DISSERTATION

A STANDARDIZED 3U CUBESAT COMMON BUS ARCHITECTURE FOR IMPROVING MISSION SUCCESS

Submitted by

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ABSTRACT

A STANDARDIZED 3U CUBESAT COMMON BUS ARCHITECTURE FOR IMPROVING MISSION SUCCESS

CubeSats are a class of nanosatellites that use a standardized form factor called “one unit” (1U). Each 1U measures 10 × 10 × 10 cm and enables scalable configurations. Common CubeSat sizes include 1U, 1.5U, 2U, 3U, and 6U; larger variants up to 12U and 27U. The small form factor of CubeSats enables low-cost science missions and on-orbit technology demonstrations. However, CubeSats have historically exhibited high failure rates and are therefore not viewed as reliable platforms for high-value science missions. This research examines whether the CubeSat failure rate can be reduced through systems engineering with the development of a 3U common bus. The 3U common bus is characterized, designed, tested, and integrated into the CubeSat design process to improve performance and enable higher reliability.

In this research, past CubeSat missions were examined to determine whether identifiable trends exist in subsystem failures, including failure mechanisms and contributing factors. Following a systems engineering methodology, each CubeSat failure was examined to determine whether deficiencies in verification and validation (V&V) and integration and test (I&T) activities prior to launch are associated with observed failures. A set of requirements was derived based on the findings, and a notional 3U common CubeSat bus design is proposed with the goal of increasing mission success and reducing subsystem failures. A baseline schedule was developed for a standard CubeSat systems engineering development process and compared to the schedule for a 3U common bus-enabled CubeSat.

The results demonstrate that CubeSat failures are concentrated within the electrical power system (EPS), communications system (COM), and on-board computer (OBC), which together account for the majority of reported on-orbit failures. Reported failure mechanisms include battery thermal degradation, solar array deployment faults, antenna deployment failures, and radiation-induced single-event upsets. These subsystem failures are strongly correlated with deficiencies in V&V and environmental testing, with as many as 65% of CubeSat missions forgoing at least one of the following vibration testing, thermal vacuum testing, or end-to-end RF testing due to schedule and resource constraints. Collectively, these findings demonstrate that CubeSat failures are driven less by inherent subsystem limitations and more by systemic gaps in systems engineering discipline, integration practices, and V&V execution gaps that a standardized, pre-qualified 3U common bus architecture is designed to address.

The development of a 3U common bus provides stakeholders with tools and methods for accelerating development while improving reliability. The practical implication is that CubeSat development can transition from bespoke bus integration to a payload-centric model, improving reliability and mission value without proportionally increasing cost. This dissertation argues that early CubeSat failures concentrate in a small set of subsystems (EPS/COM/OBC) in large part because schedule pressure compresses integration, verification, and end-to-end testing. A standardized, pre-qualified 3U common bus is proposed as the intervention to reduce custom integration variability and return schedule margin. The measurable outputs are increased testing time (environmental, functional, and RF) and reduced late-cycle integration cycles, which together are expected to reduce early on-orbit failure rates and improve mission success.

ACKNOWLEDGEMENTS

This research was made possible by the guidance, support, and encouragement of several people who helped me see it through. I am especially grateful to my advisor, Dr. Bradley, whose insight, patience, and steady direction shaped this work from the beginning to the end. His guidance and commitment to my development as a researcher were critical to completing this dissertation.

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Finally, I want to thank my family and friends for their unwavering support. Their patience, understanding, and belief in me sustained me through the most difficult parts of this process and helped me keep my sanity. This dissertation would not have been possible without them.

DEDICATION

I dedicate this dissertation to my parents, who never stood in my way and always gave me what I needed to explore my passions. I also dedicate it to my family as a whole to my Aunt Bonnie, who told me I was smart (sometimes to a fault); to my Uncle Art, who helped me through a rough patch; to my Aunt Sandara and Uncle George, who were there for me as well; and to my Uncle Jon, who encouraged me to pursue my first master's degree thank you all. To my partner, thank you for your patience, love, and steady support through the long nights and hard weeks. To my friends, thank you for showing up when it mattered most, especially Dr. Josh Kailn, who pushed me forward when I was ready to give up. And to my cats Henry, Parker, Sara, and Jenny whose uncredited contributions included walking across my keyboard at critical moments and keeping me entertained. Meow, meow.

Ad astrea per aspera.

DISCLAIMER

This dissertation was prepared by Jeremiah Gayle in his personal capacity as a doctoral student at Colorado State University. The views, analyses, and conclusions expressed are solely those of the author and do not represent the views of Colorado State University or any current or former employer. No employer contributed to, directed, reviewed, or funded this research, nor were any employer resources used in its preparation. This work was conducted independently and does not reflect the official policies, positions, or endorsements of any organization. Any errors, omissions, or interpretations contained herein are the responsibility of the author.

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CHAPTER 1: INTRODUCTION

Although CubeSats reduce barriers to entry through standardized form factors and relatively low launch costs, they may still experience extended development timelines and elevated risk of premature failure, particularly when systems engineering rigor and verification practices are insufficient. This research proposes and assesses a standardized 3U CubeSat common bus architecture intended to improve mission reliability, reduce schedule duration, and decrease the cost and risk of successfully delivering 3U payloads to orbit.

The 3U CubeSat will be considered here with a generic systems design and development process. The goal will be to apply systems engineering methodology to develop a set of requirements that will drive the design of the 3U common bus as a system. Past CubeSat missions will be examined to baseline the most common system architectures used in flown CubeSats. The payloads flown on past CubeSat missions will also be examined to baseline the size weight and power metrics (SWaP). Based on the baseline SWaP information, a set of interface control drawings will be created for the 3U common bus. The requirements will not only drive the design of the 3U common bus they will also dictate the verification and validation (V&V) and integration and testing (I&T) methods used to verify the system. Proving out the application of system engineering to derive requirements, design, and the test flow for a 3U common bus will serve as a proof of concept for larger applications. So, this methodology can be applied to the development of a SmallSat common bus in the near term, and larger spacecraft in the long-term.

1.1 Contributions of This Work

This dissertation makes four primary contributions to CubeSat systems engineering. First, it compiles a CubeSat failure dataset and organizes it into a systems-level failure taxonomy that links subsystem failures particularly in EPS, COM, and OBC to underlying systems engineering I&T and V&V issues. Second, it derives a failure and stakeholder informed requirements baseline for a 3U common bus intended to host a 3U payload which together create a 6U CubeSat. Third, it develops a schedule framework that compares bespoke CubeSat development to a common bus approach across the NASA systems engineering lifecycle, showing how bus common bus can reduce non-recurring engineering burden and return schedule to I&T and V&V activates. Finally, it defines a notional 3U common bus architecture and associated test campaign that together provide a reference template for implementing a standardized flight-ready 3U common bus which is ready to integrate with a 3U payload to form a 6U CubeSat.

1.2 Structure of This Dissertation

This dissertation and accompanying research are organized to flow from background to motivation and context to design, verification, and testing. Chapter 2 will provide background on CubeSats, and the subsystems typically found in them. Chapter 3 will introduce the problem statement and research questions and touch on the concept of a standardized 3U common CubeSat bus. Chapter 4 surveys relevant literature and practices including CubeSat standards, CubeSat mission failures, lessons learned, and systems engineering that frame the proposed approach. Chapter 5 will cover the details of the research methodology including systems engineering process, requirements development, and methods used to assess the reliability and risk.

Chapter 6 will present the notional design of the 3U common bus and will document key design features and rationale. Chapter 6 serves to validate the approach through analysis and applications, and report results against the stated research questions. Chapter 6 will go on to discuss findings and limitations. Chapter 7 concludes the dissertation and will present research contributions and propose direction for future work including broader standardization and flight demonstration.

CHAPTER 2: BACKGROUND

2.1 Motivation

At present, CubeSats are typically classified as Class D or Class E missions, which are more risk-tolerant and therefore more failure-prone than the more expensive and less risk-tolerant Class A, B, and C missions, as defined in NASA's program and project management requirements (NASA Procedural Requirements NPR 7120.5, National Aeronautics and Space Administration, 2004; Risk classification for NASA payloads NPR 8705.4, National Aeronautics and Space Administration, 2004).

Figure 1. The Europa Clipper spacecraft during testing. Europa Clipper is a NASA Class A flagship planetary science mission, representing the most conservative risk posture in NASA's mission classification framework. *Source: NASA/JPL-Caltech (public domain)*

Class A through C missions are far less risk tolerant and require more mission assurance, part reliability verification, screening of parts / systems, and subsystems and system level testing (Risk classification for NASA payloads NPR 8705.4, 2004). For example, class A through C missions require engineering models to be developed and tested prior to building flight models. Classification of CubeSats missions as Class D or E allows for simpler designs and a reduction in reliability and quality assurance measures to be used as well as limited prototyping and testing via engineering models. While simplifications of the systems engineering and testing process translates into lower costs, it also translates into a high failure rate for CubeSats (Langer et al., 2016).

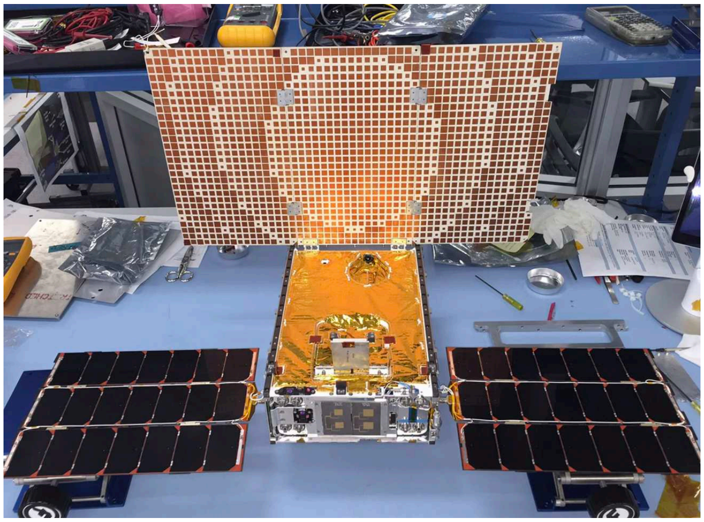

Figure 2. MarCO (Mars Cube One) was a pair of 6U CubeSats and the first to operate beyond Earth orbit. Launched with NASA's InSight mission, MarCO served as a real-time communications relay during Mars entry, descent, and landing, transmitting data back to Earth and demonstrating the viability of deep-space CubeSats. *Source: NASA/JPL-Caltech (public domain)*

A typical CubeSat shown in Figure 3 is a single integrated system-of-systems designed to enable the payload to perform its mission. At present time, the most common size for a CubeSat is 3U, measuring 10 x 10 x 34.05 cm (NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017) and weighs approximately 4,500 grams (NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017). The primary subsystems of the CubeSat are the avionics and the payload. The avionics comprise the following subsystems:

- Command and Data handling system (C&DHS)
- Electrical Power System (EPS)
- Communication System (COM)
- Attitude Determination and Control System (ADCS)
- Antenna for Uplink/Downlink

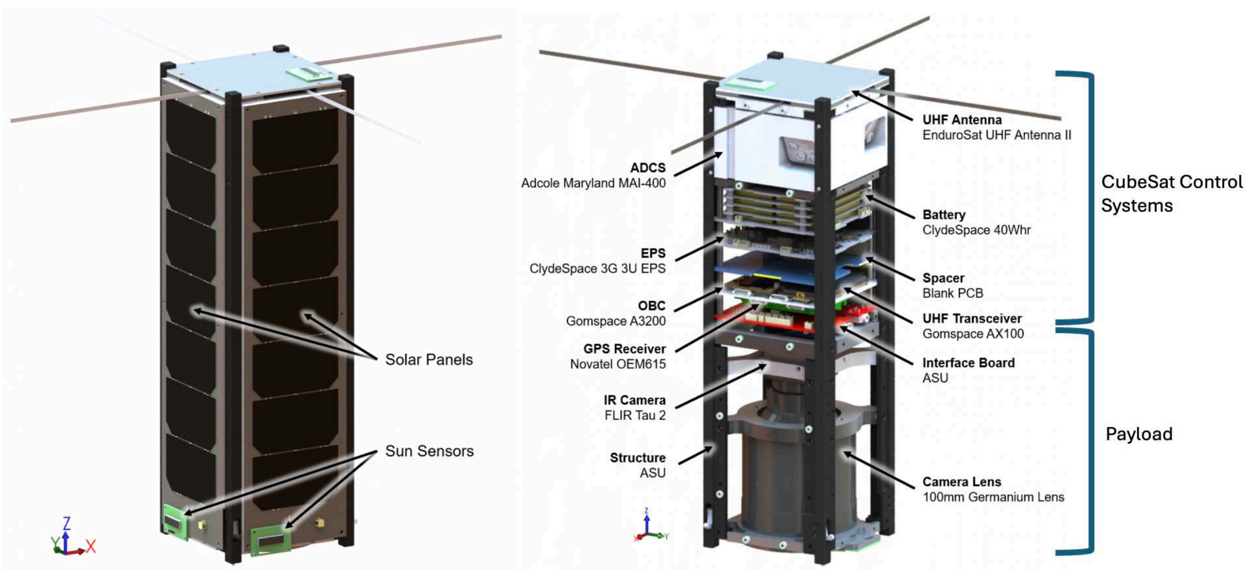

Figure 3. Typical 3U CubeSat configuration. Right: Exterior of the 3U CubeSat showing solar panels, sun sensor, and antenna. Left: Interior of 3U CubeSat showing avionics subsystems and payload. Source: ASU Phoenix CubeSat: https://phxcubesat.asu.edu

These subsystems work to support the payload the CubeSat carries and are referred to as the CubeSat Bus. The subsystems the bus is comprised of work together to provide a stable platform for the payload being flown and allow the payload to conduct its mission on-orbit.

CubeSats are being used by commercial and government operators in a wide range of applications from Earth observation to interplanetary telecommunications relay. CubeSats are being used by universities as academic tools to teach students about the intricacies of spacecraft development. Some universities work in partnership with NASA to fly new science instruments still under development. CubeSats are also used to fly low-Technology Readiness Level (TLR) systems on technology development flights. These technology development flights work to validate new technology on-orbit and to raise the TRL of the system. Maturing the system to a point where risk is reduced enough to allow it to fly on more less risk tolerant missions. In 2018, Mars CubeSat One (MarCO) (shown in Figure 2) flew with the NASA/JPL Mars Insight lander. MarCO's mission was to provide real-time communication with Insight during entry, decent, and landing (EDL). This mission marked the first time a CubeSat has been sent to another planet. MarCO helped prove that CubeSats can be reliably used outside of Earth orbit. The Artemis project has reignited national interest in the moon among the scientific community. This has spurred the development of several CubeSat missions that will be sent to the Moon to conduct science operations (e.g., NASA's Artemis CubeSats and upcoming lunar CubeSat orbiters and probes; CAPSTONE, LunIR, and LUMIO).

CubeSat development is currently in a transitional phase. This inflection point suggests that reliability is strongly influenced by the degree of investment in rigorous design, verification, and mission assurance, and that well-resourced CubeSat missions can achieve high reliability. However, CubeSats are still considered a learning tool for universities this research asserts that a solution to this problem can be devised through improved systems engineering processes. It is only in the last 5-10 years that CubeSats have started to garner the attention of NASA as a method for conducting science and exploration missions at lower cost. The primary technical constraint hindering more widespread use of CubeSats is their low reliability (Swartwout, 2016). The space industry is undergoing a transition from monolithic, billion-dollar spacecraft platform toward more agile, lower-cost, and increasingly reliable platforms such as CubeSats. These systems enable new mission concepts and provide opportunities for on-orbit technology demonstration at reduced cost, thereby accelerating technology maturation.

CHAPTER 3: RESEARCH AGENDA

This chapter defines the research problems that will be addressed in this dissertation and presents a set of research questions and tasks. Building upon the understanding of CubeSats discussed in Chapter 1, the goal of this section is to formalize the direction of the research toward improving the reliability and overall performance of CubeSats through the application and assessment of a 3U common bus architecture

3.1 Problem Statement

Even with major advances in CubeSat technology, CubeSats still continue to experience a much higher failure rate than other NASA Earth and Planetary missions. Many of the CubeSats fail either partially or completely before achieving their mission objectives. This reduces the funders', designers', and other stakeholders' confidence in CubeSats as a reliable platform for science research and technology development. Reducing failure CubeSat failure rates requires improvements to the systems engineering process used to develop and test CubeSats specifically focusing on I&T and V&V activities.

Currently most CubeSats teams design and build both the CubeSat bus and the payload simultaneously. This practice increases schedule risk, resource strain, and integration complexity. If the CubeSat bus and payload design activities, were separated and the CubeSat bus standardized via a flight qualified 3U common bus this research hypothesizes that the reliability could be improved nonrecurring engineering (NRE) costs could decline, and more schedule margin could be allocated for V&V activities and I&T activities.

This dissertation will address the following central question: How can CubeSats reliability be improved through systems engineering process enhancements and the implementation of a 3U common bus?

3.2 Research Focus

The focus of this research is improving CubeSat success by focusing on the systems engineering process with an emphasis on I&T and V&V as well as developing a notional 3U common bus. The research will evaluate whether separating the CubeSat bus development from the payload development can reduce failures by increasing time and effort available for verification and testing activities. The outcome will be a schedule-based and risk-based justification for using a 3U common bus.

3.3 Research Questions

To address the stated problem this dissertation seeks to answer three main research questions.

3.3.1 Research Question 1

3.3.1.1 What is the Current State of CubeSat Mission Systems Engineering?

CubeSats consistently demonstrate higher failure rates than larger NASA and commercial spacecraft. Understanding the underlying causes of these failures is essential for improving mission outcomes.

3.3.1.2 Literature Review and Failure-Mode Analysis

To answer this question, a comprehensive literature review will be conducted using academic, industry, and NASA sources. The goal is to identify the most common CubeSat failure modes and their underlying causes and to determine which subsystems fail most frequently. This analysis will provide a baseline understanding of current CubeSat reliability and of how systems-engineering practices affect mission success.

3.3.2 Research Question 2

3.3.2.1 Can Risk Be Reduced, and Schedule Margin Increased by Separating the CubeSat Bus Engineering From the Payload Engineering?

This research hypothesizes that operational risk can be reduced by standardizing the CubeSat bus by leveraging a flight proven platform with known characteristics (Shannon, 2020) Using a vetted bus would reduce V&V and I&T testing down to acceptance qualification testing and eliminate NRE related to the CubeSat bus. This would free up engineers to focus on payload design, I&T, V&V, and testing activities. Additionally, time would be saved since the engineers would not need to focus on bus level systems engineering and design activates (Shannon, 2020). The time saved could be reallocated to the back end of the schedule as margin, allowing for more time to conduct the associated I&T and V&V activities integrating the payload with the bus. Integrating the payload with the bus would be simplified through a standardized mechanical and electrical mating architecture allowing for schedule savings.

3.3.2.2 Schedule

To answer this question, a baseline schedule for the 3U common bus will be developed so that a comparison between the standard CubeSat and 3U common bus schedule can be conducted.

3.3.2.3 Baseline Schedule Development

A baseline schedule for CubeSat including the bus and payload will be constructed using representative missions such as Temporal Experiment for Storms and Tropical Systems – Demonstration (TEMPEST-D), Mars Cube One (MARCO), and the Arcsecond Space Telescope Enabling Research in Astrophysics (ASTERIA). The schedule will capture typical durations from Pre-Phase A to Phase D forming a benchmark against which the 3U common bus can be evaluated.

3.3.2.4 Common-Bus Schedule and Risk Comparison

A second schedule will be developed for the 3U common bus. The difference in total duration between the standard CubeSat and the 3U common bus will be used to evaluate schedule gains focusing on I&T and V&V activities. The corresponding reduction in technical risk gained from more time allocated to I&T and V&V activities will be evaluated. The results will be evaluated to see if they justify the creation of a 3U common bus to provide a more reliable CubeSat architecture.

3.3.3 Research Question 3

3.3.3.1 Can a CubeSat 3U Common Bus Design and Improved Systems Engineering Process Increase the Success Rate of a Low Earth Orbit (LEO) CubeSat Mission?

A standardized 3U common bus is proposed using state of the art technology. The goal is to offer a commercial off the shelf (COTS) solution that will have known reliability and be flight qualified. Unlike the current CubeSat development method, the 3U common bus is intended to support multiple mission types through a standardized 3U common bus using standardized mechanical and electrical interfaces to accommodate a 3U payload. The 3U common bus with be verified and tested prior to delivery and adhere to strict quality assurance (QA) and V&V procedures before delivery for integration of the 3U payload.

This research asserts that divorcing the bus design from the payload design would allow the program user to focus on the development of the payload and V&V activities needed to flight qualify the payload. No longer would a program have to be concerned with both the bus and payload systems engineering, design, and fabrication.

3.3.3.2 Notional Bus Architecture Development

To answer this question, a notional 3U common bus architecture will be developed based on derived requirements. A preliminary mechanical and electrical interface control document (MICD/EICD) and CAD model will be created to show the proposed configuration of the 3U common bus.

3.3.3.3 Integration and Verification Planning

A notional I&T and V&V plan will be developed describing how a 3U payload would be tested and integrated with the 3U common bus.

3.3.3.4 Mission Compatibility Assessment

By answering the third research question the author seeks to develop a notional 3U common bus design with documents detailing how the 3U payload interfaces with the 3U common bus. The author seeks to also develop a notional I&T and V&V plan outlining how the CubeSat 3U common bus will be tested and integrated with the 3U payload and be tested as a 6U CubeSat.

3.4 Summary

This chapter has established the problem statement, research focus, and research questions to be addressed. The following chapter presents a detailed CubeSat lifecycle derived from the NASA Systems Engineering Handbook and CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers, which together form the technical foundation for schedule modeling and the 3U common bus analysis presented in Chapters 4 through 6 (NASA Systems Engineering Handbook, 2016; National Aeronautics and Space Administration, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017).

CHAPTER 4: STATE OF THE ART OF CUBESAT ENGINEERING

This chapter seeks to answer Research Question 1, which is restated as follows: What is the current state of CubeSat systems engineering?

In response, this chapter presents a literature review encompassing academia, industry, and NASA CubeSat missions. The review examines reported CubeSat failures and, where identified, the causes of those failures. The goal of the research is to identify the most common CubeSat failure modes, determine which subsystems fail most frequently, and identify the common causes of those failures where noted.

A literature review and a set of industry interviews were conducted to determine the most common failure modes found in CubeSats. This chapter presents the introduction, literature review results, interview results, and discussion and conclusions associated with this research. Upon completion of this chapter, the author will have the information needed to answer Research Question 1.

4.1 Interview Methods

The research employed a qualitative methods approach consisting of a literature review and interviews with subject matter experts (SMEs) relevant to the dissertation topic. The literature review focused on documented CubeSat failures and the pre-launch testing activities performed prior to flight. Peer-reviewed journal articles, conference proceedings, NASA technical reports, and publicly available mission information were reviewed to characterize CubeSat reliability and identify subsystems commonly associated with premature on-orbit failure. This review established a baseline understanding of reported failure modes, as well as V&V, I&T, and environmental testing activities described in the literature.

Interviews with CubeSat SMEs were conducted to supplement the literature review. These interviews were used to obtain engineer perspectives on subsystem failures, V&V activities, I&T activities, and environmental testing decisions that are often not fully documented in formal publications. The interview data were used to place the literature findings in context and were not treated as a statistically representative sample. Detail regarding interview participants, recruitment methodology, interview structure, and response count is provided in the following sections.

4.1.1 Subject Matter Expert Participants

SME input was obtained through interviews with six professionals directly involved in NASA CubeSat flight missions. Participants included four CubeSat lead engineers and two CubeSat program managers affiliated with NASA and the Jet Propulsion Laboratory (JPL). These individuals supported an average of two to four CubeSat missions and were directly involved with per-launch V&V and I&T activities, environmental testing, and post-launch assessment activities. All participants were responsible for flight hardware and/or program execution, providing practitioner insight into CubeSat subsystem failures and pre-launch verification and testing activities. Participation was voluntary, and individual responses were anonymized to support candid discussion of mission outcomes and testing limitations.

4.1.2 SME Participant Identification and Recruitment

Participants were identified through a selection process focused on NASA CubeSat missions that experienced partial or complete on-orbit failure. Publicly available mission summaries, conference proceedings, and journal publications were reviewed to identify candidate missions. Individuals associated with those missions were identified based on their inclusion in the literature. This approach ensured that interviewees had direct knowledge of the CubeSat failure modes encountered. Participants were contacted directly. This research was approved by Colorado State University Institutional Review Board under protocol #3990, December 27, 2022.

4.1.3 Interview Structure and Questions

The interviews were conducted using a semi-structured format to enable consistent coverage of topics while allowing participants to elaborate based on mission-specific events. Interviews were conducted through a combination of phone discussions and email correspondence. Core questions addressed which subsystems failed on the CubeSat mission, whether failure root causes were identified, and what level of pre-launch testing was performed. Additional questions focused on V&V activities, I&T activities, and whether environmental testing such as random vibration, thermal vacuum, or EMI/EMC testing was reduced or omitted due to schedule or programmatic constraints. Responses were captured through notes and written communication and synthesized qualitatively.

4.1.4 Response Count and Use of Interview Data

A total of six responses were obtained and included in the analysis. The limited number of responses reflects the small population of NASA CubeSat missions with publicly acknowledged failures and the limited availability of detailed post-failure documentation. Consequently, the interview data were not treated as statistically representative but rather as expert input. The interview results were used to augment the findings of the literature review by providing practitioner insight into testing shortfalls and subsystem failures that are often underreported in formal publications. The interview results are limited by the small population of CubeSat missions with publicly acknowledged failures and the availability of expert participants with direct flight experience.

4.2 CubeSat Literature Review

4.2.1 CubeSat Schedule Literature Review

CubeSat development times vary widely depending on the organization. On average, development time for non-university CubeSats ranges from 18–24 months (Richardson et al., 2015). Development times for various organizations breakdown as follows: commercial CubeSats are completed in approximately 1.7 years, Department of Defense (DoD) CubeSats in 1.6 years, and university CubeSats in approximately 3.8 years (Richardson et al., 2015). University CubeSat development time is the longest, often taking between 36 and 48 months from concept to completion. Funding availability and technical requirements can significantly influence this timeframe (CubeSat 101, 2017). Critical phases include concept development, design reviews, hardware fabrication, testing, integration, and launch manifesting, with the latter often representing the most variable component of the schedule (Olson, 2019).

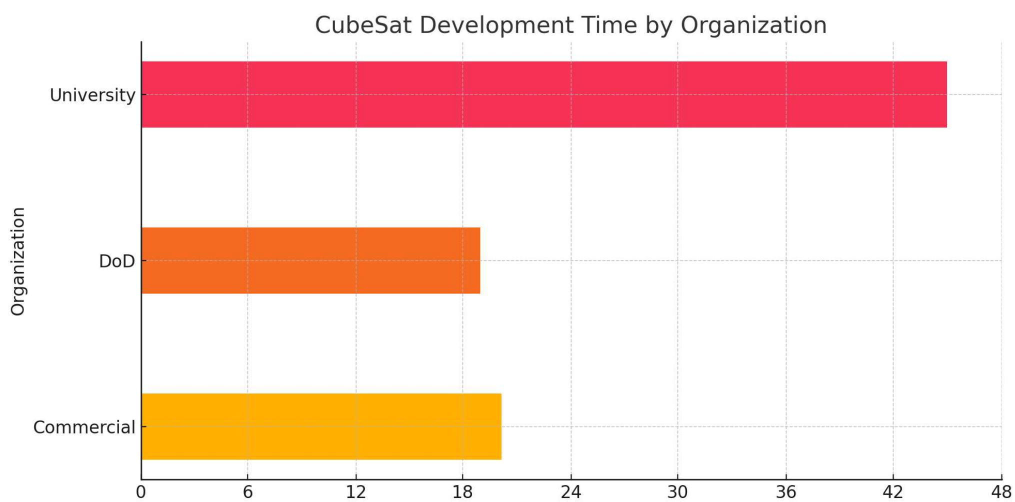

Figure 4. CubeSat development time by organization. Typical university CubeSat development time is 46 months, DoD development time is 19 months, and industry development time is 20 months on average. *Data from Richardson et al. (2015).*

For example, ASTERIA shown in Figure 5 was a CubeSat designed to conduct observations of nearby stars from low Earth orbit. ASTERIA was developed at the JPL in collaboration with the Massachusetts Institute of Technology. Development began in December 2014 and concluded in June 2017 with delivery to NanoRacks (Smith et al., 2018), resulting in a total development time of 30 months.

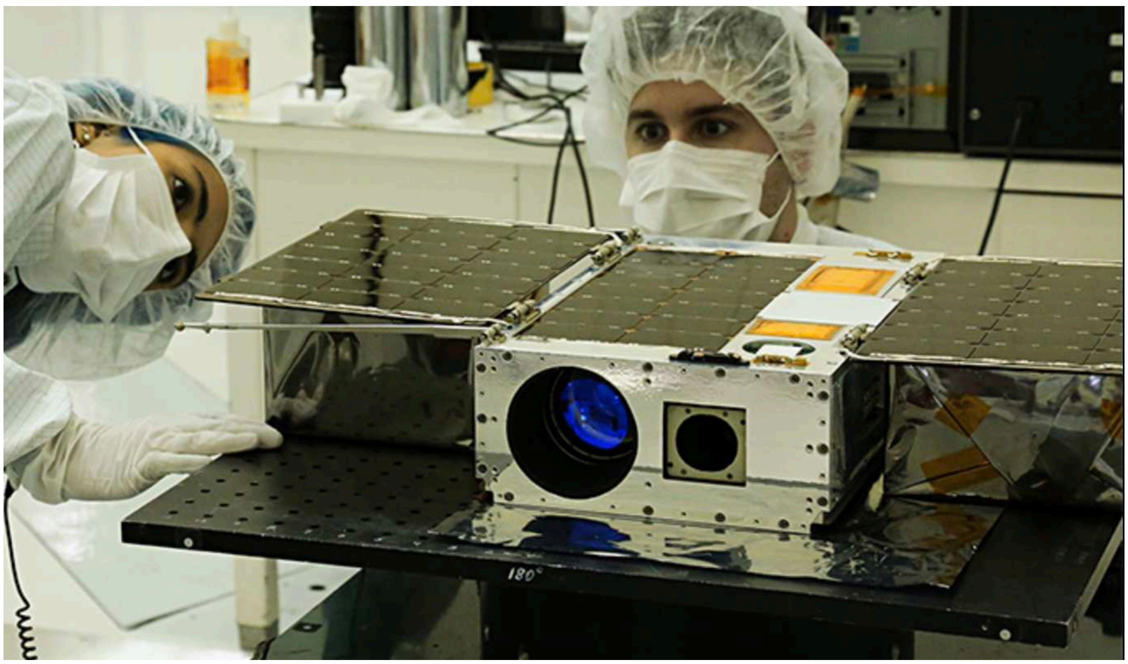

Figure 5. ASTERIA CubeSat during testing: *Source: NASA/JPL-Caltech (public domain)*

The TEMPEST-D CubeSat shown in Figure 6 was a collaborative project between Colorado State University (CSU) and NASA JPL. TEMPEST-D was led by CSU, with JPL responsible for providing the instrument and contracting the CubeSat bus. TEMPEST-D began construction in August 2015 and was completed in August 2017 for a total development time of 24 months (Reising et al., 2018).

