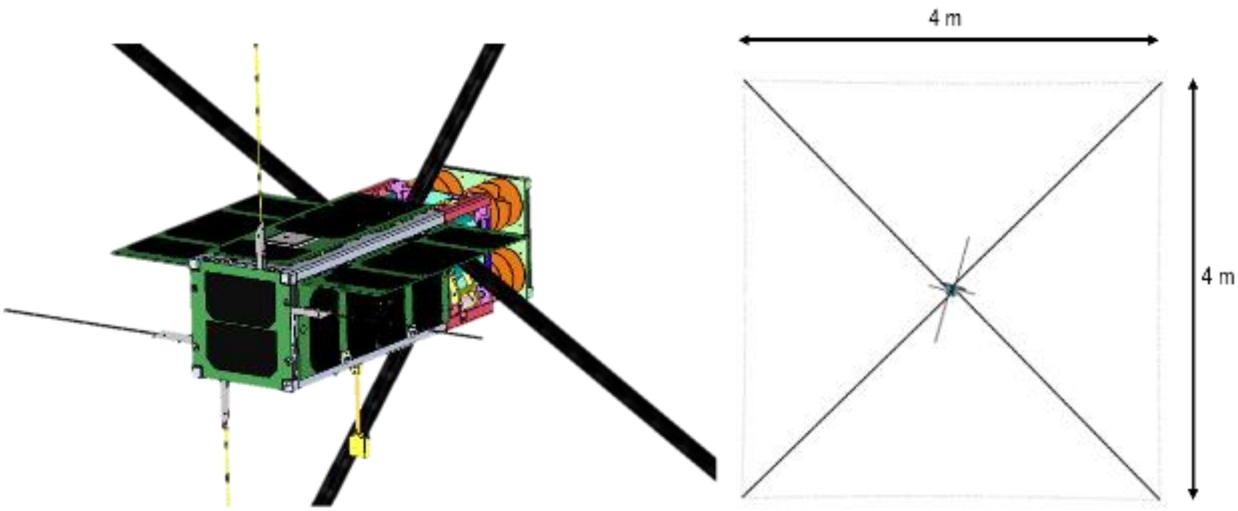


Figure 4-1. Virtual assembly of SwampSat II.

The Containerized Satellite Mission Life-Cycle has been implemented for the SwampSat II mission. In Pre-Phase I, the team members were trained on the end-to-end process since the team members may contribute to different satellite subsystems throughout the project and collectively realize the space and ground systems required to satisfy the mission objectives. This process is known as the conjunctive group task, where the group members interact with one another to produce a product [118] [135]. In a conjunctive group task, the performances of the group members are improved due to the Kohler effect. The Kohler effect occurs when the team members exert greater effort when working in groups [119]. Majority of the team members were new entrants with limited knowledge and experiences, thus, the training was required.

In Phase I, the mission definition, the mission objectives, and the external drivers (i.e., constraints) were used to identify the mission concept of operations (CONOPS).

Based on each operating mode of the mission CONOPS, the tasks and the components that perform those tasks were identified. These components were then organized into the product breakdown structure (PBS) from which the work breakdown structure (WBS) was developed. Based on the decomposition and the WBS, the requirements including subsystem and component requirements were developed and listed in the requirements verification matrix. Based on the requirements, preliminary designs for each subsystem were developed. For example, the science data transmitter's preliminary design for the telemetry, tracking, and control (TTC) subsystem was developed as follows:

- Performed analyses
    - Examined payload data generation
    - Used link analysis to examine different hardware configurations (transmit power and antenna configurations) that closed the link
- Performed trade studies
    - Used component requirements as the parameters in the weighted design matrix
    - Identified available COTS components in weighted design matrix
    - Identified best-suited design for the science data transmitter

The simulations and analyses such as orbit, attitude, power, communication, structural, and thermal were performed and based on the analyses, trade studies using weighted design matrices were conducted to develop the preliminary designs for all subsystems. Once the preliminary designs were completed for each subsystem, the system budgets were updated with contingencies. With the preliminary designs for each subsystem completed, the interfaces (internal and external) were identified and the verification and validation (V&V) plans were developed. A preliminary design review (PDR) with subject matter experts and stakeholders were conducted to assess the activities performed in Phase I. The PDR with external reviewers was successful and all the reviewers were satisfied and commented on the structure and organization of the SwampSat II project. The reviewers noted that typically reviews with academic institutions lack a structured process, however, the SwampSat II project showed that systematic approaches were implemented. With the successful PDR, the SwampSat II project transitioned to Phase II.

SwampSat II is currently in the detailed design phase (i.e., Phase II of the Containerized Satellite Mission Life-Cycle) and thus, at time of this writing, the success (or failure) of the Containerized Satellite Mission Life-Cycle cannot be fully quantified. Therefore, to assess the efficacy of the Containerized Satellite Mission Life-Cycle process, it was applied to another ongoing activity in the lab and is discussed in the next section.

DebriSat

Recently, two catastrophic on-orbit collision events occurred that significantly impacted the space debris community:

- The Fengyun 1C missile test in 2007 [75] and
- The accidental collision of Iridium 33 and Cosmos 2251 in 2009 [76].

After these catastrophic events, NASA utilized its current breakup model and compared to the cataloged SATCAT data [77] [78] . NASA’s model predictions matched well for the older satellite (i.e., Cosmos 2251), but, demonstrated significant differences for the modern satellites (i.e., Fengyun 1C and Iridium 33). The inaccuracies in the current breakup models (see Figure 1-6) were attributed to the use of modern material and development techniques in the fabrication of the modern satellites; thus, there was a need to update the breakup model for the future of manned and unmanned space missions. To update the breakup model, the DebriSat project was conceived. The DebriSat test article was a 56-kg satellite (shown in Figure 4-2) that was developed as a representative modern LEO satellite constructed from modern materials, components, and process techniques. The DebriSat test article was subjected to a laboratory hypervelocity impact (HVI) test in April 2014 [74]. The debris fragments resulting from this laboratory HVI test have been collected and are being characterized. The DebriSat project is an on-going collaborative effort by NASA, the DoD, the Aerospace Corporation, and the University of Florida (UF) to update the current breakup models.

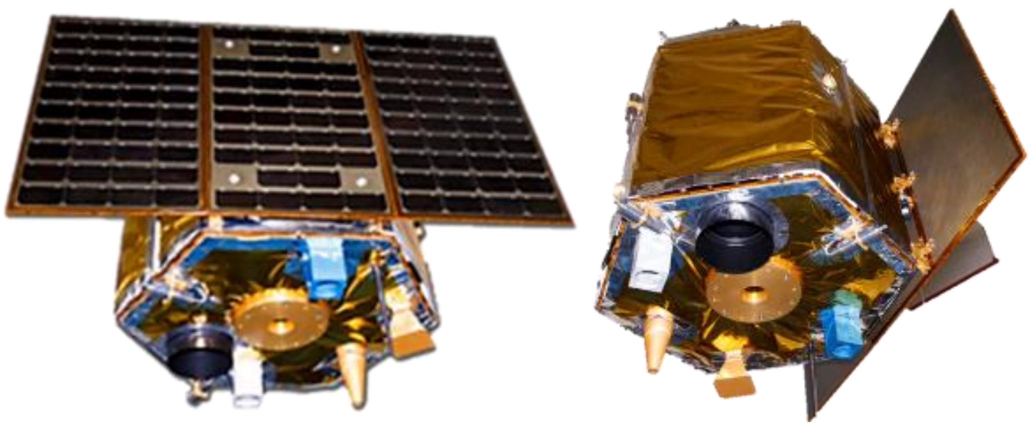


Figure 4-2. Pictures of DebriSat. Courtesy of Moises Rivero and Kunal Patankar.

The DebriSat project consisted of three distinct phases: pre-HVI test, HVI test, and post-HVI test. The pre-HVI test phase involved the design, manufacturing, and assembly of the DebriSat test article, which are the early phases of a project life-cycle. The HVI test and the post-HVI test are equivalent to the launch and post-launch operation phases, respectively. Currently, the DebriSat project is in the post-HVI test phase. The Containerized Satellite Mission Life-Cycle processes were not applied to the pre-HVI test and HIV test for DebriSat since the Containerized Satellite Mission Life-

Cycle was in development; however, the phases of the Containerized Satellite Mission Life-Cycle have been implemented in the post-HVI test phase of the DebriSat project.

Post-Hypervelocity Impact Test Activities

The Containerized Satellite Mission Life-Cycle phases have been implemented for the post-HVI test phase of the DebriSat project. In the post-HVI test phase, all debris fragments with a linear dimensions of 2 mm and greater are collected, characterized, cataloged, and stored (see Figure 4-3). During collection, the fragments are carefully extracted from foam panels that were installed in the HVI test chamber to capture the fragments during the HVI test. X-ray images of the foam panels are used to locate the embedded fragments prior to extraction. During the fragment characterization, each fragment’s size, material, shape, and color are assessed and entered in the Debris Categorization System (DCS) database. The DCS is a database solution that was designed and developed to manage the large amounts of data generated by the DebriSat project [136] [137] [138]. Once the fragment’s physical attributes are assessed, the fragment’s mass and sizes are measured. The measurements and other associated data are all cataloged in the DCS database. The fragments are physically stored in containers until they are shipped to NASA. The DebriSat post-HVI activities and the characterization efforts have been archived in References [139] and [140].

A rigorous process is required to collect, characterize, catalog, and store the estimated 250,000 debris fragments greater than or equal to 2 mm. Moreover, since the post-HVI test phase covers several years, the process needed to be independent of the operators performing the various tasks within the process since there would be several personnel changes during the period of performance of these tasks.

