Spring 2019

Program management for concurrent university satellite programs, including propellant feed system design elements

Shannah Withrow-Maser

PROGRAM MANAGEMENT FOR CONCURRENT UNIVERSITY SATELLITE PROGRAMS, INCLUDING PROPELLANT FEED SYSTEM DESIGN ELEMENTS

by

SHANNAH NICHOLE WITHROW

A THESIS

Presented to the Faculty of the Graduate School of the

MISSOURI UNIVERSITY OF SCIENCE AND TECHNOLOGY

In Partial Fulfillment of the Requirements for the Degree

MASTER OF SCIENCE IN AEROSPACE ENGINEERING

2019

Approved by:

Dr. Henry Pernicka, Advisor
Dr. Suzanna Long
Dr. Warner Meeks

ABSTRACT

Propulsion options for CubeSats are limited but are necessary for the CubeSat industry to continue future growth. Challenges to CubeSat propulsion include volume/mass constraints, availability of sufficiently small and certified hardware, secondary payload status, and power requirements. A multi-mode (chemical and electric) thruster was developed by at the Missouri University of Science and Technology to enable CubeSat propulsion missions. Two satellite buses, a 3U and 6U, are under development to demonstrate the multi-mode thruster's capabilities. Two key challenges related to these missions are the development of the feed system to support the thruster and management of the two bus programs' personnel, resources, timelines, and budgets.

The feed system was designed to support the unique needs of the thruster, within the constraints and budget of a student-designed propulsion system, while minimizing risk as a secondary payload. This resulted in the development of a unique method to pressurize propellant stored in the feed system tubing. Within the expected operating pressure range, the method was experimentally shown to provide sufficient pressure and propellant volume to the thruster to meet mission success criteria.

The 3U and 6U CubeSat buses were designed concurrently with complimentary payloads, hardware, objectives, and team structures, and required careful management of resources between the two teams. With proper management, the two programs have been able to support one another through collaboration. Lessons learned include experience with design, testing, and assembly of hardware, team training/mentoring and motivation, improved documentation practices, and risk management.

ACKNOWLEDGMENTS

My sincere thanks to my advisor, Dr. Pernicka, for his encouragement and mentorship throughout my undergraduate and graduate career. I would like to thank Dr. Rovey and Dr. Berg for their contribution of the multi-mode thruster payload to the APEX and M^3 missions, and my committee members, Dr. Long and Dr. Meeks for their feedback and time. I would also like to acknowledge the financial support from the Missouri Space Grant Consortium, Academy of Mechanical and Aerospace Engineers, and NASA Ames Research Center's education program to my education.

Thank you to the Missouri S&T Aerospace Plasma Laboratory (Alex Mundahl, Mitch Wainwright, and Andrew Taylor). I am grateful to the APEX Chief Engineers, Kyle Segobiano and Corey Dodd, for their partnership and technical guidance, Zack Fizell for his CAD support, Alex Mundahl for contributing the pressurant tank sizing code, and Matt Klosterman for his STK contributions. I appreciate that these team members (plus Andrew Kueny) continued to take me seriously during my "Veggie Sat" and knee scooter phase after ankle surgery. The contributions and support of all of the M^3 and APEX team members have made both of the satellites possible, as well as this thesis.

I would like to thank Donna Jennings for her support, willingness to listen, and for challenging me to continually improve. My parents and in-laws have encouraged me throughout my education, and my sister has provided inspiration (and grammar edits) to this thesis. I am grateful to my husband, Jaykob for sharing the graduate school experience with me and enabling me to pursue work I enjoy. Lastly, I want to thank my God who has been a source of strength for me.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

Page

LIST OF TABLES

Page

NOMENCLATURE

Symbol	Description
Δ	Change
ρ	Density
μ	Viscosity
v	Specific Volume

1. INTRODUCTION

Small satellite capabilities are developing rapidly and have the potential to change the space industry by increasing launch opportunities through both reduced cost and more efficient use of launch vehicle payload space. Figure 1.1 shows the dramatic increase in the number of small satellite missions over the last twenty years. According to NASA's Small Satellite Missions program, "Through technological innovation, small satellites enable entirely new architectures for a wide range of activities in space with the potential for exponential jumps in transformative science."[1]

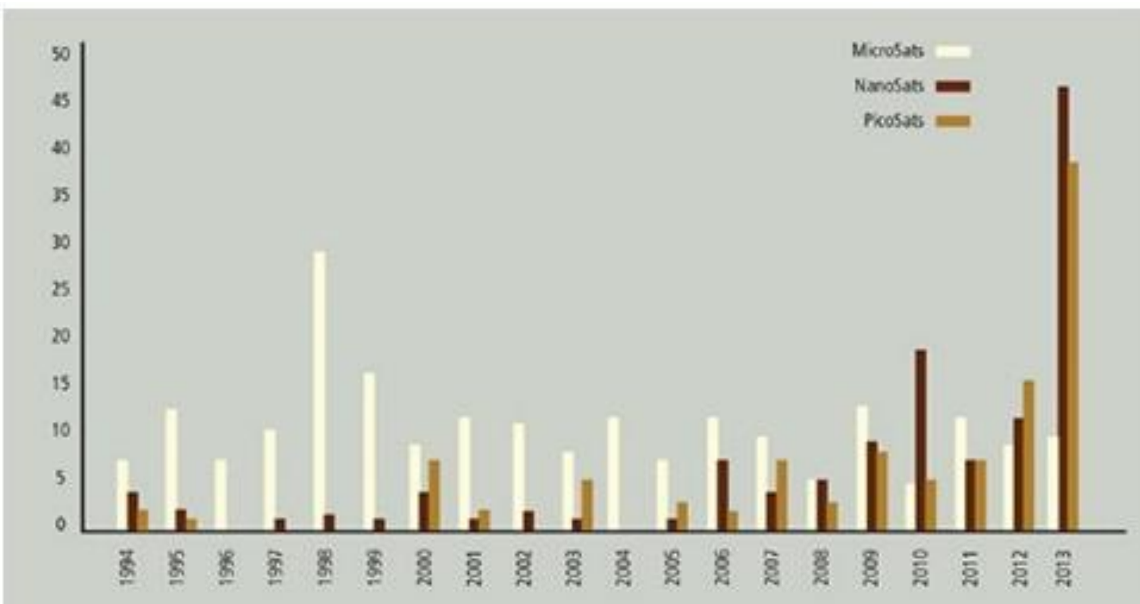

Figure 1.1. Satellite Mission Growth 1994-2013.[3]

Steve Jurczyk, associate administrator for NASA's Space Technology Mission Directorate, cites small satellites as a "paradigm shift for NASA and the larger space community."[2] This class of satellite has opened the testing ground of space to organizations, such as universities, not previously capable of supporting larger, more traditional spacecraft. CubeSats are popular among less traditional investigators. CubeSats are defined by their volume, which is composed of 10 x 10 x 10 cm “cubes” each known as units or “Us” (as illustrated in Figure 1.2). Common total volumes of CubeSats are one, three, six, or twelve Us. The standardization of these structures allows for simplified launch vehicle integration. Additionally, vendors are able to mass produce instrumentation and other necessary materials for this class in bulk, mitigating expenses associated with custom hardware for frequently used components.

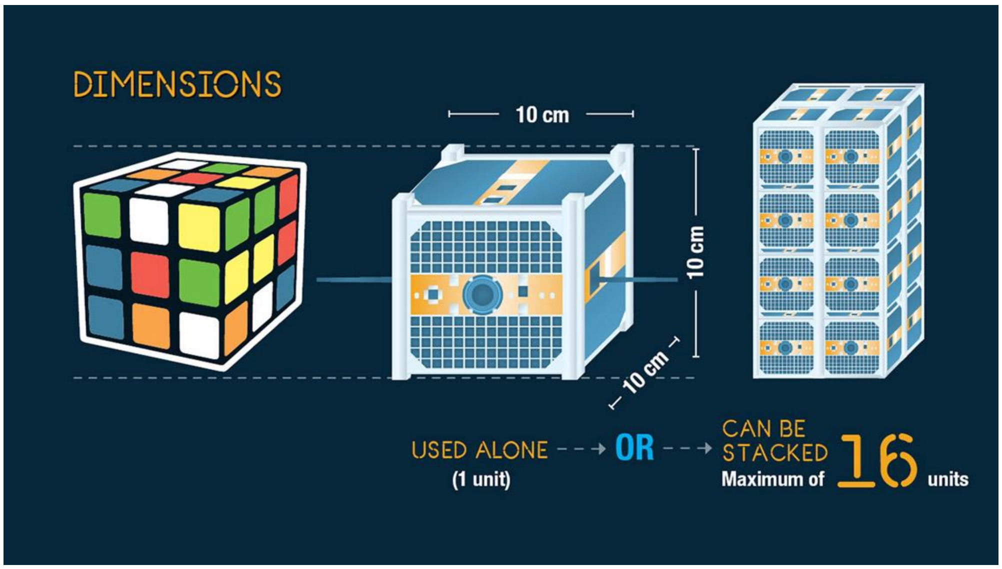

Figure 1.2. CubeSat Dimensions.[4]

However, satellites in this size category also have inherent challenges. Within the small satellite community, these challenges are frequently referred to as SWaP (Size, Weight, and Power) as mass/volume and power are often the limiting factors of a design.

In 2001, Dr. Henry Pernicka founded the Missouri S&T Satellite Research Team (M-SAT) at the Missouri University of Science and Technology. The team's first pair of satellites, Missouri Rolla and Missouri Rolla Second Satellite (MR and MRS SAT) were developed as a stereoscopic imaging demonstration mission using an innovative cold gas thruster design. MRS SAT functions as a non-cooperative object, while MR SAT contains the imaging system and cold gas thrusters. In 2015, MR and MRS SAT won first place in the Air Force Research Lab's (AFRL) University Nanosatellite 8 Program, receiving a launch opportunity. Shortly after, M-SAT, in collaboration with Missouri S&T Aerospace Plasma (AP) Lab, submitted a proposal to AFRL's University Nanosatellite 9 competition to develop a 6U multi-mode thruster technology demonstration mission. Additionally, the team submitted a proposal for a 3U version of the multi-mode thruster to NASA's Undergraduate Student Instrumentation Project (USIP) to address and increase the Technology Readiness Level (TRL) of some of the more challenging aspects of the 6U demonstration. Both proposals were accepted, and work on these satellites is on-going.

1.1. M-SAT MULTI-MODE MICROPROPULSION MISSIONS

Two micropropulsion satellites missions were chosen as an area of focus by the M-SAT team because they meet a current industry need. While the emergence of the CubeSat has enabled numerous space missions for relatively low cost and a rapid development timeline unachievable with larger satellite systems, the capability of these CubeSats has been limited by the lack of propulsive capability, particularly where multiple types of propulsion are needed. The 2015 NASA Small Spacecraft Technology State of the Art report states, "As propulsion technology matures, more small spacecraft missions will incorporate propulsion systems on board allowing for more complex mission architectures[5].” Additionally, the Air Force Technology Horizons report lists “Fractionated, Composable, Survivable, Autonomous Systems” as a technology grand challenge, and specifically calls out, “Rapidly Composable Satellite Systems: Satellites that can be assembled, tested, and launched within days of operational requirement, based on a plug-and-play/open-architecture approach using standards for self-describing components within a discoverable and autoconfiguring system”[10] as a major focus. The need for propulsion systems for smallsats is becoming more urgent as the demand for these spacecrafts and their missions expand and diversify. Khary Parker points out in NASA Goddard’s “2017 State-of-the Art for Small Satellite Propulsion Systems” presentation, “[CubeSat] uses and capabilities are growing to the point where a propulsion system is required... Current state-of-the-art for smallsat propulsion systems is rapidly evolving. However, their technology readiness level (TRL) is still relatively low.”[5] While CubeSat technology has rapidly developed, propulsion system development lags behind. Dr. Polzin from NASA Marshall Space Flight Center commented that, “There are lots of systems being developed out there, but none of them have come out clearly on top yet.”[6] Air and Space Magazine recently reported, “The consumer electronics industry has dramatically shrunk sensors and microcontrollers in the last decade, but propulsion has proven harder to miniaturize.”[7] It is therefore necessary to develop and validate the performance of such a propulsion system that can be easily integrated into a CubeSat form factor. As evidenced by the reports cited above, propulsion system technology that expands CubeSat mission opportunities is relevant and desired in many areas of the space community.

The 3U M^3 and 6U APEX seek to demonstrate a solution to this propulsive need. Students have developed a propellant feed system that integrates into both the 3U and 6U CubeSat form factor and is paired with a thruster operable in both catalytic chemical and electrospray electric modes.

1.1.1. Multi-Mode Micropropulsion Mission (M^3). M^3 is intended to be a "stepping stone" to the APEX mission. M^3 is unique in that the USIP program limits the participants to undergraduates with one graduate student exception (i.e. the author, who functions as a mentor to the team). M^3 will contain a smaller volume of propellant, perform fewer burns, and operate at a lower pressure than APEX. While the thruster aboard M^3 will be capable of operating in chemical mode, electric mode will be used exclusively. This mission scaling allows the feed system, propellant, and thruster design to be validated on-orbit while mitigating risk from a launch range safety perspective. M^3 is expected to be delivered to NASA at the end of the Spring 2019 semester with potential launch dates in early 2020.

1.1.2. Advanced Propulsion EXperiment (APEX). Lessons learned from M3 will be applied to APEX, which began development under AFRL's University Nanosatellite 9 (NS-9) competition and will continue with a potential launch opportunity under AFRL's University Nanosatellite 10 (NS-10) competition. APEX was designed as a 6U to meet the SWaP challenges associated with operating the thruster at its full capacity in both catalytic chemical and electrospray electric modes.

1.2. AUTHOR'S INVOLVEMENT

The author has been significantly involved with the design and development of both satellites. After joining the M-SAT team in the fall of 2015, the author was selected as the Program Manager for the APEX satellite (early spring 2016) and attended the University Nanosatellite 9 Program kick-off meeting in Albuquerque in January. In this capacity, the author represented the team as the primary presenter at all AFRL University Nanosatellite Program (UNP) reviews, performed Program Manager appropriate tasks - such as piloting documentation efforts, selecting and training team leads, managing the team's resources and schedule, and coordinating the design and testing process - and acted as the primary ambassador and advocate for the project. During this time, the APEX satellite was developed from concept to a functioning engineering design unit (EDU). After Phase A of ARFL's UNP NS-9 competition was completed in January 2018, the author performed appropriate close-out tasks and transitioned the team into preparation for the NS-10 program. The author assisted the Principal Investigator (PI) in preparing the UNP NS-10 proposal, which was accepted in November 2018 as one of ten entries to AFRL's University Nanosat 10 competition. Additionally, the author served as the propulsion lead for the APEX satellite from the Preliminary Design Review (January 2017) to the close-out of the NS-9 program. During this time, the author was responsible for the EDU propulsion feed system design and assembly, component selection, risk analysis, testing a novel pressurization concept, and interface planning with the thruster team. The author served as a liaison between the Space Systems Engineering Laboratory (SSE lab), responsible for the satellite bus development, and the Aerospace Plasma Lab, responsible for the payload (thruster) development.

Upon the M-SAT team's acceptance into the NASA USIP program in Fall 2016, the author was selected as the official graduate student mentor (though others donated time as unofficial mentors) and acted as a pseudo PI for the team. In this position the author mentored approximately twenty-five to thirty undergraduate students each semester through the process of designing, testing, building, integrating, and delivering a 3U CubeSat. The satellite is expected to undergo environmental testing within a year of the author's master's degree defense. The author trained and motivated team leadership, developed actionable items based on provided requirements, proofed all written deliverables, acted as a liaison between the students and collaborators, influenced design decisions, and designed the feed system. The author used lessons learned from APEX to expedite the timeline of the project and optimized the use of resources by encouraging collaboration between the APEX and M^3 teams.

Additional noteworthy activities during the author's master's program include presenting as a primary author in the Frank J. Redd Small Satellite Conference Student Paper Competition (receiving a Honorable Mention award), winning first place at the AIAA Region V Paper Conference-Masters Division, presenting at the 2017 Missouri Space Grant Consortium, receiving "2017 Best Oral Presentation" from the Academy of Mechanical and Aerospace Engineers (AMAE), attending and presenting a poster at NASA's 2017 Academy of Aerospace Quality Workshop, receiving the AMAE McGovern award, completing the Complex Systems Methodology Project Management for Research and Development course by Lory Wingate, and receiving a NASA Pathways position at NASA Ames Research Center.

2. LITERATURE REVIEW

2.1. UNIVERSITY NANOSATELLITE PROGRAM

The University Nanosatellite Program[9] (UNP) was founded in 1999 as a university-outreach program by the Air Force Research Laboratory at Kirtland Air Force Base. UNP has fostered 5,000+ students from thirty-six universities in nineteen years. UNP's main objectives are education, technology, and university laboratory development. During the course of the program, students start with a mission concept and go through a series of reviews, ending the first phase in a competition with an opportunity to launch their satellite. Common challenges for student teams include resource limitations, learning curve, improper (overly ambitious) mission scoping, insufficiently defined Concept of Operations (CONOPs), and software development. These challenges can be addressed by rigorous planning and implementation of systems engineering concepts. Critical personnel roles for a successful mission include a visionary to motivate the work, a source to drive the team's momentum, a verification role, and a manager to promote team cohesion. The timeline is often a high-risk element for universities. A delicate balance must be struck between causing overstress and allowing too much slip (time deviated from original schedule). Student-planned intermediate milestones between UNP mandated milestones can help maintain the proper pace.[8]

2.1.1. University Nanosatellite Program 4 at Missouri S&T. In 2007 a guide detailing the systems engineering aspects of small satellite development at the university level was authored by an M-SAT team member (Abbie Stewart) targeting an audience organizing such a project for the first time[9]. The M-SAT team (then known as the UMR SAT team) from 2004 to 2007 was used as a model along with participation in the UNP

Nanosatellite-4 (NS-4) Competition. Some aspects of this thesis are a continuation of this guide, specifically regarding lessons learned. In this thesis study, the M-SAT team from 2016-2018 is used as a model along with the participation in the UNP Nanosatellite-9 and -10 competitions and the NASA USIP Project. Both documents describe the process of designing and building a satellite from the ground up. This includes developing requirements, developing a design from those requirements and provided standards, testing, mitigating challenges, hardware assembly, documentation and team management, and team member/knowledge retention.

2.1.2. Nanosatellite 4 to Nanosatellite 10. The team has maintained a similar organizational structure throughout the years. The team is led by a professor (Dr. Henry Pernicka) who serves as the Principal Investigator. The professor provides technical mentorship, chooses the projects and research areas the team will pursue, oversees the budget, approves purchases (as required by the university), and provides advisory and intermediary support to the team as needed. Students fill the roles of Program Manager and Chief Engineer. These positions are usually filled by senior members of the team with prior project experience. The Program Manager (PM) manages or organizes the personnel related tasks, pilots the documentation process, runs meetings, manages the schedule, and acts as the primary team advocate to investors. The Chief Engineer is responsible for managing the technical aspects of the team. This involves attending subsystem meetings, managing interfacing between subsystems, approving hardware selections, and leading learning efforts when team knowledge gaps are discovered. It is vital to team success that the students in these two positions communicate often and work together. A Resource Manager oversees team training and outreach events. The team is broken up into subsystems with a lead overseeing each technical area. These subsystems are based on the team structure developed during the Nanosatellite-4 competition. Current subsystems for the NS-10 and USIP projects are Command and Data Handling, Communications, Guidance, Navigation, and Control (formerly two subsystems: Attitude Determination and Control and Orbit Determination and Control), Integration, Power, Propulsion, Structure, and Thermal. Payload development is overseen by the Missouri S&T Aerospace Plasma Lab, which has partnered with the M-SAT team for these missions. The Integration subsystem is unique to the M-SAT team in contrast to most other university teams. The Integration team manages interface control documents and assembly procedures and was created as a result of lessons learned early in the M-SAT program. This ensures that valuable time is not wasted in later stages due to subsystem "tunnel" mentality. Students choose which subsystems they want to participate in, but they may be asked to consider also participating in other areas, depending on the current needs of team. Leads are chosen for each subsystem, primarily from senior team members. However, students may also be recruited from outside the current team if a certain skill set is needed based on other student or faculty recommendation.

Stewart[9] asserts it is essential to select a mission that is both of interest to the team and customer. Customer interest from government organizations can be determined by strategic objectives published by the sponsoring organization. For the APEX and M^3 missions, this included the Air Force Technology Horizons Report[10] and 2015 NASA's Technology Area Breakdown[11]. Relevant objectives include:

- "Rapidly Composable Satellite Systems: Satellites that can be assembled, tested, and launched within days of operational requirement, based on a plug-and-play/open-architecture approach ...” (Section 2.27- Air Force Technology Horizons Report[10])

- “Liquid storable chemical propulsion, micropropulsion, electric propulsion, propellant storage and transfer, and power systems to enable propulsive technologies” (NASA Technology Area Breakdown for in-space propulsion 2.1.1, 2.1.7, 2.2.1, 2.4.1, 2.4.5[11])

Additionally, the Stewart thesis discusses the criticality of selecting an appropriate scope based on timeframe and budget[9]. The NS-9 program included many design decisions that were significantly impacted by budget constraints. For example, creating a feed system that was adaptable to a plug-and-play type design, similar to the thruster, would have required qualification outside of the cost constraints, so it was not included in the scope. The M^3 mission faced more schedule constraints. For example, the mission objectives had to be de-scoped to only include performance of electric mode, as the certification to transport and fly the chemical component of the propellant that enables high-thrust burns could not be completed in the given timeframe.

Also highlighted in the thesis is that conference presentations are an effective, attractive mechanism for students to improve technical writing and presentation skills. They provide feedback from industry on the project throughout its duration and can lead to potential partners with industry mentors that can aid in needed technical expertise and sometimes low-cost hardware. At such events, students also frequently receive monetary awards, internships, or job offers, as a result of their work, enhancing their career paths[9].

The guide also discusses the challenge of team turnover. While it seems that some aspects of team transition have become more structured between Nanosatellite 4 and

Nanosatellite 10, such as lead transition and improved documentation of past work, the author concurs with the Stewart thesis that a standardized form for graduating members would further improve the turnover process[9]. This form should include recently completed tasks and results, recommendations for next steps, overarching goals, physical location of relevant equipment, and who to contact with questions - at a minimum. It should continue to be emphasized to new members that questions are welcomed and encouraged. Additional program management aspects of the Nanosatellite-9 and Nanosatellite-10 Programs are discussed in Chapters 6 and 7.

2.2. MULTI-MODE (CHEMICAL AND ELECTRIC) PROPSULSION FOR SMALL SATELLITES

The following sections will provide an overview of relevant, current and past propulsion technology.