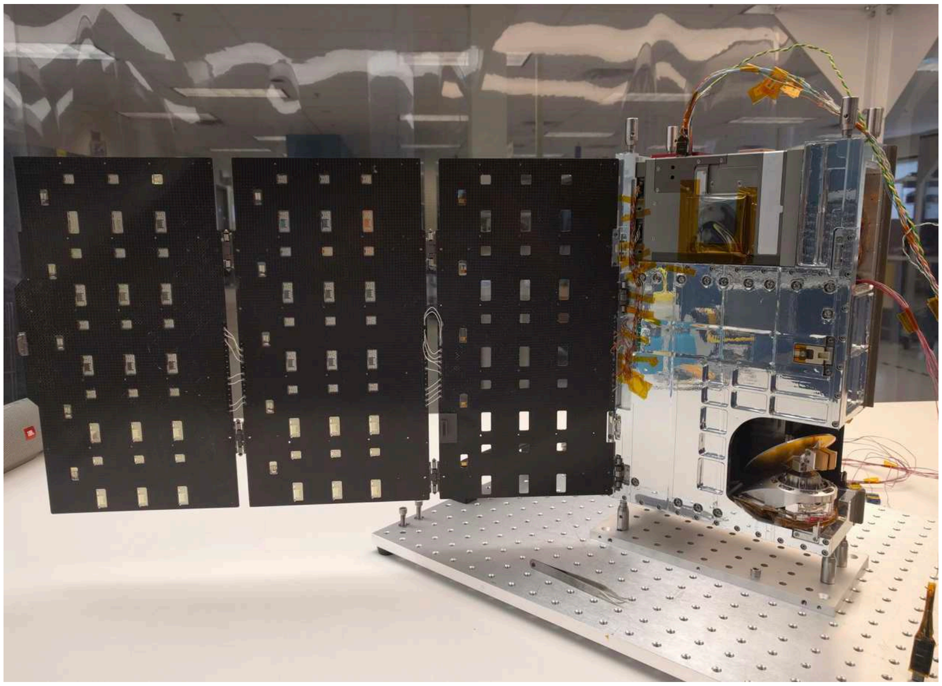

Figure 6. The completed TEMPEST-D CubeSat shown with solar panels deployed. *Source: NASA/JPL-Caltech/Blue Canyon Technologies*

The development times outlined above represent a sample of relevant CubeSat missions and illustrate the time required for CubeSat bus and payload engineering and development. In general, CubeSat design and development are schedule-constrained, which can result in testing being waived to meet delivery milestones.

4.2.2 CubeSat Failure Literature Review

4.2.2.1 Communication System

In 2016, Langer et al. surveyed 100 launched CubeSats and found that 27% of failures could be traced to communications system failure modes, including configuration or interface failures between communications subsystems. These failure modes prevented the CubeSat from making contact with the ground control station in some cases or severely limited communications capability. Other studies found that communications failures contributed to approximately 20–

30% of all CubeSat mission failures (Langer et al., 2016; Bouwmeester et al., 2022). Among the failures antenna deployment was found to impact approximately 18% of missions (Alanazi & Straub, 2018). Failure of the antenna to deploy makes communication with the ground station extremely difficult or impossible and is often mission-ending. RF interference, insufficient link margin, and inadequate electromagnetic compatibility control are well-established contributors to communication loss in small spacecraft (ECSS, 2020; Small spacecraft overview, NASA, 2019). Large-scale CubeSat mission surveys further show that communications-related failures account for a substantial fraction of overall mission losses (Langer et al., 2016; Bouwmeester et al., 2022).

In the most severe cases, the communications system fails to activate, and the CubeSat is dead on arrival (DOA) to orbit. DOA failures have been linked to inadequate pre-flight testing and were found to occur in approximately 48% of cases (Langer et al., 2016). This failure mode could be reduced if CubeSat underwent full end-to-end RF testing prior to launch to verify the communication subsystem was functioning as intended (Langer et al., 2016). While faults in other subsystems may be recoverable, failures in the communication system can end a mission before it begins. Without a functioning communication system, there is no means to transmit data to or receive data.

4.2.2.2 Electronic Power System

The EPS conditions, regulates, stores, and distributes electrical power generated by the solar arrays to the CubeSat avionics and payload and manages the charging and protection of the onboard batteries. As a result, the EPS spans multiple physical and functional interfaces, including power generation, energy storage, electrical interconnects, and thermal control, making it a critical contributor to overall mission reliability.

Surveys of CubeSat mission outcomes consistently identify the EPS as a primary source of failure. Early analyses found that approximately 14% of CubeSat failures could be traced directly to EPS related issues. Subsequent studies expanded on this finding, reporting that EPS issues contributed to up to 40% of total CubeSat mission failures and were responsible for as much as 80% of partial on-orbit failures (Alanazi & Straub, 2018; Langer & Bouwmeester 2016).

Within the EPS architecture, electrical interfaces represent a potential failure point due to their role in power distribution across subsystems. Bouwmeester et al. (2010) examined the use of PC/104 embedded connectors, which have been widely adopted as a standardized electrical and data interface between CubeSat subsystems. The PC/104 connector, shown in Figure 7, is a through-hole, board-mounted connector that is compact, mechanically robust, and expandable, enabling stacked printed circuit board configurations as illustrated in Figure 7. While Bouwmeester et al. ruled out the PC/104 connector itself as a dominant failure mode, they noted its relatively large footprint and recommended transitioning to smaller connectors to reduce volume and integration complexity. Although not a primary failure driver, EPS interconnects remain a critical element due to their role in distributing power to all subsystems.

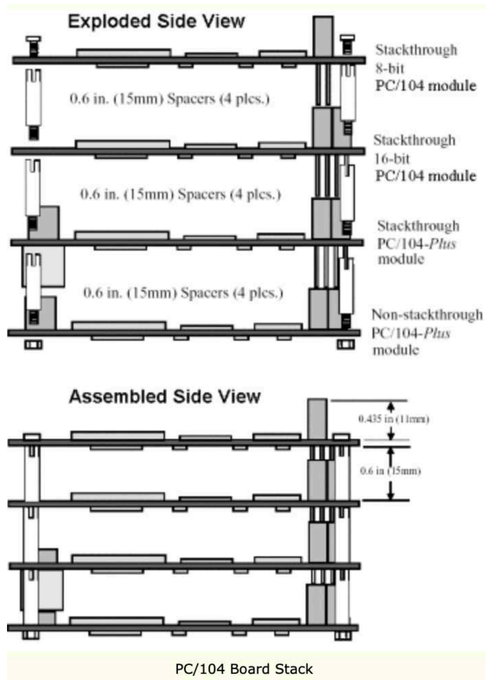

Figure 7. PC/104 connectors are widely used in CubeSats because of their small form factor and mechanical ruggedness. *Source: PC/104 Consortium (2003), PC/104 Specification, Version 2.6.*

Energy generation and storage further couple EPS performance to mission success. Batteries are a key EPS component, providing power during eclipse periods when no energy is available from the solar arrays. Following orbital deployment, CubeSats initially operate on internal battery power until the solar arrays are deployed and begin generating power. If solar array deployment fails, the CubeSat is limited to battery-only operation until energy depletion occurs, resulting in mission failure. Deployment failures have been observed in approximately 22% of CubeSat missions, directly linking solar array deployment reliability to EPS performance and mission lifetime (The CubeSat Program, Cal Poly SLO, 2022, 2020; Alanazi & Straub, 2018).

The thermal environment further exacerbates EPS vulnerability. CubeSat batteries are subjected to extreme thermal cycling due to repeated transitions between sunlight and eclipse. Inadequate thermal management can accelerate battery degradation, leading to reduced capacity, increased internal resistance, and eventual failure. Thermal degradation of batteries has been identified as a contributing factor in approximately 30% of CubeSat failures (NASA-HDBK-4002A, 2021; Langer et al., 2016), reinforcing the strong coupling between EPS design, thermal control, and mission reliability.

Despite the central role of the EPS in CubeSat mission success, system-level verification and validation are frequently incomplete. Full CubeSat-level testing, particularly thermal vacuum testing, is critical for identifying EPS-related failure modes prior to launch. However, Alanazi and Straub (2018) report that many CubeSat missions omit or reduce at least one testing activity such as random vibration, TVAC, or end-to-end RF testing as part of their V&V and I&T. The absence of comprehensive environmental testing has been associated with elevated EPS failure rates, as EPS components are especially sensitive to thermal cycling and vacuum-induced effects (Bouwmeester et al., 2022). As a result, interface, battery, and thermally driven EPS failure modes may remain undetected prior to deployment.

4.2.2.3 On-Board Computer

The OBC, sometimes referred to as the flight computer, is responsible for command and data handling by issuing commands to CubeSat subsystems and collecting, processing, and formatting telemetry for transmission to ground stations. Without the OBC, the CubeSat would be unable to function or operate as designed. The OBC contributes to approximately 15–20% of CubeSat mission failures (Langer et al., 2016; Bouwmeester et al., 2022). These failures are frequently unidentified or attributed to radiation induced effects, including single-event upsets encountered on-orbit (JPL Radiation Handbook, n.d.; Bouwmeester et al., 2022). While CubeSat programs often lack the resources to perform comprehensive system-level or component-level radiation testing, prior studies suggest that a large fraction of OBC-related failures may be associated with radiation effects and insufficient fault-tolerance mechanisms, with some sources estimating this share to be as high as 87% (Langer et al., 2016; Swartwout, 2020).

In terms of system reliability, Langer & Bouwmeester conducted a survey of 178 CubeSats from the CubeSat failure database and found the on-orbit failures at 0-, 30-, and 90-days breakdown as shown in Table 1.

Table 1. Subsystem Failure Rates at T+0, T+30, and T+90 Days

Subsystem	T+0 days	T+30 days	T+90 days
On-board computer (OBC)	16%	20%	21%
Electrical power system (EPS)	28%	44%	36%
Communications (Comms)	15%	44%	29%
Unknown failures	33%	12%	36%

In the period after launch (T+0 days, T+30 days, and T+90 days) the predominant attributable failures are in the EPS and Comms subsystems. Langer & Bouwmeester assert that the early failure rate was due to lack of or poor systems-level functional testing during the I&T of the CubeSat. Langer and Bouwmeester attribute many early on-orbit failures to insufficient system-level functional testing during I&T, particularly cases where the integrated CubeSat was not operated long enough in a flight-equivalent state to reveal system-level performance issues prior to launch. Bouwmeester (2017) found that at least 65% of the CubeSats in the survey (n = 60) were expected not to fulfill all mission objectives. While this does not imply total mission failure in every case, it highlights the need to improve CubeSat reliability and to better understand the root causes of mission underperformance and failure. Alanazi and Straub (2018) examined 270 CubeSat missions and reported that, between 2002 and 2016, 139 experienced mission failure. Although their work was based on a limited survey dataset, the findings still illustrate the need to examine methods that can improve CubeSat mission success. Alanazi and Straub found that missions exhibiting higher failure rates were more likely to report reduced system testing time, suggesting that insufficient testing and verification may contribute to mission failure. This reduction in testing time was attributed to having subsystems tested by vendors prior to delivery and the use of qualification testing and requirements analysis. It was found that typically university-led projects operate under a false assumption that the CubeSat will work as intended the first time it is integrated. This being the case, university-led CubeSats will not allocate enough time to do subsystem-level testing and integration (Swartwout, 2014). Operating like this means that university CubeSats will thus have a higher failure rate given the assumption being made that everything will run and there will be no issues. The universities that do conduct testing tend to only focus on testing of the subsystems that the CubeSat is comprised of and often omit full CubeSat-level integration testing (Alanazi & Straub, 2018). This lack of full integrated CubeSat-level testing results in failures. Multiple surveys and statistical studies indicate that many university-led CubeSat projects either reduce or skip portions of standard environmental qualification testing (e.g., random vibration and TVAC), which may contribute to the relatively high rate of mission underperformance and failure in the CubeSat community (Alanazi & Straub, 2018; Bouwmeester, 2017). Langer et al. (2016) identify DOA failures as a major contributor to early CubeSat mission loss, with many CubeSats failing immediately after deployment. This reinforces the need for comprehensive end-to-end functional testing, including full communications verification, prior to launch.

4.2.2.4 CubeSat Reliability Expert Interviews

The public literature generally does not provide mission-specific information or detailed descriptions of on-orbit anomalies. To augment the information available in the literature, interviews were conducted with NASA personnel to discuss CubeSat mission issues and their underlying causes. Two relevant mission engineers were contacted who had worked on the IceCube and FireFly CubeSats. These engineers were interviewed using an unstructured interview process to determine the root cause of the cubesat failures, and to identify procedures and strategies that might have led to CubeSat failure.

IceCube was a 3U CubeSat developed at NASA Goddard Space Flight Center to demonstrate a commercial 833 GHz cloud radiometer instrument and advance the TRL of the instrument. The CubeSat employed predominantly COTS subsystems. IceCube was categorized as a NASA Class D mission and was therefore following a Class D mission methodology (Dong, 2023). During CubeSat I&T issues with a COTS subsystem were identified that required rework and re-engineering to support mission success (Dong, 2020). Limited program resources necessitated tailoring the test methodology to meet IceCube mission requirements. This tailoring included the omission of self-compatibility EMC testing (Dong, 2023).

Prior to delivery for launch, IceCube successfully completed an end-to-end communications test with the ground station (Dong, 2023). Following deployment, the CubeSat achieved on-orbit operation despite initial challenges related to instrument noise and Global Positioning System (GPS) lock acquisition within the attitude determination and control system. IceCube was designated as a technology demonstration mission with a planned operational lifetime of 28 days; however, the CubeSat remained operational for approximately 17 months before mission termination.

FireFly was a 3U CubeSat designed to study terrestrial gamma-ray flashes, their association with lightning, and the production of energetic electrons. The CubeSat was developed at NASA Goddard Space Flight Center and employed a combination of COTS components. Full V&V testing was performed with the exception of self-compatibility EMC test (Douglas, 2020).

On-orbit, FireFly experienced a loss of communication between the onboard OBC and the gamma-ray instrument. The custom 3U bus architecture used by FireFly was susceptible to failure propagation if any attached bus component became nonfunctional (Douglas, 2020). In this case, the failure of a commercial temperature sensor led to the failure of the communications bus, which subsequently prevented communication with the instrument. As a result, commanding of the instrument was no longer possible, leading to premature mission termination (Douglas, 2020).

Together these cases demonstrate that CubeSat V&V and I&T are often neglected due to cost and schedule constraints. The result is inadequate environmental and functional testing, especially for power and communications (Alanazi & Straub, 2018; UNOOSA KiboCUBE Academy, 2022), which leads to in service failure. This research will seek to incorporate this knowledge into a conceptualized 3U common Bus.

4.2.2.5 Case Studies: NASA CubeSat Failures and Lessons Learned

This section examines a sample of recent CubeSat missions that experienced on-orbit failures and the associated failure modes. In contrast to the interviews described previously, these failures are documented in the open literature, albeit at a lower level of detail.

1. LunaH-Map (Artemis I, 2022)
    a. Failure: Propulsion system malfunction.
    b. Cause: A partially stuck valve in the propulsion system prevented sufficient propellant flow for trajectory correction.
    c. Source: TLP Network (2024). Wall (2022) reported that LunaH-Map failed to execute its planned engine burn due to a partially stuck valve in the propulsion system.

2. OMOTENASHI (Artemis I, 2022)
    a. Failure: Loss of communication and power.
    b. Cause: Solar array deployment failure led to insufficient power generation.
    c. Source: According to JAXA (2022), the OMOTENASHI CubeSat did not complete Sun acquisition and maintained unstable communications after deployment.
3. CuSP (Artemis I, 2022)
    a. Failure: Communication loss.
    b. Cause: Suspected battery malfunction affecting RF systems.
    c. Source: CuSP experienced a loss of communication shortly after deployment, and the mission was subsequently declared unsuccessful (Tangermann, 2022).
4. OCO-2 CubeSat Companion (2014)
    a. Failure: Lost communication post-deployment.
    b. Cause: Radiation-induced latch-up in an unhardened voltage regulator.
    c. Source: In some small spacecraft missions, internal charging or radiation events have caused latch-up in unprotected regulators, resulting in loss of communication or mission failure, underscoring the need for radiation-tolerant design and mitigation practices outlined in NASA-HDBK-4002A (NASA, 2015; Minow et al., 2014).
5. RadSat-g (ELaNa XIX, 2019)
    a. Failure: Solar array deployment failure.

b. Cause: Software fault in deployment sequencing; lack of thermal vacuum testing.

c. Source: TechEdSat-8 failure due to power loss after deployment, a documented CubeSat issue that illustrates similar mechanism risk (not specific to RadSat-g) (TechEdSat-8, 2019).

4.2.2.6 Common Failure Themes from NASA Sources

1. Spacecraft charging and electrostatic discharge impacts: Energetic particle interactions in the space environment can lead to internal charging and electrostatic discharge events that disrupt or damage electronic components, especially if they are not designed for such environments (NASA-HDBK-4002A, 2021).
2. Inadequate Testing:

    a) NASA GEVS provides baseline guidance for spacecraft environmental verification, including random vibration and TVAC testing, which are commonly used to validate workmanship and flight readiness. Therefore, reducing or omitting these tests decreases confidence in a CubeSat's ability to survive launch loads and the space thermal environment (GSFC-STD-7000B, 2021).

    b) Many CubeSat missions exhibit incomplete system-level verification and validation, including communications subsystem testing, contributing to a ~50% mission success rate shortfall in historical CubeSat performance data (Swartwout, 2016; Alanazi, 2018).

A recurring theme across documented CubeSat failures is the absence of comprehensive environmental and integration testing, including random vibration, TVAC, and EMI/EMC testing.

Approximately 65% of CubeSat missions do not perform one of the follow random vibration, TVAC, or end-to-end RF testing, which limits the programs ability to identify latent design and integration issues prior to launch and is associated with an increased risk of partial or complete mission failure during launch or on-orbit operations. Currently there isn't a widely used systems engineering approach tailored to CubeSats integration and testing. An ad hoc methodology for integration and testing based of the methods used for large spacecraft is used at NASA. This ad hoc method usually includes tailoring the systems engineering methods for integration and testing for each CubeSat mission based on the mission's risk tolerance level. This means that CubeSats will sometimes skip entire tests and thus misidentifying system issues. This testing is needed because each CubeSat is essentially a custom CubeSat comprised of COTS parts for common subsystems that are common in each CubeSat. One method to help with this would be the introduction of a 3U common bus incorporation all needed subsystem to support a payload. The user of the 3U common bus would then only have to focus on their payload and integration into the 3U common bus. The following sections examine these failures trends in the broader context of CubeSat development processes, systems engineering practices, and I&T methodologies.

4.2.3 CubeSat Development: A Systems Engineering Perspective

4.2.3.1 CubeSat Development Process Overview and Timeline

CubeSats generally follow a development process that is very similar to larger spacecraft. The only real differences being timeline and mission classification. According to the NASA CubeSat Launch Initiative (CSLI) on average CubeSat development requires 18-24 months to complete with the delivery to launch provider taking place 1-6 months prior to launch (CubeSat 101, 2017; NASA CSLI Proposer's Guide, 2023). Like all complex engineering projects CubeSat development follows the standard systems engineering process. This process encompasses a

System Requirements Review (SRR), Preliminary Design Review (PDR), and Critical Design Review (CDR). Each review is a gate for verification of the CubeSat design and serves as a checkpoint for verification the design will accomplish its mission.

4.2.3.2 Detailed Development Phase Breakdown

4.2.3.2.1 Detailed Development Concept Development and Funding (Months 1–6)

The first phase in CubeSat development focuses on the developing the mission objectives and the requirements needed to meet those objectives. If the NASA CSLI is going to be used to launching the CubeSat it requires proposals to show alignment with NASA's strategic goals such as technology demonstrating or Earth science research (NASA CSLI Proposer's Guide, 2023). The mission objectives and requirements development phase typically last 1-6 months during this time the team will develop the payload requirements and the needed CubeSat system requirements will fall out of this (Olson, 2019). A good example of this is the University of Colorado's QB50 CubeSat project which took 3 months to ensure their missions aligned with the QB50 consortium's goal of studying the thermosphere (University of Colorado Boulder, 2016). This approach emphasized payload stakeholder engagement to ensure the technical requirements met the scientific mission requirements.

4.2.3.2.2 Design and Procurement (Months 6–12)

During the design phase, detailed engineering is performed of the CubeSat subsystems. The designs are gated by PDR and CDR where the subsystem designs are validated against payload and mission requirements. NASA guidelines recommend the allocation of 1-6 months for this phase of the development (CubeSat 101 2017). Once the design passes CDR (in some cases after passing PDR) the procurement of COTS components starts typically taking 2-3 months. However, manufacturer's lead times can extend this process. For example, the Phoenix CubeSat mission faced a 3 month delay due to infrared sensor availability (eoPortal, 2019).

4.2.3.2.3 Integration and Testing (Months 12–18)

During integration and testing phases, the CubeSat's mechanical, electrical, and software subsystems are integrated into the CubeSat in the flight-ready configuration for full integrated systems testing and environmental testing. During integrated systems-level testing, the fully integrated CubeSat is run through full mission simulation to verify that it functions as-intended (CubeSat 101, 2017). Once the integrated systems test is completed the CubeSat will undergo environmental testing. Environmental testing protocols vary between CubeSat programs, but they can include TVAC, random vibration testing, and EMI/EMC testing (CubeSat 101, 2017). Because the constraints on launch time are so rigid for many CubeSats, the schedule often limits what types and durations of environmental testing is done on the CubeSat. To meet schedule constraints, environmental and integration testing is sometimes skipped. For example, the Phoenix CubeSat program dedicated 4 months to testing, exposing their 3U CubeSat to extreme temperatures and mechanical stresses to validate performance in low-Earth orbit. They chose to skip EMI testing (eoPortal, 2019) to make up for schedule constraints. Integration and testing activities are intended to identify critical issues prior to launch, as such issues are often not apparent until systems are fully integrated. For example, NASA's MarCO CubeSat I&T identified a propulsion system leak, which required a 6-week engineering effort to address the malfunctions (JPL, 2019).

4.2.3.2.4 Launch Preparation (Months 18–24)

Approximately 10-12 months prior to a launch, a launch readiness review (LRR) is held. During the LRR all documentation is reviewed and signed off on, and any outstanding items critical to launch are closed (CubeSat 101, 2017). After successful completion of the LRR and sign off, the delivery and integration to the CubeSat dispenser on the launch vehicle takes place. Integration with the dispenser requires coordination with the launch provider and often requires visits to the payload integration site prior to delivery of the CubeSat integration. This takes place 4-6 months prior to launch. For example, the CubeSat Radio Interferometry (CURIE) mission which launched on the ESA's Ariane 6 in 2024 required 2 days of dispenser compatibility checks and 1 day of dispenser integration on the launch vehicle (NASA, 2024). The 3 days of launch vehicle integration activities for CURIE sit comfortably within the standard practice for a rideshare participant on the Ariane 6.

4.2.3.3 Mission Definition and Requirements Analysis

CubeSat projects begin with mission payload requirements and aligning requirements with technical and budgetary constraints. NASA's CubeSat 101 emphasizes establishing clear mission requirements, including payload needs, orbital parameters, compliance with launch vehicle interfaces, and more (CubeSat 101, 2017).

4.2.3.4 Structural and Design Requirements in CubeSat Literature

CubeSat mechanical design begins with a defined set of mechanical requirements derived to ensure survivability through launch and safe deployment, while also providing sufficient structural stability and standardized interfaces to secure and support the payload. These requirements typically include limits on mass, minimum stiffness often expressed through first-mode natural frequency targets specified by the launch provider strength margins, deployer and interface constraints, and workmanship and material controls as defined in the CubeSat Design Specification Rev. 14.1*,* the NASA General Environmental Verification Standard (GSFC-STD-7000, 2021, and NASA-STD-5001B Structural Design and Test Factors of Safety for Spaceflight Hardware (The CubeSat Program, Cal Poly San Luis Obispo, 2022, GSFC-STD-7000, 2021; NASA-STD-5001B, 2021). In practice, the CubeSat primary structure (e.g., rail-based or tabbed interfaces) must comply with the mechanical envelope, rail surfaces, contact areas, and allowable protrusions defined by the selected deployment system to ensure proper integration and successful ejection (Cal Poly CubeSat Program, 2022). When a tabbed deployment system is used such as the Rocket Lab/Planetary Systems Corporation (PSC) Canisterized Satellite Dispenser (CSD) the CubeSat structure must instead conform to the tab-specific mechanical envelope, load paths, and interface characteristics defined by the dispenser provider (Rocket Lab/PSC, 2018; NASA S3VI, n.d.).

CubeSat structures are typically constructed from 6000- and 7000-series aluminum alloys with hard anodization to improve wear resistance, reduce galling, and provide electrical isolation. Structural integrity is achieved through the use of mechanical fasteners and structural adhesive bonding. These material selections and structural practices are consistent with guidance outlined in the NASA Small Spacecraft Systems Virtual Institute's state-of-the-art overview on spacecraft structures, materials, and mechanisms (NASA Small Spacecraft Systems Virtual Institute [S3VI], n.d.). The selection of alloys, test factors, and safety factors is typically performed in accordance with NASA-STD-5001**.** Payload critical alignments and load paths are typically designed to maintain stiffness and dimensional stability across thermal and dynamic environments while paying attention to keep-out zones and volume constraints within the dispenser (Cal Poly CubeSat Program, 2022).

CubeSat design verification follows accepted spaceflight environmental standard testing consisting of random/sine vibration, shock testing, and thermal-vacuum/bake-out, with notching or response-limiting where required to protect the flight hardware while still demonstrating flight readiness (GSFC-STD-7000, 2021; GSFC-HDBK-8007, 2019). For CubeSats specifically, best practice is to test with a high-fidelity prototype or engineering model of the intended dispenser to preserve interface realism and load paths (GSFC-HDBK-8007, 2019). Dynamic criteria and margin philosophies such as uncertainty factors and statistical enclosure levels are further guided by *NASA-HDBK-7005: Dynamic Environmental Criteria* and related handbooks addressing launch loads and structural dynamics (NASA-HDBK-7005, 2018). Together, these standards ensure that the mechanical design provides adequate margins, stiffness, and clean deployment to support payload pointing, thermal control, and contamination requirements, as defined in CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers and NASA Small Spacecraft Systems Virtual Institute guidance on small spacecraft integration and test best practices (CubeSat 101, 2017; NASA Small Spacecraft Systems Virtual Institute [S3VI], n.d.).

4.2.3.5 CubeSat Integration Challenges and the Role of Systems Engineering

CubeSat integration presents various challenges given their size and the complexity of the systems within them. The process of integrating subsystems with the CubeSat is particularly prone to failure due to the complexity and smaller size of the subsystems. Typically, some of the subsystems are comprised of COTS from different vendors each with potential incompatibilities and various levels of documentation and qualification quality (Bouwmeester et al., 2017).

Studies have found that the most common causes of CubeSat mission failure on-orbit are integration issues including hardware compatibility issues between subsystem manufacturers (Sukhwani et al., 2016). Secondly, integration to the power system is a major cause of failures, as variations in actual performance of COTS power systems often fail to match reported COTS specifications. These power system issues typically require extensive testing to troubleshoot and to avoid catastrophic failures (Venturini et al., 2018). Next, communication systems pose integration issues due to the potential for multidisciplinary interactions including EMI, and power system failure. A failure of the communication system on-orbit is typically a mission-ending event. Communication system failures are frequently found to be attributed to integration problems. (Bouwmeester et al., 2017).

Effective systems engineering plays a critical role in addressing integration planning and ensuring successful integration of subsystems into the CubeSat and ensuring mission success (Kaslow et al., 2017; Magone & Simske, 2025). Systems engineering is used to create a comprehensive framework to manage the complex interdependences between the subsystems through interface control documentation and systematic integration testing (NanoAvionics, 2025). Systems engineering practices encourage the implementation of proper documentation, interface control documents, and systematic verification matrices that are essential for successful integration of COTS components from multiple vendors (Garrett, 2022). Use of interface control documents when working with COTS from multiple vendors is especially important as this helps insure the fully integrated subsystems in the CubeSat will be able to interface with one another mechanically and electrically. Utilization of comprehensive systems engineering framework in educational CubeSat programs has demonstrated improved mission success rates when thorough integration testing protocols are used (Garrett, 2022).

4.2.3.6 CubeSat Functional Testing and the Importance of Systems Engineering

CubeSat-level functional testing is one of the most critical V&V processes in CubeSat development and verifies and validates all subsystems are functional and the CubeSat is operational in all flight modes before the CubeSat moves to environmental testing (GSFC-STD-7000, 2021). This V&V testing runs the hardware subsystems and software through defined testing encompassing all aspects of the on-orbit mission profile including command execution tests where all commands are verified correct (Venturini et al., 2018). Additionally, the CubeSat is run through a day-in-the-life testing regime where a typical 24-hour orbital period test is run which includes end-to-end testing of the communications, power, and compute system just to name a few. Functional testing is typically done to help identify workmanship flaws and manufacturing damage that might not be identified through other verification methods. (GSFC-STD-7000, 2021). The V&V testing does not just focus on the CubeSat subsystems it also encompasses payload functional testing meant to identify software and hardware faults. Integration tests ensure compatibility and interoperability when integrating subsystems from different manufacturers (NanoAvionics, 2024).

Systems engineering plays an important role in development of effective functional test plans and methodologies which are designed to run the CubeSat through its paces and verify the functionality of the entire integrated CubeSat system, and it helps contribute to the reduction of the 50% CubeSat failure rate through robust testing procedures (Magone & Simske, 2025).

A systematic methodology of test planning enables the development of a proper verification matrix being developed to verify requirements are met and to ensure coverage of all operational cases (Cho, 2021). The systems engineering literature provides frameworks for developing mission success approaches that establish minimum testing requirements, such as NASA's recommendation for 500-1000 hours of system-level testing time depending on mission criticality, ensuring adequate screening for workmanship flaws and design errors (NASA Systems Engineering Handbook, 2016). Furthermore, systems engineering methodologies ensure proper integration of functional testing with mission requirements through systematic verification and validation approaches that trace testing activities to stakeholder needs and mission objectives, significantly improving the probability of mission success by identifying and resolving critical issues before flight operations commence (Kaslow et al., 2017).

4.2.4 CubeSat Environmental Testing

Frequency (Hz)	ASD Level (g^2/Hz)	
	Qualification	Acceptance
20	0.026	0.013
20-50	+6 dB/oct	+6 dB/oct
50-800	0.16	0.08
800-2000	-6 dB/oct	-6 dB/oct
2000	0.026	0.013
Overall	14.1 G_{rms}	10.0 G_{rms}

Figure 8. NASA General Random Vibration Test Levels for payload 22.7 kg or less. Source: GSFC-STD-7000B (NASA Goddard Space Flight Center, 2021)

Environmental testing of a CubeSat is essential to ensure the CubeSat can survive the launch environment and the space environments. Most CubeSats are tested to the NASA general environmental specification with key testing includes random vibration (see figure 8), and TVAC testing to simulate the dynamic environment the CubeSat will see during launch and in space. The goal is to identify potential design or build flaws that could lead to failure of the CubeSat (Miller, 2016; NASA Systems Engineering Handbook, 2016). TVAC testing will expose the CubeSat to the extreme temperature swings it will encounter on-orbit with the goal being to verify the

CubeSats perform as expected and identify any defects (EnduroSat, 2024; SatNow, 2025). Full system-level testing is especially important because it can detect latent integration errors, verify proper operation across all interfaces, and ensure that the complete CubeSat perform as intended during simulated mission sequences, ultimately reducing the risk of mission failure and improving mission reliability (Kaslow et al., 2017; SPIE, 2021). Environmental testing is fundamental for qualifying a CubeSat for launch and is considered one of the best practices for achieving higher mission success rates, as defined in NASA GEVS (GSFC-STD-7000, 2021; SatNow, 2025). The typical environmental testing process for a CubeSat is outlined below:

- **Random Vibration Testing (Figure 9):** Evaluates the CubeSat's ability to survive the launch vibration environment in accordance with NASA GEVS GSFC-STD-7000. In addition to structural survivability, random vibration is a used for workmanship screening to uncover latent manufacturing defects (e.g., loose fasteners, cracked solder joints, connector issues) and is commonly paired with pre/post (and sometimes during-test) functional electrical checks to detect intermittent faults such as opens/shorts.