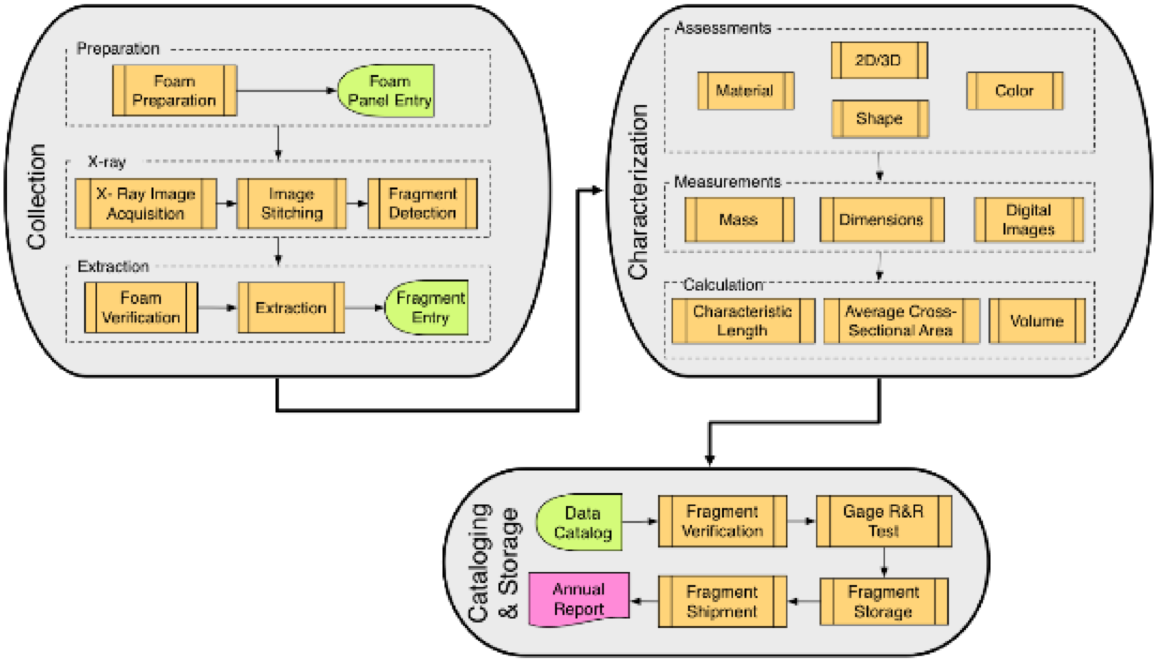


Figure 4-3. DebriSat post-hypervelocity impact test activities.

In the beginning of the post-HVI test phase, the stakeholders estimated 85,000 debris fragments would be generated from the DebriSat's HVI test. Based on their estimates, the stakeholders (NASA, the DoD, and The Aerospace Corporation) provided the objectives and requirements which were as follows:

- Objectives
  - Extract, collect, and characterize fragments
  - Recover 90% of the original DebriSat test article's mass
- Requirements
  - Characterize fragments greater than or equal to 2 mm in length
  - Measure size characteristics within 10%
  - Archive generated data and the data must have perpetuity

In addition to the above objectives and requirements, the stakeholders wanted to begin the processing and characterization of the fragments as quickly as possible, thus, many of the procedures were initially developed in an ad-hoc manner. Furthermore, the procedures used in previous hypervelocity impact tests were not applicable for the DebriSat project since many of the procedures included human-in-the-loop measurements that introduced significant human error. For example, the fragments collected from the Satellite Orbital Debris Characterization Impact Test (SOCIT) were manually measured using calipers and graph paper for size [71] [72]. Essentially, there were no previous projects and procedures that the DebriSat's post-HVI test activities could follow and implement. The tasks for the post-HVI test was to extract and collect the fragments out from the foam panels, characterize the fragments, and to archive the fragment data. In other words, the concept of operation (CONOPS) and the operating modes for the post-HVI test phase were identified as: collection, characterization, cataloging, and storage.

Prior to the implementation of the Containerized Satellite Mission Life-Cycle, procedures to extract and collect the fragment from the foam panels and data cataloging were developed while procedures for fragment characterization did not exist. However, the ad-hoc procedures for collection were hurriedly developed and not subjected to rigorous verification of the procedures, thus leading to inefficiencies. For example, the initial object detection algorithm used with the X-ray images of the foam panels was not verified and resulted in significant false positives (i.e., objects identified by the algorithm but did not physical exist); these false positives resulted in increased time spent on the fragment extraction task.

After implementation of the Containerized Satellite Mission Life-Cycle, the procedures for collection were updated and the procedures for fragment characterization, cataloging, and storage were developed. Specifically, the procedures were developed to be repeatable such that it was independent of the operator performing the tasks. Furthermore, where possible, the procedures were automated to increase productivity and efficiency while minimizing fragment handling and transcription errors (i.e., errors occurring during recording of the measurement).

Table 4-1 summarizes the pre- and post-implementation of the Containerized Satellite Mission Life-Cycle, where the columns shows the procedure statuses at the time of reviews with the stakeholders. The Containerized Satellite Mission Life-Cycle was implemented after the January 2016 review with the stakeholders and by the November 2016 review, all the procedures for the post-HVI test phase were developed. Based on the implementation of the Containerized Satellite Mission Life-Cycle, the activities for the post-HVI were clearly identified and the stakeholders were able to fully develop requirements (i.e., identify data to be archived) for each activity. After the November 2016 review, the stakeholders provided an updated list of requirements for each activity. Based on the requirements, the procedures were further updated and verified to satisfy those requirements (most right column in Table 4-1).

Table 4-1. Post-HVI test procedures pre- and post-implementation of the Containerized Satellite Mission Life-Cycle.

	Pre-implementation	Post-implementation	
Collection			
Preparation	Rev. 1	Rev. 2	Rev. 2
X-ray	Rev. 1	Rev. 2	Rev. 2
Extraction	Rev. 1	Rev. 2	Rev. 2
Characterization			
Assessment	None	Rev. 1	Rev. 2
Measurement			
Mass	None	Rev. 1	Rev. 2
Size: 2D	None	Rev. 1	Rev. 3
Size: 3D	None	Rev. 1	Rev. 2
Cataloging and Storage			
Data storage	Rev. 1	Rev. 2	Rev. 3
Verification	None	Rev. 1	Rev. 2
Gage R&R	None	Rev. 1	Rev. 2
Reviews	January 2016	November 2016	November 2017

One of the requirements for the size characterization is to determine the characteristic length of each fragment within an error of 10%. The characteristic length (L_C) is a parameter used by NASA to quantify the fragment's size and is defined as the average of the fragment's three largest orthogonal dimensions. To measure the size, two imaging systems were designed and developed, a 2D imaging system and a 3D imaging system. Both imaging systems utilize point-and-shoot cameras for object image acquisition and create representative point clouds of the fragments to compute the characteristic length. The design, development, and verification efforts for the imaging systems have been archived in Reference [141].

Implementation of the Containerized Satellite Mission Life-Cycle resulted in three revisions of the 2D imaging systems to improve accuracy and productivity. As a consequence of implementing the Containerized Satellite Mission Life-Cycle, two 2D imaging systems have been developed to increase the productivity of the size characterization process. The 2D imaging system consists of a single point-and-shoot camera and a platform with front and back lighting. Images of the fragment on the platform are taken and processed to generate a point cloud. The point cloud is then used to calculate the characteristic length.

The first revision of the 2D imaging system used a black felt curtain and a frame to enclose the camera and the platform. The first revision only measured the two largest orthogonal dimensions and assumed that the third dimension was negligible. The second revision used a blackout shroud and a right-angle prism mirror was added to provide a side view of the fragment to extract the third dimension (i.e., height). Since the addition of the mirror, the L_C calculation was updated to include the third dimension. For the third revision, an acrylic enclosure was developed to improve efficiency rather than opening and closing the shroud.

For each revision, the L_C errors were determined by comparing measurements to two calibration washers with different heights. Physical measurements of the washers were taken and used to compute the point clouds of the washers. Based on the point clouds, the characteristic lengths were determined and used as the “truth” dimensions. The percent error is calculated by using Equation 4-1, where the measured is the 2D imaging output and the actual is the “truth” dimensions. On each 2D imager, ten measurements were taken for each calibration washer and the average of the measurements was used to compute the L_C errors.

$$\% \, \mathrm{Error} = 100\left(\frac{measured - actual}{actual}\right) \tag{4-1}$$

Figure 4-4 shows the three revisions of the 2D imaging system and Table 4-2 shows the L_C errors for each revision. All the L_C errors have improved, except on Imager 2 between revision 2 to revision 3 when measuring washer 2, however, the error difference was small enough that revision 3 has been used during size measurements.

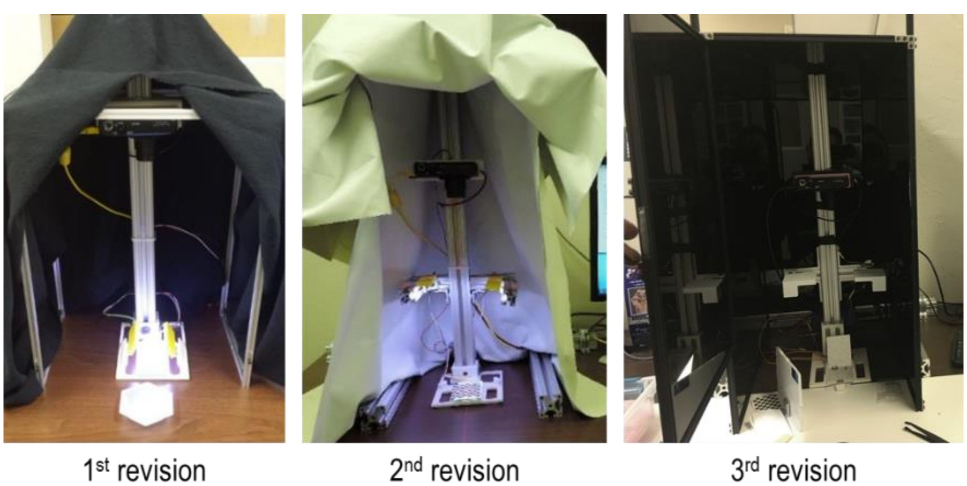


Figure 4-4. Three revisions of the 2D imaging system.

Table 4-2. L_C errors on three revisions of the 2D imaging systems.

Imager	Object	L_C Error (%)		
		Rev 1	Rev 2	Rev 3
Imager 1	Washer 1	-1.93	-1.22	0.75
Imager 1	Washer 2	-1.18	-0.90	0.89
Imager 2	Washer 1	-1.18	-0.55	-0.06
Imager 2	Washer 2	-0.54	-0.26	0.29

For the 3D imaging system, the L_C error initially did not satisfy the 10% error requirement, but after the Containerized Satellite Mission Life-Cycle implementation, the L_C measurements satisfied the requirements and were within the 10% error bound. The details of the 3D imaging system are presented later in this chapter.