2.2.1. Propulsion (EP) Overview. The first EP system was flight tested in the 1960s, however development was slowed by power limitations on spacecraft at the time and hesitancy to abandon conventional methods. EP systems are composed of the thruster structure, a power system including a power processing unit (PPU) to convert raw power to the form required by the thruster, and a propellant-feed system. According to Martinez-Sanchez and Pollard in an overview of electric propulsion provided as part of a series in the *Journal of Propulsion and Power* in 1998, “The common feature of all EP schemes is the addition of energy to the working fluid from some electrical source. This has been accomplished, however, in a large variety of physically different devices. Operation can be steady or pulsed; gas acceleration can be thermal, electrostatic, electromagnetic, or mixed; the propellant can be gas, a chemical monopropellant, or even a solid.[12]" EP systems are known to be high specific impulse (I_{sp}) and low thrust. Examples of EP thrusters include resistojets, arcjets, Hall thrusters, ion engines, pulsed plasma thrusters, vacuum arc thrusters, electrothermal, electrospray, and magnetoplasmadynamic thrusters. Descriptions of these thruster types can be found in Martinez-Sanchez and Pollard[12] and Lemmer[13]. EP systems are commonly used for station keeping, orbit adjustment, attitude control, and other repositioning maneuvers. Electrospray EP systems are of special interest to this thesis study. Electrospray systems feed propellant to an emitter with an electric field generated between the emitter tip and an opposing electrode. Surface tension and the electric field distort the propellant into a Taylor cone at the emitter tip, increasing the intensity of the electric field. Propulsion is created by emission of a droplet from the Taylor cone (colloid thruster) or ion emission (field emission electric propulsion i.e. FEEP).[12,13]

Some EP systems have achieved flight history on CubeSats. Nanospace CubeSat MEMS propulsion module on the Chinese CubeSat TW-1was an electrothermal system used to deorbit in 2015.[14] Vacuum arc thrusters (VATs), a type of pulsed thruster have CubeSat flight heritage in the form of the Micro Cathode Arc Thruster (µCAT) designed by George Washington University. µCAT flew aboard the United States Naval Academy's BRICSat-P[15] and was used to detumble the spacecraft, although mission results are not publicly available. Another VAT was flown on the Illinois Observing Nanosatellite (ION) from the University of Illinois, but no data were collected due to launch vehicle failure[16]. FEEP electrospray thrusters, known as Scalable ion Electrospray Propulsion Systems (SiEPS), were developed by the Massachusetts Institute of Technology (MIT) and were flown as part of the Aerospace Corporation's IMPACT mission in 2015 on the CubeSat AeroCube-8 [17,18]. Data from the mission were not publicly released. Busek Company developed the BIT-3, a miniature ion engine scheduled to fly on two deep space CubeSats in 2020[19].[13]

2.2.2. Chemical Propulsion Overview. Chemical propulsion systems function using the same concept as rockets and thrusters used on traditional, larger spacecraft. The chemical reaction between fuel and oxidizer creates energy by breaking molecular bonds and accelerating the by-product through a supersonic nozzle. Chemical propulsion includes monopropellants, bi-propellants, and solid rockets. To date, no chemical propulsion systems have been used on a CubeSat due to restrictions on stored energy in secondary payloads, though many have been developed in lab settings by companies such as Aerojet Rocketdyne, Busek, VACCO/ECAPS, Tethers Unlimited, Inc., and DSSP[13]. Most of these thrusters range from 0.1 to 1 N, however, the DSSP design boasts 76 N capability[13]. Hydrazine has traditionally been the propellant of choice for chemical propulsion, however, it is not ideal for CubeSats that tend to be designed as secondary payload with a “Do No Harm” mentality. Hydrazine is toxic and has the ability to auto-combust at room temperature and pressures. Some alternatives that have been considered include green monopropellants (HAN and AN-based), peroxide blends (in development), bi-propellants, solid propellants, and even water. Although each has their challenges, the most resources are being dedicated to green monopropellants at the time of writing.[13]

2.2.3. Systems with Flight Heritage To-Date. In “Propulsion for CubeSats” Dr. Lemmer states,

> “To date, only two missions have featured propulsion systems as part of the technology demonstration. The IMPACT mission from the Aerospace

> Corporation launched several electrospray thrusters from Massachusetts Institute of Technology, and BricSAT-P from the United States Naval Academy had four micro-Cathode Arc Thrusters from George Washington University. Other than these two missions, propulsion on CubeSats has been used only for attitude control and reaction wheel desaturation via cold gas propulsion systems[13]."

One noteworthy example of cold gas thruster application is the JPL MarCO mission, launched in early 2019 (after Lemmer's article was released), which contained a Vacco Micro CubeSat Propulsion Systems (MiPs) unit, capable of one to five trajectory correction maneuvers[47]. This was the first use of CubeSats for interplanetary missions. Through the APEX and M^3 missions, Missouri S&T intends to add two more satellites to the list of CubeSats with technology demonstration level propulsion systems, doubling the current number of similar missions with flight heritage.

2.2.4. Larger Satellites with Dual Propulsion Systems[20]. In 1994 a joint report between Nyma, Inc. and NASA Lewis was released entitled, "Small Satellite Propulsion Options.[21]" This report discusses the potential benefits of adding propulsion systems available at the time to existing or planned spacecraft vehicles. One such mission, the Department of Defense's (DoD's) TACSAT, was a communication satellite. The report states, "Either hydrazine arcjets or xenon Hall thrusters could be added to the satellite to perform the north/south(NSSK) and east/west station keeping (EWSK) as well as to provide rapid on-orbit repositioning."[21] This type of system was one of the earliest suggested concepts of multiple modes of propulsion systems on a spacecraft. Figure 2.1 was included in the report showing potential mass of the TACSAT with different chemical and electric thruster combinations. The total mass of the satellite with both propulsion systems ranged from 682 kg to 943 kg with a primary concern being the size of the launch vehicle required to deliver such a mass to orbit. This is compared to a base system mass (no propulsion systems) of 455 kg.

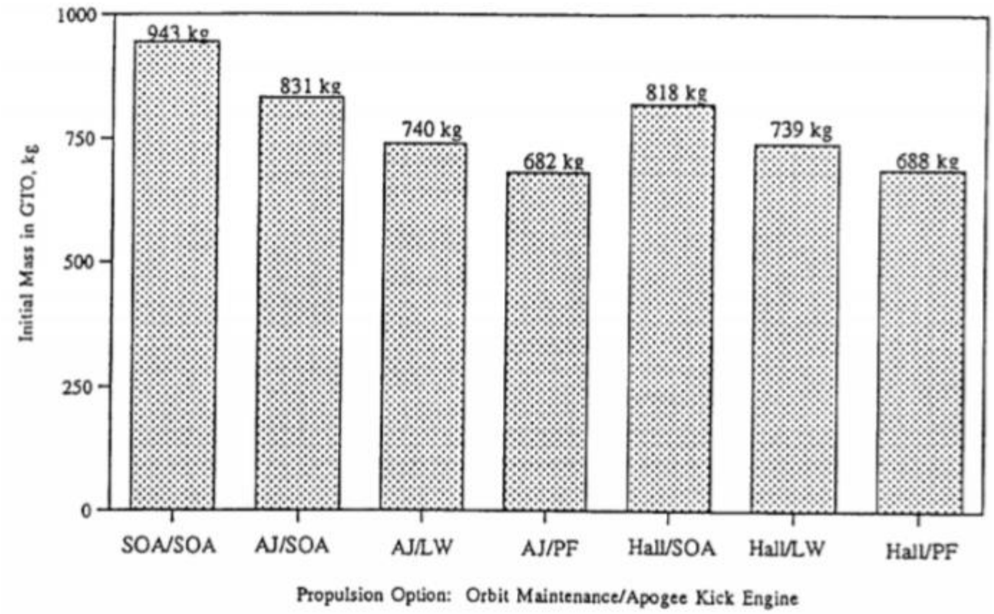

Fig. 8 Required GTO mass for a Communication TACSAT with various propulsion option combinations. SOA = state-of-art bipropellant, AJ = arcjet, LW = lightweight bipropellant, PF = pump-fed bipropellant, and Hall = Hall thruster.

Figure 2.1. TACSAT Mass with Various Propulsion Systems. [21]

NASA's proposed Mars Upper Atmosphere Dynamics, Energetics, and Evolution (MUADEE) satellite was also discussed in "Small Satellite Propulsion Options" joint report. MUADEE would contain a bipropellant chemical propulsion system and would be used to insert the spacecraft into the Martian atmosphere and perform smaller maneuvers. This mission was based on the design of the previous Venus pioneer spacecraft. Of the 800 kg allocated for the spacecraft, 350 to 425 kg would be needed for the propulsion system as shown in Figure 2.2.[21]

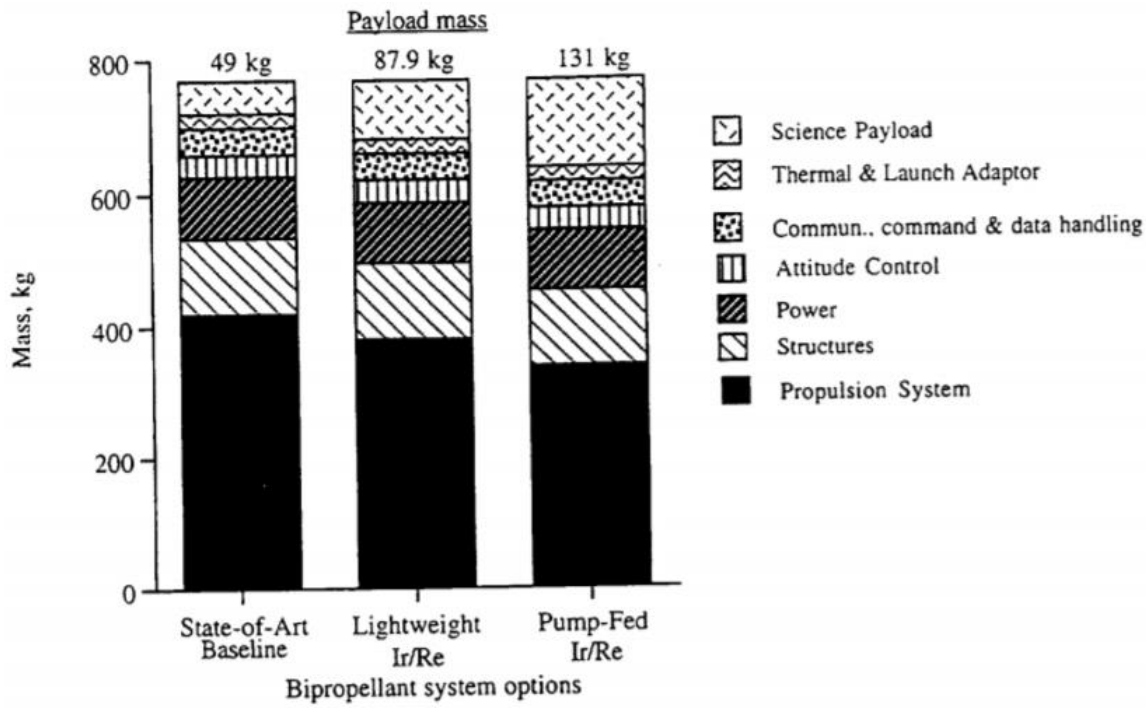

Figure 2.2. MUADEE Satellite Mass Decomposition. [21]

2.2.5. Recent Small Satellite Propulsion Developments. Technology has improved significantly, allowing modern day thrusters to be three orders of magnitude smaller than those described in 1994 with similar function. NASA's "2017 State-of-the-Art for Small Satellite Propulsion Systems" document[22] describes several thrusters currently in development within the CubeSat size envelope. Two examples are Busek's AMAC (Advanced Monoprop Application for CubeSats) project and Accion Systems' TILE-V1 project. Busek's AMAC project hosts a 1U "green" propellant thruster with a mass of 0.27 kg and a TRL of 5. Accion System's TILE-V1 project hosts an electrospray electric propulsion system with a volume slightly larger than a 1U (10 x 10 x 12.5 cm) and a mass of 0.3 kg at a TRL of 5.

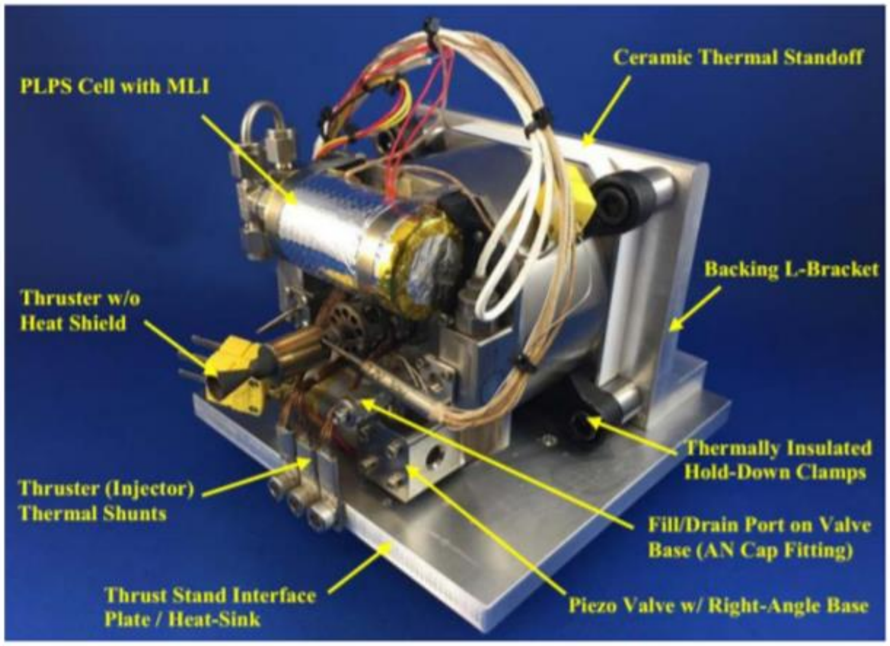

Figure 2.3. Busek's AMAC Thruster: TRL 5.[22]

Figure 2.4. Accion Systems' TILE-V1 Thruster: TRL 5. [22]

These two systems were selected for comparison because of their similarity to the APEX/M^3 thruster, which combines a green propellant, chemical system and an electrospray, electric system into a single thruster with a single feed system and propellant. The APEX/M^3 thruster is currently at a TRL 4, but is expected to reach TRL 7 by early 2020. The thruster also has a volume of 1U with a mass of 0.71 kg. Figure 2.5 compares the volumes of the missions described above. The assumption was made that the TACSAT and MUADEE missions contained no more than the standard maximum 1.33 kg per 1U.

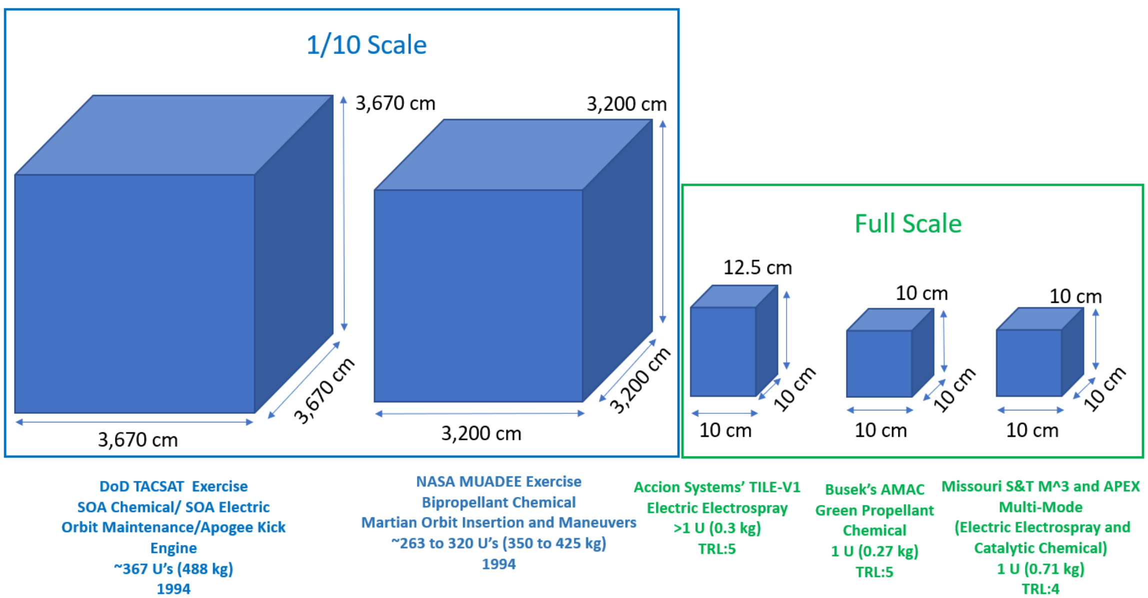

Figure 2.5. Volume Comparison.

Not only does the multi-mode thruster provide similar capability to systems that would have doubled mass and volume of the entire spacecraft twenty years ago, the thruster packages two strongly desired CubeSat propulsion technologies into a single 1U unit. Additionally, the supporting feed system is inexpensive to integrate into a 3U or 6U form factor.

2.2.6. Missouri S&T Multi-Mode Thruster. The Missouri S&T Multi-Mode thruster was developed by Dr. Steven Berg while pursuing a doctoral degree from Missouri S&T with support from Dr. Joshua Rovey. Dr. Berg developed the thruster further as a post-doctorate and founded Froberg Aerospace.

2.2.6.1. Propellant development. Rovey and Berg[23] developed a single propellant ([Emin][$EtSO_4$] mixed with a hydroxylammonium (HAN) oxidizer) and characterized it as capable of operation in either catalytic chemical or electric electrospray mode. Physical properties required for dual-mode propulsion include high density, low melting temperature, high electrical conductivity, high surface tension, and high molecular weight. This propellant is safer than hydrazine and has similar or greater specific impulse to some "green" monopropellants. The designed system attempts to take advantage of both high-thrust chemical mode and high-specific impulse electric mode characteristics. The [Emin]][$EtSO_4$] component enables electric mode operation while the HAN component enables chemical mode. The HAN component lowers the molecular weight of the mixture, thus higher power is required than other propellants to achieve electrospray mode. Highest mission capability for ΔV is for missions shorter than 150 days. The ionic liquid monopropellant mixture was tested on compatible surfaces (platinum, rhenium, and titanium) to determine decomposition and was found to decompose at the lowest temperature on platinum. The lower mass of a combined propulsion system makes this system ideal for smaller spacecraft. This work is further described in Section 3.1.[23]

A comparison was conducted of a deep eutectic 1:2 molar ratio mixture of choline-nitrate and glycerol ([Cho][NO_3]-glycerol) as a fuel component mixed with hydroxyl-ammonium nitrate (HAN) and ammonium nitrate (AN) and a formerly investigated propellant [Emin][$EtSO_4$]-HAN. Chemical rocket performance simulations predicted that this new propellant may have higher performance at lower combustion temperatures, reducing catalyst melting temperature requirements. Of the synthesized propellants, the AN mixture was found to be less reactive than HAN in atmosphere, and thus the HAN mixture underwent a greater amount of investigation. Quantitative reactivity studies indicated that [Cho][NO_3]-glycerol-HAN propellants did not have a lower combustion temperature than [Emin][$EtSO_4$]-HAN with a platinum catalyst. Thus, it was concluded that [Emin][$EtSO_4$]-HAN should continue to be used for the APEX demonstration and [Emin][$EtSO_4$] should be used for the M^3 mission. Additionally, the minimum flow rate required for the thruster to prevent flashback, assuming a design pressure of 1.5 MPa (~218 psi) and linear burn rate of 26.4 mm/s, was determined to be 0.31 mg/s for a 0.1 mm inner diameter feed tube and 3180 mg/s for a 10 mm inner diameter feed tube. It will be critical to ensure that, when the thruster and feed system design is finalized, this mass flow rate is exceeded.[24]

2.2.6.2. Computational fluid dynamics. A computational fluid dynamics analysis was performed on the catalytic decomposition of the ionic liquid, intended for use as propellant in the APEX mission, in a microtube using ANSYS Fluent. The flow in the microtube was determined to be compressible and subsonic at a Mach number of 0.0895 based on these assumptions. The analyst determined the simulation required the addition of multiphase effects and that existing simulations could not match all quantities in observed experiments. Therefore, this value should be further refined through improved boundary conditions and numerical models.[25]

3. APEX AND M^3 MISSION DESCRIPTION

Two CubeSat missions, APEX and M^3, developed at Missouri S&T will demonstrate novel propulsion technology.

3.1. PAYLOAD-MULTI-MODE THRUSTER

In order to address the propulsion needs of small satellites, a multi-mode thruster was designed to be operable in either catalytic chemical or electrospray electric mode by Dr. Steven Berg, under the direction of Dr. Joshua Rovey. Comprehensive information regarding this thruster can be found in Berg[23]. The M^3 and APEX satellite missions will be used to demonstrate the capabilities of the thruster and have the potential to result in advancing the thruster's TRL to 7. For the mission to be successful, the thruster requires integration with a compatible propulsion feed system and satellite bus.

The 1U thruster uses a single, "green" propellant and feed system. Many previous industry-developed multi-mode thruster designs included separate propellants for each mode or separate feed systems. The propellant is a mix of two ionic liquids, one fuel and one oxidizer. The thruster consists of a thousand platinum microtube emitters (fewer microtubes can be used if full system capability is not required by the mission). During the chemical mode, these microtubes are heated and act as a catalyst (in place of the more common catalyst bed). During the electric mode, a voltage is applied at the end of the microtubes to induce an electric field that extracts ions from the propellant. The chemical mode is capable of providing one newton of thrust with a specific impulse (I_{sp}) of 180 seconds, and the electric mode is capable of providing one millinewton of thrust with an I_{sp} of 800 seconds. This combined system leads to significant mass and propellant savings compared to a satellite with separate chemical and electric propulsion systems, resulting in a single system with similar capabilities. Two separate propulsion systems are not generally feasible for a CubeSat, due to mass and volume constraints. A diagram of the thruster and its position on a 6U satellite bus are shown in Figure 2.6.

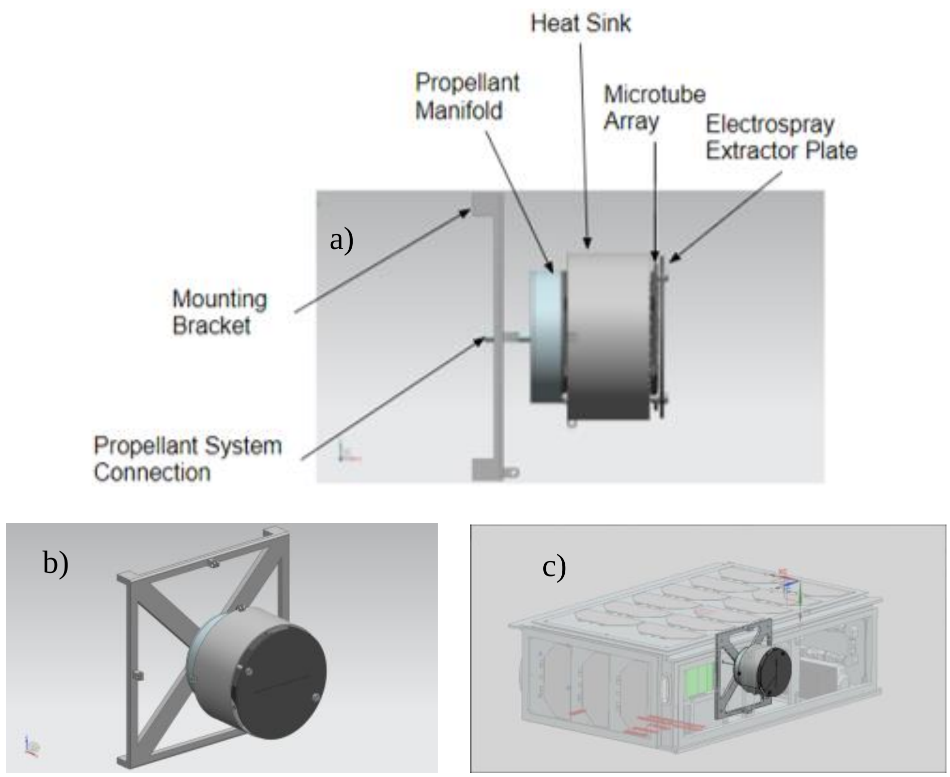

Figure 3.1. Multi-Mode Thruster a) Side-View, b) Front-View, and c) Placement on 6U Satellite Bus.

The thruster is composed of a mounting bracket to attach the thruster to the satellite frame, a propellant feed system connection (which is composed of a 1/8" diameter stainless steel tube), a propellant manifold to distribute the propellant, a heat sink to divert heat (large amounts of which are produced when the satellite is operating in chemical mode) away from the more sensitive system components, a microtube array to control the flow of the propellant, and an extractor plate energized to the voltage needed to extract ions during the electric mode.

The novel design of the thruster demands an equally unique set of system requirements. The thruster requires 3400 V, similar in magnitude to the voltage created by some power line transformers, to be supplied to the extractor plate in the electric electrospray mode. The rest of the system must be shielded. The ionic nature of the propellant limits the material selection for any system components with which contact is made. Initial work suggested that the system would need to operate at 200 psi, twice that allowed by International Space Station (ISS) regulations. Thus, the thruster is only appropriate for applications where the orbit is well-above or well-below ISS orbit.