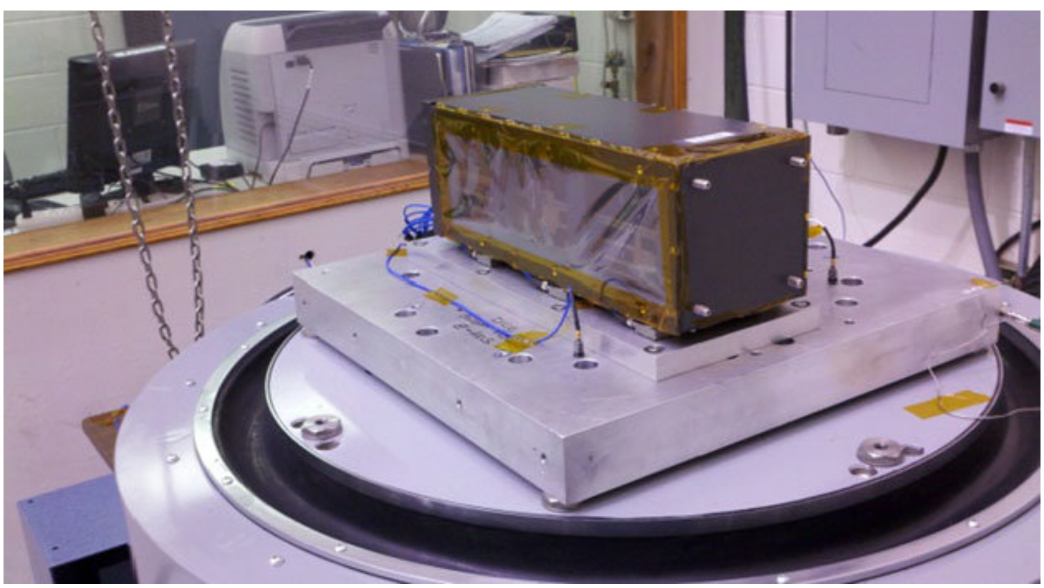

Figure 9. Dynamic Ionosphere CubeSat Experiment ready for random vibration testing *(Fish et al., 2014)*.

- **TVAC Testing (Figure 10):** Simulates the on-orbit thermal and high-vacuum environment**;** used to evaluate thermal control performance and verify electronic hardware functionality and reliability under expected hot/cold extremes in vacuum.

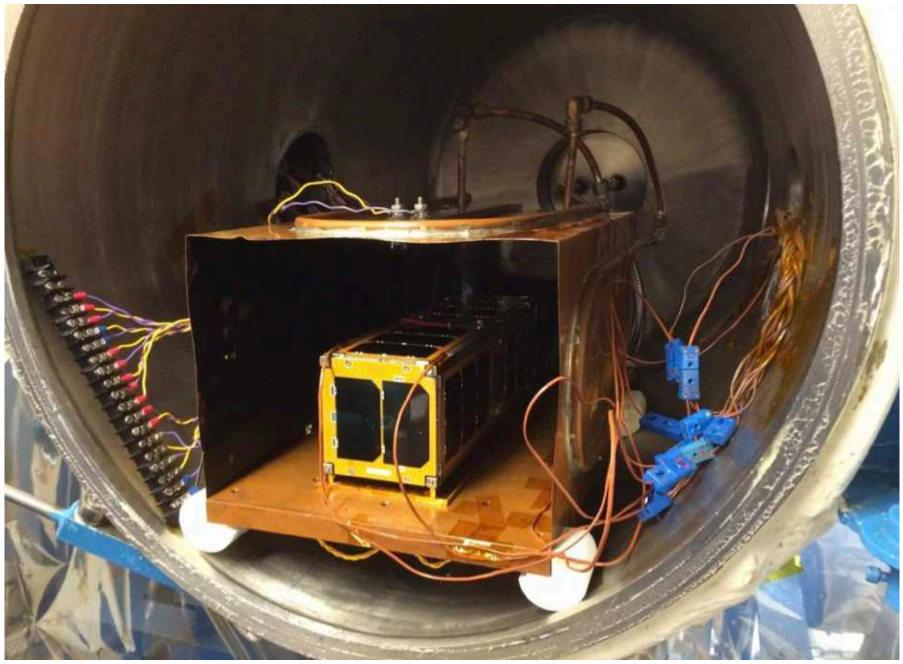

Figure 10. 3U CubeSat in TVAC instrumented for testing *(Souza et al., 2016)*.

- **EMI/EMC Testing (Figure 11):** This test ensures that CubeSat subsystems operate without generating or being susceptible to electromagnetic interference, either internal to the CubeSat or external to the CubeSat environment, consistent with the NASA Small Spacecraft Systems Virtual Institute (S3VI) Knowledge Base guidance on EMI/EMC testing (NASA Small Spacecraft Systems Virtual Institute, 2021a; Sigma Design, 2025).

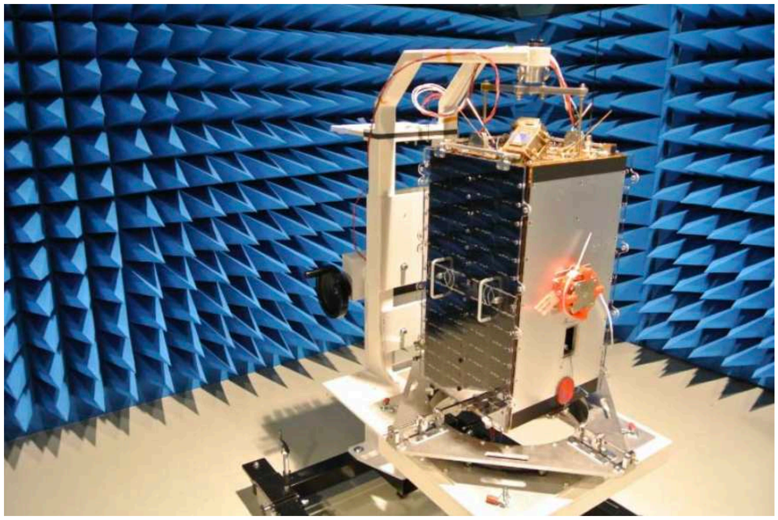

Figure 11. CubeSat in an anechoic chamber undergoing EMC testing *(IST Group, 2024)*.

Following each test a full CubeSat-level systems test is run to verify there has not been any damage to the CubeSat and to verify the systems and subsystems function as expected.

4.3 Discussion and Conclusion

The literature has pointed to three subsystems that cause the majority of failure in CubeSats the subsystems in question are as follows:

- Electrical Power System (EPS): Account for 44% of all failures after 30 days of operation
- On-Board Computer (OBC): Account for 20% of all failures after 30 days of operation
- Communication System (COM): Account for 44% of all failures after 30 days of operation

Together with failures categorized as unknown, these subsystems dominate observed CubeSat mission losses. The high prevalence of unknown failures (33% at T+0 days on-orbit) further suggests deficiencies in system-level verification, fault detection, and diagnostic telemetry, limiting the ability to attribute root cause and implement corrective actions. These weaknesses are especially consequential for university-led CubeSat missions, which commonly operate under strict schedule constraints because CubeSats are typically manifested as secondary payloads with inflexible launch delivery deadlines. As schedules compress, late-stage I&T and V&V activities are often reduced or omitted, resulting in insufficient end-to-end system-level testing prior to launch. This compression of verification represents a persistent mission assurance weakness and contributes directly to premature on-orbit failure.

4.3.1 Research Needs for CubeSat Systems Engineering

Through the process of this lit review, we have developed a set of research needs that can be used to inform future research to improve CubeSat performance in practice.

4.3.1.1 Need for Systems Engineering Discipline

CubeSat development is currently in its infancy it has matured a great deal from the introduction of the CubeSat standard but there is still room for improvement. The absence of a standardized bus architecture for subsystems is the most notable area in need of improvement. Each CubeSat mission typically develops a custom bus using various COTS systems such as EPS, communications, compute, and ADCS.

The current approach to CubeSats leads to repeated engineering efforts and increased development time as well as reduced reliability due to lack of flight heritage. The subsystems on the CubeSat may have been used by other missions but the configurations they are used in may be different. Development of a common CubeSat bus architecture would enable teams to focus on the development of the payload rather than reinventing the CubeSat bus. Without a widely adopted industry standard each development team creates a custom solution using COTS parts this compromises both schedule and reliability.

4.3.1.2 Need for 3U Common Bus Architecture

CubeSat development is currently in its infancy it has matured a great deal from the introduction of the CubeSat standard but there is still room for improvement. The absence of a standardized bus architecture for subsystems is the most notable area in need of improvement. Each CubeSat mission typically develops a custom bus using various COTS systems such as EPS, communications, compute, and ADCS.

The current approach to CubeSats leads to repeated engineering efforts and increased development time as well as reduced reliability due to lack of flight heritage. The subsystems on the CubeSat may have been used by other missions but the configurations they are used in may be different. Development of a common CubeSat bus architecture would enable teams to focus on the development of the payload rather than reinventing the CubeSat bus. Without a widely adopted industry standard each development team creates a custom solution using COTS parts this compromises both schedule and reliability.

4.3.1.3 Need for Improved Testing Methodologies

The literature reviewed constantly identified inadequate testing as the primary contributor to Cubesat mission failures. Current testing approaches often allow the omission of certain verification activities to meet launch schedules resulting in incomplete validation of critical system functions. Full mission simulation which simulates operational scenarios is also partially lacking in current CubeSat development practices. In addition, critical environmental testing is sometimes also skipped to meet launch schedule.

The need for improved testing methodologies needs to be address and it must consider schedule constraints and resource and budget limitation of typical CubeSat programs. Some suggestions to help include the development of rapid test protocols, standardize test procedures, and automated testing that can effectively validate system performance. Additionally, the implementation of a 3U common bus would allow the developers to focus more on their payload and full CubeSat I&T and V&V campaign.

4.3.2 Addressing Research Question 1

Based on the authors literature review and expert interviews conducted research question 1 can be answered through the syntheses of collected data and findings.

Research Question 1: A literature review will be conducted encompassing academia, industry, and NASA CubeSats. The review will examine the number of CubeSat failures and the causes of the failures where identified. The goal of the research is to identify the most common failure modes for CubeSats. The goal is to identify the subsystems that fail most frequently and the common causes of the failures.

4.3.2.1 Primary Subsystem Failure Analysis

The literature clearly identified three subsystems as the main source of CubeSat mission failures:

4.3.2.1.1 Electrical Power System (EPS)

The most critical failure point, accounting for 28% of failures at T+0 days, escalating to 44% at T+30 days, and maintaining 36% at T+90 days. EPS failures manifest through multiple mechanisms: battery degradation due to thermal cycling (responsible for 30% of all CubeSat failures), solar array deployment failures (affecting 22% of missions), and power system component performance mismatches where actual COTS specifications fail to meet reported values during integration.

4.3.2.1.2 Communications System (COM)

Contributing to 15% of failures at T+0 days, rising to 44% at T+30 days, and stabilizing at 29% at T+90 days. Communication failures occur through configuration or interface failures between communication subsystems (27% of surveyed missions), antenna deployment failures (18% of missions), and RF interference or inadequate link budgets (up to 40% of mission failures). Communication system failures are particularly mission-critical as it often represents mission-ending events due to the inability to command the CubeSat or receive data.

4.3.2.1.3 On-Board Computer (OBC)

Responsible for 16% of failures at T+0 days, increasing to 20% at T+30 days, and reaching 21% at T+90 days. OBC failures are predominantly attributed to radiation-induced single-event upsets (87% of OBC failures) due to inadequate radiation testing and insufficient fault tolerance design in the radiation environment of Earth orbit.

4.3.2.2 Root Cause Analysis: Systems Engineering Deficiencies

The research reveals that these subsystem failures are symptoms of deeper systems engineering process failures rather than inherent technological limitations:

4.3.2.2.1 Schedule-Driven Testing Compromises

CubeSats are built under severe schedule constraints where I&T and V&V are sometimes sacrificed to meet launch deadlines. The research found that 65% of CubeSat missions skip critical testing including EMI/EMC testing. This testing deficiency directly correlates with mission failures that could have been prevented with comprehensive end to end testing.

4.3.2.2.2 Custom Bus Development and Integration Challenges

Each CubeSat mission develops a custom bus using COTS components for common systems., This custom development approach creates failure points.

- Repeated engineering design efforts that introduce new integration risks.
- Lack of flight heritage for newly integrated subsystem combinations.
- Extensive integration and testing requirements between multiple COTS components with varying levels of documentation quality.
- Interface compatibility issues between subsystems from difference vendors that can go undetected until on-orbit operations.

4.3.2.3 The CubeSat 3U Common Bus Solution

The research shows that a standardized pre-tested common CubSat bus represents an effective solution to address the identified failure modes and the underlying system engineering causes.

4.3.2.3.1 Elimination of Custom Integration Risks

A common CubeSat bus that integrates EPS, COM, OBC, and other subsystems in a standardized pretested configuration would directly address the three primary failure modes identified in the research. By providing a thoroughly tested 3U common bus with integrated subsystems a 3U common bus could potentially eliminate integration related failures that result from COTS component combinations. Additionally, the common CubeSat bus could potentially eliminate failure cause by inadequate I&T and V&V operations at the bus level.

4.3.2.3.2 Proven Flight Heritage

The common CubeSat bus would undergo extensive qualification testing I&T and V&V and environmental testing that is sometimes skipped. This comprehensive testing establishes flight heritage for the integrated bus subsystems, directly addressing widely documented CubeSat mission assurance gaps particularly insufficient system-level functional testing which are strongly associated with DOA events and early on-orbit failures, most commonly impacting EPS, COM, and OBC subsystems.

4.3.2.3.3 Simplified Integration and Testing Requirements

The common CubeSat bus would allow the development team to focus only on the payload and payload integration with the 3U common bus. Eliminating the need to verify compatibility with the CubeSat bus subsystems. This allows the users to focus on integration of the payload to the 3U common bus and focus on full CubeSat I&T and V&V testing.

This dramatically reduces the integration testing burden from full CubeSat-level testing of custom subsystem combinations to targeted payload-bus integration testing. The payload integration testing can focus specifically on the mission-unique elements while leveraging the proven reliability of the pre-tested 3U common bus.

4.3.2.3.4 Addressing Schedule Constraints

The 3U common bus approach addresses the schedule-driven compromises that force users to skip critical testing such as EMI/EMC. Since the bus subsystems are already qualified and tested, development schedules can allocate adequate time for thorough payload integration testing and end-to-end mission verification without time needed for custom bus development and qualification. Users can also focus on payload to bus integration and testing activated to verify the functionally of the entire CubeSat system.

4.3.3 Answering Research Question 1

The research conducted to answer Research Question 1 demonstrates that CubeSat mission failures are predominantly caused by custom bus development and inadequate implementation of systems engineering methodology of integration testing of COTS subsystem combinations.

4.3.3.1 Most Common Failure Modes

- Electrical Power System failures (40%+ of all failures after 30 days).
- Communication System failures (30% of failures after 90 days).
- On-Board Computer failures (15-20% of all failures).

4.3.3.2 Root Cause and Solution

These failures result from custom integration of COTS components without adequate testing, interface verification, or flight heritage. A standardized common CubeSat bus that integrates pre-tested EPS, COM, OBC, and other subsystems directly addresses identified failure modes by:

4.3.3.2.1 Eliminating Custom Integration Risks

Pre-tested subsystem interfaces with established flight heritage.

4.3.3.2.2 Providing Comprehensive Qualification

Full environmental testing completed during bus development.

4.3.3.2.3 Simplifying Mission Development

User focus on payload integration rather than bus development.

4.3.3.2.4 Enabling Adequate Testing

Schedule resources allocated to payload-bus integration testing rather than custom bus qualification.

The common bus approach transforms CubeSat development from a custom integration challenge to a payload integration exercise, directly addressing the systems engineering deficiencies identified in the research. Users can allocate their limited schedule and resources to thorough payload integration testing and mission-specific verification, while leveraging the proven reliability of a flight-qualified 3U common bus for the three most failure-prone subsystems.

This analysis demonstrates that a standardized common CubeSat bus is not simply a convenience improvement, but rather the fundamental solution required to address the primary causes of CubeSat mission failures identified through this comprehensive research.

CHAPTER 5: BENEFITS OF A CUBESAT COMMON BUS AS EVALUATED THROUGH A SYSTEMS ENGINEERING MODEL AND SCHEDULE

This chapter compares a conventional CubeSat development approach, in which the program develops both the bus and payload, with an alternative approach using the proposed 3U common bus. The objective of this research is to quantify schedule reductions achieved through use of the 3U common bus and to identify the sources of schedule improvement across the systems engineering product development lifecycle. The CubeSat 101 reference development schedule is used as the baseline to enable direct comparison with the proposed 3U common bus schedule (CubeSat 101, 2017). Key schedule milestones are examined to assess how their timing, sequencing, and critical path are affected by adoption of a standardized CubeSat common bus architecture.

5.1 Research Question 2

Can performance risk be reduced, and schedule margin increased by separating the CubeSat bus engineering from the payload engineering?

To address this question, a baseline Cubesat development schedule model is developed and subsequently modified to represent the costs and benefits associated with adoption of a 3U common CubeSat bus. The resulting schedule model is then used to evaluate potential risk-reduction opportunities and systems engineering improvements enabled by the proposed 3U common bus architecture.

5.2 Methods

To develop the baseline schedule for a typical CubeSat, CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers is reviewed and used to create a representative design and build schedule (CubeSat 101, 2017). Systems engineering milestones are then mapped to this schedule using the lifecycle framework and review definitions provided in the NASA Systems Engineering Handbook, Rev. 2 (NASA Systems Engineering Handbook, 2016) including SRR, PDR, and CDR, are incorporated into the baseline schedule. The baseline schedule captures the full CubeSat design, fabrication, and test activities for both the CubeSat bus and payload. A Gantt chart based on this schedule is created and used as the schedule baseline.

The 3U common bus schedule is derived from the baseline schedule and modified to reflect the use of a pre-designed and pre-fabricated 3U common bus. Modifications to the baseline CubeSat schedule include removal of CubeSat bus design and fabrication activities. Systems engineering milestones defined in the NASA Systems Engineering Handbook, Rev. 2 (NASA Systems Engineering Handbook, 2016), including SRR, PDR, and CDR, are retained. The resulting 3U common bus schedule focuses on payload design and fabrication, as well as 3U common bus to 3U payload integration and test activities.

5.3 Results

5.3.1 Baseline Engineering and Schedule Model

5.3.1.1 Baseline Schedule: NASA CubeSat 101

The NASA CubeSat 101 and CubeSat launch initiative provides a recommended baseline schedule which is structured around a sequential development process, proceeding from concept through operational mission execution (CubeSat 101, 2017). This section will focus only on pre-phase A to phase D phases of that development process.

5.3.1.2 Pre-Phase A: Concept Studies

According to the NASA Systems Engineering Handbook (2016), the goal of Pre-Phase A is to produce a range of mission concepts from which future projects can be selected. For spacecraft this phase corresponds to the initial conceptualization and feasibility studies of the spacecraft mission and science or technology return. During Pre-Phase A, teams will generate multiple mission concepts that will fulfill the payload needs. Preliminary stakeholder analyses and needs assessments are conducted to ensure the spacecraft will meet the payloads mission objectives. These studies along with technology assessments help identify which mission concepts are technically achievable within the spacecraft for factor. During Pre-Phase A teams will develop a basic mission Concept of Operations (ConOps) to help identify critical technologies and associated risks and establish a rough cost and schedule (NASA Systems Engineering Handbook, 2016).

Form a CubeSat perspective CubeSat 101 frames the Pre-Phase A as having a duration of one to 6 months during which concept development takes place (CubeSat 101, 2017). The focus during this time is to define the mission purpose whether it be a scientific investigation, technology demonstration, or student training and identify potential partnerships that can increase mission feasibility and provide funding. Teams will typically align their mission objectives with the eligibility requirements of under NASA's CSLI. At this time, some early design work might take place including defining payload and bus requirements, performing rough mass and power budget analysis, and assessing regulatory issues such as Federal Communications Commission (FCC) licensing and remote sensing permissions. The goal of these activities is to assess feasibility before committing resources, and this also entails preliminary reviews to validate that the mission is achievable within the available time and budget.

5.3.1.3 Phase A: Concept and Technology Development

The NASA Systems Engineering Handbook describes that the goal of Phase A, is to develop a credible responsive mission/systems architecture that satisfies the project's expectations. Phase A begins with the development of requirements for the spacecraft and ends with a SRR where the proposed architecture and requirements are evaluated for completeness and trackability. The concept from Pre-Phase A is converted into a baseline technical approach, and the set of systems requirements are defined and validated against the mission goals. Preliminary architecture design and trades are performed, and critical technologies are matured to the appropriate TRLs. The initial project plan is laid out and covers cost, schedule, and risk are established to guide future development (NASA Systems Engineering Handbook, 2016; NASA 2024a).

For CubeSat teams, this phase typically involves securing funding from government programs such as NASA's CSLI, The National Science Foundation, the US DoD, or from private or academic sponsors. During this phase, the mission design becomes more detailed, encompassing payload requirements, bus configuration, and operational constraints. During this time teams typically prepare and submit proposal for launch opportunities through the CSLI complete with technical descriptions, management structure, and proposed budget with justifications. Partnerships with launch integrators and ground stations are also initiated at this time as is the process of regulatory compliance. An SRR is also conducted at this phase and serves as a formal checkpoint to ensure the mission's functional and performance requirements are achievable and aligned with the CubeSat capabilities (NASA Systems Engineering Handbook, 2016, L'Space Academy, 2024).

5.3.1.4 Phase B: Preliminary Design and Technology Completion

Phase B focuses on the maturing of the mission design and ensuring that the selected architecture can meet all the requirements within the project constraints. The NASA Systems Engineering Handbook emphasizes completing technology maturation, refining subsystem interfaces, and developing and implementing system-level design. During this phase the design is further defined through detailed subsystem-level design, analysis, and interface control documentation. Design dependent requirements are established, and all major subsystems undergo final design. By the end of Phase B, the design will be ready for PDR which will evaluate whether the preliminary design meets all system requirements with acceptable risk, cost, and schedule margins (NASA Systems Engineering Handbook, 2016).

For CubeSats, Phase B includes formal selection for launch under the CSLI to evaluate the launch readiness of the CubeSat (NASA CSLI Proposer's Guide, 2023). Detailed subsystem design artifacts are completed and reviewed. V&V plans are developed to ensure that testing will confirm compliance with all requirements. Regulatory licensing such as FCC experimental authorizations are typically finalized during this phase. During this time the PDR will take places and marks a critical maturity point for CubeSats. The PDR evaluates whether the preliminary design meets all system requirements with acceptable risk, cost, and schedule it will also verify that integration is feasible, and the design is ready to transition from paper to hardware (NASA, 2024c: NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017).

5.3.1.5 Phase C: Final Design and Fabrication

Phase C is the transition point from design to implementation. As defined in the NASA Systems Engineering Handbook, the objective during Phase C is to complete and document the detailed design, fabricate and procure all system components and verify each element meets its allocated requirements. During this time thermal, structural, and electromagnetic compatibility studies are performed to ensure flight readiness. Quality assurance and manufacturing control processes are established, and all hardware and software are fabricated, assembled, and tested. Phase C culminates in the CDR which confirms that the system's design maturity supports full scale fabrication and integration, and the mission can proceed confidently towards flight (NASA Systems Engineering Handbook, 2016; NASA, 2024d).

For CubeSats, Phase C consists of fabricating or purchasing all flight hardware including structures, electrical power systems, and communications as well as payload support elements. Construction or identification of ground station infrastructure and development of flight software capable of supporting mission operations also occurs. Environmental testing such as thermal vacuum testing, random vibration, and electromagnetic interference testing is conducted to verify the CubeSat will survive launch and operate on-orbit.

During Phase C the team will finalize all operational documentation, including procedures, contingency plans, and readiness checklists. The CDR ensures that subsystem integration can proceed safely, and the final design satisfies performance requirements within the CubeSat's mass, volume, and cost limitations (NASA, 2024d, NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017).

5.3.1.6 Phase D: System Assembly, Integration, and Test (AIT) and Launch

The final phase in development is Phase D and is described by NASA as assembling, integrating, verifying, and launching the spacecraft or system (NASA, 2024e). During Phase D all spacecraft hardware and software are brought together, and systems-level testing is performed to verify flight readiness this includes comprehensive V&V testing. This phase concludes with the Flight Readiness Review (FRR) confirming the spacecraft and ground systems are ready for launch and initial operation.

For CubeSat missions, this Phase D is often the most operationally intensive. Activities during Phase D includes the Mission Readiness Review which verifies that all testing, documentation, and procedures are complete and approved. The CubeSat is integrated into its deployment system and integrated with the launch vehicle. Following integration, the team will perform final functional testing with the launch service provider for vehicle integration, ground station activation, and early orbit operations. NASA's System Assembly, Integration, and Test Guidelines (2024.) emphasizes the importance of incremental testing during integration to ensure the components, subsystems, and the complete system are verified. After launch, the mission transitions into on-orbit checkout phase where early operations validate CubeSat functionality and data downlink. Lessons learned and documentation are archived to inform future missions and a process of continuous improvement (NASA, 2024f; The Aerospace Corporation, 2009).

5.3.2 Summary

The CubeSat lifecycle is defined jointly by NASA's Systems Engineering Handbook and CubeSat 101 to provide a structured and repeatable framework for small spacecraft development. The phased approach anchored by formal design reviews such as SRR, PDR, CDR, and FRR ensures that even resource-limited projects can adhere to the rigor of NASA's larger flight projects. By maintaining consistent review gates, defining clear deliverables and emphasizing risk management and verification, the process enables CubeSats to achieve mission success despite limited funding and personnel. This structured yet flexible approach has contributed to the successful development of hundreds of CubeSats through NASA's CSLI demonstrating that well developed systems engineering principles can be applied to small spacecraft (NASA Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017; NASA CSLI Proposer's Guide, 2023; NASA 2024a).

5.3.3 Strengths and Weaknesses of the CubeSat Development Process: The Impact of Dual Development

One of the defining characteristics of CubeSat development, as described in the NASA Systems Engineering Handbook and CubeSat 101: Basic Concepts and Processes for First-Time CubeSat Developers, is that the same team is often responsible for developing both the CubeSat bus and the payload. While this approach provides flexibility it often contributes significantly to the extended duration of many CubeSat projects. Unlike traditional spacecraft development programs, (where payloads are typically designed by a specialized instrument team and integrated with pre-existing or a heritage-based bus design) CubeSat missions frequently require simultaneous design, fabrication, and testing of both elements from the ground up. This parallel development, where the bus and payload/instrument evolve at the same time, introduces a level of complexity that has potential to lengthen the project life cycle. Each design iteration of the payload can necessitate an adjustment and even a redesign of the CubeSat bus requiring coordination between the subsystem engineers and the payload engineers. As a result, activities that can be developed sequentially in larger missions often become iterative and inter-dependent in CubeSat programs.

From a systems engineering perspective, this approach to CubeSat development has strengths and weaknesses. Developing the bus and payload concurrently allows for tight design integration which can lead to an optimized, compact, and exquisite designs with good mass margin. The process provides a valuable learning opportunity for students and emerging engineers by exposing them to end-to-end spacecraft design and integration rather than concentrating on subsystem work. It also enables tailored designs where the bus architecture can be customized to the payload's specific requirements (an important advantage for novel technology demonstration missions). This flexibility can yield highly efficient mission architectures especially when teams leverage testing and simulation to mature hardware and software solutions in parallel. Moreover, the close interaction between payload and bus development fosters a culture of systems thinking aligning with NASA's educational and workforce development goals under the CSLI.

However, there are weaknesses to this model which lie in its impact on schedule and resource allocation. Developing both the payload and the bus simultaneously places and enormous demand on a small team which might have limited personnel and facilities. Each subsystem's development is tightly coupled to the others so delays in payload design can ripple through the entire CubeSat development schedule. These dependencies extend testing timelines since teams must conduct repeated integration and functional tests to ensure compatibility. For educational missions, where design continuity can be disrupted by student turnover, these cycles can easily stretch a nominally 1-year effort into 3 or more years.

Another key contributor to extended development time is the CubeSat projects often lack access to a standardized or "heritage" bus that can accelerate design, integration, and testing. While some commercial providers offer CubeSat buses, many universities and small CubeSat teams cannot afford them or decide to build their own to control cost. As a result, basic and foundational subsystems must be designed and fabricated internally. This approach demands not only engineering effort but also resources for test procedure development, all of which consume schedule that would have otherwise been allocated to payload development. In larger NASA missions, the use of heritage systems allows subsystem testing and payload integration to begin earlier in the project lifecycle. In contrast CubeSat teams must often first verify the bus functionality before early stages of payload testing and integration can occur.

The parallel development of payload and bus also complicate V&V and I&T. Because the two systems are typically tested together rather than separately, failures can be difficult to locate and can require multiple rounds of subsystem-level testing to locate. Testing must often be repeated or waived after design updates, further extending the timeline. Environmental testing (especially TVAC and random vibration) can be particularly challenging when both the bus and the payload are undergoing late design updates. Access to test facilities is another limiting factor. Small programs often rely on shared or external facilities leading to scheduling bottlenecks and delays.

Despite these challenges, the parallel development model retains important benefits. It promotes innovation, allows teams to exercise full control over the design and creates a complete understanding of spacecraft systems engineering in an academic environment. However, the iterative nature of concurrent bus and payload development creates a schedule where progress depends on not just the individual subsystem maturity but on the integration readiness of the entire system.

In summary, the extended duration of CubeSat projects is not necessarily a reflection of inefficiency but rather a consequence of the normative systems engineering processes and comprehensive development scope. Building both the CubeSat and its payload demands extensive coordination, repeated testing, and continuous design iteration. While this approach produces valuable educational outcomes and fosters innovation in small-satellite engineering, it also stresses project schedules and provides higher probabilities of failure. As a result, CubeSat programs must carefully manage expectations and incorporate schedule flexibility to accommodate the additional time required for concurrent development. Future systems engineering and product development improvements such as greater adoption of modular, flight-proven CubeSat buses and standardized payload interfaces could mitigate these issues, allowing teams to retain design autonomy while significantly shortening overall mission timelines.