Since the January 2016 review, the Containerized Satellite Mission Life-Cycle has been utilized to develop and improve the productivity and efficiency of the procedures, specifically for characterization (shown in Table 4-3). The processing times per fragment for each characterization activity are shown in Table 4-3 where the Pre-implementation times represent estimates, the Post-implementation shows the average processing time per fragment, and Current shows the improved average processing time per fragment. During pre-implementation, there were two characterization time estimates, one for 2D fragments and the other for 3D fragments. The fragments are classified as either 2D or 3D based on their size, where the heights of the 2D fragments are negligible compared to their other dimensions. The assessment and mass measurement procedures are the same for all the fragments, but the size measurement differs and are based on the size classification (2D imaging system or 3D imaging system). The time estimates represent the total time for assessment, mass measurement, and size measurement for 2D and 3D fragments. Prior to the implementation, the processing times were estimates (i.e., assumptions), however, after the implementation, the actual processing times were determined. The individual processing times per fragment were computed by utilizing logs that the operators fill out during each characterization activity. The assessment, mass measurement, and size measurement times were summed to compute the fragment characterization times for 2D and 3D fragments.

Table 4-3. Comparison of characterization processing times per fragment.

	Pre-implementation**	Post-implementation	Current
Assessment	--	5.6 min	4.3 min
Measurement##			
Mass	--	6.7 min	4.5 min
Size: 2D	--	9.1 min	8.3 min
Size: 3D	--	53.2 min	50.2 min
Total: 2D fragment	10 min	21.4 min	17.1 min
Total: 3D fragment	30 min	65.5 min	59.0 min

- Represents measurement and data cataloging times
** - Initial time estimates

The characterization time estimates for the 2D fragments were 10 minutes per fragment and 30 minutes per fragment for the 3D fragments. The time estimates for the 3D fragments were higher than the 2D fragments since the 3D size measurements required more images for measurements (126 images for 3D imaging system and 2 images for 2D imaging system). These estimates were grossly underestimated compared to the processing times after the implementation of the Containerized Satellite Mission Life-Cycle, where it took 21.4 minutes per 2D fragment and 65.5 minutes per 3D fragment. The differences are attributed to the data cataloging times and unrealistic optimisms in the time estimates.

In terms of the measurement systems (mass and size), the processing times also include the time required for cataloging the measurements to the database. The database was initially setup to indirectly store images from the size measurements using dynamic links, however, the database has been updated to directly store images as binary large objects (BLOB) in the database. While the direct storage decreases performances in some cases, it guaranteed data perpetuity of the entire dataset which was a higher priority than a less reliable indirect storage.

Furthermore, the time estimates developed in the pre-implementation were unrealistic since it did not rigorously account for each characterization activity. After the implementation of the Containerized Satellite Mission Life-Cycle, each characterization activity was identified and the processing times for these activities were determined. Improvements were made to the procedures, including hardware and software improvements, to increase productivity and efficiency for each characterization activity.

The assessment per fragment time has improved by over a minute or decreased the time by 23.2%. The percent change is calculated by using Equation 4-2, where the current processing time is denoted as “new” and the previous processing time is denoted as “previous”. A negative percent change represents a decrease and a positive percent change represents an increase, therefore, in this case a negative percent change is a decrease in the processing time and is an improvement. The processing time improvement for the assessment is attributed to the addition of the USB microscopes to aid in assessing smaller fragments that are difficult to physically see.

$$\%\,\mathrm{Change} = 100\left[\frac{new - previous}{previous}\right] \tag{4-2}$$

The mass measurement algorithms were updated to be memory efficient, where redundant variables were removed and variables were stored in data structures. With the updates to the mass measurement algorithm, the mass measurement per fragment time has improved by over two minutes or decreased the time by 32.8%.

For the size measurements (2D and 3D imaging systems), hardware and software updates were made to improve the processing times. For the 2D imaging systems, the acrylic enclosures were developed rather than shrouds that required opening and closing. In addition, the algorithms were updated to be more memory efficient, thus, the processing time improved by approximately one minute or decreased by 8.8%. For the 3D imaging system, the algorithms were updated to be more memory and computationally efficient and as such, the processing time improved by three minutes of decreased by 5.6%.

Prior to the implementation, the processing times were estimates (i.e., assumptions), however, after the implementation, the actual processing times were determined. Furthermore, improvements were made to the procedures, including hardware and software improvements, to increase productivity and efficiency. The processing times are used to better plan the overall project schedule which were poorly estimated prior to the implementation of the Containerized Satellite Mission Life-Cycle.

It is also worth noting that some improvements to the processing times can be attributed to the Pre-Phase I activities; i.e., training of the operators resulted in improved performances since the operators were then knowledgeable of the task(s) to be performed.

In the following sections, the details of the implementation of the Containerized Satellite Mission Life-Cycle to the DebriSat's post-HVI test phase are presented. It should be noted that the discussion that follows, project tasks are equivalent to the satellite operating modes and the sub-tasks are essentially equivalent to the satellite subsystems and components.

Pre-Phase I: Systems Engineering Training

Implementation of the Containerized Satellite Mission Life-Cycle began with Pre-Phase I, where the operators (i.e., students) were trained on systems engineering principles and processes. The benefit of the systems engineering training is for the operators to understand the importance of the end-to-end process since the operators may perform different tasks throughout the post-HVI test phase; i.e., having contributions to different satellite subsystems in the satellite vernacular.

The post-HVI test phase was divided into tasks (e.g., collection, characterization, cataloging, and storage) and each task was further divided into sub-tasks. The operators perform these tasks/sub-tasks and their efforts collectively produce the data necessary to update the current breakup model (i.e., the product). This process is known as the conjunctive group task, where the group members interact with one another and influence one another to produce a product [118] [135]. In a conjunctive group task, the performances of the group members are improved compared to a group member working individually due to what is referred to as the “Kohler effect”. The Kohler effect occurs when the group members exert more effort to avoid being the inferior group member in the conjunctive group task [119]. This motivation gain when working in groups occurred despite the absence of any apparent task-related ability differences among the group members [135].

All the operators in the DebriSat post-HVI phase are undergraduate students that span a wide range of disciplines from non-technical (e.g. liberal arts) to technical (science, mathematics, and various engineering disciplines). Thus, the implementation of Pre-Phase I has been critical for the post-HVI test phase, since the operators have limited knowledge and experiences. All new operators to the DebriSat project are trained on the end-to-end process to ensure that they have an understanding of the importance of the task(s) they perform. An initial assessment of the collected and archived data have shown that the training has been very effective since of the 150,000 fragment entries in the database, only 59 errors (< 0.04%) were identified. However, further training is needed to ensure that the tasks are executed properly.

Phase I: Post-HVI Breakdown

The mission objectives for the post-HVI test phase are to collect, characterize, catalog, and archive fragments from the HVI test. Specifically, the goal is to recover at least 90% of the original mass of the DebriSat test article. The requirements were to

- characterize fragments greater than or equal to 2 mm in length,
- to measure the size characteristics of the fragments within 10% error,
- to archive the data, and
- to have data perpetuity in the archived data.

In Phase I, the mission, the mission objectives, and the requirements were utilized to identify the concept of operation (CONOPS), shown in Figure 4-5. The four tasks (i.e., operating modes) for the CONOPS were identified as: collection, characterization, cataloging, and storage.

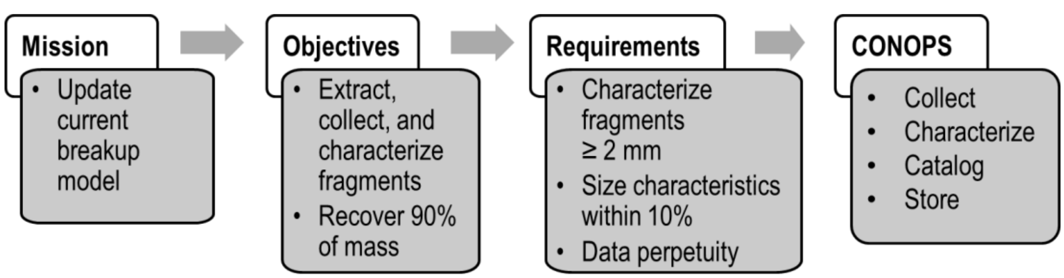


Figure 4-5. Post-HVI breakdown to CONOPS.

Each task from the CONOPS were broken down into sub-tasks and for each sub-task, the subsystems and the components necessary to execute those sub-tasks were identified. The breakdown is shown in Figure 4-6 where a more detailed breakdown of the measurement task showing the subsystems of the measurement systems (i.e., mass and size subsystems) is also shown. After the breakdown, the preliminary designs for each subsystem commenced and were matured in Phase II; development and verification occurred in Phase III, and integration in Phase IV. The 3D imaging system is used as an example to elucidate implementation of the process to the design, development, and verification process of a subsystem.

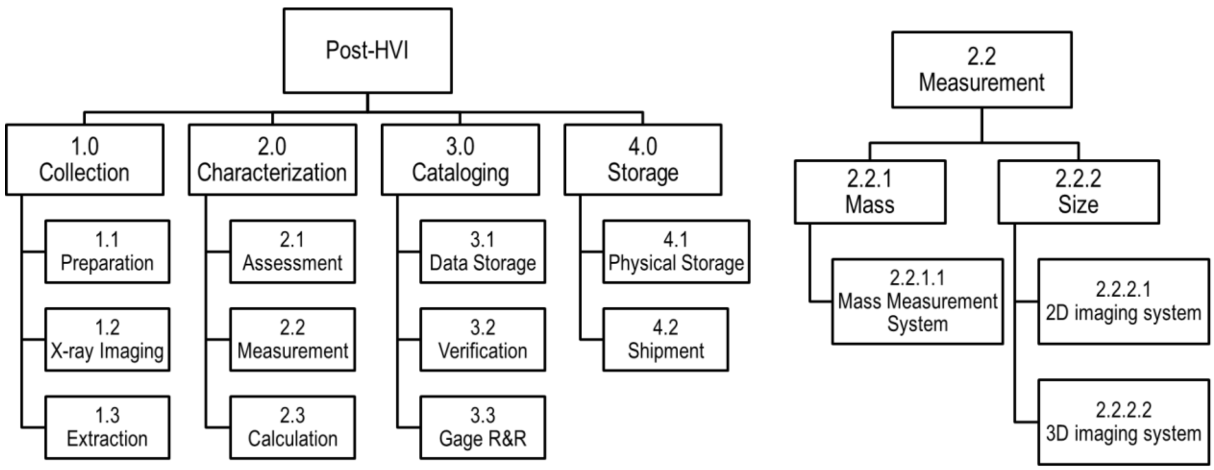


Figure 4-6. Breakdown of CONOPS.