3.2. SATELLITE BUS

The satellite bus was developed to address the unique requirements of the thruster system, including demonstrating that the thruster was capable of integration in a 3U and 6U class satellite structure. Challenges, in addition to mass and volume constraints, of integrating the thruster into a CubeSat include: generating sufficient power and voltage, determining/controlling attitude of the spacecraft on-orbit, thermal management, and validating the performance of the thruster. One key challenge is providing the power required by the thruster system and the voltage required for the electric mode without compromising power to other subsystems of the spacecraft. During the electric burn mode, the satellite requires 3400 V in a compact circuit. A second challenge of providing this voltage is reducing the size of a power processing unit such that it fits within a 3U or 6U form factor with the other satellite components while mitigating electromagnetic interference. Other “power hungry” components potentially include reaction wheels, magnetorquers, flight boards, and communications radios. To meet power requirements, deployable solar panels were considered, which would add additional constraints to the attitude, possible slew rates, and pointing of the satellite, along with those required by the communications system. Any thermal effects created during the chemical burn mode must be conducted through the structure and radiated into space. Another challenge is obtaining the hardware for a propulsion system of this size. As most propulsion systems are designed for larger spacecraft, commercial components are often not available for purchase. Finally, validating thruster performance is limited by the accuracy of the guidance, navigation, and control hardware, such as the GPS and IMU units, and the capabilities of the thruster itself. With these considerations in mind, the satellite bus designs described in 3.2.1. and 3.2.2. were produced.

3.2.1. M^3. As a 3U CubeSat, space and volume are of particular concern. The scope of the M^3 mission was tailored and limited to operating in electric mode only. The primary success criterion is defined as showing that the thruster operated but excludes characterizing how *well* the thruster operated. This reduced the amount of propellant needed, the complexity of switching between burns, communication requirements, GNC control required, and shielding required. Accordingly, the M^3 mission objectives follow as:

1) Demonstrate multi-mode thruster operation in electric mode using a single propellant;

2) Demonstrate on-orbit functionality of a multi-mode thruster in electric mode.

3.2.2. APEX. APEX is a 6U CubeSat that houses a full thruster, operable in both chemical and electric modes. Additionally, the thruster performance will be quantified during the APEX mission through both feed system data and measured orbit changes. Additional information regarding orbital maneuvers can be found in Morton and Withrow26, and thruster validation through feed system data is further discussed in Section 3.3. This validation requires significantly more Guidance, Navigation, and Control (GNC) capabilities and requires large quantities of propellant, compared to the M3 mission. Additionally, more mission flexibility and operator involvement require more complex Command and Data Handling (C&DH) and Communication (COM) systems. Additionally, operating in electric mode for longer time spans requires that the current be alternated between positive and negative charges to avoid charge build up in the plasma plume which could result in a damaging arc discharge back to the spacecraft. The APEX mission objectives then follow as:

1) Demonstrate integration of multi-mode thruster technology into a small satellite architecture

2) Determine the on-orbit performance of the multi-mode thruster during chemical and electric mode burns

3.2.3. Feed System Components. The propulsion system will be integrated into both a 3U structure and 6U structure, demonstrating that such propulsion systems can be practically implemented into most smallsats. Two main challenges must be addressed for successful integration of the propulsion system into the satellite: packaging the system to fit within the volumetric constraints of the structure and sourcing/qualifying a vessel to store the propellant. The propulsion system hardware must be as compact as possible as the CubeSat structural envelope only contains 3,000 cm^3 of total volume for the 3U (M^3) and 6,000 cm^3 for the 6U (APEX - See Figure 3.2). Current methods for conserving volume include optimizing tubing size and feed system configuration. Additionally, the feed system must be constantly adjusted to updated requirements as thruster development progresses.

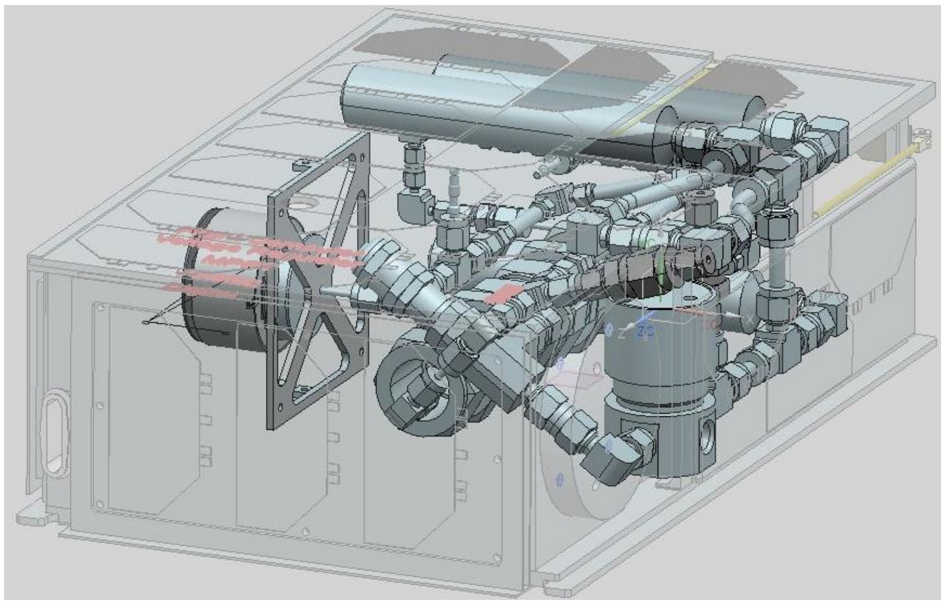

Figure 3.2. Propulsion Feed System and Thruster Layout in APEX CubeSat.

Table 3.1. Feed System Constraints

Constraint	Design Choice
Structures	
Volume (3U or 6U)	Minimize volume, leaving room for additional payload and bus components
Length of structure	Propellant storage tube could not exceed length of longest straight section of structure
Mass	Least dense materials used when possible
Payload	
Propellant	1) Non-corrosive material for all components in contact with propellant 2) Leak-proof storage/delivery
Pressure	1) Deliver 200 psi to thruster

Table 3.1. Feed System Constraints (Cont.)

Constraint	Design Choice
Pressure Cont.	2) System originally designed to withstand 1000 psi on pressurant side and 200 psi operating pressures on propellant side
Integration to thruster	Exit tubing size (1/8" diameter)
Design in-progress	Changing requirements require design decisions to be made with a large margin of safety and flexibility
Power	
Charged during electric mode	Insulated propulsion system
Thermal	
Propellant must be heated to 373 K	Appropriate heat sink path for chemical burn, heating pad on propellant storage tube
System (Internal and External)	
Cost	Customization/hardware limitations
No student-qualified components	COTS components
No welding (UNP)	SAE connections, Compression fittings
3 (NASA) or 4 (DoD) inhibits between propellant and outside of satellite	Solenoid valves
Timeline	Functioning by FSR (APEX), 18-month project timeline for USIP
Collaboration with Froberg Aerospace	Proprietary thruster specifications
Turn-Over	Graduate students involved for consistency and training

3.2.3.1. Propellant storage. A challenge was encountered in sourcing a space-qualified propellant tank that meets the two CubeSat form size constraints. For example, the original thruster design in the APEX satellite required 150 cubic centimeters (cc) of pressurant and 300 cc of propellant to achieve the desired maneuver durations and meet full mission success criteria for the 6U mission. These volumetric requirements have been updated with additional thruster development to 5 cc and 75 cc, respectively. The diameter of the propellant tank must be less than 10 cm to account for the depth of the structure. Additionally, the length of the tank is constrained by its location in the structure. The mass in the propellant tank will change throughout the mission as propellant is consumed, causing the center of mass of the satellite to change. As such, it is advisable to place the propellant tank lengthwise along the thrust axis in order to ensure that an excessive moment is not created by the changing center of mass, thereby perturbing the attitude of the satellite. This placement restricts the length of the propellant tank to be no more than 20 cm. In order to be compatible with the custom ionic propellant, the tank must be made of a non-reactive material such as stainless steel. The tank was designed to be rated to the pressurant value (in case of a regulator failure) and have a factor of safety of 4.0. A conservative design approach was used under the assumption that the CubeSat would be a secondary payload. A commercial-off-the-shelf (COTS) tank was unable to be found, therefore the team designed the needed hardware. However, it became clear after APEX's Preliminary Design Review, that the team would encounter excessive scrutiny from launch safety with a student-built pressure vessel.

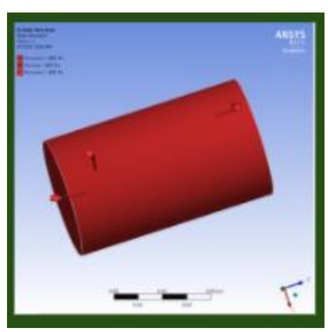

Figure 3.3. 300 cc Custom Tank Designed by M-SAT Team Member, Suzy DeWael.

As an alternative approach, the M-SAT team designed a propellant feed system that stores the propellant in the feed system lines, specifically a qualified 1" OD stainless steel tube. The pressurization process is similar to an automotive master cylinder brake system. The pressurant provides a force (analogous to the force provided by pressing on the brake) which actuates the piston head. The piston head creates hydraulic pressure on the fluid (propellant, akin to the brake fluid) that is expelled out of the storage area and provided to the thruster when the inhibits are released. This process is displayed in Figures 3.4 and 3.5.

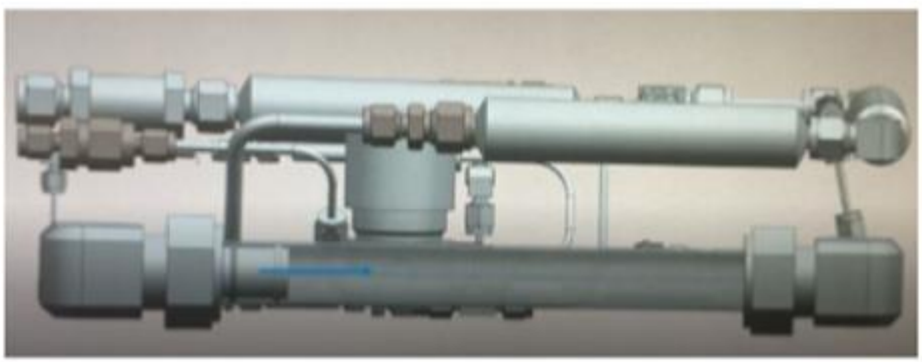

Figure 3.4. Piston Head Starting Location.

Figure 3.5. Piston Head Ending Location.

The stainless steel tube that stores the propellant in place of a traditional tank is ~10.75” in length x 1” outer diameter (OD), carrying 75 cc of useable propellant for APEX. M^3 was designed to pressurize the system in the same manner as APEX. However, the amount of propellant needed is significantly lower (5 cc minimum), as well as the operating pressure (~30 to 50 psi). The required operating pressure is significantly lower because M^3 will operate in electric mode only. M^3 leadership elected to carry extra propellant (~6.6 cc) in order to meet full, rather than minimum success criteria. Thus, the modified propellant storage tube is ~3.4” length x 0.5” OD.

If tanks composed of the necessary material are manufactured by a COTS provider in the future, students are advised to perform a trade study to evaluate if the number of connections and leak rate could be reduced with the COTS component.

3.2.3.2. Pressurant tank assembly. The original thruster design for the APEX mission required a pressurant tank assembly with an internal volume of at least 150 cc. The tank needed to be rated to 375 psi with appropriate factor of safety[33]. Stainless steel and titanium were determined to be acceptable materials to prevent potential chemical reactions and meet the factor of safety requirements.

A single pressurant tank could not be sourced meeting APEX’s requirements. Therefore, it was decided to connect three of Swagelok’s 50 cc sample cylinders (Swagelok SS4CDTW50) end-to-end as one system. They have a combined pressurant volume of 150 cc and a pressure rating of 1000 psia (Figure 3.6). The chosen pressurizing agent is nitrogen gas as it does not react with the ionic fluid propellant used for the payload and is more efficient than other considered inert gases (helium and argon).

Upon further thruster development, the operating pressure and amount of propellant needed were significantly reduced from the original, provided specifications. M^3 will use a single, 50 cc sample cylinder and is expected to operate ~50 psi. The final pressurization value will be determined after internal losses are experimentally confirmed. The final operating pressure for APEX is to be determined, based on thruster requirements. At the time of this writing, the team is optimistic that the APEX operating pressure may be reduced to 100 psi and 75 cc. For this operating pressure and volume of propellant, the pressurant tank could be resized to an internal volume of 50 cc, with an expected pressurant tank value of 151.3 psi (See Appendix B.) This value may be adjusted to account for a higher/lower operating pressure or to account for internal losses.

Figure 3.6. A Single Swagelok 50 cc Sample Cylinder.[27]

3.2.3.3. Solenoid valves. Solenoid valves should be less than 10 cm in length, rated to a minimum of 200 psia (or current operating pressure), made of stainless steel or titanium, and have AN, SAE, or Swagelok fittings.

For the APEX satellite, four solenoid valves are used to control propellant flow to the payload (thruster). A total of four valves are utilized for redundancy, ensuring that propellant does not reach the payload prior to deployment from the Planetary Systems

Corporation Canisterized Satellite Dispenser (PSC CSD). A fifth valve is placed between the propellant storage system and pressurant side and prevents the propellant from being pressurized until commanded to do so by the flight computer. Additionally, the solenoid valve and check valve ensure that if a propellant leak were to occur during launch, the propellant would not be able to reach the components of the system under higher pressurization (on the pressurant side of the system). The top selection is a derivative of Lee Company's IEPA series (IEPA1221241H, Figure 3.7) due to its size, pressure rating, and hold voltage of 1.6 V that can be supplied within the power budget. The valve is 1.3" long with a 1" port on each side and 0.25" diameter. The pressure rating is 800 psia with a proof of pressure of 1600 psia and burst pressure of 2400 psia. The solenoid valve layout for M^3 is identical to APEX with one exception: three solenoid valves will be used to control propellant flow to the payload, instead of four. The variance in the design is a result of different inhibit requirements for DoD-sponsored (APEX) versus NASA-sponsored (M^3) missions.

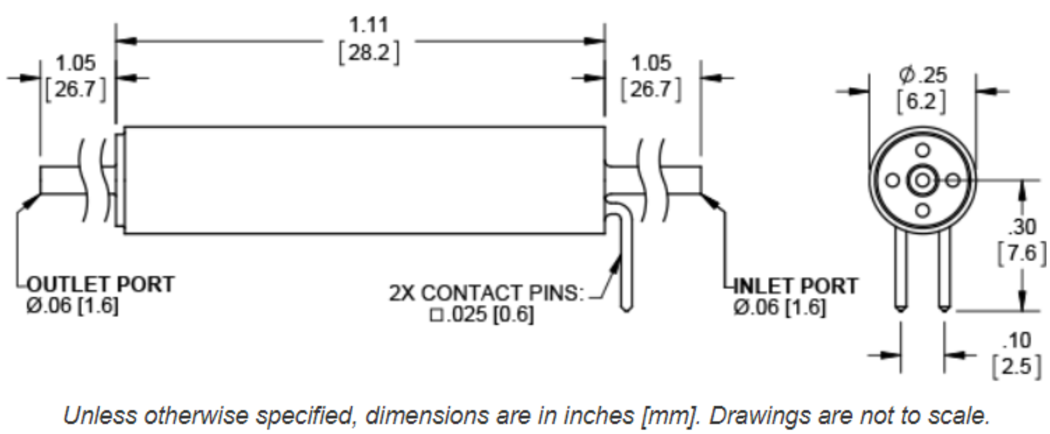

Figure 3.7. Lee Company's IEP Series Solenoid Valve.[28]

3.2.3.4. Check valves. Check valves must be rated to at least 200 psia (or current operating pressure), made of stainless steel or titanium, and have AN, SAE, or Swagelok fittings. Length should be minimized if possible.

Check valves control the fluid flow through the system, guaranteeing flow in only one direction. Two check valves (one in-line and one at a T-junction) will be used to fill the pressurant tanks and the propellant storage tube prior to launch. Both check valves will then be locked. The third check valve is in-line and will help mitigate the risk of a propellant leak, as it would prevent propellant from being able to reach components under higher pressures (on the pressurant side of the system). A fourth check valve will be placed just before the thruster as another redundancy to prevent heated propellant from flowing backwards into the propellant storage tube should a malfunction occur. Swagelok's SS-CHS2-1 and SS-CHS4-1 were chosen because these models have the ability to withstand 6000 psia of pressure, are made of stainless steel, have 1/8" and 1/4" connections (standard for the system tubing), respectively, and are relatively compact with a length of 2.14".

Figure 3.8. Swagelok SS-CHS2-1 Check Valve.[29]

3.2.3.5. Pressure relief valve. For APEX only, a pressure relief valve will be placed in the pressurant side of the system between the pressure regulator and check valve. This will mitigate pressure build up over 200 psi in the tubing and ensure that if pressure build up occurs, nitrogen will be released and contained within the launch canister instead of propellant being released. The pressure valve chosen is the Generant Vent Relief Valve High Pressure (VRVH). This valve can be factory set and locked between the range of 150 and 600 psi and has an option of either a 1/8” or 1/4” NPT connection. This is the only NPT connection in the system and was selected because of its standard connection size. This part was not included in the M^3 layout due to volume constraint and lower operating pressures.[30]

3.2.3.6. Pressure regulation. For APEX, a pressure regulator will be used to adjust the pressurant to the desired operating pressure which may differ for each mode. A pressure regulator is not required by the M^3 system because M^3 only operates in electric mode at lower operating pressures.

The inlet of the pressure regulator needs to be rated to the current pressure of the pressurant tank assembly as it will experience higher pressure values from the pressurant tank than the outlet will experience. The regulator needs to be able to regulate down to 200 psi (or the current system operating pressure). The regulator should be made out of stainless steel or titanium and use either AN, SAE, or Swagelok fittings.

The pressure regulator will adjust the pressure from the pressurant tanks to 200 psi (or the current system operating pressure). The Emerson Electronics’ mechanical regulator (BB-66PL3KEB4) was chosen to achieve this pressure drop (Figure 3.9) for the NS-9 competition. The BB-6 series of pressure regulator outputs the propellant at 220 psi and is made of stainless steel. This model was used in the first engineering design unit of APEX. Electric pressure regulators were considered but determined to add risk due to the timeline and lack of team experience. However, with more experience or a longer timeline, it is strongly suggested the team consider an electric version of regulation through either a COTS component or by implementing a “bang-bang” set-up for the Nanosat-10 competition. See Section 5.2.1 for more detail.

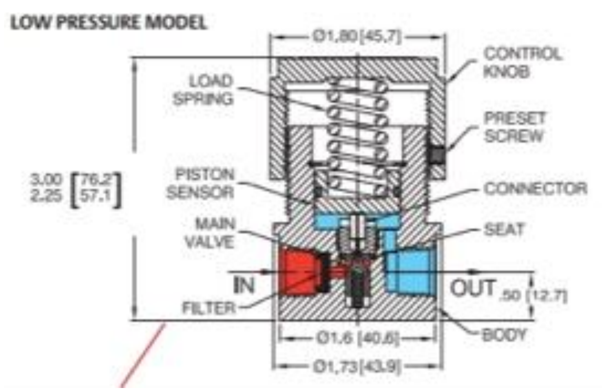

Figure 3.9. Mechanical Pressure Regulator Internal View.

3.2.3.7. Thermal sensors. Thermal sensors must be able to attach to the desired component in a way that will withstand temperature changes and vibration. The thermal sensor selected by the thermal subsystem is the Maxim Integrated model DS18S20. This model was chosen for ease of integration to the flight computer (through the Flight Computer Interface Board), small size, appropriate temperature operating range, and previous team experience. A thermal sensor will be placed on the propellant storage system to monitor the temperature of the propellant. Additionally, a thermal sensor will be placed on each solenoid valve between the propellant storage system and thruster. If the solenoid valves become heated beyond the desirable temperature range, the flight computer will send a command for the solenoid valves to close so that heated propellant cannot reach the propellant storage tube during a system malfunction.

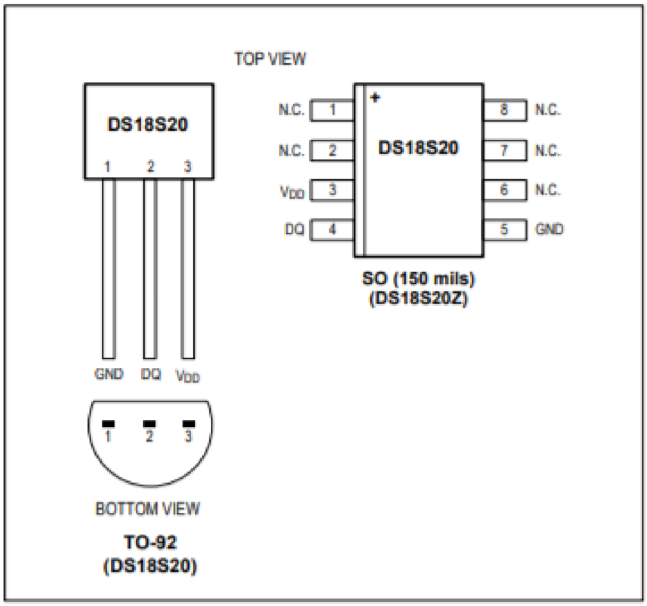

Figure 3.10. Maxim Integrated Model DS18S20 Pin Layout.[31]

3.2.3.8. Pressure transducers. Per structure restrictions, the pressure transducer should be less than 10 cm long. It should be rated to at least 200 psia (or the current operating pressure), made of stainless steel or titanium, and have AN, SAE, or Swagelok fittings.

Pressure transducers will be used to measure the pressure in the feed system. One will be placed at the inlet to the propellant storage system to measure the pressure of the stored propellant. The second transducer will be placed at the inlet of the thruster to provide pressure readings as flow enters the payload. TE Connectivity's miniature EPRB-1 Pressure Transducer has been chosen. This sensor was chosen because of its size (0.11

mm diameter diaphragm), ability to withstand high pressure (68.95 MPa), and its required supply voltage (5 V) that is consistent with the voltage supplied by APEX's power system.

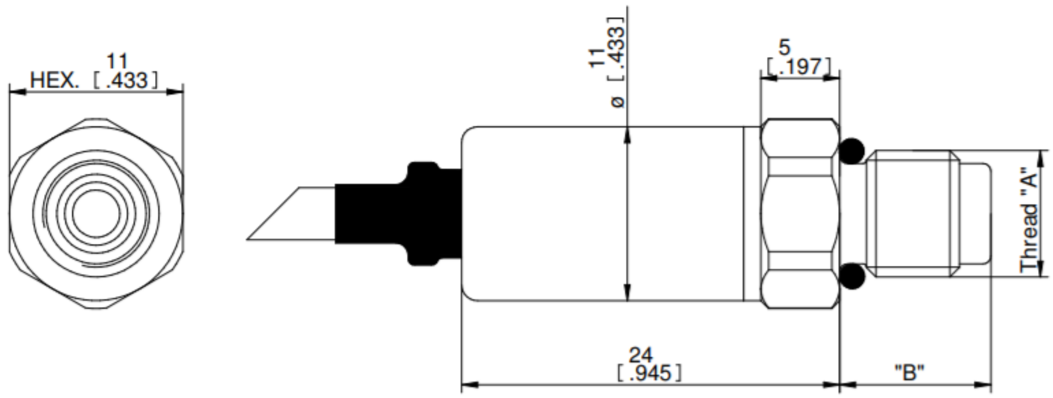

Figure 3.11. TE Connectivity's Miniature EPRB-1 Pressure Transducer Dimensions.[32]

The reader is referred to the TE Connectivity catalog for Thread "A" and length "B" options. In the APEX design, Thread "A" is a 10-32 UNF-2A and length "B" is 0.34 inches.[32]

3.2.3.9. Tubing. To avoid microfluidic effects, such as flow instability and viscosity changes, the tubing should be at least 1/8" and rated up to 1000 psia, but size and mass should be minimized. Exceptions may be made to integrate with components with 1/16" connections.

McMaster-Carr 316 stainless steel tubing will be used. A maximum pressure range is available for this tubing between 1100 and 1600 psi. Lines will be 1/8" diameter throughout for consistency except where Swagelok unions will be used to transition to the size of component connections.