5.3.4 3U Common Bus Engineering and Schedule Model

Typical CubeSat development and build times when both the bus and the payload are being built in parallel and integrated into a CubeSat are shown in Figure 12. The development and build time for the CubeSat bus and payload in a traditional approach break down as follows:

- Systems engineering requirements development for the bus/payload: 6 months
- CubeSat bus design: 10 months
- CubeSat bus fabrication: 6 months
- Payload design and fabrication: average of 15 months
- Compressed integration and test/verification and validation (I&T/V&V) of bus/payload as a fully integrated system: 3 months
- Compressed environmental testing: 1 month
- Compressed functional testing of fully integrated system: 2 months

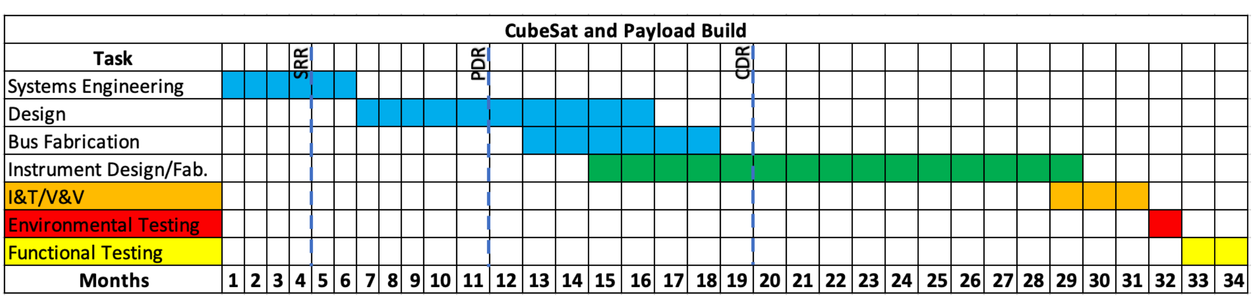

Figure 12. CubeSat bus and payload build timeline for a standard CubeSat mission.

These timelines are consistent with the CubeSat development schedules outlined in CubeSat 101 *(*CubeSat 101*,* 2017) which shows that the typical CubeSat projects require 18-24 months from concept to completion with hardware fabrication and testing taking 2-12 months depending on the mission complexity (NASA CubeSat Launch Initiative, 2017). Work by the Air Force Institute of Technology has shown that an 18-month development cycle is an achievable goal but requires careful planning and the use of a COTS CubeSat bus to both reduce risk and schedule pressure (Debes et al., 2011).

As illustrated in figure 12, the baseline development and build schedule leads to compressed I&T and V&V, environmental, and function testing phases. Only 6 of the 34 months of the project development are spend in these stages of the SE lifecycle. Compressing these activities often leaves insufficient time to fully verify the CubeSats functionality at a level required to ensure mission success. NASA Small Spacecraft Virtual Institute recommends that schedule margin should always be added when developing a schedule, particularly for critical path items like I&T which frequently overrun their allocated time (NASA Small Spacecraft Systems Virtual Institute [S3VI], 2021). In many cases, the compressed schedule has forced programs to reduce the test campaign and omit full integrated CubeSat testing such as EMI/EMC or random vibration testing of the payload prior to integration (NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017). Research on CubeSat mission failures has shown that inadequate testing is the primary contributor to on-orbit anomalies with a comprehensive study showing 70% of documented CubeSat anomalies (19 out of 27 analyzed cases) could have been prevented through more comprehensive testing (Venturini et al., 2018).

5.3.5 Schedule Benefits of 3U Common Bus Architecture

In contrast Figure 13 presents the model of payload build time for a 3U common bus CubeSat payload. This schedule estimates that using the CubeSat 3U common bus could save an estimated 4 months when compared to traditional CubeSat development approaches. In this example, schedule savings are be reallocated to activities to help ensure missions success like V&V and I&T activities required to verify and test the combined 3U common bus and 3U payload to verify mission readiness.

The proposed revised development schedule shown in figure 13 shows how using the 3U common bus approach build would take place. Since the bus is not being developed in parallel with the payload, only the payload requires a custom design and fabrication process. This eliminates the dependencies and bottlenecks that are intrinsic to the concurrent bus and payload development. Since the electrical and mechanical interfaces to the 3U common bus are pre-defined, this significantly reduces any upfront systems engineering related to powering the payload and controlling it. The 3U common bus allows the payload developers to focus on the systems engineering that applies exclusively to the payload-specific requirements, rather than addressing bus-payload integration challenges.

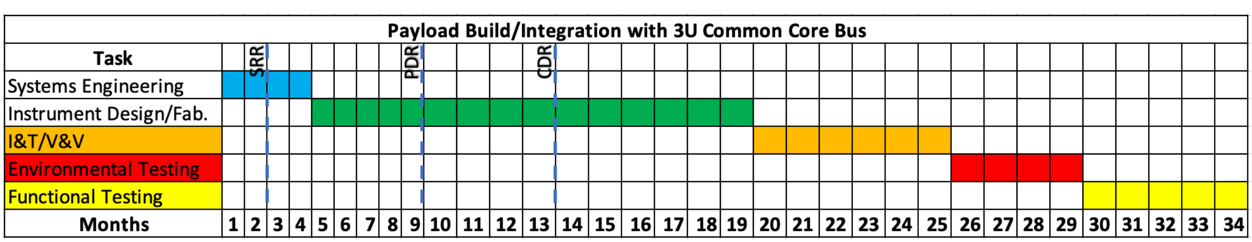

Figure 13. Payload build timeline for a CubeSat mission utilizing the 3U common bus.

The development and build time for the payload using the 3U common bus would break down as follows:

- Systems engineering requirements development for the payload: 4 months
- Payload design and fabrication: 15 months on average
- Uncompressed I&T/V&V activities: 6 months
- Uncompressed environmental testing: 4 months
- Uncompressed functional testing: 5 months
- Total development time: Approximately 34 months from start to delivery.

This research asserts that increasing the amount of time for I&T from 3 months (compressed schedule) to 6 months (uncompressed schedule) represents a major improvement in the probability of mission success as extra time will allow for more testing to verify functionality of the CubeSat. Existing literature demonstrates that CubeSat mission success is strongly tied to thorough I&T activities, and that I&T activities should be expected to take up one third to one half of the overall schedule since inadequate testing is consistently identified as a major contributor to CubeSat mission failure (Venturini et al., 2018). This study (Venturini et al., 2018) and others found that CubeSat developers universally emphasized the importance of extensive functional testing, with end-to-end day-in-the-life testing identified as the single most critical test to enable CubeSat success.

The extended environmental schedule testing allows more through and complete environmental testing. This level of testing is very difficult to achieve in a compressed schedule but becomes feasible when bus development is eliminated from the critical path. TVAC and random vibration testing have been shown to identify mission-critical issues that would not be discovered through bench top testing alone, with multiple organizations reporting that TVAC testing revealed issues that would have ended the mission but were corrected prior to launch (Venturini et al., 2018).

5.3.6 Critical Path Analysis and Schedule Trade-Offs

5.3.6.1 Critical Path Analysis

Critical path analysis the 3U common bus would fundamentally shift the critical path from the bus development to payload development and testing activities. In the traditional CubeSat approach bus design and fabrication activities consume 18 months of schedule and create dependencies that compress downstream I&T activities. By eliminating the bus development from the schedule, the critical path shifts to payload design and fabrication followed by I&T activities. Shifting the critical path provides several advantages:

- Reduced schedule risk: Payload development has fewer interdependencies than combined bus-payload development, reducing the likelihood of cascading delays.
- Enhanced test margin: The 150% increase in I&T time (from 2 to 5 months) provides substantial margin for discovering and resolving integration issues without impacting delivery dates.
- Improved mission assurance: Extended environmental and functional testing enables identification of failure modes that would remain undiscovered in compressed schedules.

5.3.6.2 Phase Bottlenecks

Traditional CubeSat development experiences bottlenecks in three primary areas:

1. Systems engineering requirements phase: Defining bus-payload interfaces when both are being developed simultaneously requires extensive iteration and often results in requirements instability late in the design phase.
2. Integration phase: Discovering bus-payload incompatibilities during integration leads to expensive redesign and schedule delays when both systems are custom developments.
3. Environmental testing phase: Compressed schedules force parallel testing of bus and payload subsystems rather than sequential qualification, reducing the effectiveness of failure isolation.

The common bus would eliminate the first two bottlenecks by providing a predefined bus with known interfaces on a flight heritage on a proven platform. The third bottleneck is resolved through an increase in time allocation for testing.

5.3.6.3 Heritage Versus Flexibility

The trade between heritage reliability and design flexibility represents a fundamental decision point for CubeSat programs and directly influences schedule risk and critical path. Custom bus development provides maximum configurability but introduces design uncertainty, interface churn, and extended verification timelines. In contrast, heritage platforms constrain flexibility but provide stable interfaces, characterized performance, and established qualification processes.

From a schedule perspective, flexibility primarily benefits payload accommodation, whereas heritage primarily benefits development predictability. Custom buses require iterative systems engineering cycles to converge payload interfaces, power allocations, and thermal architectures, frequently extending early design phases and compressing downstream I&T. A heritage platform shifts this balance by fixing bus interfaces early, enabling payload teams to design against known mechanical, electrical, thermal, specifications.

This trade determines whether the program critical path is driven by iterative co-development (custom CubeSat) or by payload design followed by I&T (3U common bus). For missions compatible with standardized interfaces, heritage platforms enable earlier design freeze and reduce schedule volatility by eliminating major sources of late-stage rework.

5.3.6.4 Heritage Benefits

A flight-proven bus provides measurable reductions in both technical and schedule risk. Heritage hardware allows programs to leverage validated designs, characterized operating margins, and lessons learned from previous missions, including failure mode identification and mitigation strategies demonstrated on orbit (Aerospace Corporation, 2018). These factors shorten qualification timelines, reduce integration uncertainty, and stabilize overall development schedules.

The proposed 3U common bus architecture enables critical path decompression by removing bus design and qualification from the schedule. This reduces development scope to payload design followed by integrated payload–3U common bus I&T and V&V on a flight-qualified platform. Rather than parallel development of the CubeSat bus and payload, the 3U common bus enables earlier payload engagement with stable interfaces, shifting schedule emphasis to payload design and system-level verification while reducing integration uncertainty. This shift expands available test margin and decouples design iteration from integration activities. For missions whose requirements fall within 3U common bus capabilities, this approach improves schedule predictability while increasing overall mission assurance.

5.3.7 Discussion and Conclusion

The analysis demonstrates that separating CubeSat payload and bus engineering provides measurable schedule benefits, particularly when using a standardized 3U common bus. In the traditional integrated development approach, tightly coupled bus and payload design activities create schedule congestion, compress the critical path, and limit the time available for V&V and I&T. In contrast, the use of a pre-qualified 3U common bus can remove approximately 16 months of bus design and fabrication from the project schedule, allowing that time to be reallocated to system-level testing and integration activities, consistent with scheduling guidance from the NASA Small Spacecraft Systems Virtual Institute (S3VI) Knowledge Base (National Aeronautics and Space Administration Small Spacecraft Systems Virtual Institute, 2021b; Venturini et al., 2018).

The proposed 3U common bus schedule reduces total program duration from 38 to 34 months while expanding I&T and V&V from 6 to 15 months this directly support higher reliability (Venturini et al., 2018). NASA and JPL schedules show that the standard CubeSat programs tend to concentrate 20% to 25% of their time in the testing phase whereas successful missions dedicate up to half of their lifecycle to I&T and V&V activities (NASA CubeSat Launch Initiative, 2017).

The broader significance of this architectural shift extends beyond schedule compression. It represents a maturation of CubeSat program into a modular, reusable paradigm parallel to larger spacecraft practices (NASA Small Spacecraft Systems Virtual Institute, 2025). By decoupling payload engineering from the spacecraft platform, CubeSat developers achieve higher mission assurance, cost predictability, and repeatable performance margins across multiple payloads. The same principle aligns with NASA's ongoing Small Satellite Platform initiatives emphasizing modularity, COTS standardization, and interface unification (NASA Small Spacecraft Systems Virtual Institute, 2025; Terran Orbital, 2022).

5.3.7.1 Addressing Research Question 2

Through comparative analysis and integration of the NASA CubeSat scheduling data, this reseaech demoonstrates that adopting a 3U common bus architecture reduces programmatic risk and inscreases shcedule margine. By eleminating bus design and fbarication activities, total project duration is sortedned by approximitely 4 months, while testing time is increased by approximately 9 monhts, enabiling the decompression of I&T and V&V phases. These trends are consistent with planning and scheudleing guiadnce from the NASA Small Spacecraft Systems Virtual Insiture (S3VI) Knowledge Base Venturini et al., 2018; National Aeronautics and Space Administration Small Spacecraft Systems Virtual Institute, 2021b).

The approach also supports risk reduction by leveraging flight-proven subsystems with known failures modes, reducing bus-level failures and isolating payload issues to functional level interfaces (Planning and management: Scheduling, NASA Small Spacecraft Virtual Institute, 2021, NASA 2021a). Mission developers benefit from a more predictable integration process and expanded testing margin which aligns with NASA recommendations to allocate at least one-third of the schedule to I&T (Venturini et al., 2018).

Therefore, the research question addressed, and the answer is yes, separating the bus and payload engineering reduces performance risk and increases schedule margin.

The standardized 3U common CubeSat bus represents an evolution in CubeSat design toward greater reliability using a heritage flight proven system while adhering to spaceflight standards (NASA Small Spacecraft Systems Virtual Institute, 2025; GlobalSpec, 2019). The goal is for future CubeSats to prioritize using a standardized bus which accepts 3U payloads to ensure higher missions success rates and predictable delivery times.

CHAPTER 6: DESIGN AND DEVELOPMENT OF A 3U COMMON BUS

In this chapter, the author addresses Research Question 3: Can a 3U common bus architecture and improved systems engineering process improve the reliability of a LEO CubeSat mission? In previous chapters, it was established that most CubeSat buses are mission-specific builds that combine varying mixes of custom and COTS subsystems for EPS, OBS, telemetry, and ADCS. While this approach can meet payload-unique needs and can, in some cases, accelerate early development, it often limits cross-mission reuse and results in inconsistent interface standards, uneven subsystem maturity, and fragmented V&V flow (NASA, Launch Services Program & The CubeSat Program, California Polytechnic State University, 2017).

This chapter defines a 3U common bus that integrates with a separate 3U payload to form a full 6U CubeSat. The intent is to standardize EPS, OBC, ADCS, and mechanical interfaces in accordance with the CubeSat Design Specification (Cal Poly CubeSat Program, 2022) while maintaining compliance with dispenser geometry and launch constraints. The design also incorporates a structural verification approach aligned with NASA GEVS to ensure the 3U common bus will survive the launch environment (GSFC-STD-7000, 2021).

To answer Research Question 3, the author compares heritage 3U and 6U CubeSat power generation, power storage, and OBC capability, derives functional interface requirements for a 3U common bus, and defines mechanical and electrical interfaces that are then validated through a V&V process. The goal is to show that a standardized 3U common bus can reduce integration complexity, improve repeatability, and increase overall mission reliability when used as the basis for a 6U CubeSat (EnduroSat, 2024; GomSpace, 2024; Space Inventor, 2024; Alén Space, 2024).

6.1 Methods

A comparative analysis of 6U CubeSat missions was conducted with a focus on power generation, power storage, and OBC resources. The missions selected represent a range of operational mission profiles and subsystem architectures, providing a reference dataset for identifying performance trends and bounding 3U common bus capabilities. The intent of this analysis is to serve as a framework for deriving a requirements set that reflects demonstrated CubeSat performance rather than assumptions or theoretical sizing. This ensures that the resulting 3U common bus is capability-driven, informed by flight-proven CubeSats, and traceable to Research Question 3.

6.2 Comparative Review of 3U and 6U CubeSat Performance

Specifications and datasheets from representative CubeSat platforms were reviewed to quantify typical subsystem performance, including power generation, battery capacity, data-handling throughput, avionics capability, mass allocation, and thermal performance for 3U and 6U CubeSats (EnduroSat, 2024; GomSpace, 2024; Space Inventor, 2024; Alén Space, 2024). This review was supplemented by general design guidance from CubeSat 101.

The analysis focused on characterizing differences in subsystem capability between the 3U and 6U form factors, not as a simple scaling exercise, but to evaluate whether a 3U common bus mechanically and electrically integrated with a distinct 3U payload module can achieve system-level performance comparable to that of a traditional 6U CubeSat. Specifically, the combined 3U bus + 3U payload configuration was assessed against typical 6U power, OBC, and environmental operating margins to identify areas where additional margin or integration support may be required.

The resulting comparative dataset provides a quantitative benchmark for evaluating the effective 6U configuration enabled by a 3U bus to 3U payload integration and informs requirements for ensuring reliable 3U common bus operation across a range of payload implementations.

6.2.1 Requirements Derivation Method

Using the values presented in Table 2, a requirements-based systems engineering approach is applied to derive system-level requirements. Functional analysis identifies mission-independent functions, including power generation, power storage, and OBC, as primary metrics of interest. Interface definition establishes MICD and EICD boundaries between the 3U common bus and the 3U payload consistent with CDS keep-out zones and deployer constraints. Performance benchmarks for power generation, power storage, and computing capability are derived from 3U and 6U heritage data (EnduroSat, 2024; GomSpace, 2024; Space Inventor, 2024; Alén Space, 2024). Requirements traceability is maintained to ensure that each requirement contributes to improved reliability, integration efficiency, or system performance (Aerospace Corporation, 2008).

Building on this derivation process, the resulting mission-independent requirements are formalized using the capability ranges and interface definitions developed in this chapter. These requirements, along with their associated compliance rationale and verification linkage, are presented in this chapter and are explicitly tied to the quantitative data and interface definitions summarized in the Chapter 6 tables.

Table 2. Typical performance specifications of 3U and 6U CubeSats.

Metric	3U typical range	6U typical range	References
Orbit-average power generation	5–10 W	8–18 W	EnduroSat; GomSpace
Battery energy storage	50–100 Wh	80–150 Wh (up to 200 Wh)	GomSpace; Alén Space
Payload power availability	≈10 W	10–20 W avg., ≤50 W peak	EnduroSat; Space Inventor
Processor type	Cortex-M3/M7 or small FPGA	Cortex-A or Zynq-class System on a Chip (SoC)	CubeSpace; Space Inventor
Data storage	≤16 GB	32–64 GB	CubeSpace; Space Inventor

6.2.2 Verification Planning

V&V has been structured into three tiers: (1) Bus qualification test performed to NASA GEVS; (2) Payload qualification testing performed by the mission team on the payload; and (3) Interface Acceptance Testing Planning (ATP) for joint bus–payload validation including continuity checks, power distribution verification, telemetry link confirmation, and safe-to-mate/inhibit logic review (CubeSat 101, 2017; Cal Poly CubeSat Program, 2022).

This framework supports a staged validation approach in which the 3U common bus is first proven independently before delivery and the payload is verified against mission-specific criteria, and the final integrated 6U CubeSat configuration is tested to ensure compatibility and functional readiness. In doing so, the V&V structure reduces integration risk, clearly assigns verification responsibility, and ensures that qualification evidence is traceable for both reuse and mission-specific certification.

6.2.3 6U Mission Evidence Dataset (2018–2025)

Ten Earth-orbiting 6U CubeSat missions were selected from a representative sample of CubeSats launched between 2018–2025. Selection was based on the availability of public design disclosures and mission documentation sufficient for extracting relevant subsystem capability data. Priority was given to missions that published or referenced power-system metrics, battery capacity, OBC, and payload resource allocation, as these elements directly inform bus-level requirement development. In cases where peer-reviewed papers identified a commercial bus provider but omitted detailed specifications, the bus manufacturer datasheet was used as a technical proxy, ensuring requirements were grounded in published, verifiable performance values (Blue Canyon Technologies, 2025; GomSpace, 2024, 2025).

This approach produced a consistent dataset for evaluating typical 6U CubeSat capability envelopes and supported requirement derivation for a modular 3U common bus intended to integrate with an additional 3U payload to form a complete 6U CubeSat.

6.2.4 Derived 3U Common Bus Requirements

The 6U mission evidence dataset presented in Table 3 helps to establish a representative design envelope for power generation, power storage, compute capability, and communications capability demonstrated by the selected CubeSat missions flown between 2018 and 2025. The mission evidence provides an empirical basis for identifying the minimum capability thresholds that a 3U common but must have in order to support a 3U payload to form a 6U CubeSat.

The 6U mission evidence dataset presented in Table 3 establishes a representative design envelope for power generation, power storage, compute capability, and communications capability demonstrated by the selected CubeSat missions flown from 2018 to 2025. The mission evidence provides an empirical basis for identifying minimum capability thresholds that a 3U common bus must have in order to support a 3U payload and form a 6U CubeSat.

Table 3. 6U CubeSat Mission Evidence Dataset (2018–2025).

#	6U Mission (Org; launch yr)	Solar panel area (m²)	Orbit Average Power Generation BOL (W)	Battery BOL (Wh)	Compute	ADCS	Reference
1	TBIRD / PTD-3 (NASA/LL; 2022)	0.40[1]	45[2]	140[2]	High-rate FPGA chain[1]	MTQ, RW[1]	1) Schieler, C., et al. (2022)., 2) NASA. (2022).
2	Starling (×4) (NASA Ames; 2023)	0.30[3]	39[3]	n/a	Xiphos Q7S + BIC[4]	MTQ, RW, CGT[5]	3) Miller, A., et al. (2024)., 4) Kashani, S., et al. (2024)., 5) NASA Ames. (2023).
3	CatSat (U. Arizona; 2024 ops)	0.36[6]	22.5[6]	77[6]	NanoMind 3200[6]	n/a	6) Chandra, S., et al. (2022).
4	Φ-Sat-1 (FSSCat/[3]Cat-5A) (ESA/UPC; 2020)	0.40[7]	50.0[7]	n/a	Myriad-2 VPU[7]	MTQ[7]	7) Pastena, M.. (2019). Φ-Sat-1 platform.
5	Φ-Sat-2 (ESA/Open Cosmos; 2024)	0.28[8]	45.0[8]	46.0[8]	CogniSat + Myriad-2[8]	MTQ[8]	8) Longépé, N., et al. (2023). Φ-Sat-2 architecture.
6	DeMi (MIT et al.; 2020)	0.20[9]	n/a	n/a	Hosted on 6U bus[9]	MTQ, RW[9]	9) Douglas, E., et al. (2021). DeMi mission.
7	TEMPEST-D (CSU/JPL 2018)	0.36[10]	n/a	80[10]	XB1 Avionics[11]	MTQ, RW[11]	10) Hirsh, M., et al. (2019). TEMPEST-D EPS., 11) Reising, S., et al. (2020). TEMPEST-D results.
8	AeroCube-16 A/B (Aerospace; 2024)	0.40[12]	12.3[12]	n/a	n/a	MTQ, RW[12]	12) Aerospace Corporation. (2024–2025). AeroCube-16 overview.
9	Cat-8 (UPC NanoSatLab; 2024/25)	0.10[13]	30[13]	86[13]	SDR-centric avionics[13]	MTQ, RW[13]	3) Contreras-Benito, P., et al. (2024). Cat-8 ADCS.
10	HyTI (UH/JPL/ESTO; 2023/24 ops)	n/s	40[14]	87[14]	Unibap iX5[14]	RW[15]	14) Nunes, J., et al. (2022). HyTI mission EPS. 15) Wright, M., et al. (2020). Unibap iX5 node.

Using the dataset as a requirements evidence base, this subsection derives a set of mission-independent bus-level requirements for the 3U common bus and documents how the design described in Chapter 6 is intended to meet those requirements. The objective is to translate observed 6U CubeSat performance into requirements baseline that can be satisfied by a standardized 3U common bus without redesign for every mission, while preserving margin and compatibility with CubeSat mechanical and launch constraints. The derived requirements are grouped by functional domain, including power, compute, thermal control, mass properties, and communications, and are payload-independent. Payload mission-specific changes (e.g., higher sustained payload power, specialized thermal stability needs, or non-baseline communications architectures) are handled at the payload level rather than through changes to the core bus design.

For each requirement set, a corresponding 3U common bus compliance statement is provided that identifies relevant design features, capability summaries, and verification activities defined elsewhere in Chapter 6. This structure links the heritage-based evidence presented in the mission dataset to the design implementation and analysis results and to the V&V approach used to demonstrate requirement closure.

6.2.5 Power Requirements and Compliance

Review of the selected 6U CubeSat heritage data shows that many missions operate in a higher orbit-average power (OAP) regime and rely on larger power storage to support mission duty cycles and payload operations. To enable a 3U common bus to host these payload classes without redesign, the bus adopts a minimum power threshold aligned with heritage capability while remaining compatible with a 3U form factor.

6.2.5.1 Baseline Derived Power Requirements.

PWR-001: Shall provide >15 W continuous power available to the payload during sunlit operations.

PWR-002: Shall provide >8 W continuous power available to the payload during eclipse for a typical LEO eclipse duration, accounting for bus loads and depth-of-discharge constraints.

PWR-003: Shall constrain sustained payload peaks to <20 W unless explicitly cleared via mission-specific thermal and power analysis and acceptance test planning.

6.2.5.1.1 3U Common Bus Compliance.

Table 4 defines the 3U common bus capability targets for power generation, power storage, and payload power availability. The bus targets approximately 22 W orbit-average power generation and approximately 132 Wh of battery storage, positioning the platform within the mid-band of common 6U capabilities while exceeding typical 3U storage. Table 4 also indicates approximately 15 W continuous payload power with bounded peak capability in sunlit conditions, consistent with PWR-001 and PWR-003. These thresholds are closed by the staged V&V approach detailed in this chapter: bench-top functional verification of power rail stability and telemetry; environmental functional re-verification across thermal-vacuum and qualification environments; and interface ATP for the bus-to-payload power path.

Table 4. 3U Common Bus Capability vs. Heritage Platforms.

	3U Common Bus	3U Typical	6U Typical	Improvement
Power gen (avg)	22 W	5–10 W	8–18 W	≈2.2–4.4× vs 3U typical (≈+120–340%) and ~1.2–2.7× vs 6U typical
Battery storage	130 Wh	50–100 Wh	80–150 Wh	≈+30–160% energy increase (in 6U range)
Payload power	15 W continuous, up to 25 W peak (sunlit)	~10 W	10–20 W	~1.5× continuous payload power (equal to low-/mid 6U capability)
Compute	Zynq-class SoC	Cortex-M	Zynq-class	High-performance OBC with real-time acceleration, SDR, autonomy

6.2.5.2 Compute Requirements and Compliance

Heritage review shows that missions increasingly depend on onboard processing for autonomy, communications management, payload data triage and compression, and robust fault handling. To ensure the 3U common bus can support these needs without redesign, the baseline

OBC must exceed microcontroller-class capability and provide a defined growth path for storage and data throughput.

6.2.5.2.1 Derived Compute Requirements

OBC-001: Shall provide an ARM-SoC-class OBC baseline (SoC-class processing capability suitable for autonomy and data handling workloads).

OBC-002: Shall provide >32 GB nonvolatile storage baseline, with an architecture that supports growth to approximately 64 GB class without redesign.

OBC-003: Shall provide reliability hooks appropriate for reuse (watchdog behavior, deterministic boot/recovery, and storage integrity mechanisms such as ECC where applicable).

DATA-001: Shall provide a standard internal command and telemetry backbone suitable for flight-like integration with link integrity verified through the staged V&V process.

6.2.5.2.2 3U Common Bus Compute Requirements Compliance

Table 4 specifies a Zynq-class SoC with 32–64 GB of nonvolatile memory, which satisfies requirements OBC-001 and OBC-002.The chapter’s bench-top functional criteria define measurable closure conditions aligned to OBC-003 and DATA-001, including stable operation under repeated reset cycles, zero unexpected watchdog resets during designated functional runs, and maintenance of nonvolatile memory integrity across repeated write/read cycles. Environmental functional re-verification in thermal-vacuum testing provides additional closure that compute and internal data interfaces maintain integrity across temperature extremes and reduced pressure conditions.

6.2.5.3 Mechanical/Electrical Requirements and Compliance

Standardization requires that payload integration be repeatable and minimally sensitive to individual interpretation. The bus-to-payload interface is therefore defined as a controlled mechanical datum scheme and a controlled electrical and data connector scheme. This enables repeatable integration and supports a consistent MICD/EICD baseline that can be carried into ATP testing.

6.2.5.3.1 Derived Interface Requirements (Baseline)

INT-001: The structure shall maintain rail-aligned geometry and meet CubeSat mechanical envelope keep-out requirements to ensure deployer compatibility and support stacked configurations.

INT-002: The payload-to-bus interface shall provide a repeatable mating interface with defined alignment features and a defined fastener set and torque specification to reduce payload integration uncertainty.

INT-003: The bus-to-payload interface shall provide a single, keyed connector for payload power and data, with defined EMI/EMC bonding provisions and documentation sufficient to control the MICD/EICD.

INT-004: The bus interface definitions shall be preserved such that a stacked 3U+3U=(6U) configuration can be assembled and tested without modification to the underlying bus interface definition.

6.2.5.3.2 3U Common Bus Mechanical/Electrical Requirements Compliance

The payload integration description defines the mechanical and electrical mating approach, and MICD/EICD key parameters, including a standardized fastener scheme and Micro-D connectors for power and data. These definitions directly support interface acceptance testing, where fit, connector integrity, and repeatable assembly are evaluated prior to the environmental campaign and integrated testing.

6.2.5.4 Structural Requirements and Compliance

In addition to electrical and data interfaces, the 3U common bus must meet a minimum structural stiffness requirement to reduce dynamic amplification risk and to support repeatable load environments across launch vehicles and deployers. Consistent with common CubeSat structural design practice, the first fundamental frequency is used as a screening metric for launch survivability and coupled dynamic response.

6.2.5.4.1 Derived Structural Requirements

STR-001: The integrated CubeSat structure shall demonstrate a first fundamental natural frequency >100 Hz under fixed-base boundary conditions at the deployer interface.

STR-002: If the first fundamental frequency is <100 Hz, the program shall coordinate with the selected launch provider to define an appropriate coupled loads/coupled dynamic analysis approach and verification plan for the specific launch environment.

6.2.5.4.2 3U Common Bus Structural Requirements Compliance

The 3U common bus structural design is intended to be sufficiently stiff to meet STR-001 at the deployer interface. Compliance is demonstrated by fixed-base modal analysis using the dispenser interface boundary condition. If early analysis indicates a first mode below 100 Hz, the design is treated as conditionally compliant pending launch-provider coordination per STR-002, with modeling assumptions and verification closure documented as part of mission tailoring and qualification planning.

6.2.5.5 Thermal Requirements and Compliance

Thermal control is a driver of CubeSat reliability. Because the 3U common bus is intended to support multiple 3U payloads, the thermal baseline addresses operational limits, survival limits, and battery charge protection. Closure is achieved through combined analysis and test, with analysis correlated to TVAC results.

6.2.5.5.1 Derived Thermal Requirements (Baseline)

THM-001 (operational): The 3U common bus shall maintain all bus components within their operating temperature limits for all nominal mission modes and environments.

THM-002 (survival): The 3U common bus shall maintain all bus components within their survival temperature limits for all non-operational and contingency environments.

THM-003 (battery charge protection): The 3U common bus shall prevent battery charging outside the battery vendor-specified charge temperature limits.

THM-004 (verification): The 3U common bus thermal design shall be verified by analysis and correlated by thermal-vacuum test (or TVAC + correlation plan) for worst-hot and worst-cold mission cases.

THM-005 (optional / tailored): The 3U common bus shall provide a payload mounting interface capable of maintaining payload baseplate temperature within [Tset +/- X C] (or within payload-specified limits) during payload operations.

6.2.5.5.2 3U Common Bus Thermal Requirements Compliance

The verification flow includes TVAC testing as an environmental verification method. THM-001 and THM-002 are closed by demonstrating, via worst-case thermal analysis, that all bus components remain within specified operating and survival limits for defined mission modes and environments. THM-003 is closed by implementing battery charge inhibit logic tied to battery temperature telemetry and verifying inhibit behavior during bench-top functional testing and during TVAC. THM-004 is closed by correlating the thermal model to TVAC test data and then re-running the correlated model for worst-hot and worst-cold cases. If a payload requires active baseplate temperature control, THM-005 is treated as a tailored interface requirement and verified by combined thermal analysis and payload level ATP.