Phase I: 3D Imaging System

One of the key sets of parameters used in updating the breakup model is the size characteristics of the fragments (i.e., characteristic length, average-cross sectional area, volume, and area-to-mass ratio). The requirements for the 3D imaging system were to measure these size characteristics and archive these size characteristics and associated metadata (e.g., images used in the characterization process) in the database. The only accuracy requirement for the measurement tasks was for the characteristic length (L_{C}) to be within a 10% error bound.

As per the process, a trade study was performed to identify the solution path for the size characterization subsystem. While measurement systems are commercially available, factors such as measurement rates, system adaptability, size characterization limitations and equipment costs presented significant challenges to the project and based on these trade studies, a 3D imaging system that utilize a space-carving technique to generate a 3D representation of an object on the turntable was developed (shown in Figure 4-7).

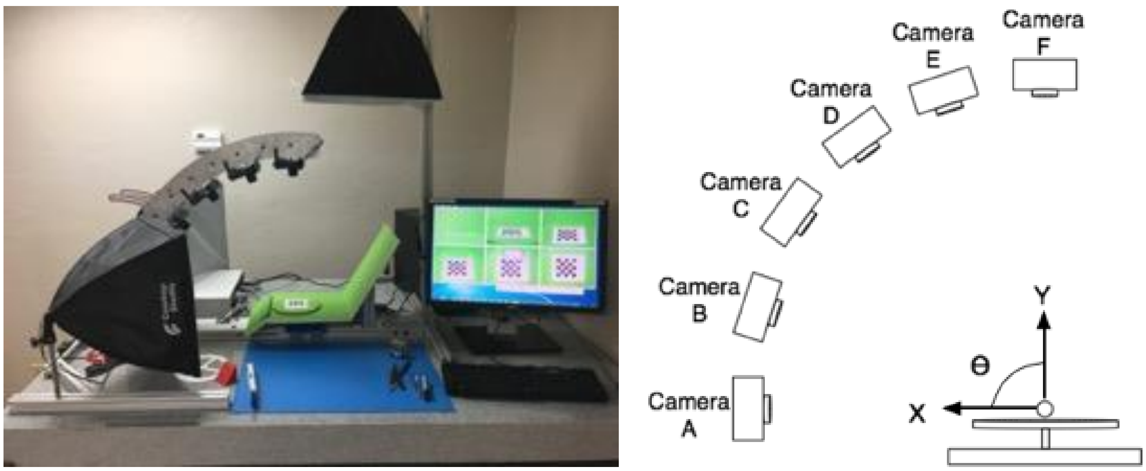


Figure 4-7. 3D imaging system setup. Courtesy of author.

The 3D imaging system was originally developed based on a 4-camera configuration (Camera B through Camera E in Figure 4-7), however, this design did not satisfy the requirement of a 10% L_C error bound. A typical space-carved result from this configuration is shown in Figure 4-8 where the carved object shows “cupping” on the top and bottom. To reduce the cupping effect, additional viewpoints (i.e., cameras) were required, thus, resulted in the addition of a horizontal camera (Camera A) as well as a nadir-looking camera (Camera F). Space-carved images from the 6-camera configuration of the same object are also shown in Figure 4-8 where it can be observed that the cupping has been drastically reduced.

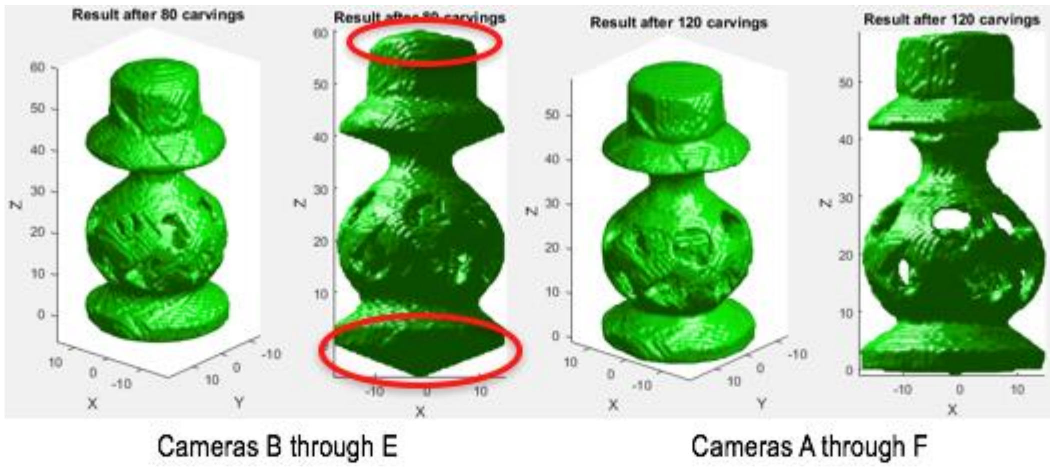


Figure 4-8. Space-carved object with 4-camera setup and 6-camera setup.

To better understand the sources of errors associated with the 3D imager, a quick primer on space carving in presented. Space-carving utilizes a volume intersection approach where it reconstructs a volume by projecting silhouettes (light rays) from corresponding viewpoints and the volume that lies outside are all removed [142] [143]. Figure 4-9 shows an example of the volume intersection approach. Based on the volume intersection, a bounding volume is generated which is then discretized into smaller volumes known as voxels. When the outline/silhouette of the object is projected to the bounding volume, the voxels that contain the outline/silhouette remains and the empty voxels are carved away. The projection and carving iterates through cameras placed around the object and the remaining volume is the output of the space-carving [144] [145] [146]. The space-carving process is shown in Figure 4-10.

The measurement procedure for the 3D imaging system consists of the following activities: camera calibration, image acquisition, image processing, and data storage. In the camera calibration activity, the poses (i.e., orientations and the locations) of the cameras are determined. In image acquisition activity, the images of the object are acquired from various directions. In the image processing activity, the images are processed via a space-carve algorithm to generate a 3D representation of the object of interest from which the size measurements are computed. In data storage, the measurements and the associated images are all archived.

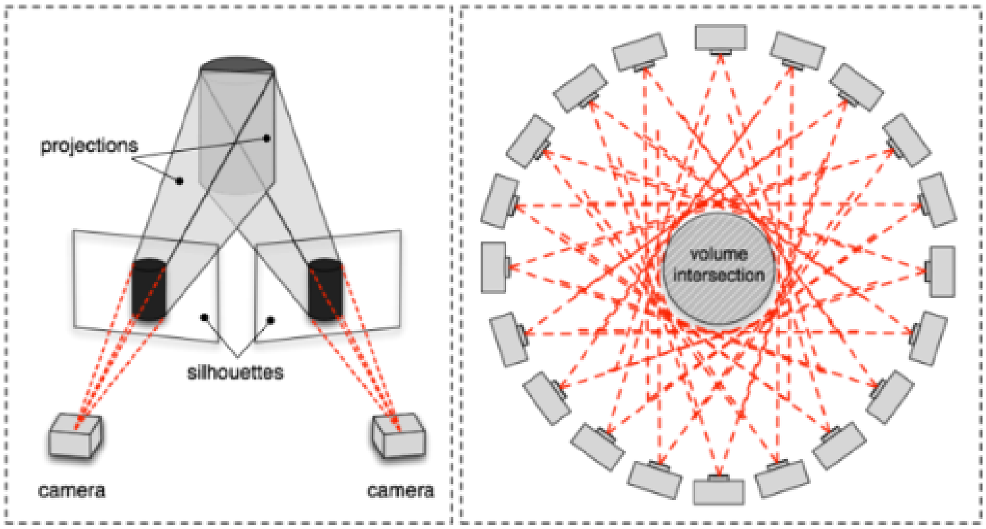


Figure 4-9. Volume intersection approach.

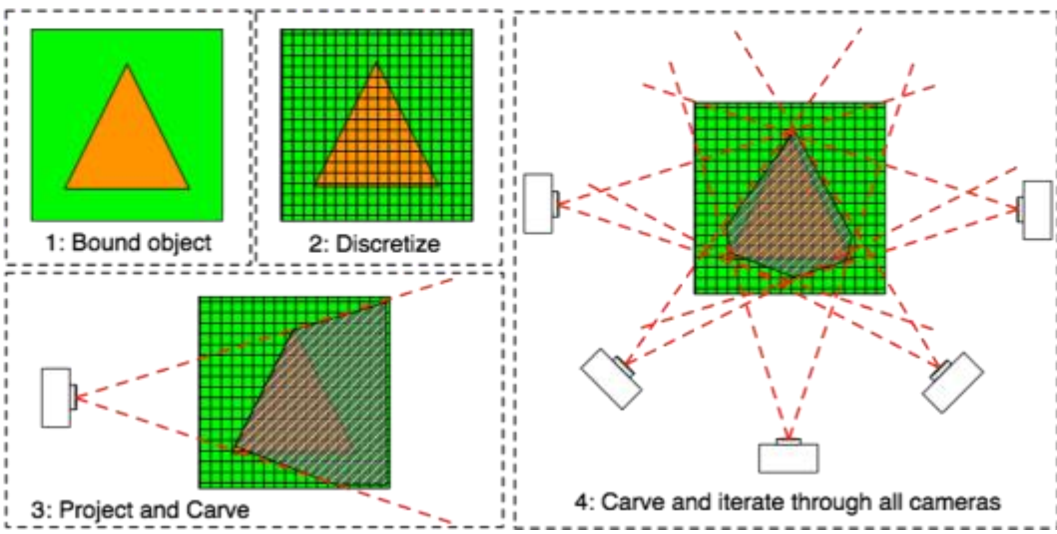


Figure 4-10. Space-carving process.

During the preliminary design, trade studies were performed to identify the best-suited options for camera calibration, image acquisition, image processing (i.e., space-carving), and data storage. For the camera calibration, the Camera Calibration Toolbox developed at the California Institute of Technology (Caltech) by Jean-Yves Bouguet [147] was identified as the best-suited option. For image acquisition, the Canon Hacker's Development Kit (CHDK) [148] was identified as the best-suited option to control the cameras. For image processing, a space-carving algorithm from [149] was determined as the best-suited option to be utilized. For data storage, MATLAB was identified as the best-suited option to archive the measurement data and associated images to the DCS database.