3.2.4. Feed System Layout. Figure 3.12 displays the layout of the entire APEX propellant feed system. The pressurant tank contains nitrogen gas, at 375 psi initially, that is regulated to 200 psi by the mechanical pressure regulator. (These values are subject to change due to updates in thruster development.) Check valves are placed at junctions for loading pressurant and propellant near the respective storage areas. Two pressure transducers are also placed at T-junctions just upstream of the propellant storage area and just upstream of the thruster to provide feed system diagnostic data. The in-line check valves and solenoid valves act as inhibits for both the pressurant and propellant, inhibiting reverse flow and ensuring the propellant is not pressurized or provided to the thruster until the appropriate commands are received from the flight computer. The quantity and placement of the inhibits are the result of a full system HAZOP analysis performed by M-SAT team members.

Prior to APEX's integration into the launch vehicle, the three pressurant tanks and propellant storage system will be filled with 150 cc of nitrogen gas at a total pressure of approximately 375 psi and 75 cc of propellant, respectively, through check valves. One check valve is located just downstream of the pressure tanks and another is located just downstream of the propellant tank. At this time, the solenoid valves will be closed to prevent any gas or fluid flow. After the tanks are filled, the feed system will be pressurized to 200 psi downstream of the pressure regulator and upstream of the first solenoid valve. Pressure in the feed system will be measured with the pressure transducer just before the propellant tank. When APEX performs a burn, power will be supplied by C&DH to open the four solenoid valves acting as inhibits and allow propellant to flow to the payload. A fifth solenoid valve will be opened allowing the pressurant to move towards the propellant storage system. The propellant feed system will cease providing propellant to the payload when power from C&DH to the solenoid valves is cut off.

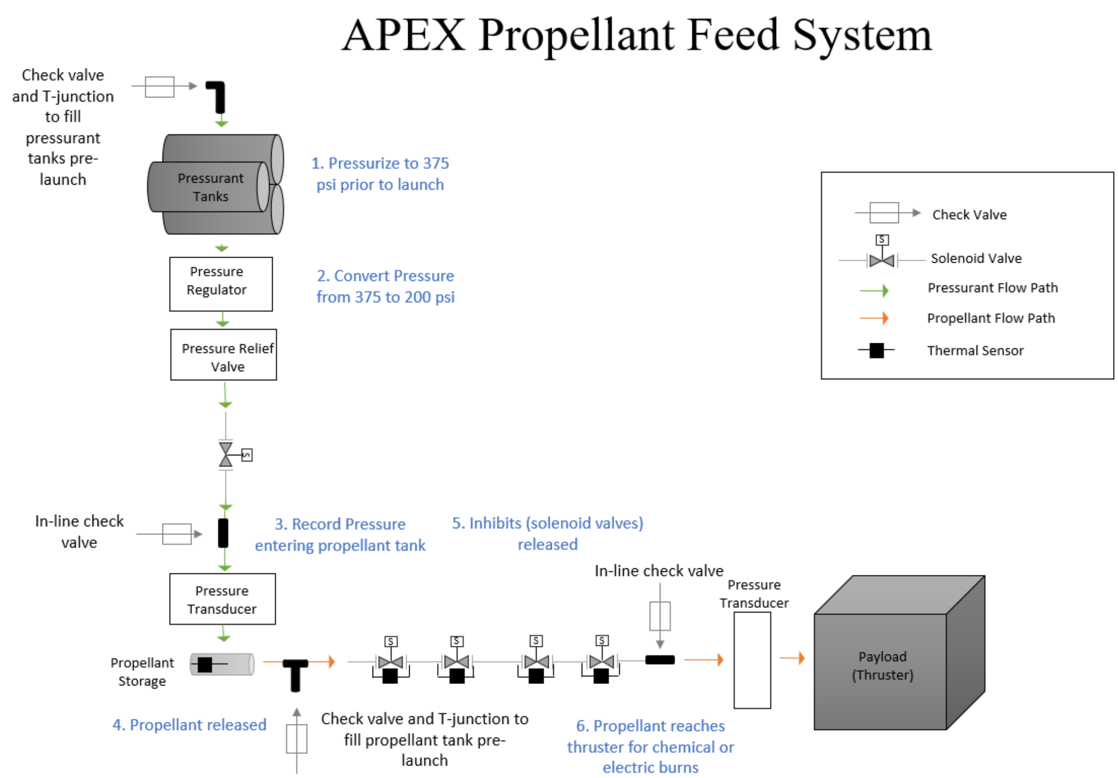

Figure 3.12. APEX Feed System Layout.

The M^3 feed system (Figure 3.13) is laid out in the same configuration, with a few alterations. The M^3 mission requires less propellant and, consequently, less pressurant. Thus, only one Swagelok sample cylinder is needed to contain the pressurant. The propellant storage tube holds 6.6 cc of propellant and is significantly shorter than its APEX counterpart. NASA versus DoD sponsorship requires three (instead of four) inhibits, and because of the lower operating pressure, the pressure regulator and pressure relief valve were omitted to meet volume constraints.

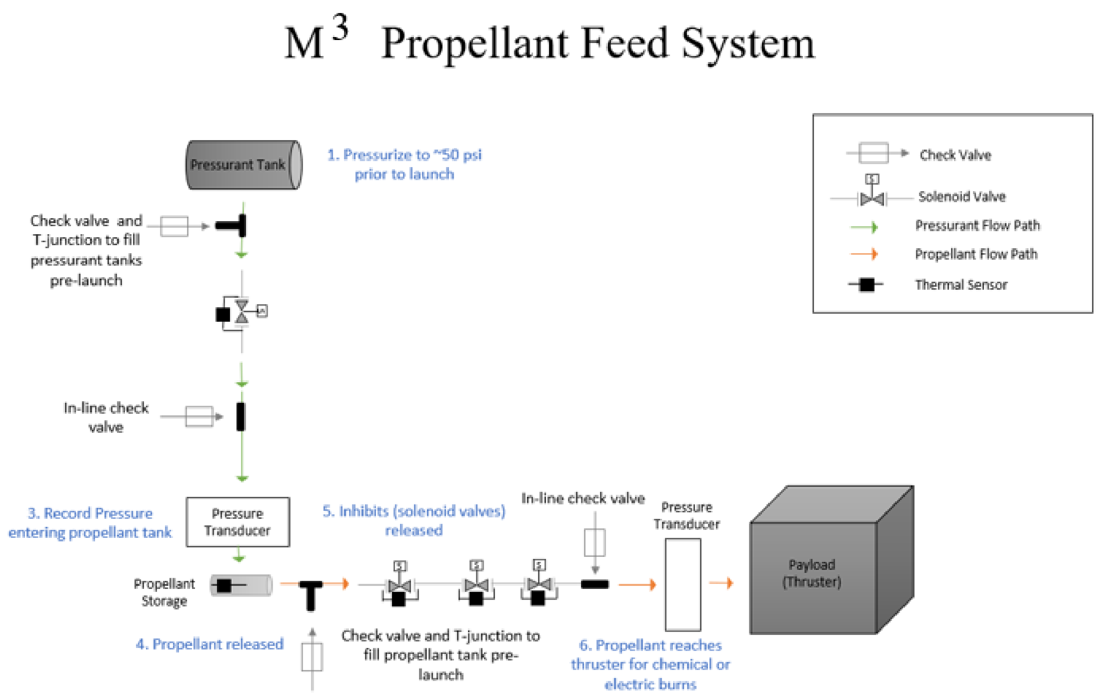

Figure 3.13. M^3 Feed System Layout.

3.3. VALIDATING THRUSTER PERFORMANCE

Pressure and temperature from the propulsion feed system, along with orbital maneuvers, will be used to qualify the thruster throughout the APEX mission. Pressure values will be recorded by pressure transducers in the feed system, and a thermal sensor will be placed on the outside of the propellant tank so that temperature values can be compared to readings from thermal couples placed in the thruster itself. These pressure and temperature readings will be compared to those obtained during ground testing.

3.4. RELEVANCE

The NASA Goddard 2017 State of the Art (SOA) for Small Satellite Propulsion Systems presentation stated that the desired qualities for SOA Smallsat Propulsion

Systems are "lowest cost possible and simple design feasibility"[5]. The feed system designed by the M-SAT team for these technology demonstration missions meets the criteria above by eliminating the propellant tank (which is often difficult/expensive to qualify) and designing a simple pressurization method that enables university teams (and other modest-budget organizations) to include propulsion systems in CubeSats.

3.5. RELAXING STUDENT CONSTRAINTS

Many design decisions were driven by additional regulations imposed on student-designed pressurized systems. When integrated into future industry-level systems not subjected to these onerous constraints, a more efficient design should be used. Changes include using customized pressurant and propellant tanks and connections, removing redundant inhibits, and integrating a more precise regulation system.

3.5.1. Component Changes. With an increased budget, technical experience, and fewer restrictions the following design alterations should be considered for an "industry version" of the APEX or M³ feed system.

3.5.1.1. Propellant storage. The propellant tank should be made of a material that is compatible with the ionic liquid such as stainless steel or titanium. Pressure vessels certified for spaceflight use must be designed with a safety factor built in. The factor of safety should correspond with the Proof Pressure which is given by

$$\mathrm{P_{proof}} = \left[\frac{1 + \text{Burst Factor}}{2}\right] \times \text{(Max. expected operating pressure)} \quad \text{or}$$

$$= 1.5 \times \text{(Max. expected operating pressure), whichever is lower.}$$ [33]

The burst factor is dependent upon material but is normally 1.5, unless otherwise stated. As long as stainless steel meets these safety requirements, it is most likely ideal for a

CubeSat mission as it is less expensive than titanium. Morton and Withrow[26] determined that 75 cc of propellant was sufficient for the APEX mission, as the mission is defined in Chapter 1. If more propellant was required, the propellant tank size would increase and the appropriate operating pressure and pressurant tank volume would need to be fitted to the mission needs (See Appendix B).

3.5.1.2. Pressurant tank assembly. The pressurant tank would also need to be constructed with the appropriate factor of safety[33]. The APEX design chose to also use stainless steel tanks to hold the pressurant, even though the pressurant is an inert gas. If a propellant leak managed to flow into the pressurant side of the system, the propellant leak would not corrode the stainless steel pressurant tank. With risk lowered by a more conventional propellant tank design or a higher acceptable risk tolerance, the designer could consider other materials, such as aluminum, to save mass. The pressurant storage area should also be one tank capable of holding the full volume capacity of the needed pressurant to reduce the number of connections and corresponding opportunities for leaks. Three tanks were connected to form one large tank in the APEX design in order to allow the use of stainless steel, COTS components with a compression fitting for the NS-9 competition. The three-tank design was replaced with a single stainless steel tank with an NPT fitting for the NS-10 competition and will undergo extensive leak-proof testing.

3.5.1.3. Number of inhibits. The APEX satellite has four inhibits (solenoid valves) between the propellant and the thruster, per UNP requirements. However, three inhibits are the industry standard.

3.5.1.4. Pressure regulation. Electrical-based regulation systems such as proportional-integral-derivative (PID) or “Bang-Bang” systems would have less mass and provide more flexibility when regulating the feed system than a mechanical regulator. While more complex, these systems would likely be more appropriate for industry use. See section 5.2.1 for more information regarding the performance of mechanical regulators versus electrically-based systems.

3.5.1.5. Revised layout. Figure 3.14 is an adaptation of Figure 3.12 incorporating the suggestions above. Depending on the risk tolerance of the mission, the check valve upstream of the propellant tank and pressure relief valve could also potentially be removed for a minimalistic design.

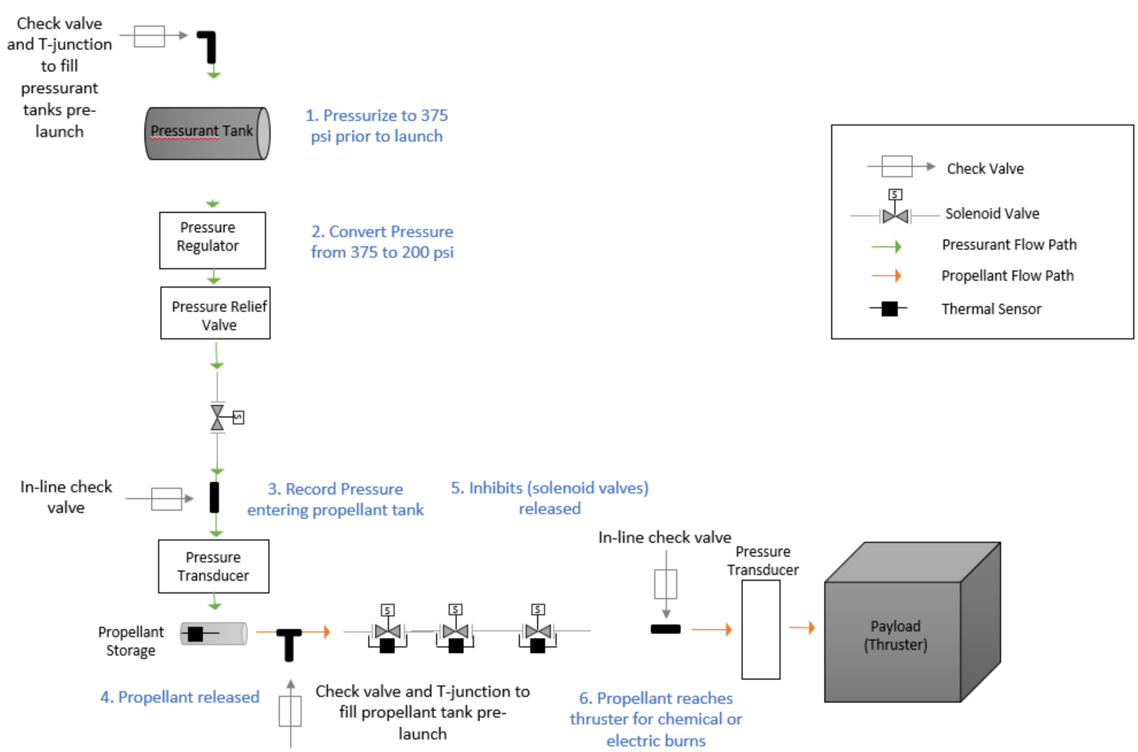

Figure 3.14. Recommended Feed System Layout for Industry Application.

Another noteworthy consideration in a revised layout is operating pressure. If the stored pressure is less than 100 psi, the satellite could be eligible for deployment from the

ISS. This would significantly increase launch opportunities and potentially mitigate additional time and cost from the launch safety approval and launch perspective.

3.5.2. Mass and Volume Savings. The current APEX feed system occupies a volume of ~3000 cc (3Us) (Figure 3.15). Fizell[34] reports that removing some of the student constraints listed above allows the system to be condensed to 2190.20 cc and 0.9857 kg. If the mission were risk tolerant and redundancies could also be removed from the system, the system would still be functional at 1428.75 ccs and 0.6369 kg (Figure 3.14). These reductions make a CubeSat mission containing the feed system and thruster with an additional primary payload feasible.

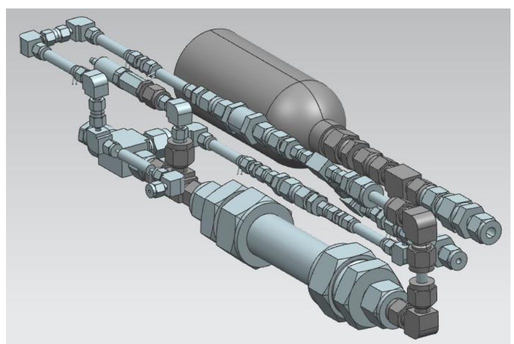

Figure 3.15. Feed System Incorporating Student Constraints.[34]

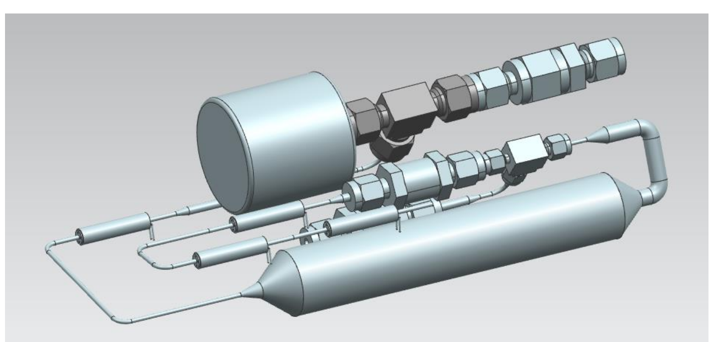

Figure 3.16. Minimalistic Feed System.[34]

4. PROPELLANT PRESSURIZATION PROOF-OF-CONCEPT

A “proof-of-concept” test set-up was designed by the author to verify the feasibility of storing the propellant in tubing and pressurizing it. Though the author considered other aspects of the feed system in testing, the scope of experimental data was limited to validating the concept of pressurizing the propellant stored in the storage tube. The key parameter measured to validate the concept was pressure at room temperature.

4.1. TEST SET-UP

The test set-up included an argon tank connected with 1/4” tubing to a cross fitting. A pressure transducer and ball valve were connected perpendicular to the flow at the fitting to measure the pressure of the pressurant (argon) upstream of the storage tube and provide a means to relieve the gas from the system. The other two branches of the fitting were used for flow in and flow out. The 1/8” diameter tubing and a 1/8” to 1/2” converter admit the pressurant into the storage tube. A custom piston head fitted with two O-rings recessed in pre-cut grooves separated the pressurant from the simulated propellant (either water, olive oil, or isopropyl alcohol) which was preloaded in the system. The storage tube contained the piston head and preloaded simulated propellant.

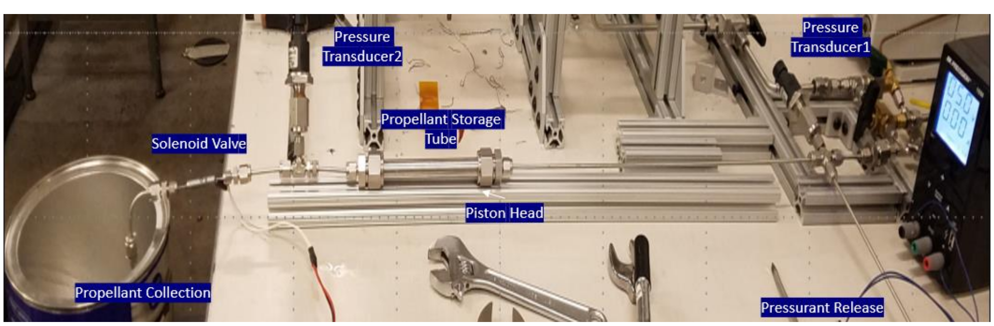

Figure 4.1. Pressurization Proof-of-Concept Test Set-Up.

The actual propellant was not used because the custom nature of the propellant made it impractical to obtain in the quantities needed for proof-of-concept testing. The 1/2” diameter storage tube was followed by a 1/2” to 1/8” reducer and a T-connection. The T-connection contained a second pressure transducer to measure the pressure of the simulated propellant downstream of the storage tube, reflecting the pressure of the propellant at the thruster interface. A solenoid valve was then used in conjunction with two 1/8” to 1/16” reducers to inhibit the flow from exiting the system until it was pressurized, at which point the inhibit was released. Finally, a drain was located at the position of the thruster (instead of the thruster itself) and allowed the simulated propellant to enter an external collection unit. The solenoid valve was connected to a power supply and required an excitation voltage of 12 V and a holding voltage of 1.6 V. The two pressure transducers were connected to a MyDAQ system, allowing pressure readings to be collected with LabVIEW. Additionally, T-slot channel bars were used to structurally support the tubing and components as needed.

4.2. SYSTEM VALIDATION (TESTING)

Concept feasibility was determined by comparing pressure and mass loss in the test set-up to predetermined acceptable levels. Mass loss was determined by pre-measuring the volume of the simulated propellant loaded into the set-up and determining the mass based on the density of the simulated propellant using

$$Mass = Volume * density \tag{1}$$

The external collection unit was placed on a digital scale under the test set-up drain, and the scale was tared. The unit collected the simulated propellant from the drain and the pre-loaded and collected mass values were compared. The pressure transducers placed before and after the propellant storage area allowed the author to determine the pressure loss due to the pressurization and movement of the piston head. Three different liquids were used as simulated propellant. Initial testing was performed with purified water; the test was then continued with isopropyl alcohol because it has a greater viscosity than the purified water. After the connections and set-up were verified with these low viscous fluids, olive oil, which is the common substance most similar in viscosity to the custom propellant, was substituted for the propellant in the system. Additionally, the pressure loss through the system was determined by the difference between the set output pressure of the pressurant tank and the pressure transducer reading near the outlet.

4.3. RESULTS

Results of the pressurization proof-of-concept experiment will be discussed.

4.3.1. Mass Loss. The graphs below show the mass remaining in the system after the propellant was pressurized at a variety of pressures. Water was used for the propellant in Figure 4.2, isopropyl alcohol was used in Figure 4.3, and olive oil was used in Figure 4.4. The main purpose of this test was to show that the propellant remaining in the system is below the acceptable loss for the mission.

The acceptable loss was determined to be 10% of the preloaded mass. This value was selected because it allows mission success criteria to still be met for the scoped mission. Figures 4.2, 4.3, and 4.4 show that the mass loss in the operating range, 150 to 200 psi, is well below the acceptable loss for each fluid tested. Additionally, the trend of the data shows a correlation between mass loss and operating pressure, as the two are inversely related. This is advantageous for the current missions (APEX and M^3) which plan to operate within the 150 to 200 psi range. The propellant side of the system can be assumed to be incompressible and steady (Appendix A). Temporarily, neglecting friction, Bernoulli's Equation can be used to characterize the behavior of the system. Bernoulli's principle states that an inverse relationship exists between pressure and velocity, thus as pressure increases, velocity decreases.

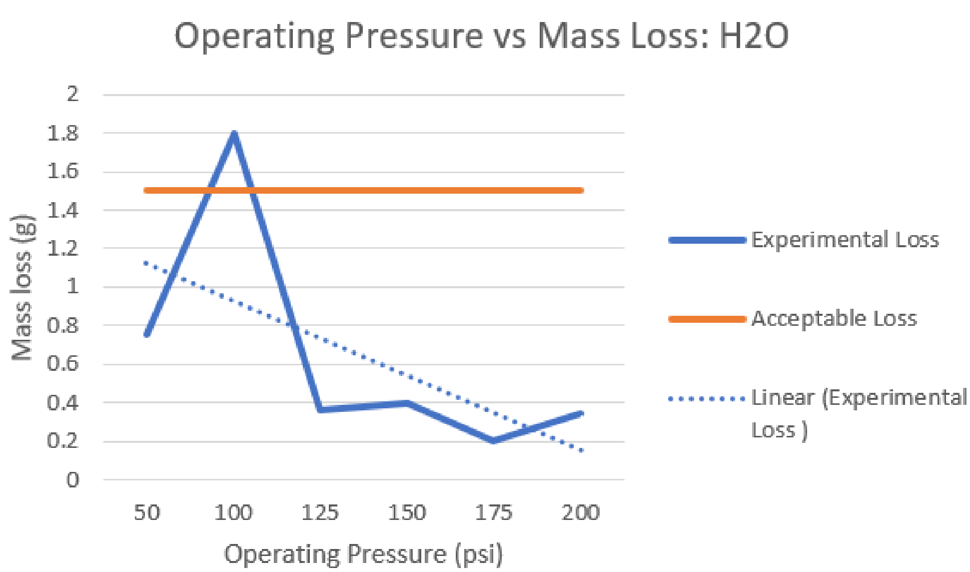

Figure 4.2. Mass Remaining After Pressurization-Water.