6.2.5.6 Mass Properties Requirements and Compliance

Mass control is a primary driver of launch compatibility and deployer compliance. To preserve payload flexibility and ensure compatibility with common 6U deployers, the 3U common bus adopts a strict upper bound on dry mass at the bus level.

6.2.5.6.1 Derived Mass Requirements (Baseline)

MASS-001: The 3U common bus dry mass shall not exceed 4,500 grams, exclusive of payload mass and deployer-provided hardware.

6.2.5.6.2 3U Common Bus Mass Properties Requirements Compliance (Chapter 6 Evidence)

The 3U common bus mass budget is controlled at the system-level and allocated across structure, EPS, avionics, and harnessing to ensure compliance with MASS-001. Compliance is demonstrated through a controlled mass properties budget and confirmed via component-level mass roll-up and final system weigh in prior to interface APT.

6.2.5.7 Communications Requirements and Compliance

Communications capability is a primary mode for data return and monitoring of the CubeSats systems. To ensure the 3U common bus supports a broad class of payloads and missions without redesign, a baseline communications capability is defined with a growth path for higher data-rate missions. These requirements address both the radio/data-rate capability and the associated antenna provisions.

6.2.5.7.1 Derived Communications Requirements (Baseline)

COM-001 (baseline): The 3U common bus shall rovide S-band downlink capability of >1 Mbps for nominal mission operations.

COM-002 (high-rate option): The 3U common bus shall provide an architecture that supports an X-band downlink option of >50 Mbps, enabled through mission-specific radio and antenna tailoring without redesign of the core bus.

COM-003 (antenna baseline): The 3U common bus shall provide an integrated S-band patch antenna suitable for baseline mission operations.

COM-004 (antenna growth provision): The 3U common bus shall provide reserve mechanical envelope, mounting features, and RF keep-out area to support optional X-band or high-gain antenna (HGA) integration without structural or interface redesign.

6.2.5.7.2 3U Common Bus Communication Requirements Compliance

The 3U common bus provides baseline S-band communications suitable for >1 Mbps downlink via an integrated S-band patch antenna. Mechanical layout and interface definitions preserve reserved area and interface accommodations for an optional X-band radio and associated high-gain antenna, enabling >50 Mbps class downlink capability through mission-specific tailoring. Verification is performed through bench-top functional link testing, antenna continuity checks, and end-to-end communications checks during integrated testing.

6.2.5.8 Requirements Verification

The requirements above are intentionally written so each can be closed by at least one tier of the V&V sequence laid out in this chapter: (1) bench top functional baseline verification (power rails, watchdog and nonvolatile memory integrity, telemetry), (2) environmental functional re-verification (functionality maintained across vibration and thermal-vacuum), and (3) integration and ATP (fit, interface repeatability, and bus-to-payload electrical and data interface integrity). Table 4 provides heritage-based traceability; the compliance statements in this subsection connect that traceability to implemented 3U common bus capability and planned verification closures.

Table 5. Compliance Matrix (derived requirements to implementation and verification).

Requirement	Requirement statement (derived)	3U common bus implementation	Verification closure plan	Evidence location in
PWR-001	Shall provide >15 W continuous payload power in sunlit operations (baseline).	Target payload continuous power ~15 W; generation target ~22 W OAP; bounded peak capability in sun.	Bench top functional rail stability and functional run; interface acceptance verifies payload power path.	Interface acceptance testing section
PWR-002	Shall provide >8 W continuous payload power in eclipse for typical LEO eclipse duration.	Battery storage target ~130 Wh supports eclipse operations with margin for bus loads.	Bench top functional + TVAC functional operation; verify EPS behavior and telemetry across thermal range.	Environmental qualification planning

Requirement	Requirement statement (derived)	3U common bus implementation	Verification closure plan	Evidence location in
PWR-003	Shall sustain payload peaks <20 W unless cleared by mission-specific analysis and acceptance planning.	Peak headroom bounded; higher sustained loads handled by tailored analysis and acceptance planning.	Analysis gate + bounded load acceptance tests; document as mission-specific tailoring item.	This subsection; V&V plan
OBC-001	Shall have an ARM-SoC-class OBC baseline suitable for autonomy and data handling workloads.	Zynq-class SoC baseline.	Bench top functional compute bring-up and stability; re-verify functionality during TVAC.	Bench functional criteria
OBC-002	Shall be >32 GB nonvolatile storage baseline with growth to ~64 GB class supported.	32-64 GB nonvolatile memory baseline.	Bench top functional write/read cycling and integrity checks; re-verify after environmental exposures.	Bench functional criteria
OBC-003	Shall have reliability hooks: watchdog behavior, deterministic boot/recovery, storage integrity mechanisms (ECC where applicable).	Watchdog and recovery criteria defined; storage integrity checks planned.	Bench top functional reset and watchdog criteria; post-environmental re-verification.	Bench functional criteria; Environmental functional re-verification
DATA-001	Shall have standard internal command/telemetry backbone (e.g., CAN with I2C support) with link integrity verified.	Internal data interfaces and telemetry backbone defined for flight-like integration.	Bench top functional interface bring-up; TVAC re-verify link integrity across temperature extremes.	Bench functional criteria; TVAC plan
INT-003	Shall have a two keyed connectors one carrying payload power and the other data; EMI bonding considerations; MICD/EICD controlled.	Keyed, EMI-bonded Micro-D connector defined at bus-to-payload interface.	Interface acceptance: mate/de-mate, continuity, and functional power/data bring-up.	Payload integration; Interface acceptance testing
STR-001	Shall be > 100 Hz first fundamental frequency (fixed-base at dispenser interface).	Stiffness-driven structure sized for deployer interface; compliance via fixed-base modal analysis.	Modal analysis: if below threshold, launch-provider coupled loads/modeling verification plan.	Structural analysis results
STR-002	If first mode <100 Hz, coordinate coupled loads/modeling. Shall verify approach with launch provider.	Tailoring path defined for low-frequency configurations; modeling/correlation plan documented.	Coupled loads analysis and model correlation as required by launch provider; acceptance criteria documented.	Launch provider coordination + verification plan
THM-001	Shall maintain all bus components within operating temperature limits for nominal modes/environments.	Thermal design supports component operating limits via conduction/radiation paths and operational constraints.	Thermal analysis + TVAC correlation; functional operation in TVAC across hot/cold plateaus.	TVAC plan; thermal model

Requirement	Requirement statement (derived)	3U common bus implementation	Verification closure plan	Evidence location in
THM-002	Shall maintain all bus components within survival temperature limits for non-operational/contingency environments.	Thermal design supports survival limits via design margins and survival constraints (as applicable).	Thermal analysis + TVAC correlation; survival soak/plateau verification (as planned).	TVAC plan; thermal model
THM-003	Shall prevent battery charging outside vendor-specified charge temperature limits.	Battery charge inhibits tied to temperature telemetry; EPS logic prevents out-of-bounds charging.	Bench top functional inhibit verification + TVAC verification at boundary temperatures.	Bench functional criteria; TVAC plan
THM-004	Shall verify thermal design by analysis correlated by TVAC for worst-hot and worst-cold cases.	Thermal model developed and correlated to TVAC data; correlated model used for worst-case predictions.	Correlation plan + TVAC test; update model parameters; rerun worst-case cases.	TVAC results + correlation report
THM-005 (optional)	Shall maintain payload baseplate temperature within [Tset ± X°C] or payload limits during ops (tailored).	Optional tailored interface: baseplate conduction paths / heater provisions as required by payload.	Tailored thermal analysis + payload acceptance/TVAC as required.	Payload ICD + tailored test plan
MASS-001	3U Common Bus dry mass shall not exceed 4000 grams (exclusive of payload and deployer hardware).	System-level mass budget allocation and controlled component roll-up.	Mass properties analysis and final system weigh-in prior to interface acceptance.	Mass budget table; acceptance test records
COM-001	Shall provide S-band downlink capability ≥ 1 Mbps for nominal mission operations.	Baseline S-band radio and patch antenna integrated on 3U common bus.	Bench top functional communications testing; integrated end-to-end link check.	Capability summary; interface acceptance testing
COM-002	Shall support X-band downlink option ≥ 50 Mbps via mission-specific tailoring.	Architecture and interfaces support optional X-band radio integration.	Tailored analysis and functional testing for selected X-band configuration.	Tailored comms test plan; payload/mission ICD
COM-003	Shall provide integrated S-band patch antenna for baseline operations.	S-band patch antenna integrated into bus layout.	Antenna continuity and functional link verification.	Interface definition; acceptance testing
COM-004	Shall reserve envelope and RF keep-out area for optional X-band or high-gain antenna (HGA).	Mechanical layout reserves area and interfaces for HGA without redesign.	Inspection and interface verification; tailored antenna test as applicable.	Mechanical layout figures; interface control documentation

6.3 Verification and Environmental Test Campaign

The verification and environmental test campaign for the 3U common bus demonstrates the bus can support a wide array of 3U payloads while reducing CubeSat-level failures through a repeatable, standards-aligned verification flow. The campaign has two objectives. First, the 3U common bus must demonstrate that its mechanical, electrical, thermal, and functional interfaces withstand the launch and space environment and exhibit predictable performance with verified margins and known behavior. Second, integrating the flight-qualified 3U common bus with a user-supplied 3U payload into a 6U CubeSat provides the end user a streamlined path to verify, through acceptance testing, that the integrated 6U CubeSat is compatible, functional, and ready for flight.

Together these tests qualify the 3U common bus and confirm that it withstands the launch and space environment. A qualified 3U common bus reduces NRE, accelerates I&T flow, and provides confidence that 3U payloads can be hosted on the bus without redesign. The result is a repeatable verification process aligned with the NASA GEVS GSFC-STD-7000, the CubeSat Design Specification Rev. 14.1, and the recommendations defined in CubeSat 101 which can be applied across multiple missions using the same 3U common bus design.

6.3.1 Test Articles and Configuration

Test articles and their configurations are defined as follows:

- 3U common bus Engineering Model (EM): flight-like configuration used for qualification-level environmental testing and design margin verification.
- 3U common bus Flight Model (FM): acceptance-tested flight hardware.
- 6U Engineering Model (EM): integrated configuration (3U bus + 3U payload) for fit checks and integration risk reduction (including random vibration, TAC, and EMI/EMC as applicable).
- 6U Flight Model (FM): acceptance environmental testing plus ATP in the integrated configuration.

6.3.2 Verification Flow and Gated Test Sequence

The verification flow is executed as a gated progression in which each stage must be passed cleanly before the hardware advances to more stressful environments. Presenting the process in a structured test-chain format supports requirement traceability, establishes what risk each step mitigates, and defines measurable pass criteria so that verification does not rely on subjective assessment.

6.3.2.1 Test Readiness Review (TRR)

Purpose: Confirm test procedures, instrumentation, configuration control, and risk controls are in place.

Risk mitigated: Running tests with incorrect setup or uncalibrated measurement hardware.

Pass criteria: No open test-blocking NCRs; procedures reviewed and approved; instrumentation calibrated and logged.

6.3.2.2 Bench Top Functional Baseline: 3U Standalone Bring-Up

Purpose: Establish a functional baseline prior to environmental exposure and define the regression reference for later gates.

Detects: EPS rail instability, watchdog resets, communication failures, Non-volatile memory errors.

Pass criteria:

- Rail ripple < 50-100 mV under worst-case load
- Telemetry loss < 0.1%
- Zero unexpected watchdog resets
- 100% NV memory integrity over 10+ write/read cycles

6.3.2.3 Environmental Qualification: 3U Common Bus Engineering Model

Purpose: Verify that the design survives worst-case NASA GEVS environments and maintains performance margins.

Detects: Solder fatigue, resonance amplification, thermal drift, EMC interference.

Pass criteria:

- Random vibration: no notching > 3 dB; no structural failure; no post-test functional degradation
- Sine survey: first mode > 100 Hz; no anomalous response in low-frequency bands
- TVAC: > 6 cycles with functional testing at hot/cold plateaus and correlation-ready data collection
- EMC/EMI (as applicable): no link margin degradation > 2 dB; no mission-impacting susceptibility

6.3.2.4 Acceptance: 3U Common Bus Flight Model

Purpose: Demonstrate reliability and workmanship integrity of the actual flight hardware.

Detects: Manufacturing variability and latent workmanship defects.

Pass criteria: TVAC > 3 cycles; acceptance vibration passes cleanly; full functional regression matches the bench top baseline.

6.3.2.5 Fit and Mechanical/Electrical Integration: Integrated 3U+3U=6U Engineering Model

Purpose: Validate mechanical and electrical compatibility of the integrated 6U configuration (3U bus + 3U payload).

Detects: Harness strain issues, misalignment, insulation damage.

Pass criteria:

- Zero harness tension during mating
- 100% continuity and insulation verification
- Correct standoff heights and structural mate alignment

6.3.2.6 6U Operational Readiness Simulation (ORS)

Purpose: Validate commandability, autonomy, and safemode response in mission-like operations.

Detects: Fault deadlock, uplink path issues, downlink degradation.

Pass criteria:

- Fault Detection, Isolation, and Recovery (FDIR) responses within timing bounds
- Uplink command execution without fault
- No safemode lockout or irreversible states

6.3.3 Requirements Traceability

The derived requirements defined in this this chapter are closed through the combination of (1) bench top functional verification, (2) qualification environmental testing on the 3U EM, (3) acceptance testing on the 3U FM, and (4) integrated acceptance testing of the integrated 3U bus and 3U payload to form the 6U FM. Thermal requirements THM-001 through THM-004 are closed through worst case thermal analysis correlated to TVAC results. Mass requirements MASS-001 are closed through controlled mass roll-up and final system weigh-in prior to interface acceptance testing. Communications requirements COM-001 through COM-004 are closed through bench top link testing and end-to-end integrated communications checks, with EMI/EMC confirming margin preservation in relevant configurations.

6.3.4 Requirements Closure

THM-001 / THM-002: Method: analysis + TVAC correlation. Closed by: correlated thermal model + TVAC testing during 3U EM qualification; regression confirmation at 3U FM acceptance and 6U FM integrated TVAC testing.

THM-003: Method: inspection + functional test. Closed by: battery charge inhibit logic verified during bench top functional; exercised during TVAC hot/cold plateaus (3U EM and 3U FM).

THM-004: Method: analysis correlation. Closed by: documented thermal model and correlation of results using TVAC data from 3U EM qualification.

MASS-001: Method: analysis/inspection (mass roll-up) + measurement. Closed by: controlled mass properties budget and component roll-up; final system weigh-in prior to interface acceptance testing (3U FM and 3U payload integrated to form the 6U FM).

COM-001: Method: test. Closed by: bench top functional link testing demonstrating >1 Mbps S-band capability; end-to-end comm checks during integrated testing.

COM-002: Method: inspection + analysis/test. Closed by: design/interface evidence showing X-band growth architecture; mission-specific verification via link test when X-band is installed.

COM-003: Method: inspection + test. Closed by: inspection/continuity of integrated S-band patch antenna + bench top RF/link verification.

COM-004: Method: inspection. Closed by: mechanical layout verification of envelope/mount features/RF keep-out; confirmed during integration fit checks.

6.3.5 Outcomes

Successful completion of the test campaign demonstrates that the 3U common bus is ready for flight and provides a streamlined path from a user-supplied 3U payload to a 6U flight ready CubeSat. Integration of the payload with a verified and acceptance tested 3U common bus provides access to a CubeSat bus with well-defined interfaces, documented design margins, and established performance characteristics.

The verification and acceptance test campaign is executed first on the standalone 3U common bus and subsequently on the integrated 6U flight configuration consisting of the 3U common bus and 3U payload. This approach follows a comprehensive verification strategy aligned with NASA GEVS, CubeSat Design Specification Rev. 14.1, and CubeSat 101 recommendations. The integrated 6U acceptance test campaign includes random vibration testing, TVAC testing, and EMI/EMC testing as applicable, and powered functional testing to confirm nominal bus and payload performance under representative launch and on-orbit environments.

Completion of the environmental and functional test campaign confirms that the integrated 6U CubeSat is flight ready. This approach reduces NRE effort accelerates I&T as well as enabling future missions to adopt the common bus architecture with minimal additional analysis or tailoring. By offloading CubeSat development activities to a verified 3U common bus architecture, mission teams are able to focus engineering effort on payload design and implementation rather than on CubeSat bus and payload development.

6.4 Results

The following section presents the result of the research and details the final configuration of the 3U common bus. The results include the mechanical layout, subsystem layout, and interface definition to support the 3U payload. In addition, derived performance requirements are provided to show power generation, power storage, and compute. Together these results represent a matured 3U common bus architecture that enables integration of a diverse range of 3U payloads.

Results are structured into four major outcome areas:

1. Structural/mechanical design: including bus geometry, mass distribution, and deployable configurations.
2. Subsystem integration: avionics placement, power system, ADCS, and payload resource sharing.
3. Interface definition: establishing mechanical, electrical, power, data, and thermal interfaces required for 3U payload integration.
4. Performance summary: expected environmental margins, operating conditions, power/thermal budgets, communication bandwidth, and overall bus capability envelope.

These results define the end-state implementation of the 3U common bus and demonstrate mechanical and electrical compatibility with externally provided 3U payloads. This outcome establishes the baseline configuration for the downstream 6U integrated architecture and provides the reference specification used to execute qualification testing and verification.

6.4.1 Mechanical Design Overview

The 3U common bus was modeled in SolidWorks 2025 to comply with Cal Poly CubeSat Program, 2022 CDS Rev. 14.1 external dimensions, rails, and keep-outs (Cal Poly CubeSat Program, 2022). The internal layout prioritizes balanced mass distribution, modularity, and assembly, enabling a stand-alone 3U common bus to be integrated with a 3U payload to form a 6U CubeSat.

6.4.2 Purpose and Scope

The following section will cover the mechanical configuration of the 3U common bus. The section will focus on the physical configuration on CubeSat configuration, including external interfaces, internal packaging, and mass properties. A strong focus is places on how the CubeSat is mechanically configured, integrated, and constrained rather than structural performance. Structural behavior, including modal and random vibration analysis is addresses separately in section 6.4.18 where results are presented in detail.

6.4.3 Overall Mechanical and System-Level Architecture

The 3U common bus is a standard 3U CubeSat form factor and complies with the external volume envelope, rail geometry, and keep-out zones as defined in the CubeSat Design Specification Rev. 14.1. The external geometry consists of four longitudinal rails that interface directly with the deployer. The rails are the primary structural interface between the CubeSat structure and the deployment system. Launch loads are carried through the rails and associated frame structures, providing a predictable load path from the deployer into the CubeSat structure. Internal avionics, batteries, and support systems are mechanically supported; however, they are treated as secondary structure and are not relied upon for transmitting primary launch loads. This division of primary and secondary structure reduces structural coupling between subsystems and simplifies mechanical verification.

The primary structure and rails are fabricated from aluminum-6061, which is commonly used in CubeSat designs and is selected for favorable structural properties, manufacturability, and compatibility with deployer interfaces. Rail surface finishes are chosen to meet typical deployer rail contact requirements without specialized coatings. The 3U common bus mechanical architecture is intended to serve as the bus section of a larger 6U CubeSat when integrated with a 3U payload. Subsystems are mechanically interchangeable within the common bus framework, enabling mission configuration flexibility while preserving the same primary structural design.

6.4.4 Internal Configuration

The internal configuration of the 3U common bus uses a vertically stacked avionics architecture integrated with a rail-based primary structure. Figure 14 shows the external interface plane and the internal arrangement of avionics and batteries in the launch configuration. Internal subsystems are mounted as non-load-bearing secondary structure and are mechanically supported by the primary frame without participating in primary load transfer. Placement is driven by mass distribution, center-of-mass (COM) control, harness routing, thermal considerations, and accessibility for integration and test. External interfaces for power and data are routed to panel-mounted Micro-D connectors located at the interface plane, enabling stand-alone bus testing and integration with a 3U payload.

6.4.5 Stack Architecture

The internal avionics "spine" of the 3U common bus consists of a PC/104 connector stacked architecture oriented along the CubeSat longitudinal axis, as shown in Figure 14. Avionics boards are arranged in a vertical stack aligned parallel to the deployer insertion direction, providing a compact and repeatable packaging approach.

The stack includes the OBC, EPS/battery boards, ADCS, radio, and a space weather monitor. Board-to-board spacing is defined by component height and controlled with threaded standoffs that provide clearance for inter-board connectors and harness routing. Boards are mounted using threaded standoffs and fasteners, providing positive retention under launch vibration while maintaining ease of assembly and disassembly. Standoff alignment features constrain board position and maintain connector alignment throughout I&T.

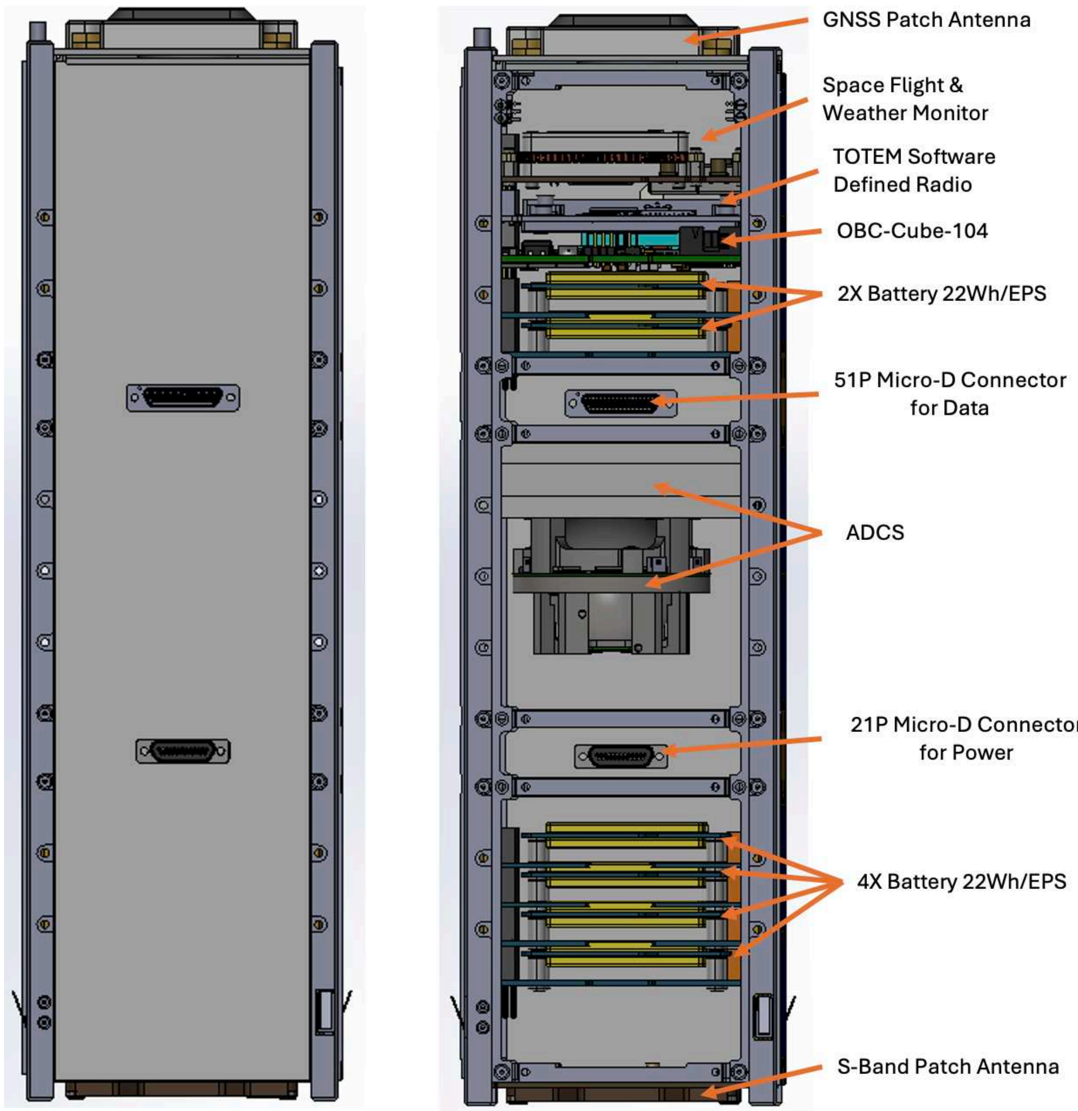

Figure 14. External and internal configuration of the 3U common bus, illustrating the rail-based structure and PC/104-style internal avionics configuration.

6.4.6 Subsystems Layout and Rationale

Subsystems within the 3U common bus are arranged in a consistent top-to-bottom configuration referenced from the deployer exit plane, as shown in Figure 14. This ordering supports balanced mass distribution, COM control, efficient harness routing, and accessibility for I&T activities. All avionics boards and internal components are mechanically mounted using the PC/104-style stack and function as non-load-bearing secondary structure.

The EPS is implemented through 6 battery boards with integrated EPS functionality, rather than a single centralized EPS board, to increase reliability. 2 battery/EPS boards are located toward the forward (deployer exit) end of the CubeSat, while 4 battery/EPS boards are located toward the aft end, positioned directly below the ADCS. Each battery/EPS board has an approximate mass of 210 grams, giving the total battery/EPS mass of approximately 1,026 grams. Because battery/EPS mass is not symmetrically distributed, a forward–aft split was selected to maintain an acceptable COM while accommodating avionics packaging constraints and preserving access to the ADCS. Locating 4 battery/EPS boards in the aft section reduces local congestion in the forward avionics bay and enables direct routing to the deployable and body-mounted solar array interfaces. Harness routing is coordinated to minimize unsupported spans and voltage drop in high-current paths while maintaining separation from sensitive signal and RF harnesses. Distributing battery/EPS boards across 2 regions also reduces local thermal dissipation buildup.

The ADCS is positioned near the geometric center of the CubeSat, immediately above the aft battery/EPS modules. The ADCS has an approximate mass of 495 grams and is mounted as non-load-bearing secondary structure using the same standoff-based architecture as the rest of the stack. Central placement reduces sensitivity to the aft-biased battery mass distribution and supports stable mass properties for attitude control. This location also facilitates alignment of inertial sensors and actuators with the CubeSat body axes and simplifies harness routing to body-mounted sensors while maintaining clearance from deployable mechanisms.

Across all subsystems, avionics boards and battery/EPS modules are mounted as secondary structure and do not contribute to primary load paths. Harness routing is designed to minimize length where it directly affects performance, avoid sharp bend radii, limit unsupported spans, and provide strain relief at connectors. Thermal dissipation is addressed primarily through subsystem placement and spacing rather than reliance on dedicated thermal straps. The resulting configuration balances power generation capability, mass distribution, thermal management, and integration efficiency while preserving the modular common-bus architecture.

6.4.7 Launch Configuration

Launch is treated as a distinct mechanical configuration of the 3U common bus. In this configuration, the CubeSat is fully stowed within the 3U envelope, with all deployable elements restrained and contained within the rail-defined volume, as shown in Figure 15. This configuration represents both the launch configuration and the configuration used for random vibration testing.

In the launch configuration, the solar panels are restrained in the stowed position using positive mechanical retention features integrated into the panel mounting hardware. The restraint method prevents relative motion of the panels under launch vibration and ensures they do not contribute unintended dynamic response during testing or launch.

In the launch configuration, all deployable systems are mechanically secured, launch loads are carried exclusively through the rails, and primary load paths rely solely on mechanically fastened structure. The launch configuration is mechanically conservative and suitable for random vibration testing and flight.

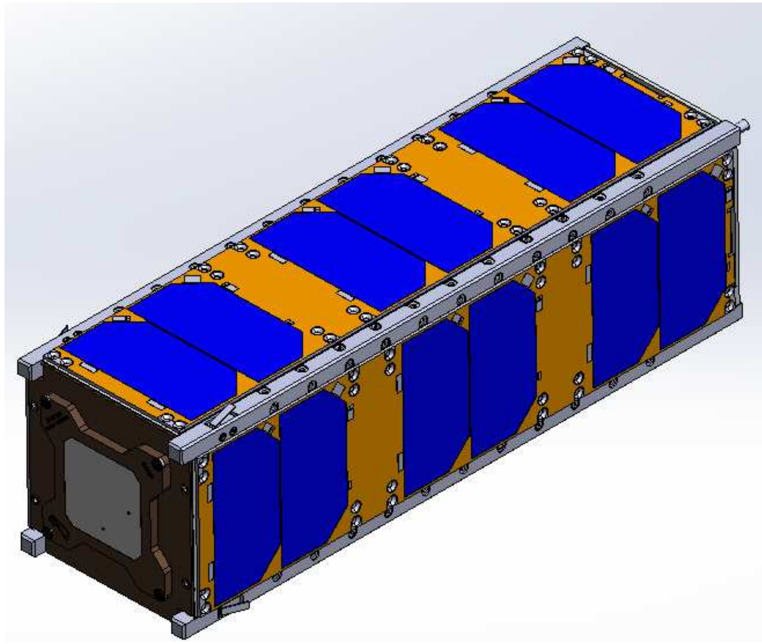

Figure 15. 3U common bus in the fully stowed launch configuration used for random vibration testing.

6.4.8 On-Orbit Configuration

Following deployment from the launch vehicle, the 3U common bus transitions to its on-orbit configuration. In this configuration, the solar arrays are fully deployed from the stowed configuration, as shown in Figure 16. The deployed array geometry represents the nominal on-orbit configuration of the 3U common bus when operated without a 3U payload. This configuration is used for TVAC and EMI/EMC testing of the stand-alone 3U common bus.

Antennas are released from launch restraints and deploy into operational positions to support communications. Antenna deployment is designed to maintain clearance from deployed solar arrays and other external features.

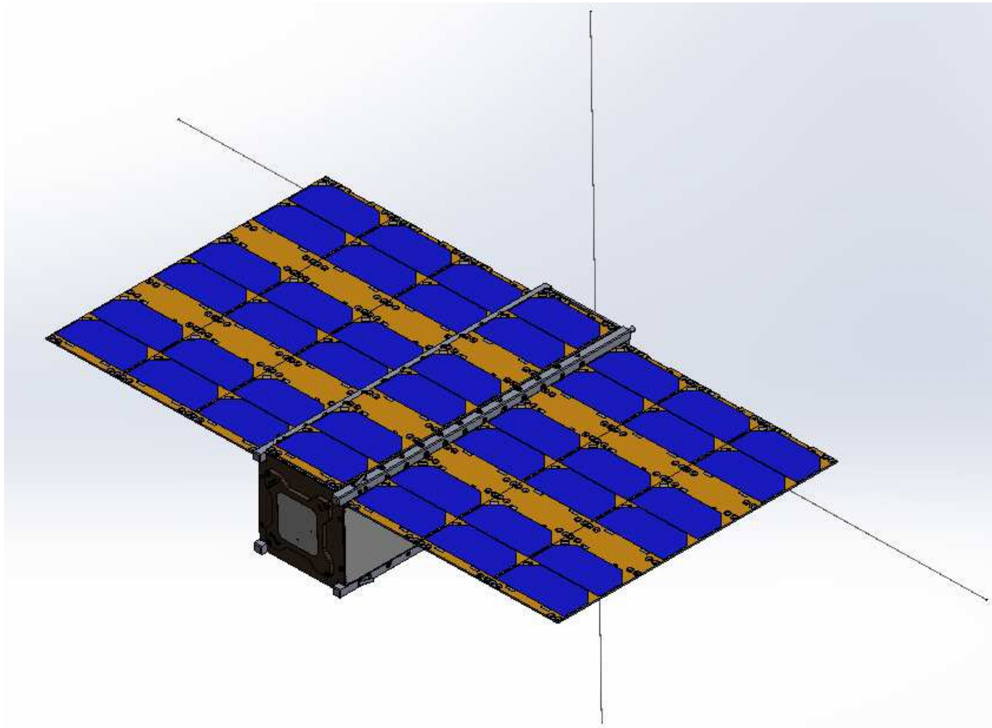

Figure 16. On-orbit configuration of the 3U common bus without an attached 3U payload, showing deployed solar array geometry representative of bus-only operation and testing.