Phase II: Design

Based on the preliminary design, algorithms for the camera calibration, image acquisition, image processing, and data storage were designed for the 3D imaging system. In addition to the software designs, a step-by-step operations plan (i.e., 3D size measurement procedure) was designed that described the size measurement procedure for the 3D fragments.

Phase III: Development and Verification

Based on the designs for the camera calibration, the image acquisition, the space-carving, and data storage, the designs were developed and tested for the 3D imaging system. The image acquisition algorithm was developed and verified first since the camera calibration and the space-carving algorithms both required images. The image acquisition algorithm was verified to ensure that 126 images (6 images at 21 azimuthal positions) were captured. To verify the camera calibration, the space-carving algorithm was needed and to verify the space-carving algorithm, the camera calibration was needed. Thus, once the camera calibration was performed, the calibration output was used in the space-carving algorithm to examine the accuracies. Specifically, the space-carving algorithm that measures the characteristic length and volume were tested by examining the accuracies.

Four different convex shapes were used in characterizing the accuracy of the 3D imaging system: rectangular prisms (referred to as prisms), right circular cones (simply referred to as cones), square pyramids (referred to as pyramids), and spheres. For each shape, three size categories, large, medium, and small were investigated. Representative images and their measured dimensions are shown in Figure 4-11 and Table 4-4, respectively.

For each prism size (large, medium, and small), 10 image sets were acquired and used to calculate the average characteristic length and average volume. The averages were then compared to the “truth” dimensions to compute the percent errors. The “truth” dimensions were determined by physically measuring the objects and using those dimensions to generate point clouds for each object. Based on the generated point clouds, the characteristic length and volume were calculated using a convex hull algorithm for the characteristic length and a built-in MATLAB function called boundary for volume. The percent error is calculated by using Equation 4-1, where the measured is the space-carved output and the actual is the “truth” dimensions.

Table 4-5 shows the average L_C and volume errors, where the Measured column represents the “truth” dimensions and the Space-carved column shows the average L_C and volume measurements. A negative percent error indicates that the space-carved outputs are smaller than the “truth dimensions”, which confirms the characteristic of the volume intersection approach.

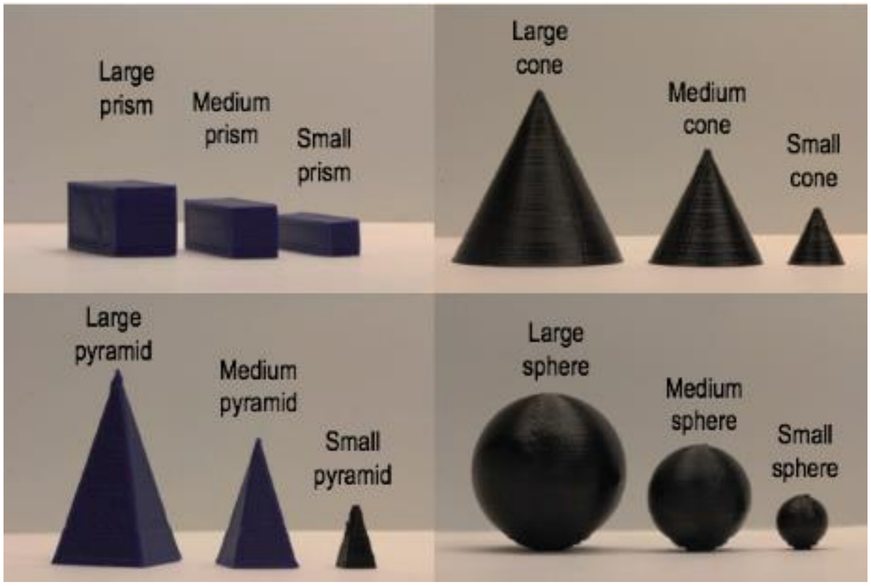


Figure 4-11. Convex objects used in characterizing 3D imaging system. Courtesy of author.

Table 4-4. Dimensions of the convex objects (in mm).

Shape	Large	Medium	Small
Prism	12.57 x 12.73 x 25.31	9.56 x 9.56 x 25.39	9.59 x 6.22 x 25.40
Cone (dia. x height)	29.97 x 29.76	19.92 x 19.74	10.00 x 9.71
Pyramid (planform x height)	15.22 x 15.23 x 29.31	10.15 x 10.11 x 19.33	4.99 x 4.97 x 9.38
Sphere	30.44 x 30.15 x 29.80	20.31 x 20.10 x 19.91	10.78 x 10.03 x 9.96

Table 4-5. Space-carved results of the rectangular prisms with 120 images

Large	Measured	Space-carved	Error (%)
L_C (mm)	26.04	25.11	-3.57
Volume (mm^3)	4048.96	3801.10	-6.12
Medium	Measured	Space-carved	Error (%)
L_C (mm)	21.94	20.77	-5.33
Volume (mm^3)	2277.33	1993.09	-12.48
Small	Measured	Space-carved	Error (%)
L_C (mm)	17.65	16.46	-6.64
Volume (mm^3)	1023.43	817.92	-20.26

The L_C errors for all the prisms were within the 10% error requirement, however, the volume error increased as the prism got smaller. While there are no volume error requirements, the volume errors were considered too high for the DebriSat project, therefore, improvements were needed. The primary source of errors is the camera pose (i.e., camera calibration), thus to improve the volume errors, the cameras had to be re-aligned. In the re-alignment process, new camera mounts were developed to minimize the motion of the cameras after calibration, a major source of errors. It should be noted that the average cross-sectional area (ACSA) errors are not included in these results since the ACSA calculations do not utilize space-carving and the point clouds. The camera re-alignment is shown in Figure 4-12A and the updated camera mounts are shown in Figure 4-12B.

A  B

Figure 4-12. Hardware updates to the 3D imaging system. Courtesy of author.

In addition to the camera re-alignments, the total number of images to be used in the 3D imaging system was reduced. Rather than utilizing 120 images (6 images for each 20 azimuthal positions), the total number of images was reduced to 101 images (1 image, 01 azimuthal position from Camera F rather than 20), thus, the space-carving algorithm was updated.

After the updates, the 3D imaging system had to be re-verified in Phase III to ensure the measurement accuracies were within the requirement. The L_C and volume accuracies were re-examined using the prisms and for each prism, 10 image sets were acquired and used to calculate the L_C and volume averages. Figure 4-13 and Table 4-6 shows the space-carved output of the prisms using 101 images.

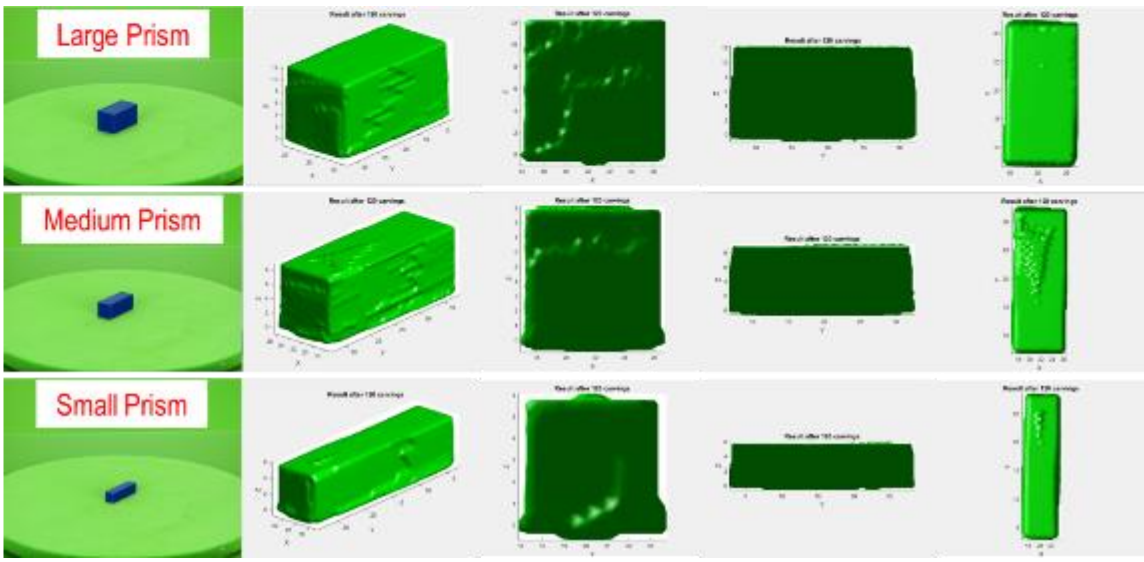


Figure 4-13. Space-carved results of the rectangular prisms with 101 images

Table 4-6. Space-carved outputs of rectangular prisms with 101 images

Large	Measured	Space-carved	Error (%)
L_C (mm)	26.04	25.24	-3.07
Volume (mm^3)	4048.96	3864.13	-4.56
Medium	Measured	Space-carved	Error (%)
L_C (mm)	21.94	21.21	-3.32
Volume (mm^3)	2277.33	2160.05	-5.15
Small	Measured	Space-carved	Error (%)
L_C (mm)	17.65	16.88	-4.36
Volume (mm^3)	1023.43	962.29	-5.97

With the cameras re-aligned and using 101 images, the L_C and volume errors improve significantly. The percent change in the L_C and volume errors from 120 images to 101 images are shown in Table 4-7, where the percent change is calculated using

Equation 4-2. A positive percent change indicates that the space-carved output is bigger with the camera re-alignments and in this case is an improvement. The previous L_C and volume errors were larger (i.e., more negative) than the updated L_C and volume errors due to "over-carving" and indicate that the space-carved output was smaller. Thus, a positive percent change indicates the updated space-carved output is bigger and is an improvement. For L_C, there is an improvement of up to 38% and for volume, the percentage improves by over 70%. Based on these results, it was decided to utilize 101 images on the 3D imaging system for space-carving. The space-carving results shown from here on are all based on 101 images.