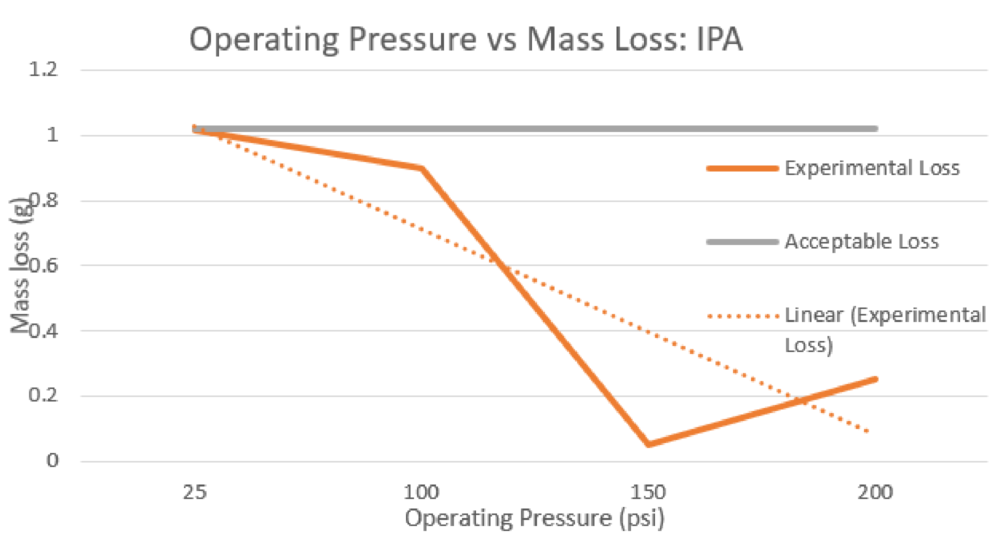

Figure 4.3. Mass Remaining After Pressurization-IPA.

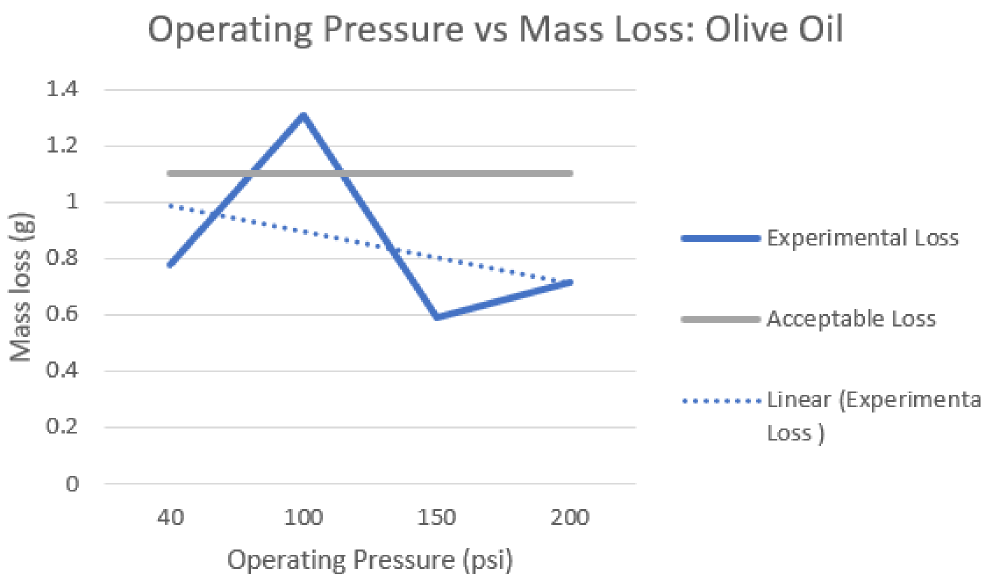

Figure 4.4. Mass Remaining After Pressurization-Olive Oil.

Starting with Bernoulli's equation,

$$P_1 + \frac{1}{2}\rho v_1^2 + \rho g h_1 = P_2 + \frac{1}{2}\rho v_2^2 + \rho g h_2 \tag{2}$$

and assuming no changes along the vertical direction and constant density,

$$(P_2 - P_1) = \frac{1}{2}v_1^2 - \frac{1}{2}v_2^2 = -\frac{1}{2}(v_2^2 - v_1^2) \tag{3}$$

i.e., due to the relationship between the pressure change and velocity change, an increase in pressure will result in a decrease in velocity.

The length required for laminar flow to fully develop is determined by

$$\frac{l}{D} = 0.06Re_D \tag{4}$$

where *l* is the length the flow needs to fully develop, *D* is the diameter of the tube, and Re_D is the Reynold's number associated with the diameter of the tube defined by

$$Re_D = \frac{\rho v D}{\mu} \tag{5}$$

where ρ is density, *v* is velocity, *D* is diameter, and μ is viscosity.

In laminar incompressible flow, as pressure increases and velocity decreases, Reynold's number decreases and the length required for the flow to fully develop decreases. With fully developed flow along a greater length of the pipe, less mass will be "left behind" in the system. Thus, it follows that at higher pressures, it is expected that less mass is "left behind" due to stagnation than at lower pressures.

The experimental data are not linear (but have a linear trend). Reynold's number was used to determine if transition or turbulent flow accounted for increased loss. However, it was found that all three fluids used to simulate the propellant resided in the laminar range by a large margin at test conditions. (See Appendix A.) Sensor error was considered, but as the change is reflected in both pressure transducer (see Section 4.3.3) and weight scale values, this was determined to be unlikely. The author has determined that the changes are most likely part of a larger trend and that additional data are needed if fully characterizing the trend was required to minimize propellant remaining in the system. At the time of writing, resources were not available to gather such data, and the author addresses this in the Future Work sections. Considering the magnitude of the difference, tool ability/performance, influence of external factors, and human factor should all be taken into account.

4.3.2. Future Work-Mass Loss. The author suggests that the data set be expanded to confirm repeatability and that a sensitivity analysis of thruster performance be completed. Based on the results of such analyses, team members should then consider if it is necessary to record more data points to further characterize the range and trend of the data. Once the full system is analyzed, the velocity of the fluid in this critical area should be re-examined. The author does not expect speeds to approach transition.

However, if transition flow is deemed unacceptable (determined by thruster restrictions) and the propellant approaches this flow region, the team will need to consider adjusting the amount of propellant loaded to account for additional loss or reconfigure the tubing in a manner that would slow the flow but still allow the system to operate within the updated requirements determined through thruster testing. Possibilities for inducing slower speeds include adding a bend in the tubing or expanding the area to slow down the fluid and increase the pressure, similar to a diverging nozzle, before the test section to avoid losses. Further considerations for a larger data set would also need to be made if the operating pressure was significantly reduced by on-going thruster development or range safety concerns. The author advises further characterization of the operating pressure range, including high or low, when the range is finalized by Froberg Aerospace.

4.3.3. Pressure Loss. Figure 4.5 shows the experimental loss in the entire pressurization test set-up (yellow line in Figure 4.5), the experimental loss from the piston head accelerating the simulated propellant only (green line in Figure 4.5), and an analytical loss (calculated using head loss in pipe flow equations) at a constant experimentally determined velocity for only the acceleration of the piston head (blue line in Figure 4.5). The experimental data sets use purified water as the simulated propellant. The magnitude of the propellant's velocity is difficult to determine from current experimental data, leading to a significant error between the analytical and experimental pressure loss values. This will be further discussed in this section.

4.3.3.1. Analytic analysis. The analytical value could not be calculated more accurately without the availability of velocity measurements at each tubing segment. Therefore, a single constant value of 9.12 cm/s was used (an experimentally approximated value for flow in the storage tube) to approximate the magnitude of loss due to major and minor head losses in the pipe. This velocity value corresponds to a constant pressure loss of 0.02731 psi. The author prepared a Matlab code for future students to more completely determine the analytical solution, when velocity values can be determined using numerical simulation of the flow in the feed system. (Appendix B.)

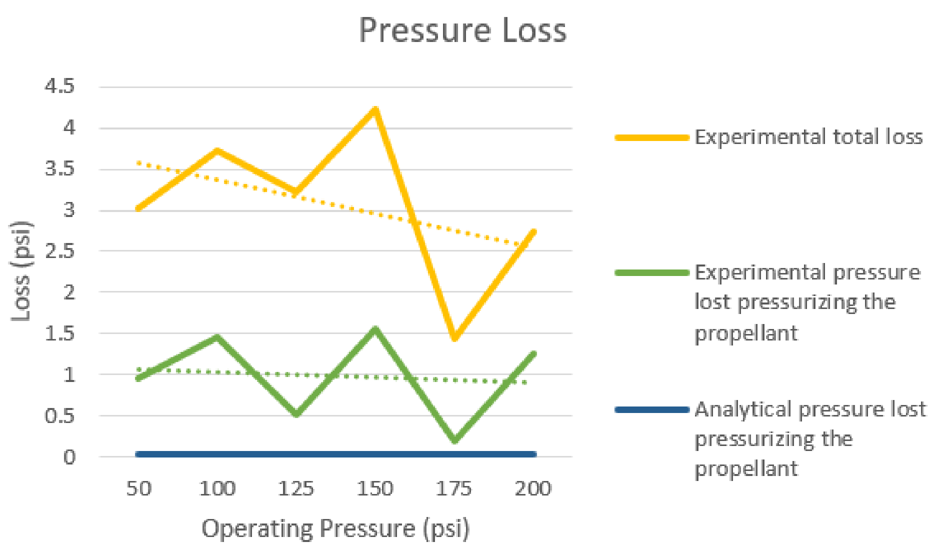

Figure 4.5. Pressure Loss in Pressurization Demonstration.

The analytical calculation was done using two sets of assumptions. The pressurant side of the system was filled with argon gas for the experiment. Therefore, compressible flow assumptions were imposed. A derivative of Bernoulli's equation for compressible, isothermal flow[35] was used to determine pressure loss using

$$w^2 = \frac{A^2}{v * f * \frac{L}{D} + 2\ln\left(\frac{p_1}{p_2}\right)} * \frac{p_1^2 - p_2^2}{p_1} \qquad \text{(solved for } p_2\text{)} \qquad (6)$$

$$\Delta p = p_1 - p_2 + \rho g h_{L_{Min}} \tag{7}$$

$$\text{where} \quad h_{L_{Min}} = k * \frac{u^2}{2g} \tag{8}$$

where *w* is mass flow rate, *A* is area, *p* is pressure, *v* is specific volume, *f* is friction factor, *L* is length, and *D* is diameter.

The second set of assumptions characterize the propellant side of the system where the flow was known to be incompressible and laminar. An experimental velocity of 9.12 cm/s was assumed to be constant throughout the flow. This velocity was determined by measuring the time for olive oil to exit the propellant storage tube at 200 psi (selected because it was the most similar to the propellant at expected operating pressure). The distance from the piston head to the tube was divided by the time. This assumption was justified by a sensitivity check using velocity derived for the Taylor simulation[25], 12 cm/s, and maximum laminar velocity for the chosen fluid. Major head loss, composed of friction effects, and minor head loss, composed from the effects of fittings, bends, the inlet, contractions, and expansions in pipe diameter, were determined and then used to derive pressure loss using

$$h_{L_{Maj}} = f * \frac{u^2}{2g} * \frac{L}{D} \tag{9}$$

$$h_{L_{Min}} = k * \frac{u^2}{2g} \tag{10}$$

$$\Delta p = \rho g (h_{L_{Maj}} + h_{L_{Min}}) \tag{11}$$

where $h_{L_{Maj}}$ is major head loss, $h_{L_{Min}}$ is minor head loss, *f* is a friction factor, *k* is a minor loss coefficient, *u* is velocity, *L* is length, and *D* is diameter. "

Once the pressure loss was determined in both the pressurant and propellant side of the system, the pressure losses were combined. Using a constant velocity was sufficient to show the analytical and experimental calculations were the same order or magnitude but was insufficient for more robust analysis (which would require velocity values at each segment).

4.3.3.2. Experimental analysis. Experimentally, the pressure lost throughout the entire test set-up ranges from 3.04 to 8.36 psi, with the largest loss occurring at an operational pressure of 150 psi. However, pressure lost due to the pressurization process (composed of the storage tube, propellant, and piston head with associated connections) only accounts for 0.42 to 1.48 psi in the chosen operational pressure range of 50 to 200 psi. This corresponds to the difference in the reading between the two pressure transducers in the system. The pressure loss over the entire system corresponds to the difference between the set operating pressure at the argon tank and the last pressure transducer in the system. These losses are well below the defined acceptable limit which is 10% of the current operating pressure. This limit was defined by identifying the allowable pressure loss without significantly affecting performance of the feed system based on experimental testing. In summary, the pressurization method of using a piston head to "push" the propellant to the thruster with regulated pressurized gas is viable from a pressure loss perspective.

Additionally, it is noteworthy that as the operating pressure increases the pressure loss trend decreases in Figure 4.5. This trend is consistent with Bernoulli's principle that indicates an inverse relationship between pressure and velocity. As pressure increases, velocity decreases. If p1 and v1 are the initial pressure and velocity (in this case at rest) and p2 and v2 are the pressure and velocity measured at a point in the flow farther down the pipe, a smaller change between initial velocity (v1) and measured velocity (v2) corresponds with a smaller difference between initial and measured pressure (p1 and p2) . For example, using Bernoulli's equation with no vertical change and negligible change in density,

$$(P_2 - P_1) = \frac{1}{2}v_1^2 - \frac{1}{2}v_2^2 = -\frac{1}{2}(v_2^2 - v_1^2) \quad (12)$$

Given $v_1 = 0$ m/s, $$\Delta p = -\frac{1}{2}v_2^2 \quad (13)$$

If $v_2 = 2$ m/s then, $\Delta p_{mag} = 2$ Pa. However, e.g., if v_2 is reduced to $v_2 = 1$ m/s, then $\Delta p_{mag} =$ 0.5 Pa.

Lower pressure loss at higher operating pressures is reflected in the experimental data sets in Figure 4.5. The experimental data show a linear trend but are not exactly linear. Like the mass data, these pressure values deviate from the linear trend because of tool ability/performance, influence of external factors. Once thruster sensitivity is determined, further characterization should be done in the pressure regions of interest to optimize the set operating pressure.

4.3.3.3. Velocity at a single point. At a single point in the system, where the force from the piston creates work on the fluid as it exits the propellant storage tube, the velocity can be more precisely determined with current information. Using a range of operating pressures (adjusted ~2psi for experimental loss in the upstream tubing), assuming negligible temperature change, and assuming infinitesimally small tubing length where the propellant exits the storage tube area, the differential form of the energy equation for quasi 1-D flow can be used to approximate the velocity.

$$C_p * dT + udu = \delta \mathrm{W} + \delta q \quad (14)$$

Assuming no heat transfer or temperature change,

$$u_2(u_2 - u_1) = W_{1 \rightarrow 2} = \text{F*distance} \tag{15}$$

where u is velocity, W is work, and F is force. Defining u_1=0 when the system is static equilibrium (just before the solenoid valve is opened),

$$u_2^2 = \left(\frac{P_{operating}}{A_{cross\ sectional}}\right) * distance \tag{16}$$

u_2 would be available to the thruster if no inhibits, redundancies, or sensors were included.

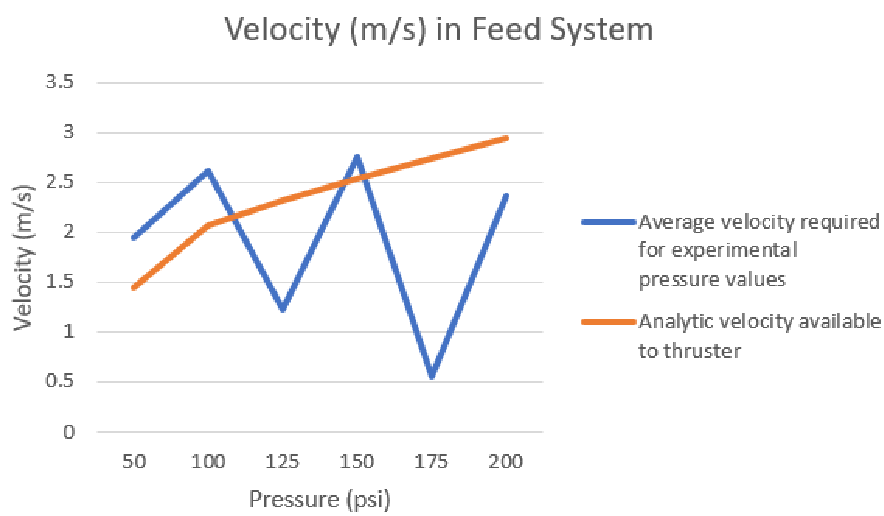

Figure 4.6. Feed System Velocity Versus Pressure.

While the exact experimental velocity (rather than the average) would create a more robust comparison to the analytical velocity at the propellant storage tube exit, fewer unknown parameters results in a comparison between the analytical and experimental portions of the experiment with less error. The trend in the analytical data is logical, as more work (and thus more pressure) will be required to move the fluid more quickly.

In summary, with an experimental set-up readily available, the magnitude and trend of the experimental pressure data was considered more reliable to confirm the viability of the pressurization method concept than the analytical (due to unknown critical values) at this time. The smaller error between the velocity values at a single point indicates that more precise velocity values at each segment would improve the comparison between the analytical and experimental pressure values. Before the pressurization method can progress from concept to operation, the analytical calculation will need to be supported by additional velocity information to produce a more robust model.

4.3.4. Future Work: Expanding Pressure Loss to Entire System. In addition to increasing the robustness of the analytical model with velocity data from experiment or simulation and showing repeatability of the experiment, the analytical and experimental methods will need to be expanded to address pressure loss over the entirety of the feed system. This test focused specifically on the pressure loss due to the pressurization of the propellant by the force exerted on the piston head from the pressurant; future team members will need to determine the full pressure loss expected in the system due to the change in tubing diameter, bends, friction, and connections in the entirety of the feed system. This should be done experimentally by measuring the differences in the set operating pressure and pressure transducer closest to the thruster and also analytically using pressure change due to head loss equations. These values should be compared to verify lab results and show repeatability. The starting operating pressure can then be adjusted to compensate for loss as needed. As the required inhibits contain 1/16 inch diameter tubing, it should also be confirmed that microfluidic effects do not prevent the thruster from being supplied with the minimum required operating pressure in either mode.

4.3.5. Minimum Operating Pressure. Additionally, the author determined the lowest possible operating pressure for a functioning feed system. This was done by gradually lowering the operating system starting pressure while the system was visually inspected to confirm that the piston head reached the end of the storage tube.

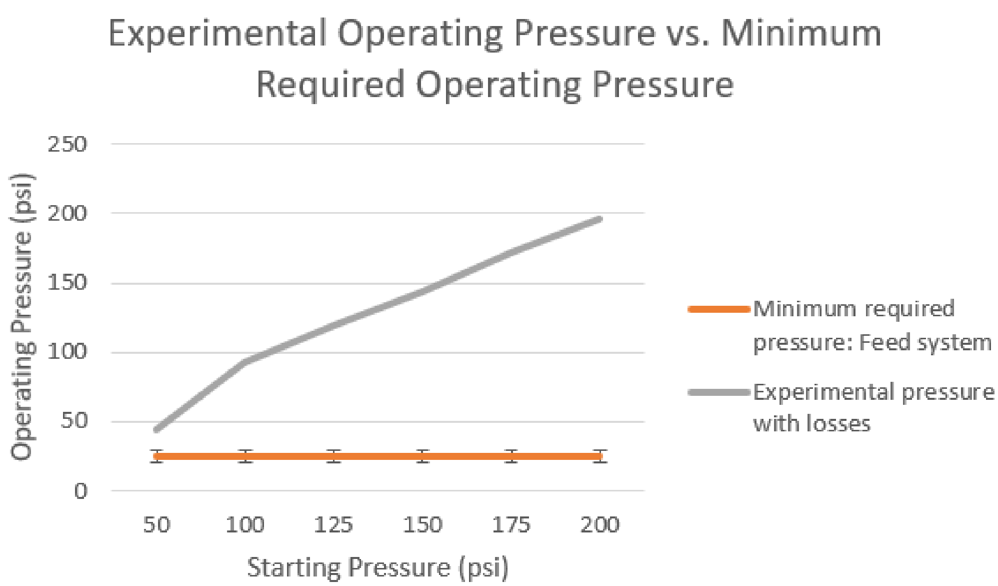

Figure 4.7. Minimum Operating vs Experimental Pressure for Feed System: Olive Oil.

A value of 25 ± 5 psi was determined to be the minimal operating pressure at which the piston head would empty the storage tube of simulated propellant (olive oil). Figure 4.7 shows that this is well below the experimental operating pressure of the system for a starting pressure range of 50 to 200 psi. This confirms that the experimental set-up would be able to supply the thruster with fuel even with significant pressure losses. However, the author suggests a margin of error above this minimum value if it were decided to operate the system in a lower pressure range. This is to account for vacuum and microgravity effects and additional losses exhibited at low pressures.

4.3.6. Component Testing. In addition to testing the custom pressurization method, propulsion subsystem team members were required to validate the performance of COTS components.

4.3.6.1. Completed component testing. Individual COTS components were also tested during the building and testing of the EDU for the FSR event. This testing process suggested some modifications. A different pressure relief valve was selected due to misinterpretation of manufacturing designs. Through testing and research, it was determined that the solenoid valves required an additional component to regulate the voltage going into the valves. Providing a constant voltage of 3.3 or 5 volts caused the valves to overheat and fail open. This was corrected by supplying a 12 V excitation voltage, followed by a lower holding voltage of 1.6 V through additional circuitry. (Experimentally it was determined the valves could be opened more slowly at a voltage above 2.5 V.) The pressure transducers were successfully integrated to the motherboard and functioned according to manufacturer specifications. It was determined through MR and MRS SAT testing and further definition of mission requirements that a pressure regulator was unnecessary for M^3 operating at a single pressure, and that a mechanical pressure regulator was insufficient for APEX as APEX operates in two modes. More information on this decision can be found in Section 5.2.1. During the Fall 2019 semester, propulsion systems sensors and solenoid valves were successfully integrated with the flight computer. Payload testing is on-going by members of Froberg Aerospace.

4.3.6.2. Future component testing. Acceptance into AFRL's Nanosat-10 competition has renewed project funding and will prompt the purchase of additional components to resolve open issues from the Nanosat-9 competition process. Each new component will be evaluated before the system can be fully integrated and characterized.

4.3.7. Next Steps-Testing. Leak-proof testing began for the USIP feed system, during the Summer 2018 and Spring 2019 semesters. After finalizing any changes to the APEX feed system for the Nanosatellite-10 competition, both teams will need to complete rigorous leak proof and vibration testing in a cyclical manner. If possible, the full test set-up should also be subjected to testing in the AP Lab vacuum chamber after initial leak proof and vibration testing to provide more extensive results. Additionally, stress tests should be conducted with relevant connections and seal material to mitigate risk. Lastly, extensive testing should be completed once the feed system is integrated with the thruster under the guidance of Dr. Rovey and Froberg Aerospace.

4.3.8. Risk Analysis. The author contributed to an informal HAZOP analysis of the feed system. Six to eight students met weekly over the course of a semester to ensure the analysis was comprehensive. The feed system was modeled as a simplified layout consisting of five regions: pressurant tank, pressurant transport line, propellant storage tube, propellant transport line, and thruster feed.

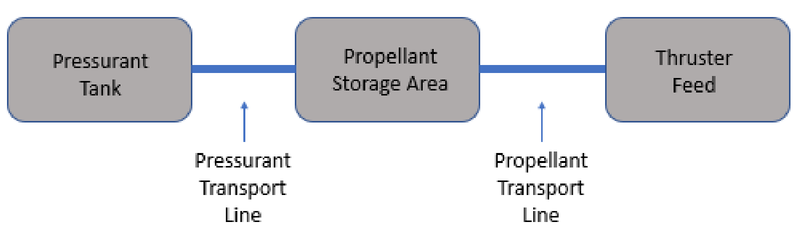

Figure 4.8. Simplified Feed System Diagram.

The team then considered the implications of the system pressure being too much, too little, or having a complete lack of pressure, temperature, containment, and flow. Once each condition was considered, the source of the undesirable effect was considered as well as possible safeguards and actions required as a result. This table can be found in Appendix C. Actions required included extensive leak proof and vibration testing, possible refortification of external connections, direct action added into flight computer code to monitor system health and close valves as needed, thermal coating research, possible addition of heaters, a pressure relief valve, and the ability to cut power to the propulsion system. The severity of each condition was informally evaluated by team members and indicated which actions needed immediate attention. In the near future, severities need to be formally documented.

5. FEED SYSTEM LESSONS LEARNED AND PROPOSED APEX IMPROVEMENTS

5.1. LESSONS LEARNED

The following lessons were learned during APEX feed system development. Future team members are advised to consider these lessons in future design decisions.

5.1.1. Hardware Selection. There are several additional design constraints derived from the propellant composition and the nature of a student-designed pressurized system. It is important to understand the impact of these constraints so that any updates to the feed system design will be made with full understanding of the original design. These constraints include material selection, cost, customization, leak potential, and size.