6.4.9 Mass Properties

A subsystem-level mass budget is maintained for the 3U common bus to verify compliance with deployer mass limits while maintaining margin for late-stage mass growth. Masses are assigned based on vendor data and prior CubeSat heritage design information. Mass growth allowances are applied at the subsystem-level, with higher-uncertainty items assigned larger margins consistent with common mass management practice.

The mass budget is tracked as a dataset and summarized by subsystem to enable verification of total CubeSat mass and margin. Subsystems with mechanically complex or harness-intensive implementations are assigned higher growth allowance, while lower-uncertainty structural components are assigned smaller margins. The resulting total mass of the 3U common bus, including applied margins, remains within the allowable deployer mass limit, demonstrating compliance with 3U launch constraints.

6.4.10 Mass Budget

The total estimated mass of the 3U common bus prior to margin is 3,867 grams shown in table 5. After application of subsystem-level Mass Growth Allowances (MGA), the total mass with margin is 4,217.3 grams, which remains below the 4,500 grams mass limit for a 3U CubeSat. The mass budget demonstrates compliance with deployer constraints while preserving margin to accommodate late-stage design changes.

Table 6. Subsystem-level mass summary for the 3U common bus, including applied mass growth allowances.

Subsystem	Estimated Mass (g)	MGA (%)	Magin (g)	Mass w/ Margin (g)	Notes	Reference
3U Structure	320.00	5%	16.00	336.00	aluminum frame	COTS 3U frame (Al 6061/7075); low uncertainty.
Solar Panels	1,050.00	10%	105.00	1,155.00	body mounted + deployable	Larger multi-panel wings sized for higher OAP to support 6U loads + bigger battery.
Battery 6X	1,260.00	5%	63.00	1,323.00	larger, higher Wh	Higher-capacity Li-ion pack (22Wh ea. integrated EPS) 132Wh total, enabled by larger deployable array.
Space Weather Monitor	128.00	5%	6.40	134.40	Monitors space environment	
OBC / CDH	75.00	5%	3.75	78.75		Single-board OBC; small PCB-level mass variation.
ADCS	495.00	15%	74.25	569.25	Includes sensors & reaction wheels, torque rods	Sized to control full 6U inertia + big wings (larger wheels, robust mounts, star tracker).
Radio + Antennas	259.00	10%	25.90	284.90		GNSS + UHF/VHF + S-band/X-band + antennas and RF cabling
Harness / Fasteners / Misc.	280.00	20%	56.00	336.00	Slightly increased harness for larger array + higher-power system; highly growth-prone.	
Total 3U Bus Mass	**3,867.00**	**-**	**-**	**4,217.30**	**≤ 4,500g for 3U**	

6.4.11 Center of Mass

The center of mass (COM) of the 3U common bus is controlled through subsystem placement, as summarized in Table 5. The COM is maintained near the geometric center of the 3U envelope in both the longitudinal and lateral directions to support stable deployer release and minimize tip-off rates.

CubeSat layout decisions are driven by COM control considerations. Higher mass subsystems are located near the longitudinal midpoint and close to the CubeSat centerline, while lower mass subsystems are distributed symmetrically about the centerline to avoid localized offsets. This arrangement reduces sensitivity to mass growth in higher-uncertainty subsystems and preserves COM stability as the design matures.

Compliance with deployer COM limits is verified through subsystem-level mass accounting reflected in the CubeSat mass budget.

6.4.12 Structural Analysis

The purpose of the structural analysis is to provide first order verification that the mechanical design of the 3U common bus satisfies structural requirements prior to qualification testing. The analysis evaluates the mechanical configuration described in Section 6.10, with focus on the primary structure and integrated subsystems in the fully constrained launch configuration.

Structural performance is evaluated using Finite Element Analysis (FEA) in ANSYS Mechanical Workbench. Modal analysis is used to determine natural frequencies and mode shapes, while random vibration analysis is used to assess structural response under launch vibration loading. The launch environment is represented by the NASA GEVS PSD curve (GSFC-STD-7000, 2021). Use of the NASA GEVS PSD is appropriate when a specific launch vehicle has not been selected and launch vehicle-specific environments are unavailable. The CubeSat is required to demonstrate a first natural frequency of at least 100 Hz to reduce coupling risk with the deployer and to reduce dynamic interaction with the launch vehicle during ascent.

The analysis is subject to limitations. Linear elastic material behavior is assumed, and only the launch configuration is evaluated. On-orbit structural behavior, post-deployment configurations, nonlinear effects, detailed fastener behavior, and deployable mechanism dynamics are outside the scope of this study. This intentionally limited scope supports a focused assessment of launch survivability aligned with the objectives of this work.

6.4.13 Finite Element Model Description

The FEA model represents the 3U common bus in the fully stowed launch configuration and includes the rail-based primary structure and internal avionics components. The primary structure is modeled to capture global stiffness and load transfer through the rails, while internal avionics boards and components are represented as simplified solid bodies to capture mass.

All modeled components are assigned aluminum-6061 material properties for this first pass structural analysis. A uniform material model simplifies the analysis and is appropriate for early verification focused on global behavior rather than local stress concentrations.

Interfaces between components are modeled as bonded contacts; discrete fastener elements are not included. Bonded contacts represent mechanically rigid interfaces and prevent relative motion, providing a conservative stiffness representation for modal and random vibration assessment.

Avionics mass properties are represented by adjusting the effective density of simplified solid bodies to match subsystem mass estimates. This approach preserves inertial properties and overall mass distribution while maintaining computational efficiency.

6.4.14 *Boundary Conditions and Load Case*

Boundary conditions and load cases are defined to represent the mechanical interface between the CubeSat and the deployer in the launch configuration. Constraints are applied consistent with the rail-based structural load path**.**

6.4.15 *Boundary Conditions*

As shown in Figure 17, one rail end was assigned a fully constrained condition (A) to arrest rigid-body motion. The remaining rail ends (B and C) are constrained in a single translational direction only, allowing deformation in the unconstrained directions. This constraint scheme permits realistic global bending and torsional response while eliminating non-physical rigid-body modes.

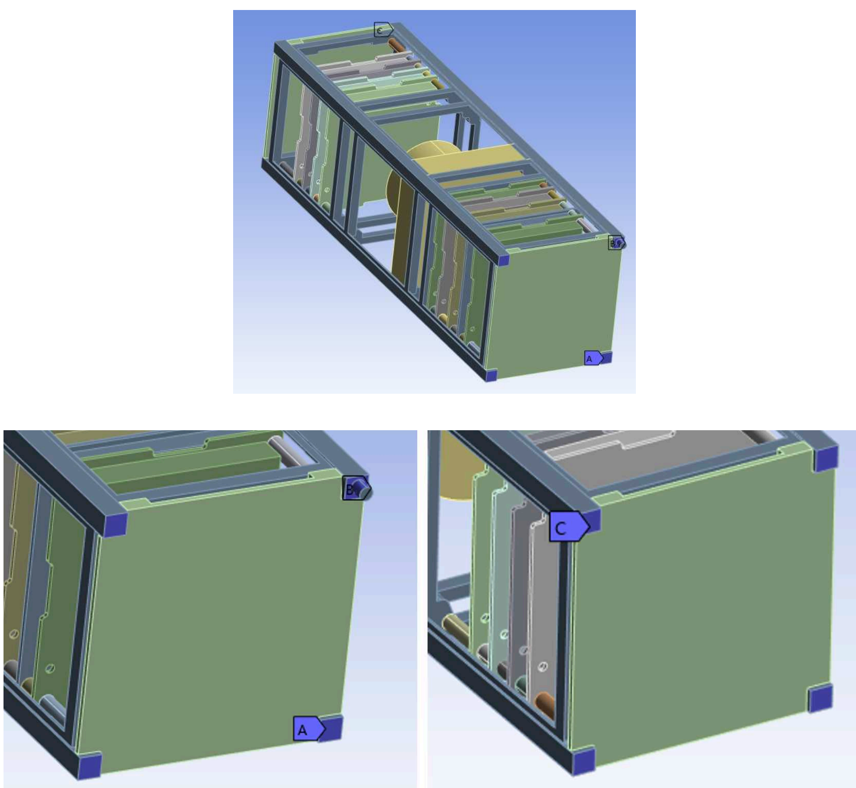

Figure 17. Finite element model boundary conditions applied at rail ends. One rail end is fully constrained to prevent rigid-body motion (A), while remaining rail ends are constrained in a single translational direction (B, C) to allow conservative global structural response.

Constraining only the ends of the rails represents a conservative boundary condition. In flight hardware the deployer provide distributed support along their length, modeling support only at the ends reduces artificial stiffness and ensures that structural response is not underestimated. All deployable elements are modeled in the fully stowed configuration and are bonded to the primary structure to prevent relative motion during launch.

6.4.16 Applied Load Cases

Two structural analyses were performed to evaluate the behavior of the 3U common bus in the launch configuration: modal analysis to characterize the natural frequencies and mode shapes, and random vibration analysis representative of the launch environment.

For the modal analysis no external loads are applied to the model. The analysis is based only on the mass and stiffness properties of the CubeSat model with boundary conditions applied at the rail interfaces as described in the previous section. Natural frequencies and mode shapes are determined from the system dynamic model. This approach is appropriate for modal characterization because the objective is to identify the inherent dynamic characteristics of the structure rather than its response to applied forces.

Random vibration response is evaluated using the NASA GEVS random vibration PSD. In the absence of a selected launch vehicle, the GEVS PSD curve for payloads with mass less than 22.7 kg is used as a representative launch environment, as shown in Figure 18. This curve provides a conservative, widely accepted proxy when launch-vehicle-specific environments are unavailable.

Frequency (Hz)	ASD Level (g^2/Hz)	
	Qualification	Acceptance
20	0.026	0.013
20-50	+6 dB/oct	+6 dB/oct
50-800	0.16	0.08
800-2000	-6 dB/oct	-6 dB/oct
2000	0.026	0.013
Overall	14.1 G_{rms}	10.0 G_{rms}

Figure 18. NASA General Random Vibration Test Levels for payload 22.7 kg or less. *Source: GSFC-STD-7000B (NASA Goddard Space Flight Center, 2021).*

For random vibration response prediction, structural damping is modeled using uniform modal damping of 2% ($\zeta = 0.02$) across the frequency range of interest. NASA GEVS recommends that damping values be selected based on test measurements or prior experience however test-derived damping data were not available for the CubeSat configuration analyzed in this study (GSFC-STD-7000, 2021). A damping ratio of 2% was therefore selected as an experience-based value consistent with common spacecraft structural analysis practice. Published aerospace structural dynamics references report typical damping ratios in the range of approximately 1–3% for lightweight metallic structures when experimental correlation data are unavailable (Blevins, 2001; Craig & Kurdila, 2006).

6.4.17 Mesh and Model Fidelity

The FEA model employs a three-dimensional solid mesh applied uniformly across the model structure and internal components shown in Figure 19. A single global element size is used and provides sufficient resolution to capture overall stiffness, mass distribution, and global dynamic behavior of the CubeSat in the launch configuration.

No local mesh refinement was applied to the CubeSat. The global mesh adequately resolves the rails, frame, and avionics components elements to represent their contribution to the load transfer and overall structural response.

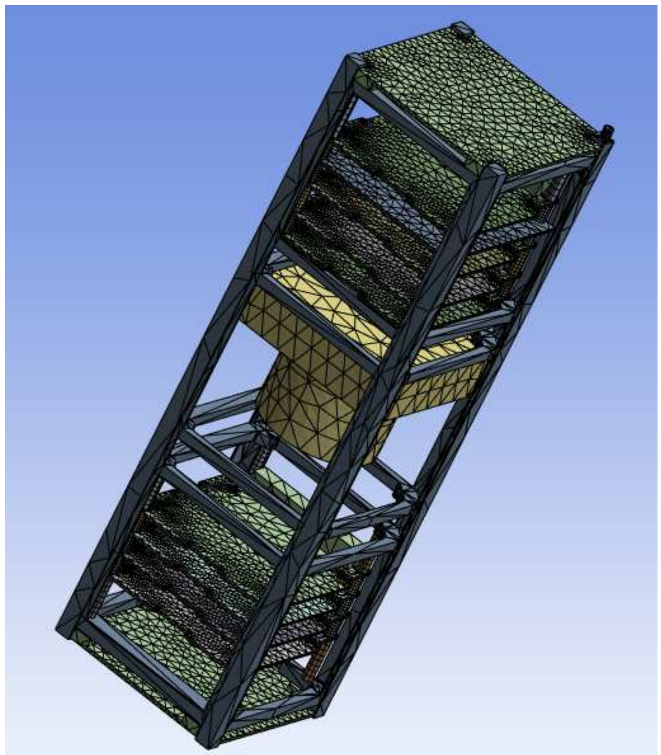

Figure 19. Finite element mesh of the 3U common bus showing global mesh density used for first-pass structural analysis.

Mesh convergence studies are not performed for this analysis. The selected mesh density is considered sufficient for modal characterization and random vibration response assessment, as the analysis objective is to evaluate global structural behavior rather than localized stress concentrations. The resulting modal frequencies and mode shapes are stable and consistent with expected CubeSat structural response, indicating that further refinement would not meaningfully change first order results.

6.4.18 Modal Analysis Results

A modal analysis was performed to determine the natural frequencies and corresponding mode shapes of the 3U common bus in the launch configuration. The first four modes are summarized in Table 7 and represent the dominant dynamic behavior of the structure.

Table 7. Primary structural modal frequencies of the 3U common bus demonstrating compliance with the 100 Hz minimum requirement.

Mode	Frequency (Hz)	Meets Requirement
1	144.80	PASS
2	209.29	PASS
3	366.41	PASS
4	686.15	PASS

The first mode occurs at 144.8 Hz and corresponds to a global bending mode of the primary structure with maximum deformation occurring near the mid-span of the frame and minimal deformation at the rail interface planes. This mode shape indicates that launch loads are effectively carried through the external rails, consistent with the intended structural load path. The first mode shape is shown in Figure 20.

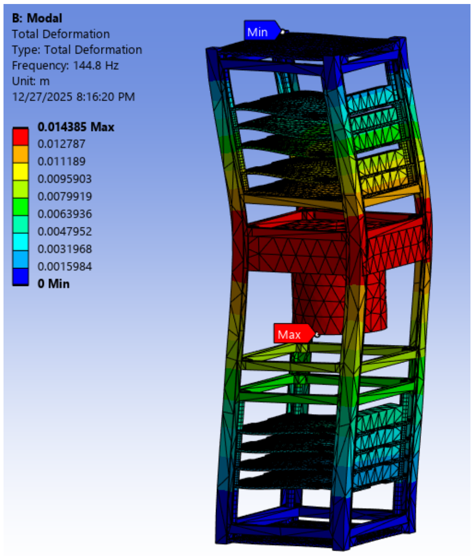

Figure 20. First natural mode shape of the 3U common bus at 144.80 Hz, showing global bending behavior in the launch configuration.

The second mode, at 209.29 Hz, represents a higher-order global bending response with increased participation of the frame members. Higher modes at 366.41 Hz and 686.15 Hz exhibit increasingly localized deformation and torsional characteristics, reflecting stiffening of the structure at higher frequencies and reduced global participation.

All identified primary modes exceed the baseline CubeSat requirement of 100 Hz for the first natural frequency, providing adequate separation from deployer and launch vehicle excitation frequencies. The results demonstrate that the mechanical design satisfies dynamic stiffness requirements for launch.

Based on the modal results, the 3U common bus demonstrates sufficient global stiffness and meets the minimum natural frequency requirement for launch. The structure is therefore considered dynamically compliant for subsequent vibration analysis and qualification testing.

6.4.19 Stress Analysis Results

The structural stress under the launch environment was evaluated using the NASA GEVS random vibration PSD curve and applied independently along the X, Y, and Z axes. Equivalent von Mises stress results were extracted for each axis to identify peak stress locations.

Stress concentrations identified in the contour plots occur at localized geometric discontinuities and constraint interfaces and are attributable to mesh singularities inherent to the bonded-contact, first-pass finite element model. These localized concentrations do not represent meaningful stress states of the model. To determine representative structural stress, the author evaluated stress levels in regions surrounding the singularities by probing adjacent elements and extracting stabilized stress values away from the singularities.

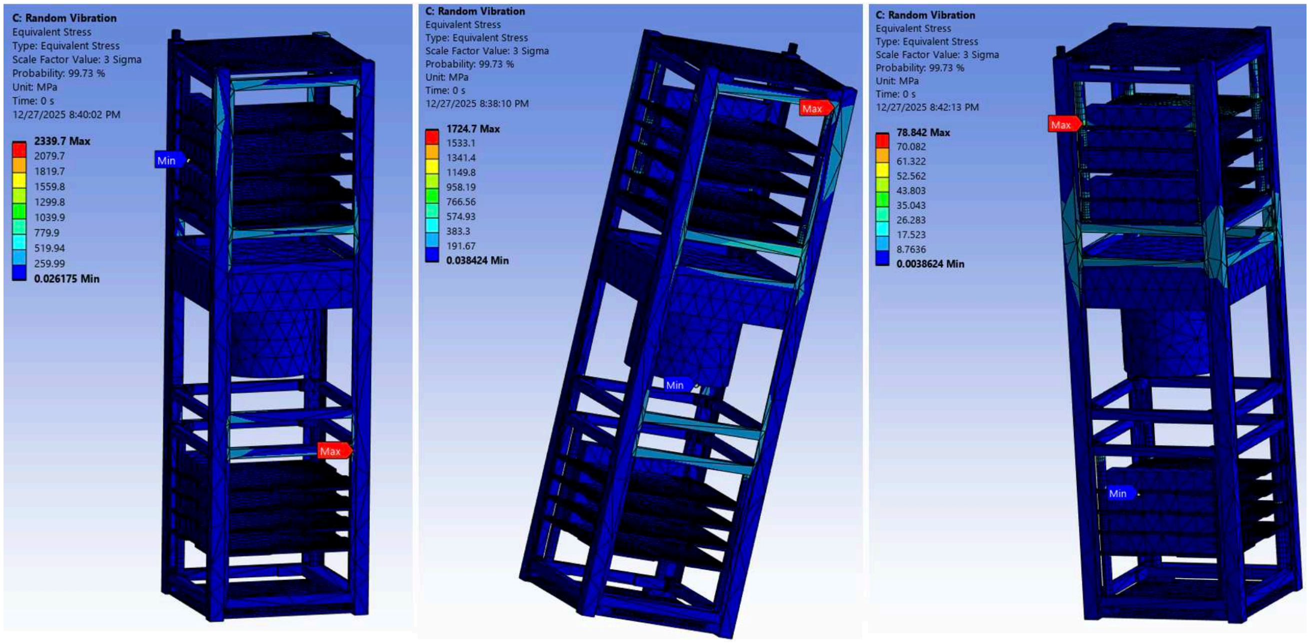

Figure 21. Random vibration equivalent (von Mises) stress results for the 3U common bus under NASA GEVS excitation. Left: X-axis input; center: Y-axis input; right: Z-axis input.

The structural analysis shows the random vibration stress response for each axis evaluated are as follows. For the X-axis the maximum von Mises stress is 122.93 MPa Figure 21. For the Y-axis the maximum stress is 146.5 MPa shown in Figure 21 which represents the governing load case for the structure. For the Z-axis the maximum stress is 17.22 MPa shown in Figure 21. These values are taken as the true structural response and exclude non-physical singularity-driven peaks.

All stresses remain below the yield strength of aluminum-6061. Using a minimum yield strength of approximately 276 MPa for aluminum 6061-T6, the resulting margins of safety are positive for all axes. The minimum margin of safety occurs under Y-axis excitation and exceeds zero, indicating no yielding under worst-case launch loads.

The stress response demonstrates that launch loads are effectively carried through the rail-based primary structure, consistent with the intended structural load path. Internal avionics and secondary structure do not experience stresses approaching material limits. The analysis confirms that the structure maintains positive margin under all evaluated random vibration load cases.

No yielding is predicted under worst-case loading conditions, and positive structural margins are maintained in all directions. Based on these results, the 3U common bus is structurally adequate for launch vibration environments and suitable for progression to qualification-level testing.

6.4.20 Summary

The structural design of the 3U common bus meets mechanical requirements for launch. Modal analysis shows the first mode exceeds the 100 Hz requirement, and higher-order modes exhibit appropriate global and localized behavior. Random vibration stress analysis shows positive margins of safety with no yielding predicted under worst-case launch environments.

The mechanical design satisfies geometry and mass constraints while preserving margin. Launch loads are carried through external rails via a predictable load path, and internal avionics function as non-load-bearing secondary structure. Based on the combined modal and stress analysis results, the 3U common bus is structurally compliant and suitable for launch. The design is therefore appropriate to progress to qualification-level testing and flight implementation.

6.5 3U Payload Integration

This section defines the mechanical and electrical interfaces provided by the 3U common bus to support integration of the 3U payload. The internal layout of the bus, established in earlier sections, is treated as a fixed configuration for the purposes of payload accommodation. Emphasis is placed on standardized mounting features and electrical interface points that enable repeatable I&T and efficient V&V during build and test.

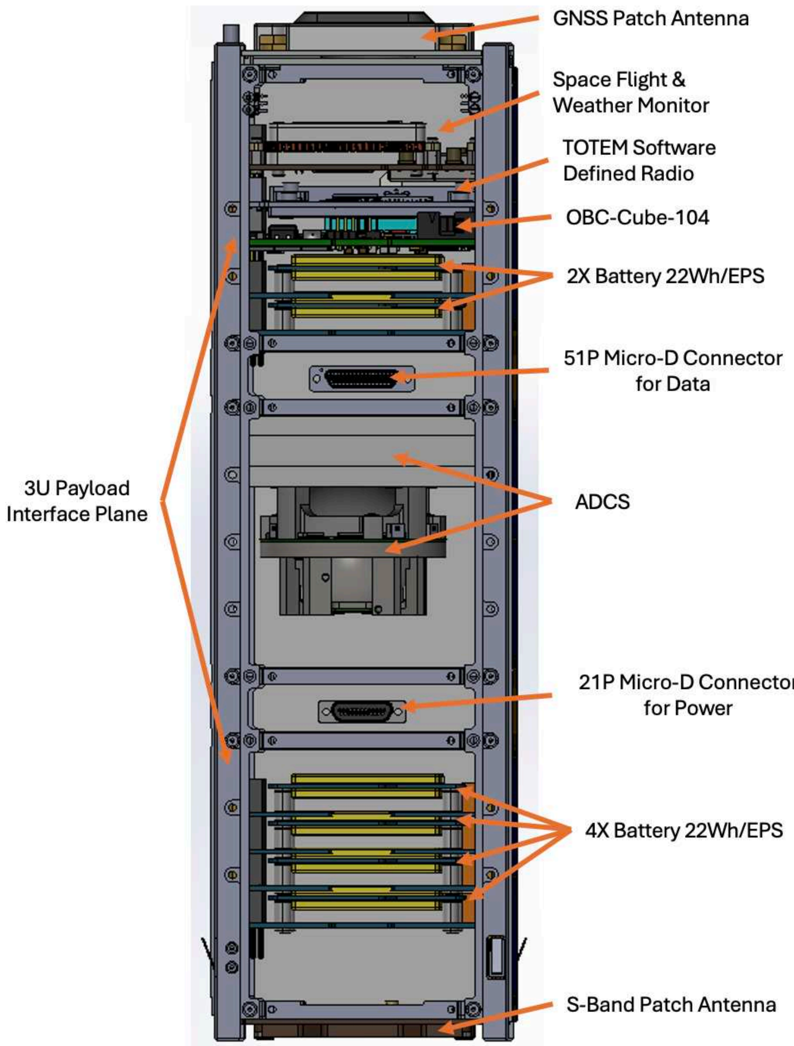

Figure 22. Internal subsystem configuration of the 3U common bus and is provided for reference to the payload interface location only; the internal layout is not redefined in this section.

The 3U common bus interfaces mechanically and electrically with the 3U payload at a defined interface plane shown in Figure 22. Mechanical attachment is achieved using rail-aligned fasteners and alignment features to ensure repeatable positioning. Electrical integration is accomplished through a 51-pin Micro-D connector for data and a 25-pin Micro-D connector for power, providing standardized electrical connectivity between the bus and payload (Glenair, n.d.).

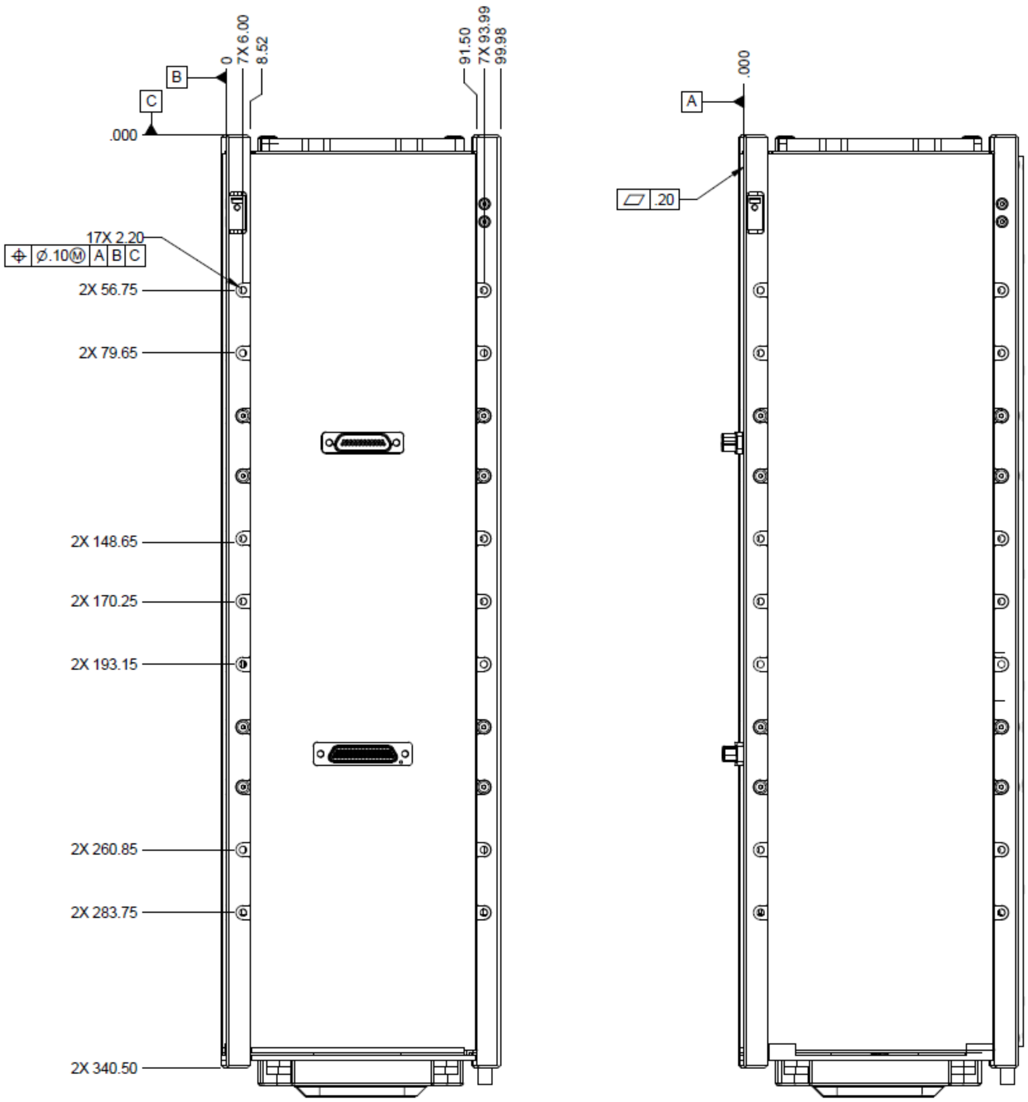

Figure 23. The MICD for the 3U common bus payload interface, showing the standardized payload interface plane, locations of the 14 M2 fasteners, and alignment pin features used to govern mechanical integration between the bus and payload.

Mechanical integration between the 3U common bus and 3U payload is controlled with an MICD. The MICD defines the standardized 3U payload interface plane and specifies mounting hole locations, fastener locations, alignment features, as well as allowable mechanical tolerances required to ensure repeatable and reliable integration. Mechanical attachment is accomplished using 14 M2 fasteners distributed around the payload interface perimeter and installed into self-clinching (PEM-type) nuts in the 3U common bus structure to provide uniform clamp load and interface stiffness. Repeatable alignment is ensured through the use of 2 alignment pins, one providing full in-plane location and the second alignment in a single axis to prevent over-constraint while maintaining positional repeatability. These interfaces establish a controlled mechanical boundary between the bus and payload independent of payload-specific internal designs. By standardizing the mechanical interface, the MICD reduces integration variability, minimizes the potential for fit-check issues, and supports consistent assembly, I&T, and V&V during build and test activities. Figure 23 shows the MICD for the 3U common bus, explicitly showing the locations of the M2 fasteners and alignment pins used to govern mechanical integration between the bus and payload.

Electrical integration between the 3U common bus and payload is governed by the EICD shown in figure 24. The EICD defines the physical locations and orientations of the payload electrical interface connectors on the bus. The payload electrical interface consists of a 51-pin Micro-D connector (J1) used for data interfaces and a 25-pin Micro-D connector (J2) used for power distribution. The EICD is provided to document connector placement keepout zones and do not define signal assignments, electrical characteristics, or internal harness routing.

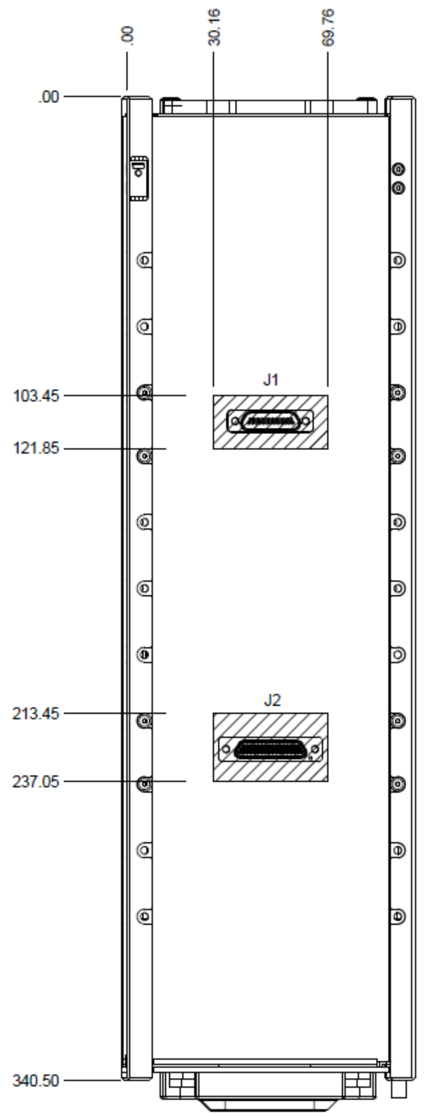

Figure 24. The EICD for the 3U common bus payload interface, showing the locations and orientations of the 51-pin Micro-D data connector (J1) and 25-pin Micro-D power connector (J2).