Table 4-7. Percent changes/improvements from 120 images to 101 images

	Large	Medium	Small
L_C	14.0 %	37.7 %	34.3 %
Volume	25.5 %	58.7 %	70.5 %

In addition to the rectangular prisms, the other convex shapes (cone, pyramid, and sphere) with three different sizes were used to examine the measurement accuracies on the 3D imaging system. For each object, 10 measurements were taken and the averages were used to compute the L_C, volume, and average cross-sectional area (ACSA) measurement errors (shown in Table 4-8). All of the measurement errors were within 10% for all the objects.

Once the measurement accuracies were all verified, the remaining algorithms (including data storage) were developed and verified. During the verification for the data storage, it was determined that to archive the measurement data and the associated data took over 20 minutes for each fragment. Based on this, the 3D measurement procedure was updated to batch process the images, where the image processing and data storage are conducted overnight. Once each of the 3D measurement process was developed and verified, it was integrated and verified to ensure that the overall 3D measurement process (calibration, image acquisition, image processing, and data storage) were operational.

Table 4-8. Average characteristic length, ACSA, and volume errors for convex shapes.

L_C errors (%)	Large	Medium	Small
Cone	-1.42	-1.32	-2.09
Prism	-3.01	-3.34	-4.55
Pyramid	-4.51	-4.10	-8.68
Sphere	-1.44	-2.35	-2.69
Volume errors (%)	Large	Medium	Small
Cone	-3.60	-4.04	-9.25
Prism	-4.56	-5.15	-5.97
Pyramid	-7.45	-6.81	-7.86
Sphere	-7.02	-7.40	-9.36
ACSA errors (%)	Large	Medium	Small
Cone	4.52	4.62	6.28
Prism	0.15	2.40	2.73
Pyramid	1.04	3.11	7.82
Sphere	0.31	0.49	-2.02

Similar design, development, and verification processes were implemented for the other subsystems and sub-tasks. For each subsystem, the components were developed and verified first. Once the components were verified, the components were integrated into subassemblies/subsystems and verified through testing. The subsystems were then used to verify each sub-task and task in the post-HVI test phase. Once each task of the post-HVI test was verified, the post-HVI test phase progressed into Phase IV.

Phase IV: System Verification

Once each task for the post-HVI test phase were verified, all of them were integrated in Phase IV. To verify the integrated system, a test version of the DCS database was utilized since all of the data generated throughout the post-HVI test phase is archived in the DCS database. The test DCS database is identical to the DCS database but is only used for testing purposes. The integrated tasks were verified and upon verification, the post-HVI test phase progressed into Phase V.

Phase V: Operations

At the time of writing of this document, the DebriSat project is in the operations phase of the post-HVI test. Table 4-9 summarizes the status of the DCS database in terms of the number of debris fragments collected, characterized, cataloged, and stored.

Table 4-9. Fragment count in DCS database as of July 17, 2018.

	Fragment count in database
Collection	151,990
Characterization	26,786
Cataloging and Storage	3,927

As the operations phase continues for the post-HVI test phase, the procedures for each task and sub-task are monitored for productivity and efficiency. For example, it was observed that the camera calibration for the 3D imaging system requires over 2 hours, therefore, automation efforts are on-going to improve the camera calibration time. Another example occurred during fragment extraction. In this case, the tables originally used during extraction were not ergonomically suitable as they required the operators to bend over during the extraction process. New tables with adjustable heights were implemented and subsequent monitoring of the procedures and obtaining feedbacks from the operators the improvements assisted in increased productivity.

Data from the database are regularly sent to the stakeholders and recently, a subject matter expert found database errors during data analyses. Upon investigation, it was determined that the errors were due to operators utilizing the actual database to performing validation testing rather than the test database. The errors were easily identified since the procedures for the post-HVI test were developed in a systematic way by following the Containerized Satellite Mission Life-Cycle and provided traceability. With the source of the error identified, actions are being developed to prevent these errors in the future.

Summary

The Containerized Satellite Mission Life-Cycle presented in Chapter 3 was implemented on the DebriSat project, specifically for the post-HVI test phase. After the implementation of the Containerized Satellite Mission Life-Cycle phases, an end-to-end process for the post-HVI test was developed and validated. It has been found that the training that occurs in Pre-Phase I is necessary since the operators are all undergraduate students (new entrants) with limited experiences and knowledge. The design, development, and verification of the set of procedures to collect, characterize, catalog, and store the debris fragments generated during the laboratory HVI test followed a structured approach as outlined in Phases I through IV. The implementation continues in Phase V as the post-HVI test activities continue.

Since the post-HVI test activities were systematically developed by following the Containerized Satellite Mission Life-Cycle phases, feedback from the operators and stakeholders are easily implemented. The use of the Containerized Satellite Mission Life-Cycle processes to systematically develop the procedures for the post-HVI test phase has been quite instrumental in the identification the source of errors that have occurred to date.

CHAPTER 5
CONCLUSION

Since Sputnik-1 was launched in 1957, thousands of satellites have been launched to space. In recent years, there have been a shift in paradigm, where the small satellites have become more popular due to their shorter development times and lower costs (development and launch). Specifically, within small satellites, the CubeSat class satellites have become extremely well-liked by academia and new space entrants. These CubeSat class satellites are typically launched as secondary payloads and are deployed into orbit through the use of deployment containers. These deployment containers (i) interface one or more of these satellites to the launch vehicle and (ii) prevent harm to the launch vehicle (and other satellites). The advancement of these deployment containers enables these “containerized” satellites to be launched in swarms. As a result, the number of containerized satellites launched to space have significantly increased.

When structured procedures are not utilized in the development of these containerized satellites, they are developed in an ad-hoc manner which may influence the mission success. Structured processes exist for larger traditional monolithic satellites, however, the existing structured processes are overly burdensome and not easily adapted to this class of smaller and compact containerized satellites.

A comprehensive project life-cycle for containerized satellites, referred to as the Containerized Satellite Mission Life-Cycle was developed by leveraging existing project life-cycles and engineering activities performed by the satellite community. The Containerized Satellite Mission Life-Cycle is adaptable and flexible, and can be applied to containerized satellite missions as well as non-satellite missions. Furthermore, since containerized satellite projects in academia are inherently staffed by new students, it is highly recommended that they adopt a structured approach into their programs.

Two application of the Containerized Satellite Mission Life-Cycle were presented, one to SwampSat II (a containerized satellite mission) and the other to the post-HVI test phase of the DebriSat project (a non-mission project). The Containerized Satellite Mission Life-Cycle is being implemented on the SwampSat II mission project. At the time of this writing, the SwampSat II mission project has just completed preliminary design and thus, assessment of the Containerized Satellite Mission Life-Cycle cannot be fully quantified.

Therefore, to assess the efficacy of the Containerized Satellite Mission Life-Cycle, it was applied to the post-HVI test phase of the DebriSat project. Prior to its application, the post-HVI test phase of the project was essentially conducted in an ad-hoc manner. After its implementation, an efficient end-to-end process was developed and validated. With the implementation, where possible, procedures were automated to increase productivity and efficiency while minimizing fragment handling and transcription errors. In some instances, the productivity and efficiency improved by over 30%. Furthermore, through rigorous training (Pre-Phase I), the implementation has enabled the operators (all undergraduate students with various disciplines) with limited experiences and knowledge to learn and execute each post-HVI test activity. The implementation for the DebriSat project continues to monitor and improve the project's productivity and efficiency.

The implementation of the Containerized Satellite Mission Life-Cycle for the SwampSat II mission will continue to be monitored to assess its efficacy throughout the process. As the SwampSat II project progresses and matures, the Containerized Satellite Mission Life-Cycle's success (or failure) will be fully quantified. The structured process enables traceability and based on the outcome of the implementation, improvements to the Containerized Satellite Mission Life-Cycle can be made. It is intended that the Containerized Satellite Mission Life-Cycle will be institutionalized in the University of Florida's future containerized satellite missions.

APPENDIX A
LIST OF CONTAINERIZED SATELLITES

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
1	11/24/2002	STS-113 Endeavour Shuttle	MEPSI 1A	USA	1	SSPL
2			MEPSI 1B	USA	1	SSPL
3	6/30/2003	Eurorockot	QuakeSat	USA	3	P-POD
4			CUTE-1 (Oscar 55)	Japan	1	CSS
5			CubeSat XI-IV (Oscar 57)	Japan	1	T-POD
6			CanX-1	Canada	1	P-POD
7			DTUsat-1	Denmark	1	P-POD
8			AAU CubeSat	Denmark	1	P-POD
9	10/27/2005	Kosmos-3M SSETI Express	CubeSat XI-V (Oscar 58)	Japan	1	T-POD
10			NCUBE-2	Norway	1	T-POD
11			UWE-1	Germany	1	T-POD
12	2/21/2006	JAXA M-V-8	Cute-1.7+APD	Japan	2	CSS
13	7/26/2006	DNEPR-I	SACRED	USA	1	P-POD
14			ION	USA	2	P-POD
15			Rincon 1	USA	1	P-POD
16			ICE Cube 1	USA	1	P-POD
17			KUTESat	USA	1	P-POD
18			NCUBE-1	Norway	1	P-POD
19			HAUSAT-1	S. Korea	1	P-POD
20			SEEDS-1	Japan	1	P-POD
21			CP-2	USA	1	P-POD
22			AeroCube 1	USA	1	P-POD
23			MEROPE	USA	1	P-POD
24			Mea Huaka'I (Voyager)	USA	1	P-POD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
25			ICE Cube 2	USA	1	P-POD
26			CP-1	USA	1	P-POD
27	12/10/2006	STS-116 Discovery Shuttle	MEPSI 2A	USA	1	SSPL
28			MEPSI 2B	USA	1	SSPL
29			RAFT 1	USA	1	SSPL
30			MARScom	USA	1	SSPL
31	12/16/2006	Minotaur-I	GeneSat-1	USA	3	P-POD
32	4/17/2007	DNEPR-II	CSTB1	USA	1	P-POD
33			MAST	USA	3	P-POD
34			AeroCube 2	USA	1	P-POD
35			CP-4	USA	1	P-POD
36			CAPE-1	USA	1	P-POD
37			CP-3	USA	1	P-POD
38			Libertad-1	Colombia	1	P-POD
39	4/28/2008	PSLV-C9	COMPASS-1	Germany	1	X-POD
40			AAUSAT-II	Denmark	1	X-POD
41			Delfi-C3	Netherlands	3	X-POD
42			CanX-2	Canada	3	X-POD
43			SEEDS-2	Japan	1	X-POD
44			CanX-6	Canada	Other	X-POD
45			Cute-1.7 + APD II	Japan	Other	CSS
46	8/3/2008	Falcon-1	PREsat	USA	3	P-POD
47			NanoSail-D	USA	3	P-POD
48	11/15/2008	STS-126 Endeavour Shuttle	PSSC-Testbed 1	USA	2	SSPL
49	5/19/2009	Minotaur-I	PharmaSat	USA	3	P-POD
50			CP-6	USA	1	P-POD
51			AeroCube 3	USA	1	P-POD
52			HawkSat I	USA	1	P-POD
53	7/15/2009	STS-127 Endeavour Shuttle	BEVO 1 (DRAGONSAT 1)	USA	1	SSPL