5.1.1.1. Fittings. There are many types of pipe fittings. Common fitting types encountered for satellite propulsion systems include: Pipe butt weld, National Pipe Thread (NPT) fittings, Society of Automotive Engineers (SAE) threaded fittings, Army/Navy (AN) fittings, and compression fittings (e.g., Swagelok, Yor-Lok, etc. based on brand). Compression fittings, SAE, and AN fittings are all commonly used for M-SAT applications. Compression fittings are encountered most frequently in pipe-to-pipe connections, while SAE fittings are common for component connections. Connections that adapt SAE to compression fittings are common and usually easy to source. NPT fittings should be avoided if possible, as past team experience shows that NPT fittings are more likely to leak (compared to compression or SAE fittings) and require the use of Teflon tape as a seal. This tape can become fodder if it breaks off in the system. However, these considerations should be taken within context. For example, it is possible that trying to avoid using a single NPT fitting would require the use of multiple fittings.

All fittings have the capacity for some leakage; therefore, the use of more compression fittings could potentially pose a higher probability of leaking than the use of a single NPT fitting. (This would need to be determined by placing the system in the fume hood in both configurations.) It is also important to note that the repercussions of a slow leak of (inert) pressurant are not nearly as significant as a propellant leak in the APEX/M^3 systems. Butt welds should not be used for UNP projects, because UNP strongly discourages welding on student systems. If welding is the only option, one could consider requesting a waiver to allow outsourcing of the welding.

5.1.1.2. Inhibits and flow regulation. Significant consideration has been spent on selecting and operating the valves to be used as inhibits and the most effective way to regulate fluid flow in the feed system. UNP-sponsored missions require four inhibits between the propellant and the thruster. NASA-sponsored missions require three inhibits between the propellant and thruster. Refer to Sections 5.2.1 regarding pressure regulation and 5.2.2 for more information regarding inhibits.

5.1.2. Materials. Not all materials are compatible with the propellant. Common metals that are compatible include stainless steel and titanium. Generally, stainless steel was used in the USIP and APEX designs because it is less expensive than titanium. Most seal materials are compatible with the propellant, but all new materials introduced to the design should be verified by Froberg Aerospace if they come into contact with the propellant. Both the pressurant and propellant sides of the feed system were designed with stainless steel to negate the repercussions of leaked propellant internally or externally. However, if this were to significantly affect the volume or components available, the team could discuss if using noncompatible materials (such as aluminum) on the pressurant side of the feed system is within the risk tolerance of the mission. The author only recommends this if it reduces the risk of the overall design.

5.1.3. Student Constraints. The student-based constraints that significantly affect the design on the feed system are limitations on qualifying custom components, welding, cost, and complexity. Refer to Section 3.5 for more information.

5.1.4. Size. The thruster is expected to require 1U of volume on the satellite. The feed system currently consumes more volume than any other main system on APEX; together the thruster and feed system use approximately four of APEX's six Us. A large volume is expected due to the large volume of the pressurant and propellant tanks and other required components, however, a significant portion is due to limitations resulting from the use of COTS connectors. Many sections of tubing have to be "stepped up" or "stepped down" to account for the varying connection sizes of the COTS components instead of being able to use one custom connection to transition tube sizing. This means that it is important to try to avoid extra "steps" if multiple connections options are available for a component to optimize the overall layout.

5.1.5. Training-Hardware. A significant portion of M-SAT hardware training occurs student-to-student. At the beginning of the author's program "sufficient" training included a more experienced student instructing a new member on how to perform a task and possibly showing them how to complete the task. This led to the team having a small group of "experts" who gained experience mostly through trial-and-error or by consulting other team members. When these "experts" graduated or were diverted to another project, their experience left with them. The author noticed a significant increase in "mistakes" during the gaps of experienced members leaving and new member training, even when experienced members could have been consulted. Because of this, the author employed a training method based on Admiral David Marquet's "Intent Based Environment".[36] The author implemented this training method with the USIP and APEX propulsion subsystem members and saw a decrease in tasks that had to be corrected and in damaged hardware. The first step is for the experienced team member to perform the task while the trainee(s) watch and the trainer explains why each action is performed. The second step is for the experienced team member to perform the task but to ask the trainee(s) what the trainer should do next before executing an action. This gives the trainer the opportunity to correctly identify any missed information and add any clarifications. The third step is for the trainee to perform the task, but only after verbally stating what action they plan to take and why (if relevant) and receiving confirmation from the trainer before proceeding. If the trainee has questions, it is helpful for the trainer to inquire how the trainee would respond to the question (Example: "What do you do if you do not feel the click in the torque wrench?") and then add to or correct the response. This method invokes learning by visual, physical, aural, and verbal methods and includes repetition rather than just visual and aural learning. Additionally, students are encouraged to be "self-thinkers" and have more ownership of the training process. This method may not be feasible for tasks that are performed a limited number of times over the course of the project, but it produces a more reliable work force for tasks that are consistently required.

5.1.6. Diagnosing the Problem: Hardware/Software. One critical skill that is not commonly encountered in the classroom is diagnosing anomalies encountered during lab testing and how to resolve them. Most student interaction with hardware systems traditionally occurs during labs when students have limited time and there are instructors available to address “bugs.” This is often not the case in industry, and while experience makes the process more efficient, the author has determined that the following questions can help students start the troubleshooting process:

- “Did I follow the plan?”

Here the student reviews the instructions and determines if a task was missed, incomplete, or misunderstood.

- “Was there an error in execution?”

Here the student reviews their work. For hardware, adjustments may include retaking a measurement, tightening a connection, resetting an instrument, etc. For software, students may need to check the sequence of their loops, search for mistypes and overwrites, or have misplaced parentheses. The student should “retrace their steps.”

- “What assumptions did I use?”

This is the first step that requires the student to “think outside the box.” The student may need to re-evaluate their test plan, check that a sensor is still calibrated correctly, asses if there are any external factors that could be affecting the outcome, determine if the system is being used in its intended environment, etc. In software, the coder will need to assess what assumptions pre-coded functions make, check units, etc. In summary, the student needs to determine why they did what they did.

- Does each component perform as expected?

This includes evaluating the performance of individual components based on the manual (if they are COTS components) and expectations based on knowledge of the physics/system. If the component works according to expectations, but the system is not performing as expected, the interface should be analyzed. If the component behaves differently than expected, the student needs to determine if their expectation (rather than the data) may be incorrect or if the component is faulty. This may require seeking out others with more experience or speaking with the manufacturer.

This list is not comprehensive. Above all, the student should use common sense. Although students in STEM fields are trained to address problems methodically on paper, this is not always the automatic response when the problem is tangible. Lastly, students should not be afraid to ask for help after attempting to address the challenge independently. Sometimes gaining the required experience includes watching a more experienced individual methodically address the challenge, so the mentor's method can be imitated in the future.

5.2. FEED SYSTEM IMPROVEMENTS

The second APEX proposal was successfully selected, allowing the team to compete in AFRL's Nanosatellite-10 competition starting in January 2019. With a working flatsat and mostly-functioning subsystems that resulted from the NS-9 round, the team will need to focus on improvements from lessons learned and implement the thruster for NS-10. The author advises that the following items in the feed system should be further considered before the Flight Selection Review for NS-10: pressure regulation, swarm formation application, valve testing, risk analysis, numeric and analytical analysis.

5.2.1. Pressure Regulation. A Tescom BB-6 mechanical pressure regulator was selected for the NS-9 EDU. Material and operating pressure constraints significantly reduced the options for mechanical regulation. The BB-6 selection was made under the assumption that the thruster design by Froberg Aerospace would regulate the flow and account for the different flow rates required to execute chemical and electric mode.

Additionally, it was thought that once the regulator was set, it would regulate the pressurant to a constant value for any flow condition. However, through testing for MR and MRS SAT, it was determined that mechanical regulators of the selected style may be reset due to vibration during launch. Additionally, through APEX component testing, it was revealed that the flow must be steady and continuous for the regulator to maintain the expected pressure output. Inconsistent flow (created with a partially opened ball valve) caused the regulator to output lower pressures than expected. Additionally, once the propulsion system as a whole was more completely understood, it was determined that regulating pressure primarily upstream for the electric and chemical modes would be a more efficient design. Therefore, the author recommends the APEX team explore electric pressure regulation options such as a COTS electric regulator with the relevant specifications, PID control, or a “bang-bang” set-up designed in-house. A “bang-bang” set-up[37] would use the opening and closing of solenoid valves to control the fluid.

5.2.2. Inhibits. Another component that was difficult to select for the NS-9 EDU was an appropriate valve to act as the inhibit between the propellant and the thruster. Solenoid valves and latch valves were both considered. Solenoid valves were chosen over latch valves because of the differences between the valves’ resting states. Latch valves require an input voltage to both open and close, whereas solenoid valves require a constant input voltage to remain open. Theoretically, if a valve were to fail, a latch valve would remain in its most recent state, but a solenoid valve would close without a power supply. The latter is more ideal from a range safety risk perspective. Additionally, sufficiently small latch valves were more difficult to source than were solenoid valves. Lee Company’s IEPA1221241H solenoid valve was selected because it was the smallest valve that met team requirements, and the M-SAT team had experience with this model from MR and MRS SAT. Testing revealed the valves were very fragile. The team experienced complications with the provided tubing separating from the valve, overheating resulting in valves failing in an open position, and bending of the 1/16" diameter tubing inducing stress on the component. The company was contacted regarding the tube separation and overheating challenges and more extensive training was conducted to prevent additional stress on the components. Component age was determined to be a possible contributing factor to the original overheating issue, however more precise regulation of the voltage to the valves was also needed to eliminate the issue. The chief engineer found "Micro-Thruster Development: Propulsion System for the DelFFi Mission[38]," a thesis written at Delft University of Technology, where another team experienced similar difficulties. This thesis was used as a reference for the team to correct the circuitry needed to use the valves effectively. Though significant improvement had been made in this area, the author recommends continued experimentation with the valves before flight to ensure complete understanding of hardware limitations.

5.2.3. More Risk Analysis. A risk analysis team was formed to manage risks associated with flying an innovative propulsion system as a secondary payload. The team did the analysis with the "Do No Harm" principle in mind. The team began with an informal HAZOP analysis of the propulsion system and considered high-level factors from other subsystems. (Refer to Section 4.3.4 for more information.) The author recommends using the USIP and MR and MRS SAT launch preparation experience to gather more specific expectations from a range safety perspective (which will vary from launch to launch and vehicle to vehicle) and to take preemptive measures to address risk. Additionally, other subsystems' risks should be studied more in depth. Mitigation of risks in all subsystems should be tracked.

5.2.4. Numeric and Analytic Analyses. The author has performed high level analysis to determine how closely the analytical pressure losses in the feed system compare to experimental pressure losses. This analysis should be expanded in the future to the entire system, along with additional experimental testing, to ensure that the pressurant tank is set at the correct pressure. Additionally, team members should continue to work with Froberg Aerospace to ensure that once a final mass flow rate is determined for each mode the feed system can produce these flow rates with consistency. The team also needs to further study the effect of the propellant on the materials in the system over a period of time (such as the time the satellite can expect to be on the launch pad, at a minimum). Lastly, the author co-wrote a paper describing validation of the thruster including pressure and temperature data from the feed system and orbital maneuvers (see Morton thesis[39]). Following further thruster development, these orbital analyses should be explored more extensively, and the team should collaborate with Froberg Aerospace to finalize an on-ground test scheme to compare thruster and feed system performance.

5.2.5. Swarm Formation Application[20]. Multi-mode Micropropulsion enables flexible missions and complex missions where both high thrust and high specific impulse are required. Primary feedback from the Nanosatellite-9 competition indicated that reviewers would have appreciated additional focus to application of the thruster to a specific scenario. One such scenario that would benefit greatly from this type of propulsion system is satellite swarm technology. This application was discussed on a systems level in Withrow and Klosterman's, "Mulit-Mode Micropropulsion Systems Enabling Swarm Technology" paper that placed first at the 2018 AIAA Region V Paper Competition in the master's category. The main concept is that many space applications in the next few decades will require systems of cooperative satellites. While there has been significant study into the GNC and programming required for such formations, they are currently limited by propulsive capabilities.

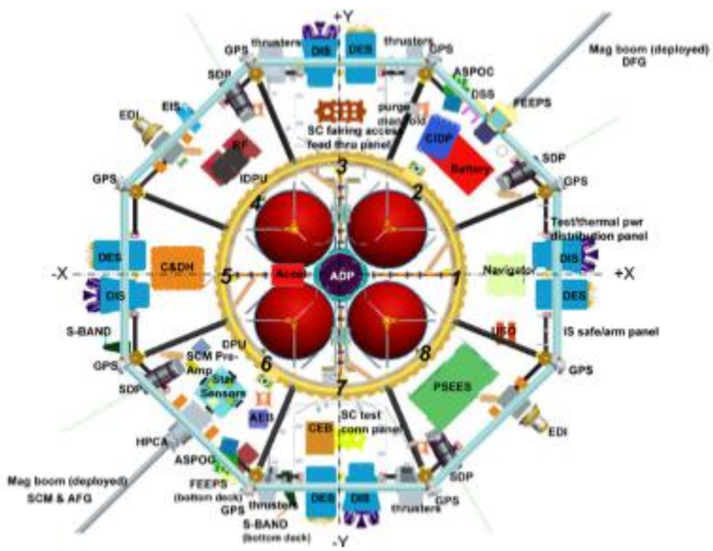

Figure 5.1. MMS Satellite Layout.[40]

NASA's Magnetospheric Multiscale Mission (MMS) and Germany's TanDEM-X mission are examples of large satellites that have successfully flown formation missions. NASA's MMS mission flew four satellites. Each were 3.5 meters wide, 1.2 meters tall, and had a mass of 1250 kg[40]. The overall mission cost was $1.1 billon[42]. Propellant alone accounted for 360 kg of mass per satellite[40], and as shown in Figure 5.1, the propellant tanks consume a large portion of the internal volume of the satellite. If a state-of-the-art

(SOA) thruster and its feed system could be miniaturized to fit within a CubeSat, significantly less propellant would be required, and distributed swarm technologies would enjoy the same benefits that the CubeSat standard provides.

As a next step in APEX's development, the author contributed to the Nanosatellite-10 proposal. As a result of the previously mentioned feedback, application to swarm formation was included as a key component of the proposal. While it is not reasonable to attempt to add specific swarm maneuvers with another vehicle to APEX's CONOPS, the analytical and simulation work can be expanded upon to validate the use of the APEX thruster for such applications. Klosterman and Withrow showed, using "leader and follower" formation in LEO in STK software, that it would be possible to perform a rendezvous maneuver followed by station keeping with the current 6U design and propellant onboard the spacecraft.

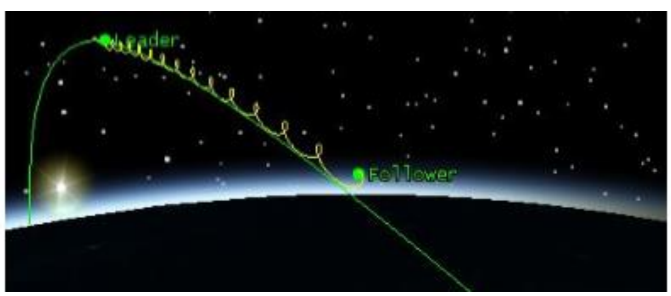

Figure 5.2. Rendezvous Simulation in STK.

A theoretical 3U with 30 cc of propellant was also considered. The amount of propellant required to effect rendezvous and possible burn time after rendezvous can be found in Tables 5.1 and 5.2. Table 5.2 shows the burn time available for station keeping after the chemical burn for the rendezvous if the satellite is able to transition to electric mode using a multi-mode thruster compared to a the satellite that had a chemical thruster alone and was required to use chemical burns to perform station keeping maneuvers.

Table 5.1. Propellant Usage to Rendezvous

	Initial Propellant Mass	Propellant Used in Rendezvous	Propellant Remaining
6U	106 grams	66 grams	40 grams
3U	43 grams	32 grams	11 grams

Table 5.2. Station-Keeping Capabilities

		Performance After Rendezvous	
	Propellant Remaining	$\Delta V(\frac{m}{s})$	Burn Time
6U Electric	40.425 grams	39.5571	88.1265 hours
6U Chemical	40.425 grams	8.9003	71.3825 seconds
3U Electric	10.57 grams	20.711	23.0426 hours
3U Chemical	10.57 grams	4.660	18.6645 seconds

For the Nanosatellite-10 competition this work should be expanded to include more satellites in various configurations and optimize the amount of propellant needed on board.

6. DELIVERING SATELLITES EFFICIENTLY AND ROUTINELY

6.1. DIFFERENCES BETWEEN INDUSTRY AND UNIVERSITY SATELLITE DEVELOPMENT

The innovation of small satellites and CubeSats has opened the unique environment of space to non-traditional groups, such as universities, in addition to industry. While both groups produce orbit-ready spacecraft, there are large differences between the two groups regarding budget, schedule, risk, and expectations. Universities are typically given eighteen to twenty-four months to go from concept to a functioning spacecraft, a budget of ~$100K - $150K, and work primarily with a part-time volunteer workforce. The Aerospace Corporation reported that industry averages over three years[43] in just the manufacturing stage, has a budget on the order of millions of dollars, and has a full-time dedicated workforce. The author has concluded that these differences mostly center around "expectations."

A university satellite is expected to survive the space environment, perform a quantifiable function on-orbit, and "Do No Harm" as a secondary payload. The goal is for science or technology to be advanced through the mission, but if full success is not achieved, the mission can be considered partially successful due to its educational benefits. The "reward" of full success is future opportunities and credibility for the university and students. However, in industry, the stakes are higher. The goal of an industry mission is to improve the state-of-the-art, further science, secure national defense, and/or make a profit. Therefore, the accountability is much higher. A government mission is accountable to the taxpayers, and a private mission is accountable to the customer/shareholders. This generates the different environments between the two groups.

Universities can complete satellites on smaller budgets because they are not paying their "employees" (except for gained experience), have lower overhead cost, and can often make use of/alter donated hardware to achieve their mission. Industry is responsible for overhead, must pay their employees according to their experience level, and often use customized mission hardware. Universities usually have a larger risk tolerance than industry, inside of the "Do No Harm" constraint. This is driven by the expectation and accountability of the mission outcome. This is a significant contributor to the gap in timelines from concept to launch between universities and industry. Ultimately, a university must design and test a satellite to the certainty that it will not harm the primary payload (as a worst-case scenario). An industry satellite must be designed and tested to the certainty that its success can be justified to the stakeholders, which typically includes bearing launch expenses as a primary payload. However, it would not be fair to conclude that building satellites at universities is simpler than building satellites in industry because universities lack the reputation, technical expertise, available resources, budget, and schedule that many members of industry enjoy. These should all be considered when discussing methods of successfully and efficiently delivering satellites. The descriptions and recommendations below may have application in industry settings, but they are primarily written with universities in mind.

6.2. TRAINING

Training is essential to ensure the satellite is built to the appropriate level of quality and to transfer the project from one group of students to the next.

6.2.1. Team Training. Team training is managed on the M-SAT team by the Resource Manager because training exercises are common to multiple projects. This training is generally conducted by more experienced team members, but expert training is also utilized when possible. Upon joining the team, each member is asked to complete training with the document writing and management system (LaTeX and the repository), ESD training, and relevant lab equipment. Additionally, it is encouraged for new members to read the subsystems Conceptual Design Documents (CDDs) in which they are interested. The CDDs are updated every semester with a general description of the system, completed tests, and current progress. Additional training such as soldering and CAD are offered as needed. It is generally helpful for these trainings to occur during regular meeting times if possible, as more team members are likely to be able to attend. Tiger teams are small groups formed to address specifically challenging areas where the team lacks experience. Another tool that was implemented during NS-9 and the USIP program is a document written by graduating seniors that describes any information that is needed for another team member to continue their work. This document includes items such as important criteria which were considered in decisions regarding specific design components and the location of tools that are shared with another lab. Seniors should be prompted to begin this document before other graduation commitments begin, so that an appropriate amount of time will be devoted. The transition document has been implemented to reduce loss of knowledge and skills despite annual team turnover.

6.2.2. Leadership Training. The idea of using deputy leads is not a new idea for the M-SAT team, but for the APEX and M^3 missions, the timing of choosing a deputy lead was altered. Multiple projects and the sophomore-level BalloonSat course open more opportunities for students to get involved before their junior or senior year in spacecraft design, prompting leads to begin earlier in their school careers. Additionally, more projects allow more students to continue with the team into graduate school. As a result, leads have more semesters to work with their replacement before passing on the position.

Ideally, leads ought to select their deputy with a minimum of one semester, and preferably two semesters, remaining until they plan to leave the team. While this is not always executable, transition is the most effective when this timeframe is used. This allows one semester for the lead and deputy to work together, and one semester for the deputy to lead the subsystem with the former lead still available on campus to assist with the transition. One-to-one training continues to be the most effective method for ensuring successful transition.

Managing multiple satellite projects under the same program has unique challenges, such as sharing the talent pool, lab space, equipment, mentors, and other resources. It has been necessary with three mature programs for PMs, CEs, and Resource Managers to begin regular meetings with the PI known as “Executive Meetings.” Executive Meetings help the projects support each other, helps prevent attitudes of segregation and unhealthy competition between project teams, and allows the executive team to address big picture concerns. Big picture concerns include matters such as future team goals, how to best utilize members among the three projects, how to share lab space, and areas for improvement. With many projects, there are many decisions that must be made each day, and it is vital that executive members have a unified position when addressing team challenges.

6.3. INCENTIVES FOR UNIVERSITY STUDENTS

One of the biggest differences between university programs and industry programs is that students are primarily volunteer and balance full-time class schedules in addition to satellite research. At most some students receive credit for senior design or graduate course credit. With this in mind, it is very important for the PM to create an atmosphere that encourages team members to invest in the satellite project. However, it should be noted that incentive does not fall solely on the PM; students will get out of the team what they put into the team.

Creating an engaging atmosphere is done primarily by making team members feel welcome and inspired by the significance of their contribution - both in context of their contribution to the team and to the mission. The atmosphere should encourage enthusiastic contribution and strengthen a sense of community amongst team members. The PM significantly influences this atmosphere through activities like guiding students in discovering niches of personal interest related to the project and providing opportunities for team members to connect outside of the lab environment. Voluntary events such as movie nights and athletic intramural activities can help team members learn to communicate better and relax. One such event that was implemented during the NS-9 program was a team dinner to celebrate the completion of a review. The PM and CE (with advice from the PI) selected one individual from the team who went “above and beyond” and would pay for that individual’s meal. This provided friendly competition as well as an opportunity for team members to relax/bond. Other “rewards” included opportunities to go to conferences which provide experience and networking opportunities. In addition to a sense of camaraderie and external motivators, an engaging atmosphere is built as students realize the significance of their personal work to the overall mission. The significance of context should not be overlooked as a motivating factor. Students are more invested if they understand how their current task actively moves the team closer to launching the satellite. Students are often willing to even volunteer extra time over breaks if it means seeing milestones reached during their time on the team. This was demonstrated during the break just before NS-9's Final Selection Review. With the exception of one or two students, all core team members could be found in the lab a week before the team departed. Work is significantly more efficient with team members in close proximity to each other to collaborate. If this seems to be difficult to accomplish, never underestimate the power of pizza!

6.4. SATELLITE ASSEMBLY, INTEGRATION, AND TESTING (AIT)

AIT is an area often overlooked at the beginning of a program by universities. One unique aspect of the M-SAT team is that it has an integration subsystem to help mitigate potential neglect in this area. Prior to the NS-9 program, the integration team was used for matters that covered multiple subsystems as well as to assist areas that were falling behind. At the beginning of the Nanosat-9 Program, Interface Control Documents (ICDs) were managed by each subsystem. Later in the program, a standard template was created, and the leads worked with Integration to write the ICDs. Transferring the control of the ICDs to the Integration subsystem allowed a central group to actively look for discrepancies between interfacing, which preemptively addressed mistakes that would have consumed valuable time and resources to correct later. Also, although not implemented in Nanosat-9, the author recommends having the Integration team review test plans. Testing is an iterative process and integration is extremely complex. Therefore, it is helpful to have more than one perspective on a test or assembly plan.