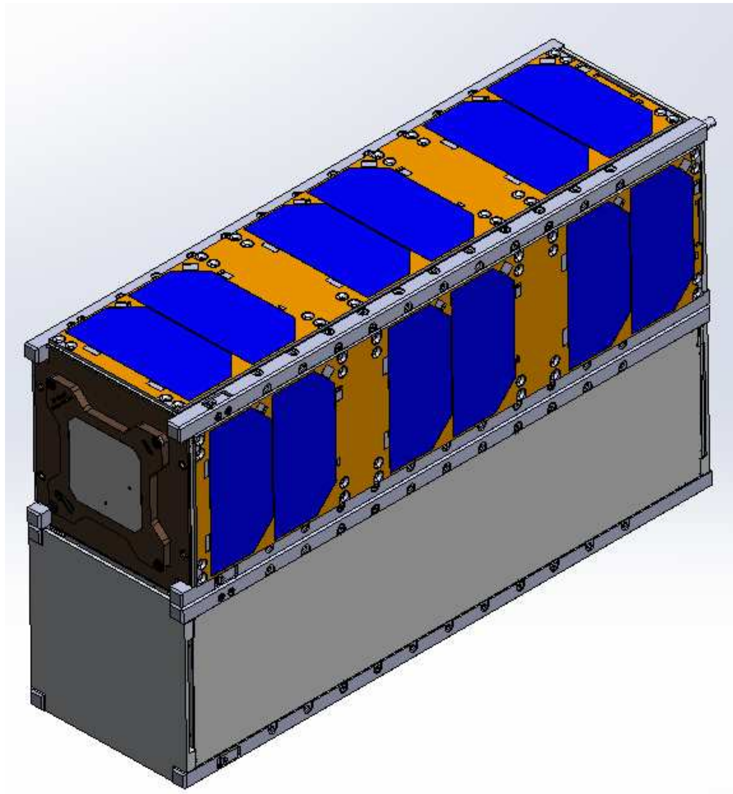

Figure 25. Integrated 6U launch configuration of the 3U common bus and a notional 3U payload. The figure depicts the integrated bus and payload in its launch configuration, illustrating the standardized mechanical and electrical interfaces controlled by the MICD and EICD that enable repeatable payload integration and deployer compatibility.

Figure 25 shows the integrated 6U CubeSat in launch configuration showing the 3U common bus mated to a notional 3U payload. Following deployment, Figure 26 shows the on-orbit configuration with antennas and solar panels deployed. Together, these figures illustrate how the standardized mechanical and electrical interfaces defined by the MICD and EICD enable direct payload integration. The MICD and EICD provide a complete, controlled payload interface for 3U payload integration by standardizing mechanical mounting and alignment features as well as data and electrical connections, while treating the internal 3U common bus configuration as a fixed architecture. These representations are intended to demonstrate system-level integration and interface compatibility and are not intended to define internal subsystem layouts or detailed harness routing. By formalizing the payload interface through controlled interface documentation, the 3U common bus supports repeatable payload integration, reduces integration risk, and strengthens downstream assembly, I&T, and system-level V&V.

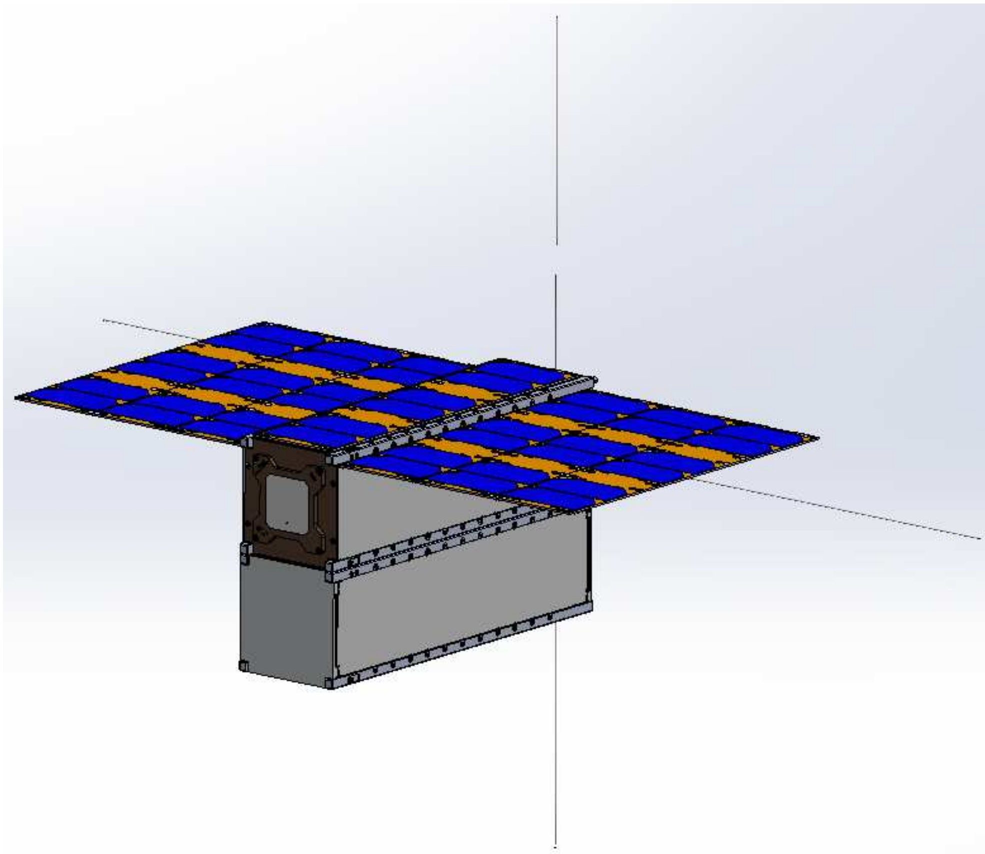

Figure 26. Integrated 6U on-orbit configuration showing the 3U common bus mated to a notional 3U payload, illustrating the resulting CubeSat configuration enabled by the standardized mechanical and electrical interfaces defined by the MICD and EICD.

6.5 Power and Compute Capability Summary

As summarized in table 8, the 3U common bus design supports capabilities consistent with and in several areas exceeding heritage 3U platforms. The bus incorporates a 6U CubeSat class deployable solar array configuration that enables an orbit-average power generation of approximately 48 W, representing a ~2–4× increase over conventional 3U CubeSats and approaching the lower end of 6U performance. Battery capacity is approximately 132 Wh, exceeding typical 3U ranges (50–100 Wh) and approaching the mid-band of 6U designs, enabling higher duty-cycle payload operations and increased eclipse margin. The bus provides approximately 15 W continuous payload power with bounded peak capability in sunlit conditions. Compute and storage utilize a Zynq-class SoC with 32–64 GB nonvolatile memory, enabling autonomy, SDR-centric missions, and on-orbit reconfiguration (EnduroSat, 2024; GomSpace, 2024; Space Inventor, 2024; Alén Space, 2024; CubeSpace, 2023).

Table 8. Comparison of 3U common bus capabilities with heritage CubeSat platforms.

	3U Common Bus	3U Typical	6U Typical	Improvement
Power Gen (avg)	48W	5–10 W	8–18 W	≈4.8× vs 3U typical (≈+120–340%) and ~1.2–2.7× vs 6U typical
Battery Storage	132 Wh	50–100 Wh	80–150 Wh	≈+65% energy increase (from baseline 6U range)
Payload Power	15 W continuous, up to 25 W peak (sunlit)	~10 W	10–20 W	~1.25× continuous payload power (equal to low-/mid 6U capability)
Compute	Zynq-class SoC	Cortex-M	Zynq-class	High-performance OBC with real-time acceleration, SDR, autonomy

6.6 CubeSat Interface Acceptance Testing

The 3U common bus meets primary mechanical objectives for mass balance and launch survivability in LEO environments (GSFC-STD-7000, 2021). The 3U common bus and 3U payload meet dimensional requirements per CubeSat Design Specification Rev. 14.1 and form a 6U CubeSat. Emphasis is placed on configuration-managed mechanical interfaces: the bus provides a standardized mounting pattern, alignment datum, and power and data connection scheme reducing fit-up uncertainty and simplifying 3U payload integration. These interfaces are fully documented through MICD/EICD and verified through an interface ATP that confirms clearance, alignment, connector engagement, and mechanical access. Together, these results establish a repeatable, configuration-controlled integration pathway suitable for future payloads and higher-volume manufacturing (Aerospace Corporation, 2008; CubeSat 101, 2017).

Table 9. Requirements traceability to 6U heritage, author–year citations.

3U "Shall/Should"	6U Evidence (author–year; page)	Rationale / Scaling
PWR-001 (≥15 W to payload in sun; ~25 W array)	XB6 arrays 92–108 W (Blue Canyon Technologies, 2025, p. 1); GomSpace 54/103 W (GomSpace, 2025, p. 2); Starling on BCT bus (Miller et al., 2024)	Half-stacked 3U with single-side deployable ~25–30 W OAP; ≥15 W to payload after bus load.
PWR-002 (≥8 W in eclipse)	Battery from 77 Wh (GomSpace, 2024, p. 1); XB6 6.8–20.4 Ah (~50–150 Wh) (Blue Canyon Technologies, 2025, p. 1)	8 W for 30–35 min eclipse ≈4–5 Wh; feasible with scaled 3U capacity and DoD.
PWR-003 (≤20 W sustained peaks unless cleared)	TBIRD ~100 W payload (Schieler et al., 2022, pp. 1–3)	3U cannot host TBIRD-class loads; cap aligns with thermal/power limits.
COM baseline (S-band ≥1 Mbps; X-band ≥50 Mbps option)	XB6 S-band 2 Mbps; X-band to 10 Mbps (Blue Canyon Technologies, 2025, p. 1); CatSat ~50 Mbps (Chandra et al., 2022; University of Arizona, 2023)	S-band for TT&C; ≥50 Mbps option justified by 6U demos.
Antennas (S-band patch; reserve area for X/HGA)	Starling S-band and CommPack (Miller et al., 2024; Kashani et al., 2024); CatSat inflatable HGA (Chandra et al., 2022)	Common 6U layout; reserves growth for high-rate antennas.
OBC class (ARM-SoC ≥~2000 DMIPS; 32+ GB NVM; ECC)	Q7S on Starling (Miller et al., 2024); Myriad-2 on Φ-Sat-2 (Longépé et al., 2023)	3U baseline with mezzanine upgrade path.
ADCS hooks (≤1° control; ≤0.5° knowledge; upgrade ≤0.1°)	XB6 fine-pointing/star-tracker options (Blue Canyon Technologies, 2025, p. 1)	Aligns with COTS 6U capability; growth for X-band/optical.

These results demonstrate that standardization, not scaling alone, can deliver 6U operational capability using a 3U common bus reducing integration variability, improving verification consistency, and strengthening overall mission reliability.

6.7 Environmental Qualification of the 6U CubeSat

Environmental qualification testing evaluates whether the integrated 6U CubeSat (3U common bus + 3U payload) meets launch vehicle and on-orbit operating environments. Testing follows NASA GEVS acceptance levels and environmental screening requirements outlined in Cal Poly CubeSat Program, 2022 CDS Rev. 14.1 (GSFC-STD-7000, 2021; Cal Poly CubeSat Program, 2022).

Random vibration testing shall be performed at acceptance levels using the NASA GEVS spectrum. The test article is integrated in a deployer-equivalent mounting interface to replicate load paths. Objectives are to (1) demonstrate the 6U CubeSat can withstand launch vibration loads and (2) verify post-test functional continuity, including connector retention and bus–payload data link stability.

Following random vibration, a full functional test shall confirm no damage, intermittent connections, or interface degradation occurred. Attention is placed on power regulation margins, OBC–payload command exchange throughput, and telemetry channel stability (Aerospace Corporation, 2008; CubeSat 101, 2017).

Thermal-vacuum cycling simulates the space thermal environment and validates stable electrical and thermal performance across operational temperature ranges. The CubeSat undergoes multiple hot/cold cycles in vacuum with thermocouples on key components, including battery cells, EPS converters, payload processor, and radio front-end. Operational checks will be conducted at thermal extremes to verify the following:

- EPS power conversion and battery charge/discharge behavior
- OBC reset stability, boot timing, and watchdog performance
- Payload command execution under thermal stress
- Link-layer integrity on CAN/I²C across temperature swing
- Thermal control of the stacked 3U+3U configuration

Completion of TVAC provides confidence the integrated 6U CubeSat can withstand the orbital environment and sustain operation on-orbit. Performance is documented through sensor-logged data, functional telemetry packets, and configuration-managed test reports contributing directly to flight readiness determination (GSFC-STD-7000, 2021).

6.8 Summary of Transition to V&V

The results in this chapter demonstrate that a systematically engineered 3U common bus grounded in heritage design and standardized interfaces can reduce integration risk and improve mission reliability. The final CAD design and derived requirements establish the hardware baseline for environmental qualification and interface acceptance testing.

6.7 Discussion and Conclusions

The results of the design, analysis, and V&V planning presented in this chapter demonstrate that a 3U common bus can support mission performance traditionally associated with 6U CubeSats. Comparative data show that by incorporating 6U-class deployable solar arrays, increased battery storage, and a Zynq-class OBC, the 3U common bus achieves performance near the lower-to-mid band of 6U CubeSats. This supports the conclusion that a 3U common bus mated to a 3U payload can deliver 6U-class operational capability without increasing mass beyond 3U deployer limits and without requiring a mission-specific bus redesign.

Mechanically, MICD and EICD documentation reduces payload integration variability by providing repeatable mounting points and standardized electrical and data interfaces. When combined with a rigorous verification and environmental test campaign, the 3U common bus supports modular stacking into a functional 6U assembly while ensuring qualification evidence exists at both the bus level and the integrated-system-level. The result is an architecture that reduces late-phase integration risk a consistent contributor to CubeSat mission failures and enables faster payload bring-up through configuration-controlled interface definitions.

Findings associated with Research Question 3 therefore support the conclusion that a reusable 3U common bus architecture can improve CubeSat reliability when paired with repeatable integration processes, robust interface documentation, and defined environmental acceptance testing. The 3U common bus does not eliminate all integration risk; however, it meaningfully shifts failure likelihood away from bus subsystems by providing known interfaces and verified performance. This enables missions to allocate engineering effort toward payload design and integration rather than bus redevelopment.

CHAPTER 7: CONCLUSIONS, CONTRIBUTIONS, AND FUTURE WORK

The objective of this dissertation was to investigate if CubeSat reliability can be improved through the application of structured systems engineering principles and the development of a flight ready 3U common bus which a 3U payload will be integrated with forming a 6U CubeSat. CubeSats enable rapid low-cost access to space, yet they remain challenged by high failure rates driven by inadequate I&T and compressed delivery schedules. Using a combination of literature review, schedule modeling, and systems-level bus design, this research evaluated three questions summarized as follows: 1) What subsystems are driving the failure rate of CubeSat missions? 2) Can schedule risk and inadequate testing be reduced by separating the bus and payload development? and 3) How does a 3U common bus improve reliability and mission success?

This chapter synthesizes the technical results presented in chapters 4, 5, and 6 and describes how these findings address the research questions and discusses the contributions this work makes to CubeSat engineering. The chapter concludes with proposed future work required to transition the 3U common bus from a design framework into flight hardware.

7.1 Research Contributions

Case studies in existing CubeSat literature acknowledge that a common architecture may offer cost development and reliability advantages. However, the systems engineering implications of common bus adoption (specifically how standardization effects failure risk, schedule compression, and V&V and I&T) remain largely unexplored. This research addresses that gap by evaluating the trades between performance, reliability, and schedule by demonstrating how a disciplined systems engineering product development framework enables improvements in the reliability of a CubeSat by utilizing a 3U common bus. This dissertation makes the following contributions to CubeSat systems engineering:

1. **A survey and review of CubeSat failure datasets and systems-level failure classification.**

   A dataset of CubeSat failure modes across missions, organizations, and platforms is compiled from published studies and other publicly available data. This dataset is used to identify recurring CubeSat subsystem failures in EPS, communications, and OBC subsystems, as well as a "unknown" failures. These outcomes are organized into a systems-level failure classification that links technical failures to upstream drivers such as schedule compression, limited environmental and functional testing of subsystems and the integrated CubeSat. This framing provides a traceable foundation for requirements allocation and risk mitigation in future CubeSat missions.

2. **A failure informed stakeholder informed set of requirements for a 3U common bus.**

   Using the failure classification, stakeholder needs, and a curated set of recent 6U CubeSat missions, the dissertation derives a set of requirements for a 3U common bus intended to mate with a 3U payload. These requirements explicitly target mitigation of the most frequent failure modes and define expectations for subsystem maturity, MICD/EICD's, and I&T. The result is a structured requirements baseline for a 3U common bus and its interfaces.

3. **A CubeSat product development schedule model that compares bespoke and common-bus development.**

   A schedule framework is developed that compares traditional CubeSat development where teams design or heavily customize their own bus to a development that reuses a pre-qualified 3U common bus. The comparison is aligned with the NASA systems engineering lifecycle and highlights where system-level reuse reduces non-recurring engineering burden and returns schedule margin to I&T and V&V activities. This schedule-centric view gives mission formulation teams a concrete way to evaluate the benefits and trade-offs of adopting a 3U common bus.

4. **A 3U common bus architecture; example implementation and test campaign for 6U-class missions.**

   The dissertation defines a 3U common architecture that balances performance and manufacturability, integrates state-of-the-art CubeSat avionics, and provides standardized MICDs and EICDs. The architecture is sized to support a representative set of 6U missions when combined with a 3U payload. A corresponding example environmental and functional test campaign is outlined to show how the bus can be pre-qualified and then reused with minimal redesign. While illustrative, this design serves as a reference architecture and systems-engineering template for organizations implementing a common-bus approach. The author is unaware of other CubeSat common bus designs that describe a 3U common bus which would be mated to a 3U payload creating a 6U CubeSat.

The 3U common bus is designed to be delivered as a flight-qualified, ready-to-use CubeSat bus with standardized, well-defined interfaces and associated I&T and V&V tailored procedures. Following V&V it will serve as a reliable baseline for mission-specific payloads, reducing infant mortality failure rates and increasing the engineering bandwidth available for science- and technology-driven experimentation missions.

Collectively, these contributions advance CubeSat design methodology beyond one-off, custom builds toward a standardized, testable, and repeatable 3U common bus, enabling future missions to scale capability while controlling schedule risk and improving reliability.

7.2 Positioning and Novelty of this Research

The Doctor of Engineering in Systems Engineering is a doctoral degree that seeks to emphasize research activities that use both "applied" and "translational" research methods. Where applied research seeks to generate new knowledge applied to the domain of the systems engineering lifecycle processes, translational research activities are concerned with the investigation and development of new knowledge in the context of the Systems Engineering Practice. The D. Eng. is a degree program that seeks to work in collaboration with industry and other SE enterprises to learn about and define industry-relevant research problems, to study how SE is practiced in industry, and to develop novel and improved practices for SE that can be directly translated into industry. In the D. Eng., the University and industry work together to develop SE expertise, thought leadership, and competitive innovation to improve the state of the practice of SE.

In that context, the research activities completed in this dissertation research effort constitute both an applied and translational research contribution. This dissertation's applied research contributions lie in integrating existing reliability data, failure trends and flight results in practice into learnings that can be applied to a practical, repeatable systems-engineering framework informed by CubeSat reliability data. First, it explicitly links published subsystem-level failure statistics and mission-assurance guidance to underlying systems-engineering and schedule drivers and then evaluates a 3U common bus architecture as one concrete intervention to address those drivers. Second, it develops a schedule-based comparison between bespoke and common bus that is aligned with NASA systems engineering lifecycle.

This dissertation's translational research contribution comes in translating these learnings about the challenges and failures of systems engineering processes as documented and exercised in the CubeSat engineering processes into requirements for a novel design and architectural contribution to the field of CubeSat design. This translational research contribution comes in the form of a design description for a 3U common bus reference architecture, supported by interface definitions and a proposed I&T and V&V test campaign. This material is provided as an open, systems-engineering-oriented template, rather than a proprietary product specification. Together these elements provide an incremental but meaningful contribution by translating the high-level concept of "using a common CubeSat bus to improve reliability" into a traceable, repeatable systems-engineering approach and artifact adaptable to different organizations and mission needs.

7.3 Future Work

Although this dissertation advances CubeSat reliability and provides a detailed blueprint for a 3U common bus reference architecture, additional technical work remains before the design can be fully validated and transitioned to a flight-qualifiable implementation. Future work includes the following activities, which will mature the 3U common bus from a reference architecture into a validated, flight-qualifiable 3U CubeSat common bus. Remaining tasks include:

Completion of hardware-level structural and thermal FEA: While preliminary stiffness targets and PSD loading conditions have been established, modal frequency, margin evaluation, and hot/cold operational behavior should be verified using detailed, flight-like geometry and representative material properties.

Fabrication and test of the 3U common bus engineering model (EM): A physical build is required to support environmental testing, power-on testing, and long-duration stability testing.

Execution of NASA GEVS-level qualification on the 3U bus flight model (FM): Random vibration, thermal-vacuum (TVAC) cycling, and EMI/EMC acceptance testing must be completed to finalize the bus as certifiable, flight-ready hardware.

Hardware-in-the-loop (HIL) fault-tolerance and autonomy testing: The system should demonstrate safemode recovery, EPS protection response, command continuity across anomalies, and watchdog resilience, including tolerance to single-event upsets (SEUs) and other anomalous conditions.

End-to-end payload integration testing using a mission-representative 3U payload: Interface acceptance testing and integrated 6U environmental qualification will validate that the architecture supports real payload geometry, load profiles, and data-handling requirements without redesign.

Long-duration functional operation testing: Multi-week testing of the stacked 3U+3U configuration will characterize reset behavior, thermal performance, battery performance, and watchdog stability over mission-representative duty cycles. Completion of these activities will demonstrate that the 3U common bus performs as modeled under environmental, electrical, and operational stress. Once completed, the 3U common bus can be treated as flight-qualified hardware capable of supporting 3U payloads.

Fly a demonstration mission using the 3U bus as a reference baseline: This will generate an on-orbit performance dataset, raise the system TRL, and support broader adoption through demonstrated flight performance.

Develop standardized documentation packages and ICD distribution artifacts: These documents will lower the integration barrier for institutions with limited budgets and limited prior CubeSat experience.

Long-term scalability: Extend the methodology to 6U and 12U common buses to enable modular CubeSat architectures in which mission-unique hardware is contained primarily within the payload volume, while the bus is treated as reusable architecture rather than a recurring custom design. Ultimately, a flight-qualified common bus family could elevate CubeSats into dependable science platforms capable of supporting missions in climate science, plasma physics, lunar science, and L1/L2 heliophysics.

Model-Based Systems Engineering (MBSE) integration: Develop a SysML-based system model of the 3U common bus architecture using the CubeSat System Reference Model Profile (CSRM) (https://www.omg.org/spec/CSRM/1.1/About-CSRM?utm_source)as the baseline reference model for architecture definition. The CSRM baseline would improve requirements traceability, interface definition, and verification planning across subsystems and payload interfaces. A computer-based system model would allow future developers to perform architecture trade studies, verify interface compatibility, and maintain configuration constancy across bus and payload. It would also provide a more structured framework for managing changes as the architecture is adapted and updated for different mission concepts or payloads.

Agile CubeSat development methodologies: Evaluate how the bus-payload partitioning described in this research could support more iterative development approaches. Future work could examine the use of incremental subsystem maturation, rapid prototyping, and partial or full parallel development of engineering model (EM) and flight model (FM) development to reduce development schedules while still maintaining acceptable mission level reliability. Doing these things would help assess whether standardization enable a more responsive development process without introducing unacceptable technical or programmatic risk.

Mission suitability analysis: Characterize the type of missions and payloads that can be supported within the 3U payload volume and by the 3U common bus. Candidate payloads include Earth observation sensors, communications demonstrations, technology demonstration payloads, and compact scientific. Additional analysis of payload mass, power, thermal dissipation, and data-rate requirements would help define the operational envelope of missions that can be supported without major modification to the baseline 3U common bus architecture.

Accessibility for university-led missions: Evaluate whether the proposed 3U common bus architecture can be fabricated, integrated, and tested using commercially available components and university resources. Such an assessment determines whether the architecture meaningfully lowers the system integration complexity and development barriers for student-led CubeSat programs. Additionally, this work also examines documentation needs, integration workflows, and test and infrastructure requirements to determine how feasible the architecture is for budget constrained or first-time university CubeSat development programs. To further increase accessibility, reproducibility, and wider adoption, the CAD model, dissertation, interface documentation, and related design artifacts generated through this research are being made available through a public GitHub repository [https://github.com/jgayle17-a11y/3U_CubeSat_Common_Bus.git]. The availability of this documentation enables reproducibility while enabling broader distribution of the architecture to university teams and other emerging CubeSat developers. This approach fosters community adoption, facilitates adaptation to mission-specific needs, and promotes continued maturation of the architecture through collaborative refinement.

Cost and programmatic assessment. Evaluate how the use of a standardized CubeSat bus architecture influences development cost, schedule, and NRE effort compared to traditional mission-unique CubeSat designs. Such an assessment would help evaluate the practical programmatic value of standardization for government, commercial, and university-led missions. Future analysis could also compare the architecture against conventional CubeSat development approaches in terms of schedule compression, integration burden, and the potential for recurring cost savings across multiple missions.

7.4 Conclusion

The results of this dissertation are new knowledge that support three major areas of the systems engineering body of knowledge.

First, the results of this research support the assertion that CubeSat reliability challenges are predominantly systemic and procedural, rather than purely technical. As demonstrated in Chapter 4, most early mission failures originate in EPS, COM, and OBC subsystems; however, the underlying failure mechanisms frequently trace to insufficient functional testing, interface uncertainty, and subsystem integration performed too late in the life cycle. Omitted TVAC testing, incomplete EMI/EMC testing, and inadequate end-to-end system verification are recurring contributors to DOA outcomes and early on-orbit faults and failures. Overall, the evidence indicates that systems-engineering breakdowns, not individual component performance, represent the dominant driver of early CubeSat failures.

Second, schedule modeling presented in Chapter 5 shows that decoupling CubeSat bus development from the payload development path reduces schedule compression and increases available I&T margin. Eliminating redundant bus-level V&V and mission-unique subsystem redesign shortens the critical path, returning schedule capacity to late-phase integration and test, where CubeSat reliability is most sensitive. When procurement of a flight-qualified 3U common bus replaces full bus development, payload stakeholders can reallocate schedule time into I&T activities, increasing functional test depth without extending overall program duration.

Third, the results and analysis presented in Chapter 6 demonstrates that a standardized 3U common bus can meet low-to-mid-band 6U CubeSat performance envelopes in a 3U form factor, providing a repeatable integration foundation for diverse mission payloads. The resulting system supports higher average power, increased energy storage, and modern SoC-class onboard compute relative to heritage CubeSat buses, while a 6U CubeSat class deployable solar array enables power levels suitable for moderate-duty payloads. When paired with structured MICDs/EICDs and a staged NASA GEVS aligned qualification flow, the architecture reduces integration uncertainty.

The overarching conclusion of this research is that a 3U common bus architecture, combined with disciplined systems engineering practices, can improve CubeSat mission success by reducing I&T variability, improving V&V consistency, and enabling payload-focused development. This represents a fundamental shift in CubeSat methodology from custom bus fabrication toward a reusable architecture that supports a repeatable, certifiable bus-plus-payload assembly.

CubeSats in their current state are prone to premature failure, limiting their effectiveness as science platforms. Improving CubeSat reliability would enable lower-cost, more robust, and more frequent missions in low Earth orbit and beyond. Significant opportunity remains to improve CubeSat design and construction practices. Development of an architect adaptable and reliable 3U common bus would reduce early failures and help CubeSats realize their full potential as a credible platform for scientific missions.

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APPENDIX A – 3U COMMON BUS CAD AND INTERFACE CONTROL DRAWINGS

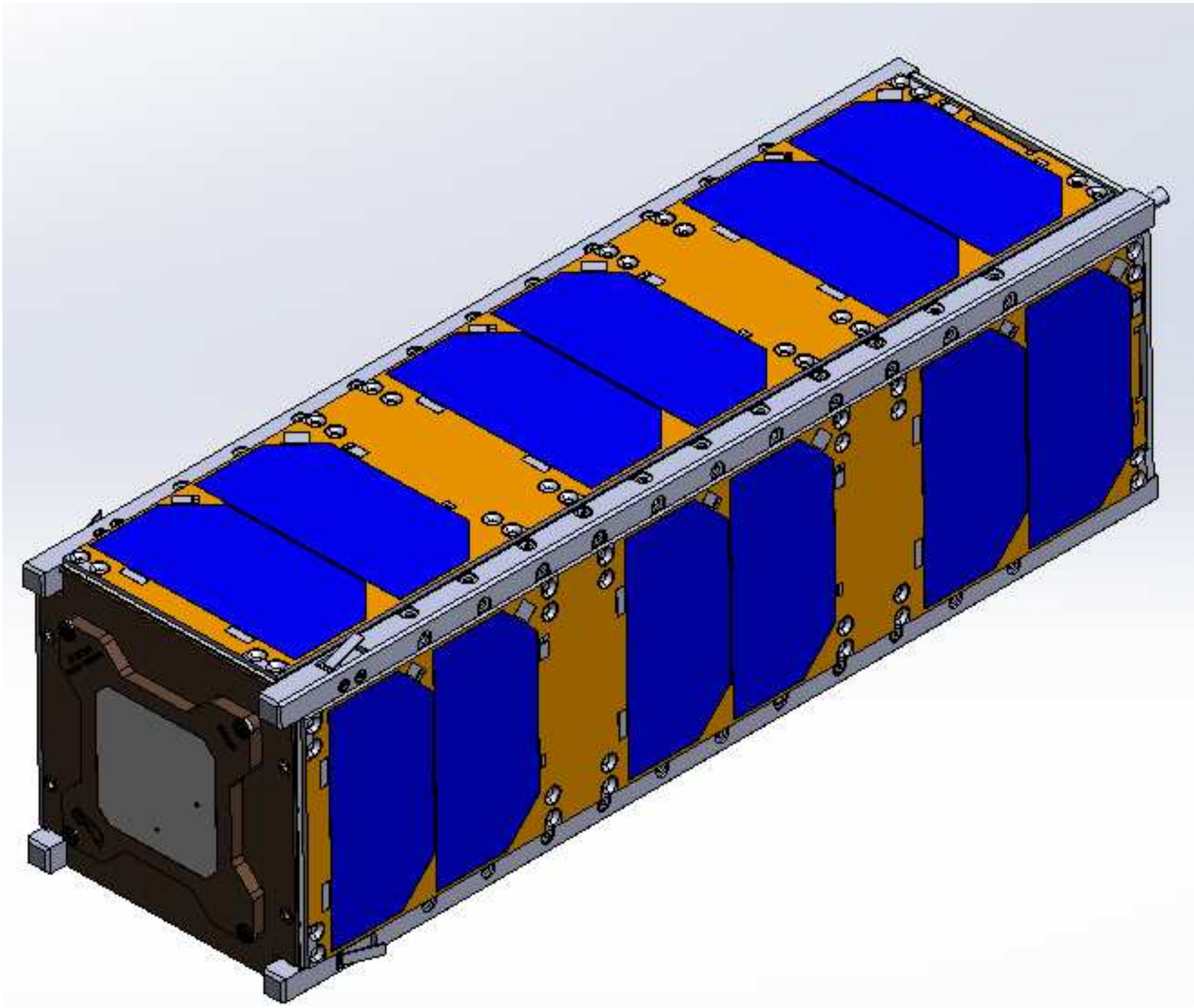

Figure 27. 3U common bus in launch configuration without the 3U payload installed, showing the external mechanical configuration and integration-ready payload cavity. Rails and external envelope are consistent with CubeSat form-factor requirements. Multi-layer insulation (MLI) is omitted for clarity.