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
54			AggieSat (DRAGONSAT 2)	USA	1	SSPL
55	9/23/2009	PSLV-C14	SwissCube-1	Switzerland	1	SPL
56			BeeSat	Germany	1	SPL
57			UWE-2	Germany	1	SPL
58			ITU-pSat1	Turkey	1	SPL
59	3/27/2010	Terrier Mk.70 Improved Malemute	ADAMASat	USA	2	P-POD
60			Cal Poly 1U	USA	1	P-POD
61	5/20/2010	JAXA H-IIA	KSAT (Hayato)	Japan	1	J-POD
62			Negai	Japan	1	J-POD
63			Waseda-SAT2	Japan	1	J-POD
64	7/12/2010	PSLV-C15	Tisat-1	Switzerland	1	X-POD
65			StudSat	India	1	Own
66			AISSAT-1	Norway	Other	X-POD
67	11/19/2010	Minotaur-IV	O/OREOS	USA	3	P-POD
68			RAX-1	USA	3	P-POD
69			NanoSail-D2	USA	3	P-POD
70	12/8/2010	Falcon-9	Perseus 000	USA	1.5	P-POD
71			Perseus 001	USA	1.5	P-POD
72			Perseus 002	USA	1.5	P-POD
73			Perseus 003	USA	1.5	P-POD
74			QbX1	USA	3	P-POD
75			QbX2	USA	3	P-POD
76			SMDC-ONE	USA	3	P-POD
77			Mayflower-Caerus	USA	3	P-POD
78	3/4/2011	Taurus XL	KySat-1	USA	1	P-POD
79			Hermes	USA	1	P-POD
80			Explorer-1 (PRIME)	USA	1	P-POD
81	7/8/2011	STS-135 Atlantis Shuttle	PSSC-Testbed 2	USA	2	SSPL

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
82	10/12/2011	PSLV-C18	Jugnu	India	3	Own
83	10/28/2011	Delta-II	AubieSat-1	USA	1	P-POD
84			DICE-1	USA	1.5	P-POD
85			DICE-2	USA	1.5	P-POD
86			Explorer-1 (HRBE)	USA	1	P-POD
87			RAX-2	USA	3	P-POD
88			M-Cubed (COVE)	USA	1	P-POD
89	2/13/2012	ESA Vega	Xatcobeo	Spain	1	P-POD
90			UNICubeSat	Italy	1	P-POD
91			ROBUSTA	France	1	P-POD
92			e-st@r	Italy	1	P-POD
93			Goliat	Romania	1	P-POD
94			PW-Sat	Poland	1	P-POD
95			MaSat-1	Hungary	1	P-POD
96	7/21/2012	JAXA H-IIB to ISS	F-1	Vietnam	1	J-SSOD
97			TechEdSat	USA	1	J-SSOD
98			WE WISH	Japan	1	J-SSOD
99			RAIKO	Japan	2	J-SSOD
100			FITSAT-I (NIWAKA)	Japan	1	J-SSOD
101	9/13/2012	Atlas V NROL-36	SMDC-ONE 2.2 (Baker)	USA	3	P-POD
102			AeroCube 4.0	USA	1	P-POD
103			AeroCube 4.5A	USA	1	P-POD
104			AeroCube 4.5B	USA	1	P-POD
105			Aeneas	USA	3	P-POD
106			CSSWE	USA	3	P-POD
107			CP-5	USA	1	P-POD
108			CXBN	USA	2	P-POD
109			CINEMA	INT	3	P-POD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
110			Re (STARE)	USA	3	P-POD
111			SMDC-ONE 2.1 (Able)	USA	3	P-POD
112	2/25/2013	PSLV-C20	AAUSAT-III	Denmark	1	X-POD
113			STRaND-1	UK	3	ISIPOD
114			UniBRITE (CanX-3A)	Canada	Other	X-POD
115			TugSat-1 (CanX-3B)	Austria	Other	X-POD
116	4/19/2013	Soyuz-2	BeeSat-2	Germany	1	ISIPOD
117			BeeSat-3	Germany	1	ISIPOD
118			Dove-2	USA	3	ISIPOD
119			SOMP	Germany	1	ISIPOD
120			OSSI-1	S. Korea	1	FlyMate
121	4/21/2013	Antares 110 A-ONE	PhoneSat 1.0 (Graham)	USA	1	ISIPOD
122			PhoneSat 1.0 (Bell)	USA	1	ISIPOD
123			PhoneSat 2.0.beta (Alexander)	USA	1	ISIPOD
124			Dove-1	USA	3	ISIPOD
125	4/26/2013	Long March 2D	NEE-01 Pegaso	Ecuador	1	ISIPOD
126			TurkSat-3USat	Turkey	3	ISIPOD
127			CubeBug1	Argentina	2	ISIPOD
128	5/7/2013	ESA Vega	ESTCube-1	Estonia	1	ISIPOD
129	8/3/2013	JAXA H-IIB to ISS	ArduSat-1	USA	1	J-SSOD
130			ArduSat-X	USA	1	J-SSOD
131			PicoDragon	Vietnam	1	J-SSOD
132			TechEdSat-3	USA	3	J-SSOD
133	9/29/2013	Falcon-9	POPACS	USA	3	CSD
134	11/19/2013	Minotaur I	Ho`oponopono-2	USA	3	P-POD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
135			Vermont Lunar	USA	1	P-POD
136			TJ3Sat	USA	1	P-POD
137			KySat-2	USA	1	P-POD
138			NPS-SCAT	USA	1	P-POD
139			CAPE-2	USA	1	P-POD
140			DragonSat-1	USA	1	P-POD
141			Trailblazer	USA	1	P-POD
142			COPPER	USA	1	P-POD
143			SwampSat	USA	1	P-POD
144			ChargerSat-1	USA	1	P-POD
145			PhoneSat 2.4	USA	1	P-POD
146			SENSE SV 1	USA	3	P-POD
147			SENSE SV 2	USA	3	P-POD
148			Firefly	USA	3	P-POD
149			Prometheus 1.1	USA	1.5	NLAS
150			Prometheus 1.2	USA	1.5	NLAS
151			Prometheus 2.1	USA	1.5	NLAS
152			Prometheus 2.2	USA	1.5	NLAS
153			Horus	USA	3	NLAS
154			ORSES	USA	3	NLAS
155			ORS Tech 1	USA	3	NLAS
156			ORS Tech 2	USA	3	NLAS
157			Prometheus 3.1	USA	1.5	NLAS
158			Prometheus 3.2	USA	1.5	NLAS
159			Prometheus 4.1	USA	1.5	NLAS
160			Prometheus 4.2	USA	1.5	NLAS
161			Black Knight	USA	1	P-POD
162	11/21/2013	DNEPR I	VELOX-P2	Singapore	1	ISIPOD
163			NEE-02 Krysaor	Ecuador	1	ISIPOD
164			PUCPSAT-1	Peru	1	PEPPOD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
165			iCUBE-1	Pakistan	2	PEPPOD
166			FUNcube-1	Netherlands	1	ISIPOD
167			Delfi-n3Xt	Netherlands	3	ISIPOD
168			ZACUBE-1	South Africa	1	ISIPOD
169			Dove-3	USA	3	ISIPOD
170			Dove-4	USA	3	PEPPOD
171			Triton 1	Netherlands	3	ISIPOD
172			CINEMA2	INT	3	ISIPOD
173			CINEMA3	INT	3	ISIPOD
174			OPTOS	Spain	3	ISIPOD
175			CubeBug2	Argentina	2	ISIPOD
176			GOMX1	Denmark	2	X-POD
177			HiNCube	Norway	1	ISIPOD
178			HumSat-D	Spain	1	PEPPOD
179			First-MOVE	Germany	1	ISIPOD
180			UWE-3	Germany	1	ISIPOD
181			WNISAT-1	Japan	Other	X-POD
182			BRITE-PL-1 (Lem)	Poland	Other	X-POD
183	12/5/2013	Atlas V	FIREBIRD A	USA	1.5	P-POD
184			FIREBIRD B	USA	1.5	P-POD
185			AeroCube 5A	USA	1.5	P-POD
186			AeroCube 5B	USA	1.5	P-POD
187			ALICE	USA	3	P-POD
188			SNAP-3	USA	3	P-POD
189			M-Cubed-2 (COVE2)	USA	1	P-POD
190			CUNYSAT-1	USA	1	P-POD
191			IPEX	USA	1	P-POD
192			SMDC-ONE (David)	USA	3	P-POD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
193			TacSat-6	USA	3	P-POD
194			SMDC-ONE (Charlie)	USA	3	P-POD
195	1/9/2014	Antares 120 to ISS	SkyCube	USA	1	NRCSD
196			UAPSAT	Peru	1	NRCSD
197			ArduSat-2	USA	2	NRCSD
198-225			Flock-1	USA	3 (x28)	NRCSD
226			LitSat 1	Lithuania	1	NRCSD
227			LituanicaSat 1	Lithuania	1	NRCSD
228	2/5/2014	Soyuz-U to ISS	Chasqui 1	Peru	1	Hand
229	2/27/2014	JAXA H-IIA	OPUSAT	Japan	1	J-POD
230			ITF-1 (Yui)	Japan	1	J-POD
231			INVADER (Artsat-1)	Japan	1	J-POD
232			KSAT2 (Hayato-2)	Japan	1	J-POD
233	4/18/2014	Falcon-9	SporeSat	USA	3	P-POD
234			TSAT	USA	2	P-POD
235			PhoneSat 2.5	USA	1	P-POD
236			ALL-STAR/THEIA	USA	3	P-POD
237			KickSat	USA	3	P-POD
238	6/19/2014	DNEPR	ANTELSAT	Uruguay	2	P-POD
239			AeroCube 6	USA	1	P-POD
240			LEMUR-1	USA	3	P-POD
241			DTUSat-2	Denmark	1	QuadPack
242			Duchifat-1	Israel	1	QuadPack
243-253			Flock-1C	USA	3 (x11)	QuadPack
254			NanoSatC-Br 1	Brazil	1	QuadPack
255			PACE	Taiwan	2	QuadPack