Extra time should always be allowed for testing and assembly, as they tend to be iterative processes. It is common in research to obtain results outside of the expected range or to not be fully aware of system requirements in the beginning. It also improves the chances of completing the tasks in the expected time frame if a layout is done and a plan is reviewed with the team members who plan to conduct the test prior to testing. This provides sufficient time to make alterations as necessary.

6.5. DOCUMENTATION MANAGEMENT AND REVIEWS

Documentation management is done primarily through the M-SAT team repository. The PM is responsible for collecting and organizing the documentation for each review and the PM and CE work with the PI to ensure that the documents are prepared with the expected quality. Good documentation is essential for program success. UNP staff members frequently use the figurative phrase, “If it’s not documented-it didn’t happen.” to express the importance of documentation to university teams. The author learned the following lessons regarding preparing for a review:

1) Start early.

It is tempting for documentation to become an afterthought. To ensure quality documentation, track the progress of documentation throughout the semester and assign and enforce document due dates as they should be completed, not just in time for the review. Spreading out deliverables will help to avoid the “last-minute rewrites” or the decision to exclude a document.

2) Give your team a buffer.

It is much more effective to say, “The document is due to person X by date Y,” than to say, “The document is due to the sponsor by date Z.” If the option is available, it should be assumed that team members will wait until just before the deadline to submit a document - so make the deadline early. During the editing process, most documents will inevitably be sent back to the original author for lack of quality or additional information (as a consequence of reviewing related documentation). As a PM, it is important that something is not so time critical that the editor must "just do their best to fix it" if someone else could rewrite it better or benefit from implementing the corrections themselves.

3) Provide accountability to the team.

Part of the "team atmosphere" is that every member is doing their part to support the others. The author found that team members are more responsive to the team than to an individual. Therefore, letting team members know they will be expected to update the entire team on the status of their documents as a review approaches is appropriate. This provides extra incentive for team members to meet their deadlines and an opportunity to take responsibility and suggest a plan or ask for help if they are falling behind schedule. Team members should be given ample opportunity and encouraged to ask for help if they do not think they will meet a deadline. Also, be certain to ask follow-up questions to status reports. They will often reveal potential schedule or technical risks.

4) It is okay to ask for team support.

The APEX and USIP team have a policy that the PM or CE attempts to review every document before it is submitted for a review. The PI also reviews critical documents. However, there are too many documents for this review to be a first-draft review. Before submitting a document to the PM/CE/PI, subsystem members should review documents for quality and grammar, and subsystem leads should "approve" documents before the documents leave the subsystem. The author found it beneficial to assign each lead one or two major documents from other subsystems to review as well. This allowed the document writer to get feedback from another perspective outside the subsystem while they still had the opportunity to make adjustments. The PM and CE should be able to focus on checking quality and cohesiveness across the subsystems by the time they review the documents. It is reasonable to ask for team participation when it comes to written documentation because the documentation is the representation of the work that the team has done to the reviewers and sponsors, and it may be the only part of the project that the reviewers or sponsors actually see.

6.6. SCHEDULING

General guidance is provided by the sponsoring organization (AFRL or NASA) on expectations regarding milestones and deliverables. However, to deliver the fully completed project on time, a significant amount of effort must go into dividing the deliverables into smaller tasks, identifying project-specific deliverables (in addition to expectations provided by the sponsor), and making sure that tasks are allotted appropriate time and order of completion.

6.6.1. "Working Backwards." In research it is difficult to estimate how long tasks will take to complete. "Working backwards" is the concept of writing out all tasks that must be completed by a certain milestone with an estimate of how long each task will take. Then the tasks are ordered chronologically starting with the last task. Once the last task is placed, the scheduler determines which tasks need to be completed before the last task, and places them on the timeline. This process is repeated until all the tasks have been placed. Once the timeline is filled with tasks, the scheduler can easily identify if the current time budgeted exceeds the time until the milestone, where critical sub-milestones are (and the associated "critical paths"), adjust the time for each task to be completed as needed, and track progress. The general idea is to set deadlines with known ramifications of slip rather than simply asking what the next step should be. This concept is similar to using a network diagram (discussed in Section 6.6.4), but it is easier to explain to team members without an Engineering Management or Systems Engineering background. It is also easy to then convert the timeline into a Gantt chart (discussed in Section 6.6.3). The author implemented this technique for all subsystems but focused on monitoring the implementation for the Propulsion subsystem during the NS-9 cycle. Compared to the progress prior to its implementation, the author found that team members were more motivated to make up slip early on, understood the “big picture” (leading to less slip), and that slip could be managed more effectively as a known risk.

6.6.2. Assigning Tasks Versus Managing Tasks. At the beginning of the NS-9 Program, the author, as the Program Manager, created a list of deliverables with deadlines. The author found that it was difficult to enforce these deadlines, and that enforcing the deadlines became even more difficult as the number of deliverables increased throughout the program. The author found that it was significantly more productive to work with the lead of each subsystem to create a schedule. The PM would create a general list of tasks with each lead based on the sponsor-required deliverables and the tasks the lead knew needed to be completed to reach the big picture goal for that period, such as a flatsat, EDU, etc. This was usually done in the presence of other subsystem leads to identify which tasks overlapped subsystems. Then the lead was given the list and asked to provide the PM with a schedule for the semester. Once the PM had the schedules, they were checked to make sure that they were reasonable and did not overlap with one another in a way that would make subsystems compete for resources. They were also consolidated into a master schedule for use by the PM, CE, PI, and sponsors to quickly track progress. This process had the benefit of creating more realistic timelines because the leads had more technical knowledge and created more ownership of the schedule from the lead's perspective, which increased their accountability. When a deadline slipped, rather than simply calling out the lead, the PM could simply draw attention to the slip and work with the lead to determine the cause and plan to adjust for it. While this required more up-front work for the PM, it replaced "nagging" with an opportunity for the lead to step up and work with the PM to come up with a better plan than either individual could on their own. In this way the team went from having a single leader, to having many leaders with a single manager. Although it was unknown to the author at the time, this strategy is very similar to Admiral David Marquet's "Intent-Based Environment[36]" in which the primary leader communicates intent instead of direct directions to their team members to take advantage of the group's combined knowledge.

6.6.3. Gantt Charts. Gantt charts are a visual tool used to quickly track progress. The expected start and completion dates, progress, and critical path can be easily identified. Figure 6.1 shows an example from the APEX satellite.

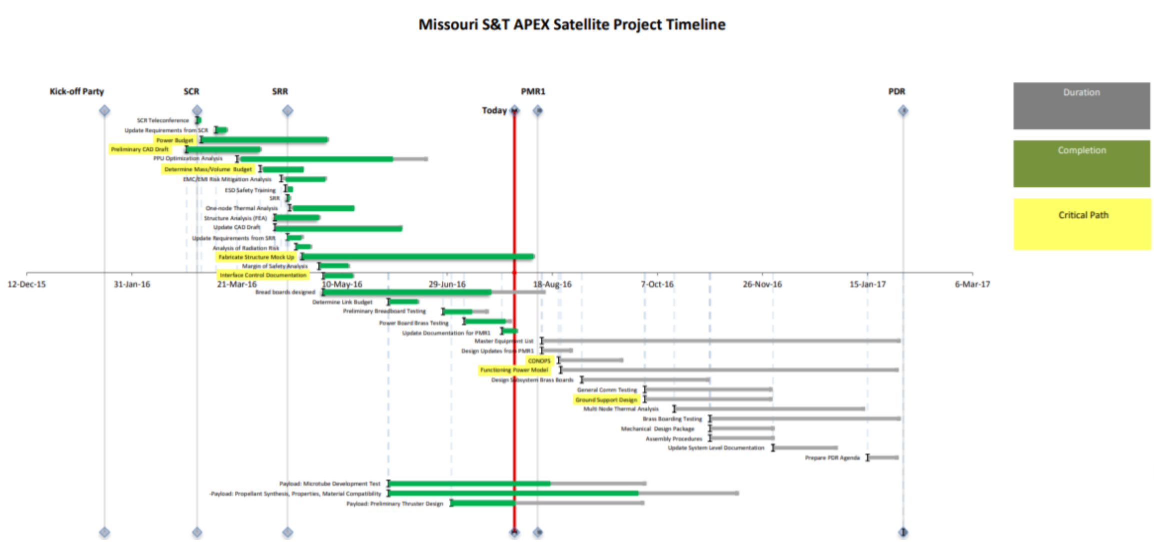

Figure 6.1. Gantt Chart of APEX for the First Year.

6.6.4. Network Diagrams[46]. Network Diagrams can be used to adjust a program for slip or evaluate if resources are being used efficiently. Each block in a Network Diagram follows the following format:

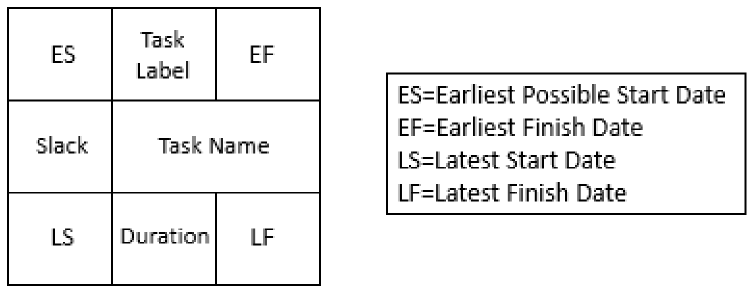

Figure 6.2. Network Diagram Block.

Creating a Network Diagram begins by identifying which tasks must be completed before the next task can begin. Place each Task Name and Label in a block. A chronological "map" of the tasks is created, with a block designated for each task (as shown in Figure 6.3).

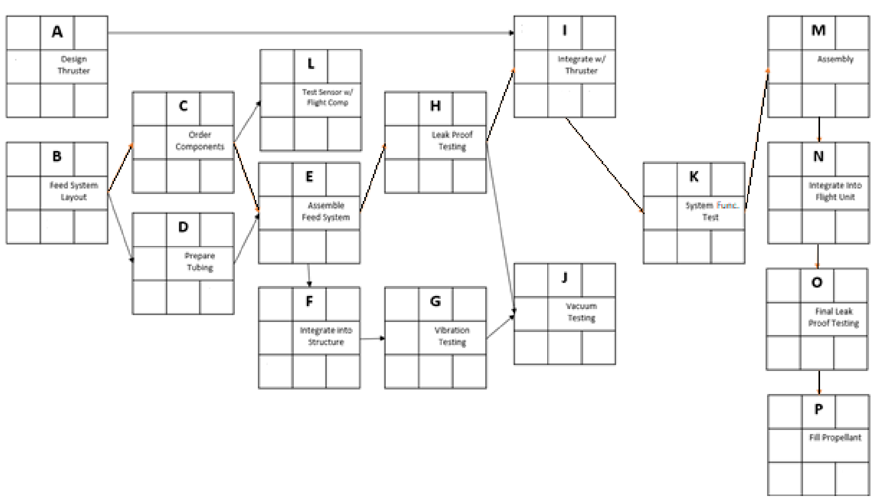

Figure 6.3. Map of Task Blocks.

Next, the earliest start date for the first task is filled in. The earliest start date and the duration determines the earliest finish date for each task. The earliest finish date then becomes the earliest start date for the next task. If the new task is dependent on more than one task, the latest of the earliest finish dates should be used. This process is repeated until the top row of each block is filled. For the last task, the earliest finish date will be identical to the latest finish date. The duration of the task can be subtracted from the latest finish date to determine the latest start date. The latest start date then becomes the latest finish date for the task(s) before it. The process is completed until the bottom row of each block is filled. The last gap, the middle row in the first column is the slack and is determined by the difference between the earliest start and latest start date. This is the amount of time that the schedule can slip without affecting tasks after it. If a task is late, slack can be used to identify which task timelines can be altered for the project to get back on schedule. The tasks that have zero slack are "critical tasks." The "critical path" is determined by the tasks that drive the longest overall duration. There will be no slack along this chain of tasks.

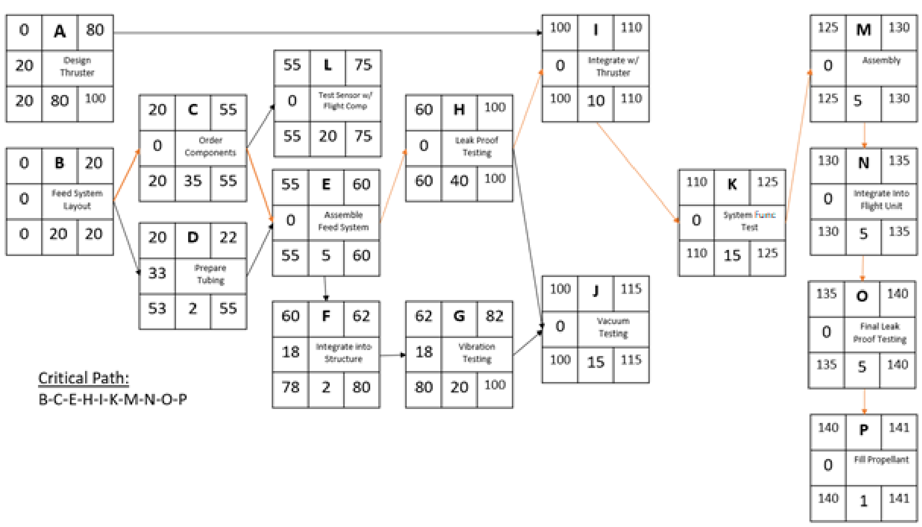

Figure 6.4. Network Diagram of the Propulsion Subsystem Development in Industry Environment.

Microsoft Project is a software tool commonly used in industry to manage schedules and work load. This software is useful for developing tools such as Gantt charts and Network Diagrams. Figure 6.4 shows an example from of a Network Diagram from the Propulsion subsystem with the adaptation of the system being designed and tested in an industry environment. The duration of each task in Figure 6.4 is measured in days.

6.6.5. Flexibility. Although it is important to create and actively maintain a schedule, it is equally as important to know how to manage risk to the schedule, project, and budget when key elements are not achieved as planned. In a research environment, deviation is inevitable and sometimes even beneficial. Tools such as Gantt charts and Network diagrams help to manage this risk.

7. CONCURRENT PROGRAMS

7.1. ONE PROJECT VERSUS MULTIPLE PROJECTS

The author has concluded that the process of developing M^3 and APEX was more efficient because of their complementary nature/payloads and shared timeline. This chapter compares a single, focused project to a program with multiple, complementary projects.

7.1.1. Resources/Personnel. One project allows team members to focus all their efforts and resources towards completing that task. This simplifies the prioritization process and creates a unified team mind frame. However, a team with a single project is limited to the resources designated specifically to that project.

In contrast, a program with multiple projects possesses the ability to share resources and personnel. This can be beneficial, if carefully managed, by fully utilizing all team members' knowledge and work hours and limiting the purchase of redundant hardware. For example, APEX personnel can easily assist M^3 personnel with CAD work while the APEX structure is being machined. Also, if both satellites plan to use the same solenoid valves, the valves may be shared between the engineering design units, and new ones can be purchased for both flight units. The challenges are mostly programmatic including adding complexity to scheduling team members and hardware use. Additionally, design changes and delays can affect both projects. However, for universities with well-managed teams, where experienced team member time and funding are often limiting factors, the benefits seem to outweigh the risks. The most important aspect from a programmatic perspective is to avoid over scheduling team members or resources and clearly defining the priorities placed on tasks. This also creates a training ground for students that is more similar to industry where team members will likely be asked to manage work on multiple projects simultaneously. Currently, 65% of M-SAT team members participate in multiple satellite projects.[44] Another challenge can be sponsor concern that the team has overcommitted themselves. This can usually be managed by providing information regarding previous team performance and current team composition as long as the current roster and team abilities are compatible with the current missions.

M^3 and APEX’s collaborative style is also similar to the concept of rolling inventory used most commonly for managing warehouse activities. The concept of rolling inventory is to increase efficiency by eliminating handling time of materials. A derivative of this concept has been employed in the aerospace industry by Northrop Grumman Innovation Systems (NGIS), formerly Orbital ATK, to quickly produce and deliver their ESPAStar vehicle. NGIS employees order components for the next ESPAStar vehicle before receiving the contract based on knowledge of previous ESPAStar missions. This enables NGIS to approach the customer with a fixed budget and schedule, allowing them to significantly reduce the time from concept to launch.[45] While it is not practical for a university team to take an identical approach because of limited budgets and space, similar concepts allowed M^3 and APEX to be developed more quickly than previous M-SAT missions. M^3 and APEX shared many components due to the missions’ similarities. Examples include heating systems, propulsion feed system hardware, power requirements, integration procedures, and structural mounting. This allowed the M-SAT team to maximize the use of time and resources. M^3 was given an original timeline of eighteen months from awarded contract to delivery. Compared with previous M-SAT mission development timelines, like MR and MRS SAT that were in development for over a decade, this might not seem practical. However, the mission's similarity to APEX made a significant impact on development time. While troubleshooting for APEX, team members were often effectively troubleshooting M^3 as well. Team members could use APEX hardware for a proof-of-concept run before purchasing additional components and flight hardware. This reduced the time waiting on materials to arrive before testing that can be on the order of months for non-standard hardware, aggravated by the tendency for design and testing to be cyclical processes.

There are three main types of organizational structures: Dedicated Project, Functional, and Matrix. Dedicated Project organizations tend to give the most control to the Project Manager[46]. Team member focus and resources are all devoted to one project, creating a strong team culture. Disadvantages include limitations on external assistance and complications in team member transition during the close-out phase. Functional organizations give the most control to the functional manager (technical lead) who divide their team members according to their skill set across projects. For example, the structural lead may be in charge of a team working on three projects. This is helpful when expertise is high in demand. Disadvantages include the tendency for certain workers to be spread too thin and the program manager lacking the authority to guide the project to success. A matrix organizational structure is a combination of the two. Team members report both to a project manager (PM) and functional lead (subsystem lead). This allows team members to be utilized as needed but focus on an individual project. The most prominent disadvantage is the potential conflict of reporting to two separate individuals. It is very important in this structure for the project manager and functional lead to communicate often and clearly. Matrix organizations can be labeled as a project matrix or a functional matrix if the organization leans more towards one than the other.

M-SAT employs a matrix organizational structure, more specifically a project matrix for M^3 and APEX. The satellites are not dedicated project structures because team members and resources are shared between projects, however, they cannot be labeled as purely functional organizational structures because team members volunteer to work on a specific satellite, each of which has unique goals, budgets, and timelines. M-SAT is a project matrix organizational structure because it has project managers (PMs and CEs)- rather than functional managers (subsystem leads) in a functional matrix organization- that have the final authority. Benefits of this are mentioned above, such as reallocating resources between the two programs when appropriate. Disadvantages include balancing priorities and providing team members with the appropriate level of work.

7.1.2. Learning. One of the most common challenges for university teams is knowledge retention. Graduation commonly produces a four to six-year turnover rate, with even less time for team members who join after their freshman year. In recent years programs such as UNP, which help train students in the satellite field, have attempted to assist with this challenge through mentorship opportunities and workshops. However, efforts must also be made internally for a team to transition projects between student groups successfully.

With a single project, leadership positions are limited. A fifty-person team may have fifteen leadership positions. With only 30% of the team in leadership roles, it becomes tempting for the other team members to become reliant on this minority. Additionally, if the lead (usually the most experienced member of a subsystem) encounters a subject matter with which they are unfamiliar, the team may be stalled until an external expert is consulted or the new subject area is self-taught.

However, with multiple projects, more leadership positions are available. A fifty-person team with two projects may require thirty (or more) leadership positions. Thus, 60% of the team is actively involved in making day-to-day decisions with significant impact and will be more engaged in the projects. (~59% of current M-SAT team members hold leadership positions.[44]) If team members know they will have an opportunity (or may be asked) to have a leadership position, they are more likely to attempt to increase their productivity and knowledge to fill the leadership vacuum before being placed in a position of responsibility. Additionally, unfamiliar challenges are not as intimidating to team members when it is easy to collaborate with members of other projects, who may be more experienced.

One of the most significant resource deficits of a university project is team member knowledge. Most learning is done through hands-on experience. While it is not guaranteed that multiple projects will provide a team member with experience in a certain area, it does significantly increase the chance of a team member having knowledge or having encountered a given challenge previously. It often saves the project significant time and money if a team member can be “borrowed” by another project to teach another member the material; this also increases the number of team members that possess a given knowledge set. Additionally, with multiple projects, team members are more likely to get a well-rounded perspective than with a single project which is more beneficial for their future careers. For example, consider a student who joins M-SAT to fulfill senior design requirements for two semesters. If the student only works on the M^3 GNC subsystem, they may gain design experience. However, if they join both the APEX and M^3 projects they may gain experience with design and hardware, as well as have a better understanding of GNC overall because they will have experienced two missions with similar goals but different requirements. The concept of team members learning from one another was so integral to the M^3 and APEX development process that the Propulsion, Thermal, Structure, C&DH, and COM subsystems all chose to combine their weekly meetings until the satellites diverged in the building stage of their projects.

7.1.3. Reputation. The direct effect of reputation on the success of a team is difficult to quantify. The author conducted a poll which indicated ~15% of M-SAT surveyed team members claimed that prior team success was a motivating factor for joining the M-SAT team.[44] However, it is logical that a team with multiple successes- whether that be an experience, competition result, or launch- is more likely to attract more team members and add credibility to proposals, papers, investors, etc. However, the opposite is also true that a bad reputation can negatively affect team growth, credibility, etc. on other projects. When multiple projects are involved, it becomes imperative that members of leadership are able to differentiate and effectively communicate that the team is building on its successful experience and learning from previous mistakes.

7.1.4. Schedule. A program with a single project can devote all resources towards meeting that project's deadlines. However, a major challenge is accurately estimating how long each task will take and avoiding lags when technical challenges, funding gaps, or other obstacles arise.

A program with multiple projects must carefully manage resources in order to meet the deadlines of each project. However, resources for one project can be redirected to another when lags occur. Additionally, experience from one project can be used to decrease the number of technical/programmatic challenges in another. For example, the APEX engineering design unit was built during the design and test phase of M^3. However, due to funding allocations, the M^3 flight model was constructed before the APEX flight model. Because the programs were similar, lessons learned from one program reduced delays in the other. Also, early in the program, deadlines for M^3 were met easily because the construction of APEX was driving the development speed. M^3's timeline drove the timeline in later phases, allowing the team to be ahead of deadlines and deliver more than the "minimum expected" in later stages of APEX development. APEX was able to "borrow" M^3 personnel to assist with challenging propulsion and C&DH hardware leading up to the final NS-9 competition. In return, many APEX team members were able to be "loaned" to M^3 to meet pressing deadlines in between the NS-9 and NS-10 funding phases.

7.1.5. Summary. In conclusion, at the university level, a team with multiple projects (and adequate resources), as opposed to just one, has the advantage of fully utilizing resources such as personnel, cost, time, training, and reputation at the expense of more complexity from a management perspective.

APPENDIX A.

STATE OF FLOW IN EXPERIMENTAL FEED SYSTEM PRESSURIZATION SET-UP

To determine if flow in the test set-up was within laminar range, the maximum velocity for laminar flow was determined and compared to experimental and simulation data. To solve for velocity,

$$u = \frac{Re * \mu}{\rho D}$$

Reynold's number for maximum velocity in the laminar flow range is 2300. Thus,

$$u_{max,laminar} = \frac{2300 * \mu}{\rho D}$$

For water, μ=0.0010005 Pa/s and ρ=998.21 kg/m^3. Therefore, maximum laminar flow for water in the storage tube is 18.15 cm/s.

For isopropyl alcohol, μ=0.00204 Pa/s and ρ=786 kg/m^3.Therefore, maximum laminar flow isopropyl alcohol in the storage tube is 47 cm/s.

For olive oil, μ=0.085 Pa/s and ρ=908.7 kg/m^3. Therefore, maximum laminar flow for olive oil in the storage tube is 1694.04 cm/s. Experimental testing estimated the olive oil moving in the storage tube at 9.12 cm/s, which is significantly less than 1694.04 cm/s.

For propellant, μ=0.1 Pa/s and ρ=998.21 kg/m^3. Therefore, maximum laminar flow for propellant in the storage tube is 1814.27 cm/s. Assuming the flow in the pipe will be on the same order of magnitude as Taylor's velocity of ~ 12 cm/s[25], the flow in the storage tube will be subsonic.