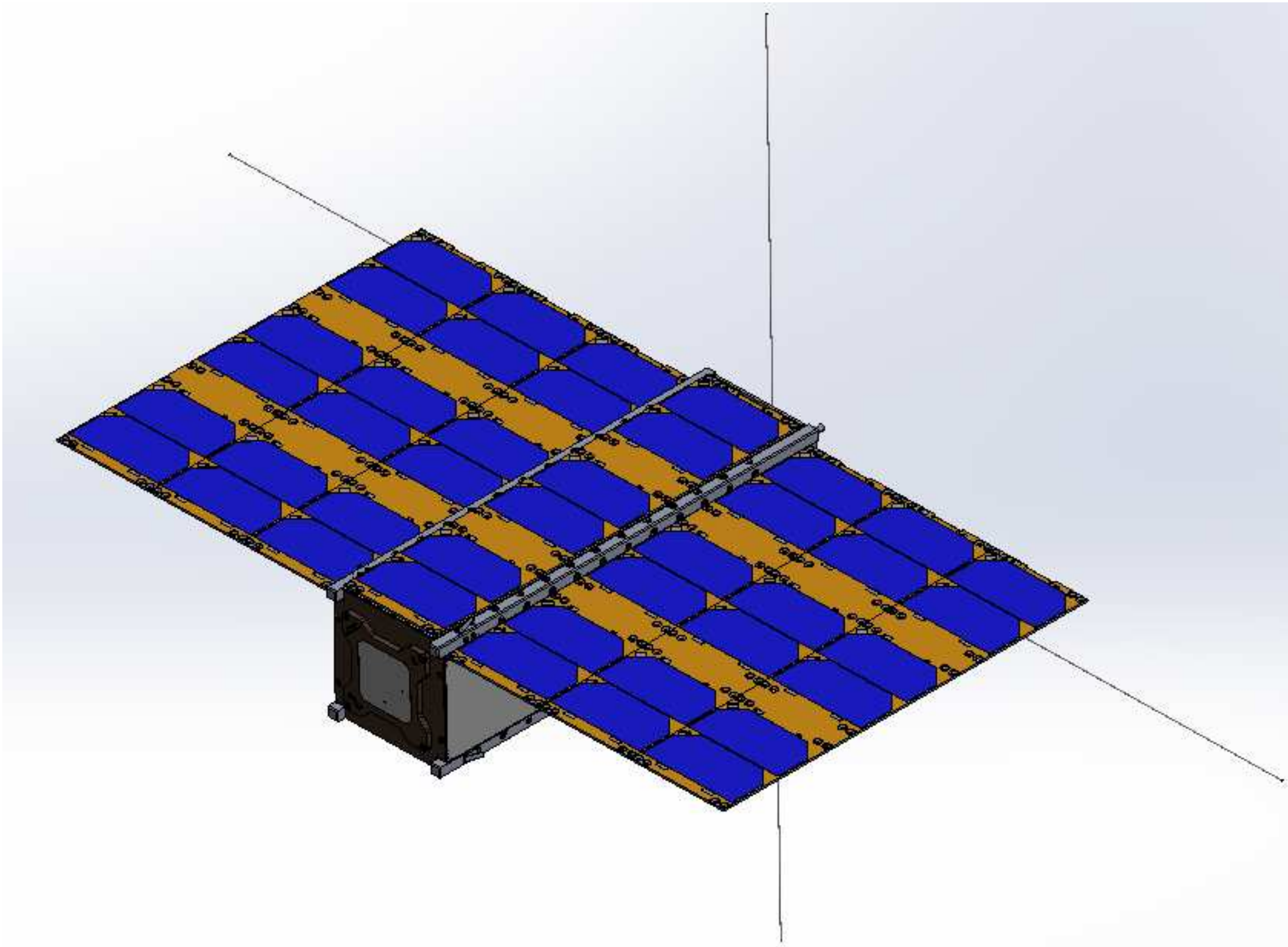

Figure 28. On-orbit configuration of the 3U common bus without an attached 3U payload, showing deployed solar array geometry representative of bus-only operation and testing. Multi-layer insulation (MLI) is omitted for clarity.

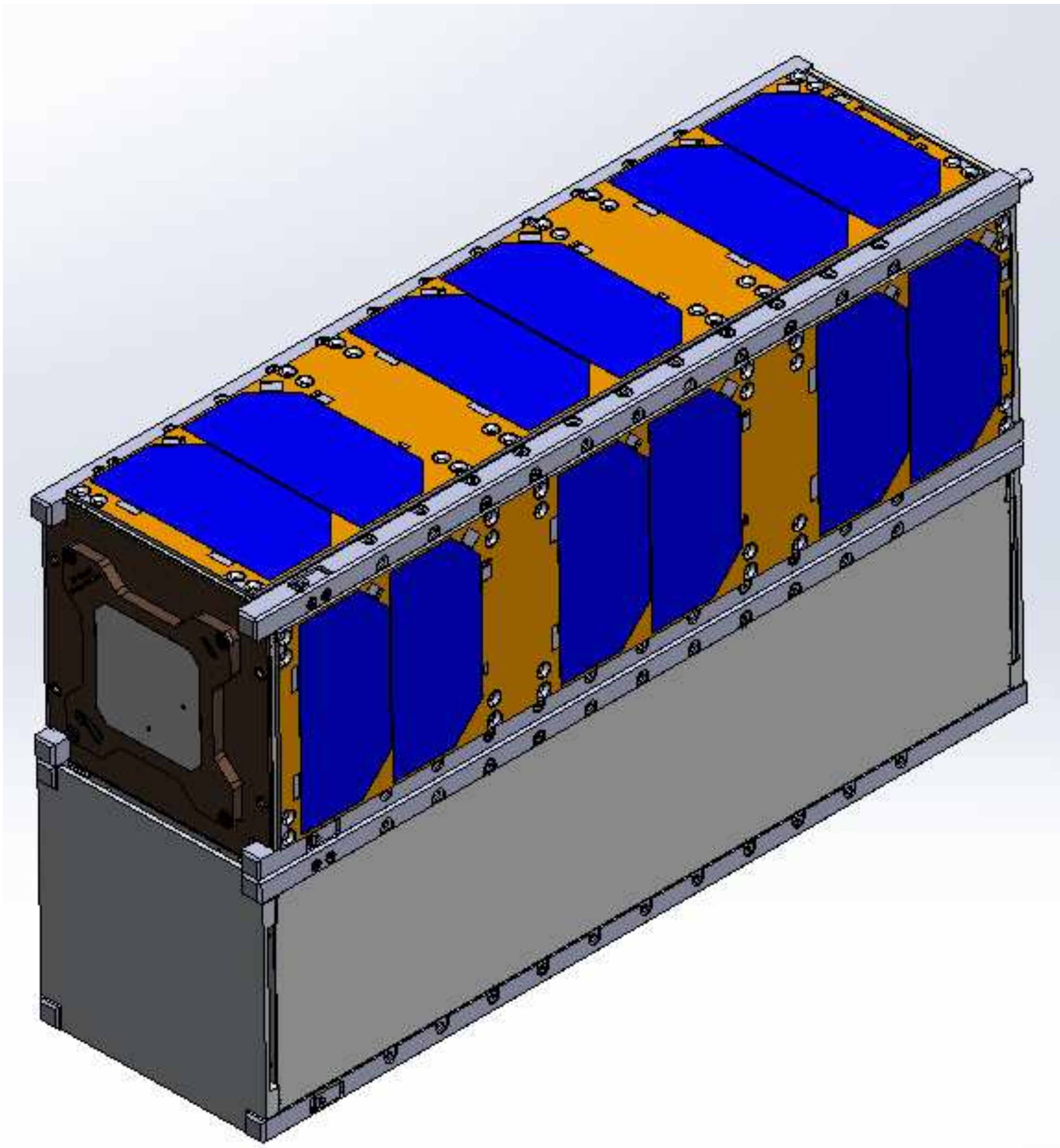

Figure 29. Integrated 6U launch configuration of the 3U common bus with a notional 3U payload. The figure depicts the integrated bus–payload assembly in its launch configuration and illustrates the standardized mechanical and electrical interfaces controlled by the MICD and EICD, enabling repeatable payload integration and deployer compatibility. Multi-layer insulation (MLI) is omitted for clarity.

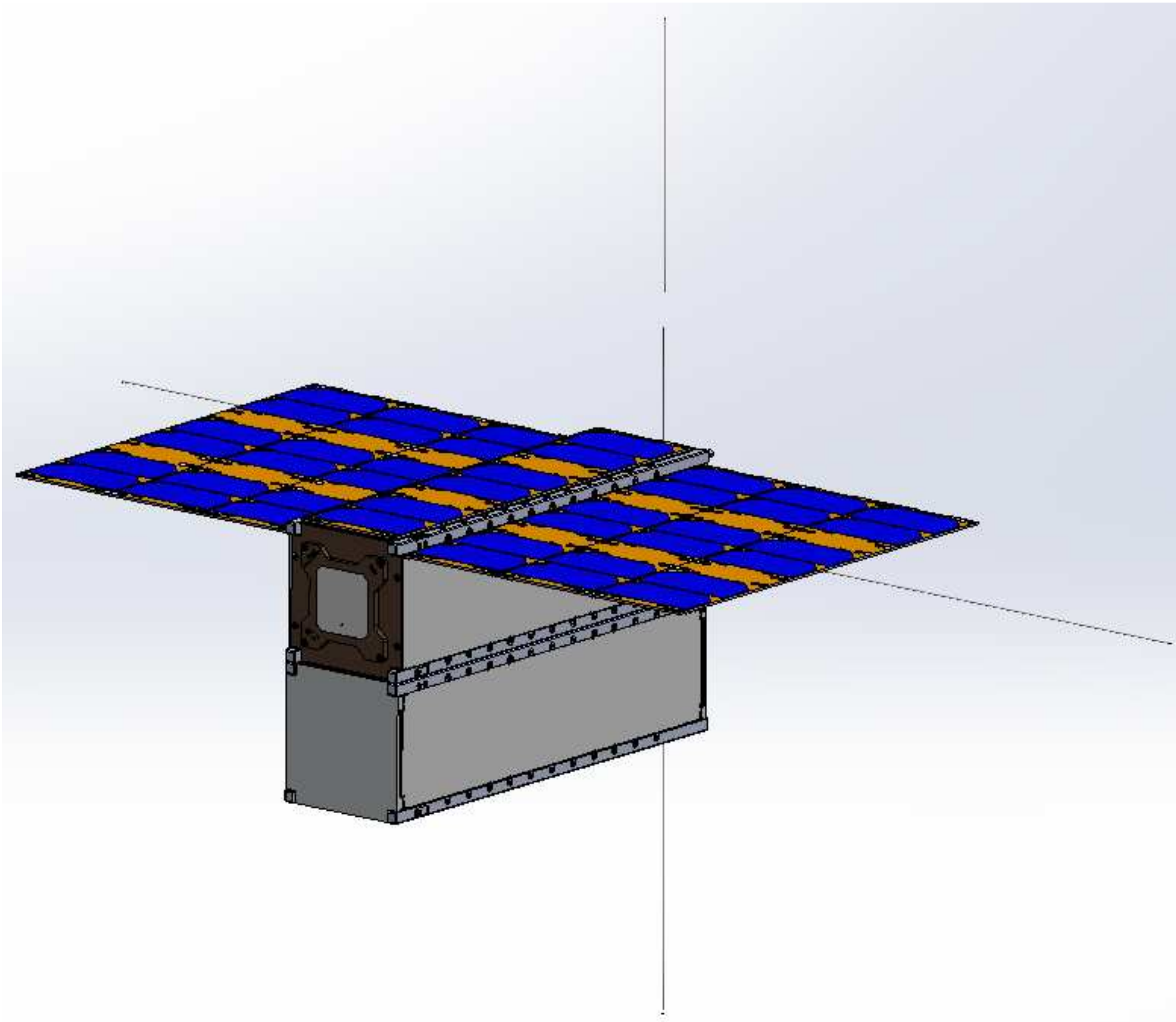

Figure 30. Integrated 6U on-orbit configuration showing the 3U common bus mated to a notional 3U payload, illustrating the deployed CubeSat configuration enabled by the standardized mechanical and electrical interfaces governed by the MICD and EICD. Multi-layer insulation (MLI) is omitted for clarity.

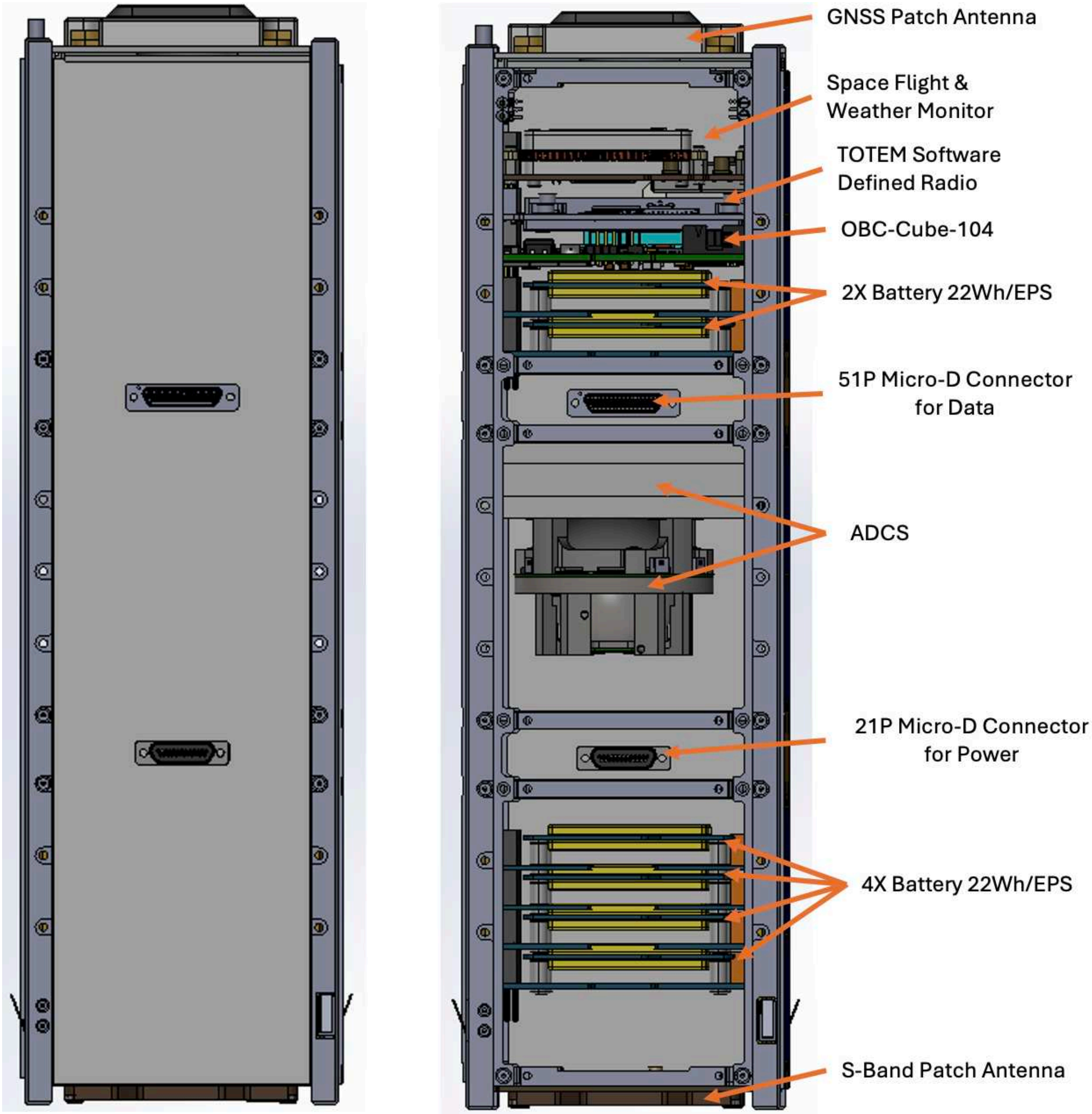

Figure 31. 3U common bus internal subsystem configuration (top-down view) showing nominal component placement, including the GNSS antenna, space weather monitor, OBC, Battery/EPS modules 1–2, 51-pin Micro-D data connector, ADCS, 25-pin Micro-D power connector, Battery/EPS modules 3–6, and the S-band antenna. Layout shown is representative for packaging and interface definition.

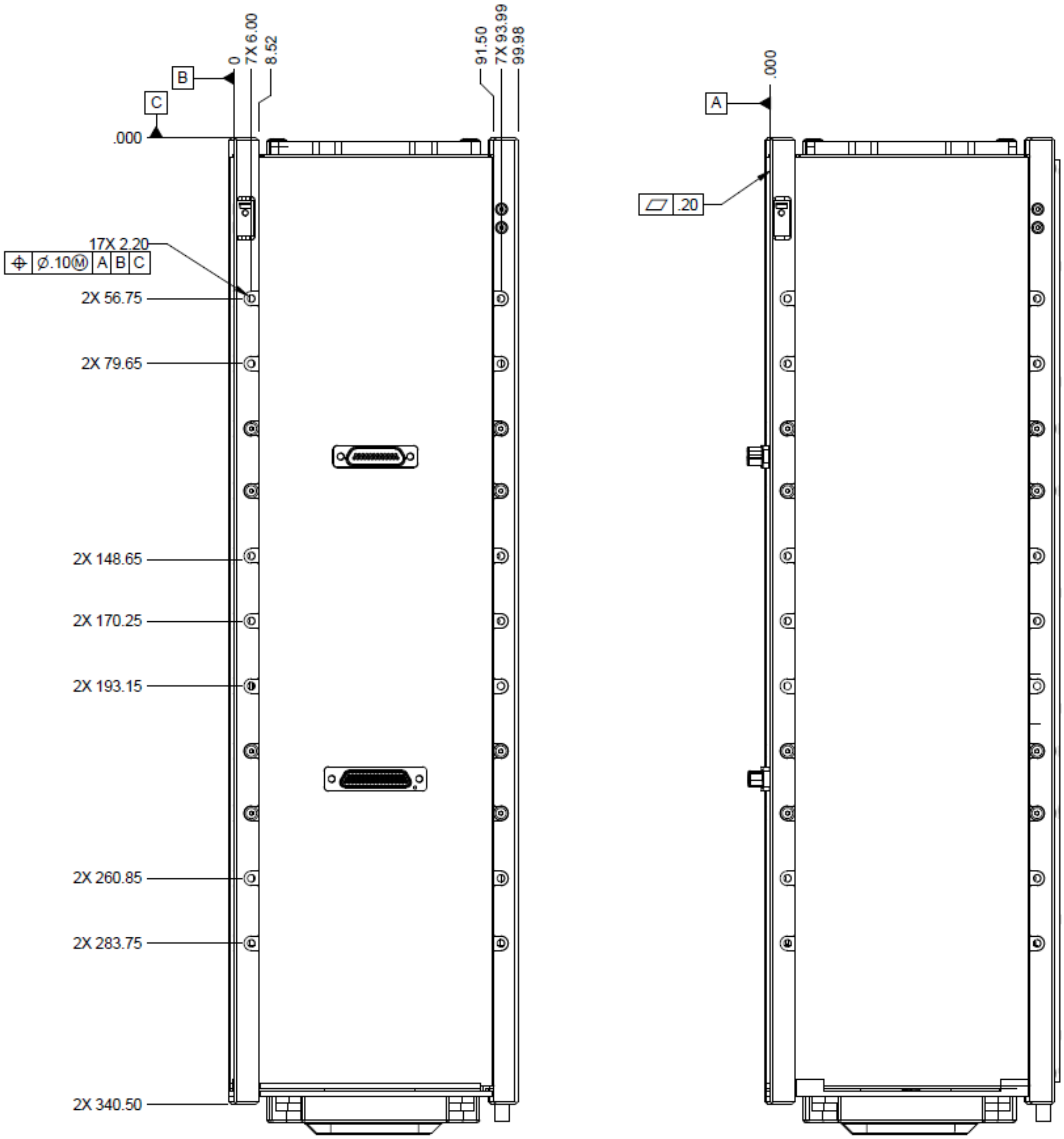

Figure 32. MICD for 3U payload interface. Locations denote M3 self-clinching fastener location (PEM nut)

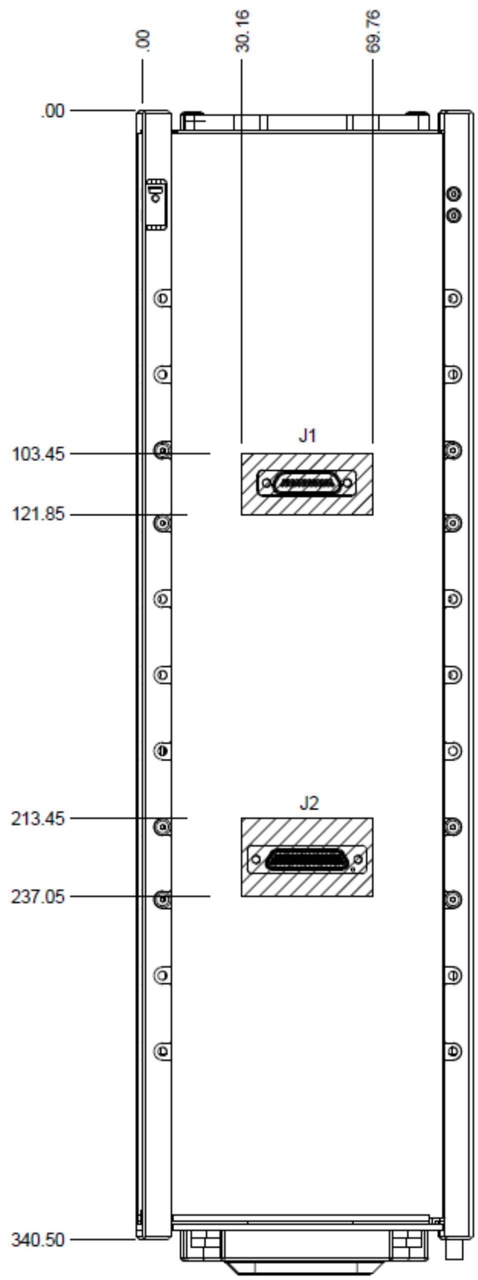

Figure 33. The payload connectors shall be free-wired (pigtail) connectors to allow positional flexibility for mating with the panel-mounted connectors on the 3U common bus. This EICD defines the power and data interfaces to the payload. The hatched region represents the cutout required in the 3U payload volume to accommodate connector access and cable routing.

The Micro-D connectors on the 3U common bus are panel-mounted. The payload-side connectors shall be pigtailed and left unsecured to enable alignment and mating during integration.

J1 is a 51-pin Micro-D data connector, and J2 is a 25-pin Micro-D power connector.

APPENDIX B – CUBESATE DESIGN SPECIFICATION REFERENCE DRAWINGS FOR 3U AND 6U FORM FACTORS

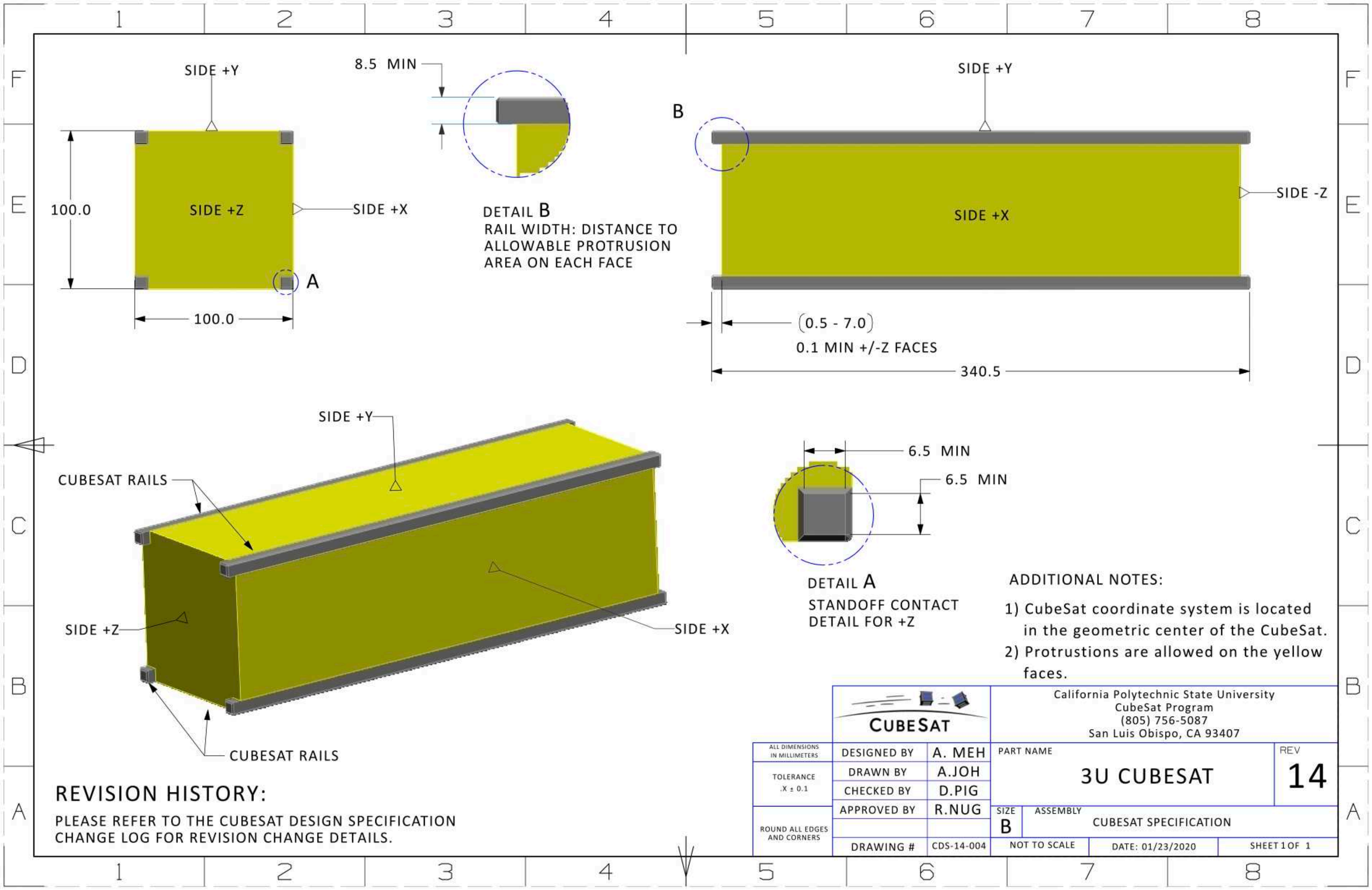

Figure 34. CubeSat Design Specification (CDS) reference drawing defining the 3U CubeSat form factor, including external dimensions and rail interface geometry. *Source: CubeSat Design Specification Rev. 14, The CubeSat Program, Cal Poly SLO.*

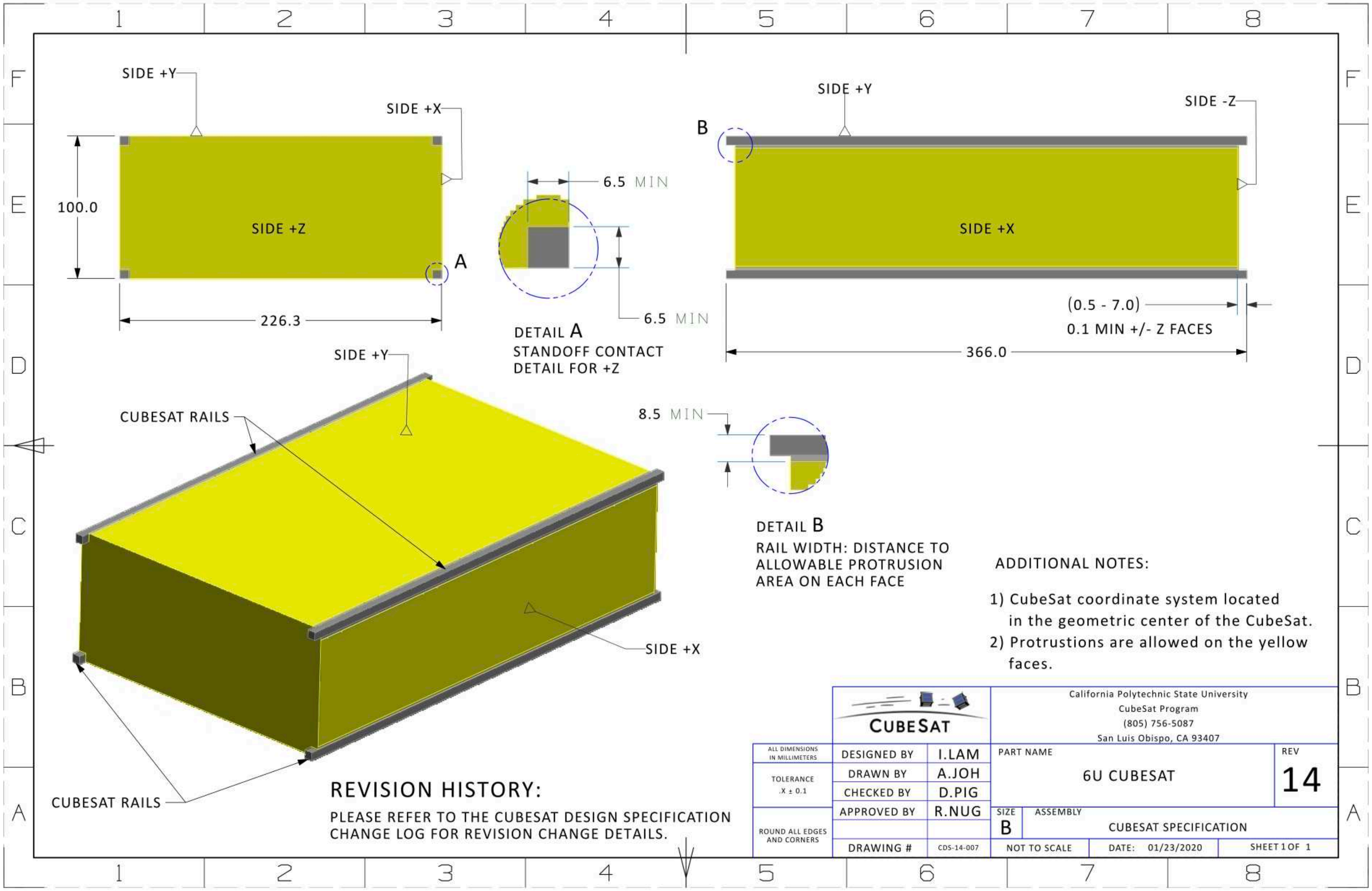

Figure 35. CubeSat Design Specification (CDS) reference drawing defining the 6U CubeSat form factor, including external dimensions and rail interface geometry. *Source: CubeSat Design Specification Rev. 14, The CubeSat Program, Cal Poly SLO.*