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
256			Perseus-M1	Russia	6	QuadPack
257			Perseus-M2	Russia	6	QuadPack
258			PolyITAN	Ukraine	1	QuadPack
259			POPSAT-HIP 1	Singapore	3	QuadPack
260			QB50P1 (E0 79)	INT	2	QuadPack
261			QB50P2 (E0 80)	INT	2	QuadPack
262			Tigrisat	Iraq	3	P-POD
263			BRITE-CA-1	Canada	Other	X-POD
264			BRITE-CA-2	Canada	Other	X-POD
265	6/30/2014	PSLV-C23	VELOX-1	Singapore	3	Own
266			CanX-4	Canada	Other	X-POD
267			CanX-5	Canada	Other	X-POD
268	7/8/2014	Soyuz-2-1b	UKube1	UK	3	P-POD
269			AISSAT-2	Norway	Other	X-POD
270-297	7/13/2014	Antares 120 to ISS	Flock-1B	USA	3 (x28)	NRCSD
298			MicroMAS	USA	3	NRCSD
299			TEchEDSat-4	USA	3	NRCSD
300			Lambdasat	Greece	1	NRCSD
301			GEARRSAT	USA	3	NRCSD
302	8/19/2014	Long March 4B	BRITE-PL-2 (Heweliusz)	Poland	Other	Dragon
303-328	10/28/2014	Antares-130 to ISS	Flock-1D	USA	3 (x26)	NRCSD
329			Arkyd-3	USA	3	NRCSD
330			RACE	USA	3	NRCSD
331			GOMX2	Denmark	2	NRCSD
332	1/10/2015	Falcon-9 to ISS	Flock-1D A	USA	3	NRCSD
333			Flock-1D B	USA	3	NRCSD
334			AESP-14	Brazil	1	NRCSD

Index	Launch Date (UTC)	Launch Vehicle	Satellite Name	Country	Size (U)	Dispenser
335	1/31/2015	Delta-7320	ExoCube (CP10)	USA	3	P-POD
336			GRIFEX	USA	3	P-POD
337			FIREBIRD 3A	USA	1.5	P-POD
338			FIREBIRD 3B	USA	1.5	P-POD
339-352	4/4/2015	Falcon-9 to ISS	Flock-1E	USA	3 (x14)	NRCSD
353			Arkyd-3 Reflight	USA	3	NRCSD
354			Centennial 1	USA	1	NRCSD
355	5/20/2015	Atlas-5	GEARRSAT 2	USA	3	NPSCuL
356			LightSail A	USA	3	NPSCuL
357			OptiCube1	USA	3	NPSCuL
358			OptiCube2	USA	3	NPSCuL
359			OptiCube3	USA	3	NPSCuL
360			USS Langley	USA	3	NPSCuL
361			AeroCube 8A	USA	1.5	NPSCuL
362			AeroCube 8B	USA	1.5	NPSCuL
363			BRICSAT-P (Psat B)	USA	1.5	NPSCuL
364			Psat A	USA	1.5	NPSCuL
365-372	6/28/2015	Falcon-9 to ISS	Flock-1F	USA	3 (x8)	NRCSD
373	7/10/2015	PSLV-XL	DeorbitSail	UK	3	ISIPOD

APPENDIX B
SURVEY QUESTIONS

The flow of the survey is shown in Figure B-1 and all the survey questions are shown in Table B-1 and Table B-2.

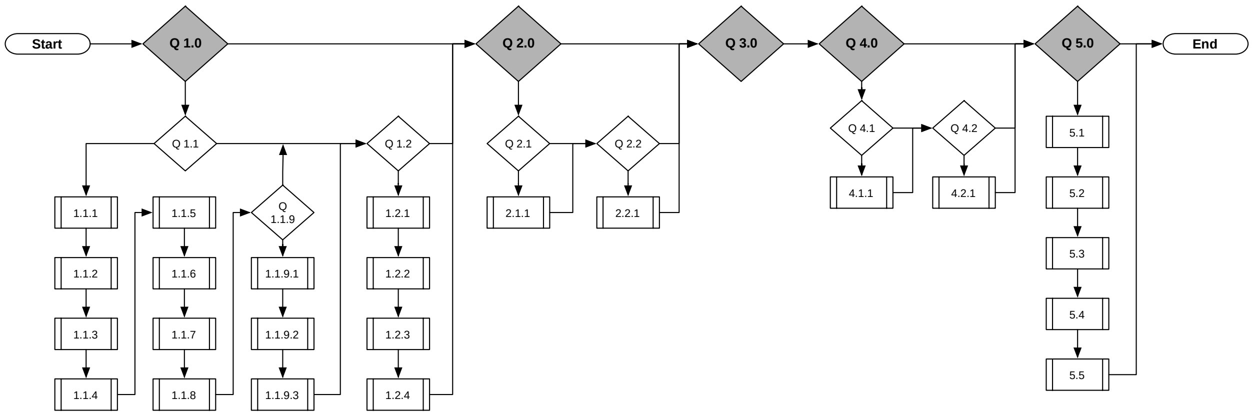


Figure B-1. Survey question flow chart

Table B-1. Survey questions from Section 1

Number	Question	Number	Question
1	Is your organization a designer, developer, or manufacturer of containerized satellites?	1.1	Have any of your organization's containerized satellites been launched?
1.1.1	What container(s) have you deployed your satellite(s) from? Select all that apply.	1.1.2	Please note the month and year of the first and the most recent launches of your organization's satellite(s). If there was only one launch, please note either row.
1.1.3	Please indicate the number of satellite(s) that fall within the mass ranges below.	1.1.4	Please indicate the form-factor(s)/size(s) of your satellite(s). Use Other box(es) to define non-CubeSat based satellite(s).
1.1.5	Does your containerized satellite(s) have any deployable appendages?	1.1.6	What are the perigee altitude(s) of your launched containerized satellite(s)?
1.1.7	Please identify the status of each of your launched containerized satellite(s).	1.1.8	On average, how frequently does your organization actively track its own satellite(s)?
1.1.9	Has your organization ever received a conjunction notification?	1.1.9.1	From whom?
1.1.9.2	How many conjunction notifications have your organization received in total?	1.1.9.3	Does your organization have procedure(s) in the event that a conjunction notification is received? If Yes, please briefly describe the procedure(s).
1.2	Is your organization planning to have containerized satellite(s) launched within the next two years (e.g., before the first quarter of the 2017 calendar year).	1.2.1	Please indicate the number of satellite(s) planned for launch that fall within the mass ranges below.
1.2.2	Please indicate the form-factor(s)/size of your satellite(s). Use Other box(es) to define non-CubeSat based satellite(s)	1.2.3	Will any of those containerized satellite(s) have any deployable appendages?
1.2.4	What are the intended perigee altitude(s) of your containerized satellite(s)?		

Table B-2. Survey questions from Section 2 through Section 5

Number	Question	Number	Question
2	The “25 Year Rule” requests that satellites in LEO should de-orbit within 25 years of the completion of their experiment or mission. Is your organization familiar with this rule?	2.1	Does your organization have a procedure in place to be in compliance with the "25 Year Rule"? If Yes, please describe the procedure.
2.2	In your opinion, should the 25 year post-operational period be changed for containerized satellites? If Yes, how long should it be changed to?		
3	Please select the system engineering processes that your organization has conducted to assure mission success. Select all that apply, and briefly describe the details. If None, please select Other and enter None.		
4	Does your organization use Commercial Off-The-Shelf (COTS) components in its satellite(s)? COTS refers to items that are available in the commercial market, which may or may not be designed for space usage.	4.1	Does your organization have procedure(s) in place to verify COTS components for quality assurance (i.e. metrology)? If Yes, please briefly describe the procedure(s).
4.1.1	What percentage (by component quantity) of your organization's satellite(s) (launched or otherwise) are COTS components? Please note the number of satellite(s) that fall within each of the ranges provided.	4.2	Did your organization manufacture any components in-house (e.g., PCBs, machining, etc)? If Yes, please briefly describe.
4.2.1	What percentage (by component quantity) of your organization's satellites (launched or otherwise) components are manufactured in-house? Please note the number of satellites that fall within each of the ranges provided.		
5.1	What is your organization's affiliation?	5.2	What mission(s) are your satellite(s) designed for?
5.3	How much does your organization assess that its satellites cost for each mass range?	5.4	Which of the following factors are included in the overall cost?
5.5	Please enter your contact information (if applicable)		

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BIOGRAPHICAL SKETCH

Bungo Shiotani was born in Osaka, Japan and moved to Hong Kong where he first learned English. He moved back to Japan and graduated high school from Marist Brothers International School in Kobe, Japan. After graduating, he moved to Florida and attended the Jacksonville University. He enrolled in a 5-year dual degree Bachelor of Science engineering program where the first 3 years was at Jacksonville University and the latter 2 years at the University of Florida. He earned a Bachelor of Science in aerospace engineering from the University of Florida and a Bachelor of Science in engineering physics from the Jacksonville University. After taking a year off, Bungo joined Dr. Fitz-Coy and his Space Systems Group to pursue a Master of Science in aerospace engineering. For his master's research, Bungo performed reliability analyses on SwampSat to identify potential failures and documented his work in this thesis, Reliability Analysis of SwampSat. Upon obtaining a Master of Science, he continued his academic career as a doctoral student with Dr. Fitz-Coy and successfully launched SwampSat into space in November 2013. During his doctoral studies, Bungo went to a 3-month internship to NESTRA in Tokyo, Japan, where he developed three microsatellites, UNIFORM-1, Hodoyoshi-3, and Hodoyoshi-4. All three microsatellites have successfully launched into space in 2014. After his return from the internship, Bungo has obtained a Systems Engineering Certificate from the University of Florida and has contributed to other small satellite projects such as SwampSat II and DebriSat. Bungo graduated with a Doctor of Philosophy in aerospace engineering from the University of Florida in the summer of 2018.