To determine if the flow of propellant was fully developed during pressurization, the length required for flow to fully develop must be determined.

$$Re_D = \frac{\rho v D}{\mu}$$

$$\frac{l}{D} = 0.06Re_D$$

Assuming olive oil as the simulated propellant,

$$Re_D = \frac{908.7\frac{kg}{m^3}*0.0912\frac{m}{s^2}*0.0127\ m}{0.085} = 12.38$$

$$\frac{l}{0.0127\ m} = 0.06 * 12.38 => l = 0.0094\ m = 0.37\ inches$$

With olive oil as the simulated propellant, the flow is fully developed at 0.37 inches in the propellant storage tube.

APPENDIX B.

MATLAB CODES USED IN DESIGN AND TESTING

Below is a pressurant vessel sizing code provided by Missouri S&T AP Lab. The internal volume of the propellant tank (V_prop) and operating pressures (p_prop) can be adjusted to determine the needed pressurant storage volume associated with a range of operating pressures for nitrogen, argon, and helium pressurants.

```
close all
clear all
clc

%%%%%%%%%%%%%%%%%%%%%
% Conversions %
%%%%%%%%%%%%%%%%%%%%%
psia2pa = 6894.76;   %[Pa/psia]
cm32m3  = 1/(100^3); %[m^3/cm^3]
m32cm3  = 1/cm32m3;  %[cm^3/m^3]
pa2psia = 1/psia2pa; %[psia/Pa]
tol = 1*10^(-6);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Known Parameters %
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
pmax = 1000;     %[psia]
pmin = [350;250;200;150]      %[psia]
% p_prop  = 300;   %[psia]
p_prop  = [300;200;150;100];   %[psia]
R_u  = 8314;     %[J/kmol-K]
% V_prop = 75;     %[cm^3]
V_prop = 15;     %[cm^3]
V_ig   = V_prop; %[m^3]
T_i    = 25;     %[degC]

% Sat_name = {'APEX'};
Sat_name = {'USIP'};
%%%%%%%%%%%%%%
% For He %
%%%%%%%%%%%%%%
gamma_He = 1.66;
MW_He    = 4.003; %[kg/kmol]

%%%%%%%%%%%%%%
% For Ar %
```

```
%%%%%%%%%%
gamma_Ar = 1.67;
MW_Ar    = 39.94; %[kg/kmol]

%%%%%%%%%%
% For N2 %
%%%%%%%%%%
gamma_N2 = 1.40;
MW_N2    = 28.02; %[kg/kmol]

% %%%%%%%%%%
% % For Air %
% %%%%%%%%%%
% gamma_Air = 1.4;
% MW_Air    = 28.97; %[kg/kmol]

gamma = [gamma_He,gamma_Ar,gamma_N2];%,gamma_Air];
MW    = [MW_He,MW_Ar,MW_N2];%,MW_Air];
gas_name = {'Helium';'Argon';'Nitrogen';'Air'};

%%%%%%%%%%%%%%%%%%%%%
% Proper Units [SI] %
%%%%%%%%%%%%%%%%%%%%%
pmax   = pmax*psia2pa;
pmin   = pmin*psia2pa;
% p_prop = p_prop*psia2pa;
p_prop(:,1) = psia2pa.*p_prop;
V_prop = V_prop*cm32m3;
V_ig   = V_ig*cm32m3;
T_i    = T_i+273;

% p_i    = linspace(pmin(1,1),pmax,2000);

% %%%%%%%%%%%%%%%%%%%
% % Testing Numbers %
% %%%%%%%%%%%%%%%%%%%
% p_i    = 21000000;
% p_prop = 938000;
% T_i    = 273;
% V_ig   = 2.153+3.513;
% V_prop = V_ig;
% R_u    = 8314; %[J/kmol-K]
c = 0;
for b = 1:size(p_prop,1)
   p_i    = linspace(pmin(b,1),pmax,2000);
   for k = 1:size(gamma,2)
```

```
        c = c+1;
        for i = 1:size(p_i,2)
            j = 0;
            V_total_u = V_ig;
            T_f       = T_i*(p_prop(b,1)/p_i(1,i))^((gamma(1,k)-1)/gamma(1,k));
            m_press   = 1.05*p_prop(b,1)*V_total_u*MW(1,k)/(R_u*T_f);
            V_press   = (m_press*R_u*T_i)/(p_i(1,i)*MW(1,k));
            V_total_g = V_press+V_prop;
            while abs(V_total_g-V_total_u) >= tol
                V_total_u = V_total_g;
                T_f       = T_i*(p_prop(b,1)/p_i(1,i))^((gamma(1,k)-1)/gamma(1,k));
                m_press   = 1.05*p_prop(b,1)*V_total_u*MW(1,k)/(R_u*T_f);
                V_press   = (m_press*R_u*T_i)/(p_i(1,i)*MW(1,k));
                V_total_g = V_press+V_prop;
                j = j+1;
            end
            V_press_mat(i,c) = V_press;
            m_press_mat(i,c) = m_press;
        end
    end
end
p_i1    = linspace(pmin(1,1),pmax,2000);
p_i2    = linspace(pmin(2,1),pmax,2000);
p_i3    = linspace(pmin(3,1),pmax,2000);
p_i4    = linspace(pmin(4,1),pmax,2000);
figure(1)
plot(pa2psia.*p_i1,m32cm3.*V_press_mat(:,1))
hold on
plot(pa2psia.*p_i2,m32cm3.*V_press_mat(:,4))
hold on
plot(pa2psia.*p_i3,m32cm3.*V_press_mat(:,7))
hold on
plot(pa2psia.*p_i4,m32cm3.*V_press_mat(:,10))
hold on
xlabel('Pressure [psia]')
ylabel('Volume [cm^3]')
title([char(gas_name(1,1)),' for ',char(Sat_name)])
lgd1 = legend('300','200','150','100');
title(lgd1,{'Propellant Tank';'Pressure [psia]'})

figure(2)
plot(pa2psia.*p_i1,m32cm3.*V_press_mat(:,2))
hold on
plot(pa2psia.*p_i2,m32cm3.*V_press_mat(:,5))
hold on
plot(pa2psia.*p_i3,m32cm3.*V_press_mat(:,8))
```

```
hold on
plot(pa2psia.*p_i4,m32cm3.*V_press_mat(:,11))
hold on
xlabel('Pressure [psia]')
ylabel('Volume [cm^3]')
title([char(gas_name(2,1)),' for ',char(Sat_name)])
lgd2 = legend('300','200','150','100');
title(lgd2,{'Propellant Tank';'Pressure [psia]'})

figure(3)
plot(pa2psia.*p_i1,m32cm3.*V_press_mat(:,3))
hold on
plot(pa2psia.*p_i2,m32cm3.*V_press_mat(:,6))
hold on
plot(pa2psia.*p_i3,m32cm3.*V_press_mat(:,9))
hold on
plot(pa2psia.*p_i4,m32cm3.*V_press_mat(:,12))
hold on
xlabel('Pressure [psia]')
ylabel('Volume [cm^3]')
title([char(gas_name(3,1)),' for ',char(Sat_name)])
lgd3 = legend('300','200','150','100');
title(lgd3,{'Propellant Tank';'Pressure [psia]'})
```

Below is the code used to determine the pressure drop associated with head loss (friction and obstructions to the flow) in the feed system pressurization experiment. It is assumed that the pressurant side of the system is compressible and laminar and the propellant side is incompressible and laminar. This code can be expanded with velocities for each segment to determine the pressure drop due to head loss of the entirety of the system. This code can also be used if pressure drop is known, to approximate an average velocity to achieve the pressure drop.

```
close all;
clear all;
clc;

g=9.81; %m/s^2

%Experimental velocity
```

```
u=0.0912; %m/s
%u=.12; %m/s %simulation
%u=.1452143; %m/s max laminar
w=.0001640334; %kg/s mass flow rate
v=0.6036784; %m^3/kg
p1(1)=1379000; %Pa=200 psi
%p1(1)=1207000; %Pa=175 psi
 %p1(1)=1034000; %Pa=150 psi
% p1(1)=861845; %Pa=125 psi
% p1(1)=689476; %pa=100 psi
% p1(1)=517107; %pa=75 psi
% p1(1)=344738; %pa=50 psi
% p1(1)=172369; %pa=25 psi

m=2 %indicates which fluids are in the system
for j=1:2

if j==1
   %Argon
   rho=5.704; %kg/m^3
   mu=0.0000223; %Pa*s

%Water
if j==2;
mu=0.0010005; 22.%Pa*s
rho=998.21; %kg/m^3

% %IPA
% elseif j==3;
% mu=0.00204; %Pa*s
% rho=786; %kg/m^3
%
% %Olive Oil
% elseif j==4;
% mu=0.085; %Pa*s
% rho=908.7; %kg/m^3
%
% %Propellant
% else j==5;
% mu=0.1; %Pa*s
% rho=1422; %kg/m^3
 end

%number of segments
   n=9;
   for i=1:n
```

```
%    if i==1;
%    %tank to vertical
%    %D=1/4in;
%    D=0.00635; %m
%    L=0.08; %m
%    k=0.08+.4+.5; %threaded union, 45% elbow,square inlet
%
%    elseif i==2
%    %vertical to table
%    %D=1/4in;
%    D=0.00635; %m
%    L=0.5; %m
%    k=1.5; %threaded 90 degree elbow
%
%    elseif i==3;
%    %horz to cross
%    %D=1/4in;
%    D=0.00635; %m
%    L=0.74; %m
%    k=0.08+1; %threaded union, cross    %%%approximated k due to cross

   if i==4;
   %cross to storage tube
   %D=1/8in;
   D=0.003175; %m
   L=0.475; %m
   k=0.065+0.08; %gradual contraction,threaded union

   elseif i==5;
   %Prop storage tube
   %D=1/2in;
   mu=0.0010005; %Pa*s
   rho=998.21; %kg/m^3
   D=0.0127; %m
   L=0.19; %m
   k=0.325+0.08; %gradual expansion,threaded union

   elseif i==6;
   %Storage tube to tee
   %D=1/8in;
   D=0.003175; %m
   L=0.6183; %m
   k=0.0825+0.9; %gradual contraction, threaded line flow at tee
%
%    elseif i==7;
%    %tee to S.V.
```

```
%     %D=1/8in;
%     D=0.003175; %m
%     L=0.015; %m
%     k=0.08; %threaded union
%
%     elseif i==8;
%     %S.V.
%     %D=1/16in;
%     D=0.0015875; %m
%     L=0.02; %m
%     k=0.065+0.08; %gradual contraction, threaded union
%
%     elseif i==9;
%     %S.V. to outlet
%     %D=1/8in;
%     D=0.003175; %m
%     L=0.015; %m
%     k=1.5; %threaded 90 degree elbow
   end

%Major losses due to friction
%assuming laminar, smooth pipe flow
Re_D(i)=rho*u*D/mu;
%density of fluid
%u=velocity
%D=pipe diameter
%mu=viscosity of fluid;
f(i)=64/Re_D(i);

%if i==1 || i==2 || i==3 || i==4 && j==1
if i==1 && j==1
hL_min(i)=k*(u^2)/(2*g);
A=pi*(D/2)^2;
% fun=@(p2) (((A^2)/(v*(f(i)*L/D+2*log(p1(i)/p2))))*((p1(i)^2-p2^2)/p1(i)))-w^2;
% %p2=fzero(fun, p1(i));
% p2=fzero(fun, 1379000);
iiii=1;
for p2=1207000:0.01:1379000
fun=(((A^2)/(v*(f(i)*L/D+2*log(p1(i)/p2))))*((p1(i)^2-p2^2)/p1(i)))-w^2;
if abs(fun)<=.0005

   p2_new(iiii)=p2 ;
   iiii=1+iiii;
end
end
```

```
for jjj=1:1:length(p2_new);
del_p(i)=(p1(i)-p2_new(jjj))+rho*g*hL_min(i); %pa
p1(i+1)=p1(i)-del_p(i);
p1_count(jjj)=p1(i+1);
end
AA=max(p1_count);
[G,H]=find(AA==p1_count);

p2_new_FINAL=p2_new(H);
del_p_FINAL(i)=(p1(i)-p2_new_FINAL)+rho*g*hL_min(i); %pa
else

    hL_maj(i)=f(i)*((u^2)/(2*g))*(L/D);
%u=velocity
%g=acceleration due to gravity
%L=length of pipe segment

hL_min(i)=k*(u^2)/(2*g);

del_p_seg(i)=rho*g*(hL_maj(i)+hL_min(i)); %pa

end
    end
end


    j=j+1;

end
 del_p_tot=0.0001450377*(sum(del_p_seg)+sum(del_p_FINAL))
```

APPENDIX C.

RISK MITIGATION

A table summarizing risks and resulting actions associated with the feed system can be found on the following pages. This table is based on the HAZOP format, but does not contain a measure of severity for each risk.

Table C.1. Risk Assessment

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
1	Pressurant tank	Pressure	More	High pressure in the pressurant tank	Heat expansion within the pressurant tank	Increased flow rate until regulator	None	None	Effects from over pressurizing the tank will be stopped at the pressure regulator.
2	Pressurant tank	Pressure	Less	Low pressure in the pressurant tank	Leaks; Temperature drop	Slow propellant flow; low pressure at thruster feed	None	Leak proof testing; follow torque specification during assembly; Consider insulation or heating element	A loss of pressure in the pressurant tank will be caused by leaks or condensing gas and will reduce the pressure provided to the thruster.
3	Pressurant tank	Pressure	No	No pressure in pressurant tank	Large leak in the pressurant tank	No flow through the system; propulsion failure	None	Vibration testing; follow torque specs; Consider external connection reinforcement	A large containment failure in the pressurant tank could be caused by launch vibrations loosening a connection or causing disassembly. The disassembled component could shear off a connection or puncture the pressurant tank. Vibration testing on the satellite will help mitigate the likelihood of such an event. External connection reinforcement would mitigate the effects.
4	Pressurant tank	Temperature	More	Inert gas in pressurant tank too hot	Heat from batteries and other internal components; sunlight; heat from thruster	High pressure in tank; High downstream temperature	Piston head would protect propellant from increased temperature	Consider insulation to reduce heat absorption from surroundings	The problems caused by the pressurant tank overheating will be expressed downstream in increased flow rate and temperature of the pressurant. The piston head would protect the propellant from the increased temperature, but the propellant may be affected by increased flow rate.
5	Pressurant tank	Temperature	Less	Low temperature in the pressurant tank	Heat loss due to radiation	Condensing gas; slow flow; sputtering flow; slow flow downstream	Heater	Consider insulation to reduce heat loss	A drop in temperature will result in a drop in pressure. Since the thruster will still operate at reduced pressures (just less efficiently), the major concern associated with a temperature drop would be if the nitrogen were to condense. This is unlikely since the critical temperature of nitrogen gas is -146.9C and the satellite is unlikely to reach that temperature.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
6	Pressurant tank	Containment	Small failure	Small leak in the pressurant tank	Loose fitting; improper tightening; valve failure;	Low tank pressure	None	Leak proof testing; follow torque specification during assembly	A small leak will cause a loss of pressure in the pressurant tank and will reduce thruster efficiency.
7	Pressurant tank	Containment	Large failure	Large leak in the pressurant tank	Tank crack; launch vibrations could cause a connection to become loose	No flow; thruster failure	None	Significant vibration testing; follow torque specs; Consider external connection reinforcement	A large containment failure would be caused by launch vibrations either causing a connection to be loose, or creating a loose part that would then shear off a connection or puncture the tank.
8	Pressurant transport line	Flow	More	Flow from pressurant tank to fuel storage is too fast	Pressure regulator failure	High downstream pressure; downstream flow too fast	Relief valve (APEX)	None	The APEX satellite contains a relief valve to prevent over pressurization; the USIP satellite is too small to fit such a valve. Proper piping assembly will prevent a moderate pressure increase from causing a leak.
9	Pressurant transport line	Flow	Less	Flow from pressurant tank to fuel storage is too slow	Low tank pressure; condensed gas; press regulator failure; containment failure	Low downstream pressure; downstream flow too slow	None	Leak proof testing; follow torque specifications during assembly; vibration testing	Low pressure in this line will be caused by either a leak or the nitrogen condensing. A leak can be prevented by the same testing and prevention procedures defined in entries 6 and 7.
10	Pressurant transport line	Flow	No	No flow from the pressurant tank to the fuel storage pipe	Tank containment failure; valve fails closed; flow obstruction	System failure;	None	Follow leak prevention outlined in entries 6 and 7	A solenoid valve will fail closed if power is lost. It can be reopened if power is restored. If the tank or a connection experiences a large leak flow may also cease, proper assembly and leak testing should prevent this.
11	Pressurant transport line	Flow	Reverse	Reverse flow in the pressurant transport line (backwards flow from the fuel storage tube)	Upstream containment failure	System failure	Check valve	Shut all valves if a rapid pressure drop is detected	Reverse flow in the pressurant transport line can be caused by a large containment failure. The inclusion of the piston head in the fuel line prevents fuel from entering this line so reverse flow here will have minimal effects, however there is still a check valve included in the design.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
12	Pressurant transport line	Pressure	More	High pressure in the pressurant transport line	Pressure regulator failure; rise in temperature, piston head ceases movement	High downstream pressure; extra stress on pipe connections	Relief valve (APEX)	Follow torque specifications during pipe assembly to prevent over pressurization from causing a leak	The APEX satellite contains a relief valve to prevent over pressurization. The USIP satellite is too small to fit such a valve. Proper piping assembly will prevent a moderate pressure increase from causing a leak.
13	Pressurant transport line	Pressure	Less	Low pressure in the pressurant transport line	Pressure regulator failure; containment loss; condensing gases	Low downstream pressure	Pressure regulator	Follow leak prevention outlined in entries 6 and 7	Low pressure would most likely occur due to containment or pressure regulator failure. Containment failure can be prevented with proper pipe assembly. If the regulator fails low, the thruster will still operate at reduced pressures.
14	Pressurant transport line	Pressure	No	No pressure in the pressurant transport line	Large containment loss;	System failure; downstream reverse flow	None	Follow leak prevention outlined in entries 6 and 7; program the flight computer to shut all valves if the transducer detects a rapid pressure drop	No pressure in the line from the pressurant tank to the fuel storage tube will be caused by a large containment failure in the pressurant tank or a pipe connection, which could result in low flow. Other than prevention techniques, reverse flow could be mitigated by including controller logic that will shut all valves if the pressure transducer detects a rapid pressure drop.
15	Pressurant transport line	Temperature	More	High temperature in the pressurant transport line	Temperature too high in the pressurant tank; heat from: batteries, solar radiation, thruster combustion	Higher pressure; high downstream temperature	Heat sink in thruster	Consider thermal coatings; Consider shortening burn times to reduce heat absorbed by feed system	High temperature in this line may be caused by a temperature increase in the pressurant tank or by heat from batteries, the thruster, or from the sun. A temperature increase here may effect pressures and flow rates, but at this stage the biggest issue is that deviations will be passed downstream. Prevention may take the form of thermal coatings to control heat transfer or shortening burn times to reduce the rate at which heat is produced.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
16	Pressurant transport line	Temperature	Less	Low temperature in the line from the pressurant tank to fuel pipe	Heat loss due to radiation; low temperature in pressurant tank	Lower pressures; condensed gas; low downstream pressure	None	Use heater or batteries to warm the line	Cooling may lead to a pressure drop in the line- deviation will carry on down the system.
17	Pressurant transport line	Containment	Small failure	Small Leak	Loose fitting; improper tightening; valve failure	Pressure drop; downstream pressure drop	None	Follow torque specs during construction; Consider external connection reinforcement	A small leak in this line will cause a pressure drop downstream. If the leak is small enough the pressure regulator may be able to compensate- at least initially until the pressurant runs out, shortening the mission.
18	Pressurant transport line	Containment	Large failure	Large Leak	Crack; launch vibrations loosen a connection; connection sheared off by debris caused by launch vibrations	System failure, reverse flow downstream	Check valve	Program the flight computer to shut all valves if the pressure transducer detects a large drop. Vibration testing of the satellite	A large leak could be caused by launch vibrations loosening a connection, or by an unassembled part from elsewhere on the satellite shearing off a connection or puncturing the pipe. This could result in reverse flow downstream. The check valve and solenoid valve would prevent reverse flow.
19	Fuel storage tube	Flow	Reverse	Reverse flow in the fuel storage line	Large upstream containment failure	Thruster non-operational; potential fuel combustion	Check valve; piston head	Program the flight computer to shut all valves if the pressure transducer detects a large drop.	In order for a large containment failure to cause reverse flow in the fuel tube either the emergency valve closure and the check valve must fail or the leak must occur between the last upstream valve and the fuel pipe. In the event of reverse flow in the fuel pipe the piston head will prevent the fuel from flowing upstream by stopping at the pipe entrance. This is doubly advantageous because the more fuel is in the line the less reverse flow is possible. (this only applies to reverse flow initiated by leaks in this line).

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
20	Fuel storage tube	Flow	More	Flow in the fuel storage tank too fast	Upstream overpressure; upstream flow too fast; downstream leak; temperature too high	Run out of fuel early; downstream overpressure; downstream fast flow	Pressure relief valve (APEX)	Experimentation to find fuel viscosity's temperature dependence	A flow rate increase in the fuel storage pipe may be caused by upstream overpressure, an upstream flow rate increase, a downstream leak, or a temperature increase. Upstream overpressure and flow increase are dependent on simultaneous pressure regulator and relief valve failure. Experimentation should be done to find the dependence of the fuel's viscosity on temperature changes to see if temperature will have a significant impact on flow rate.
21	Fuel storage tube	Flow	Less	Flow in the fuel storage pipe too slow	Upstream leak/under pressure; upstream low flow; temperature too low	Downstream low flow; downstream under pressure	None	Experimentation to find fuel viscosity's temperature dependence; Consider heaters	A drop in flow rate at the fuel storage line could be caused by an upstream drop in flow or under-pressure, which can be traced back to the failure of the pressure regulator. It could also be caused by an upstream leak. If the difference in fuel viscosity caused by changing temperature is significant then a temperature drop will also slow the flow. This will most likely be prevented by the heater.
22	Fuel storage tube	Flow	No	No flow in the fuel storage pipe	Solenoid valve failure; upstream leak; piston head movement ceases	Thruster non-operational; downstream no flow	None	Follow leak prevention outlined in entries 6 and 7	Flow stopping in the fuel pipe may be caused by upstream leaks, which may be prevented according to the methods outlined in entries 6 and 7. Since adding more valves reduces the likelihood of unwanted flow, but increases the likelihood of unwanted flow stoppage (all valves need to fail to create unwanted flow, only one needs to fail to stop flow) there is a tradeoff between these two modes of failure.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
23	Fuel storage tube	Pressure	More	High Pressure in the fuel storage pipe	Upstream overpressure; High temp	Puts extra stress on connections, could cause a leak; downstream high pressure	Pressure relief valve (APEX)	Follow proper assembly procedures so an overpressure will not cause a leak	High pressure in the fuel storage pipe could be caused by an upstream overpressure or an increase in temperature and would result in high pressure fluid being sent downstream.
24	Fuel storage tube	Pressure	Less	Low pressure in the fuel storage pipe	Upstream leak/under pressure; pressure regulator failure	Downstream slow flow; downstream low pressure	None	Leak testing and follow proper assembly procedures	Low pressure in the fuel storage line can be caused by an upstream leak or under-pressure and would result in low pressure flow being sent down the system.
25	Fuel storage tube	Pressure	No	No pressure in the fuel storage pipe	Large upstream containment failure	Downstream no pressure; no flow; downstream no flow; downstream reverse flow	None	Leak Testing; Consider external connection reinforcement	No pressure in the fuel storage tank will be caused by a large upstream containment failure and will result in a cessation of flow and pressure downstream, with the possibility of reverse flow downstream.
26	Fuel storage tube	Temperature	More	High temperature in the fuel storage pipe	Batteries, solar radiation, radiation from thruster combustion	High pressure; high downstream pressure; high downstream temperature; fast flow, downstream fast flow	Pressure relief valve (APEX)	Thermal coatings; experimentation to find fuel viscosity's temperature dependence	A high temperature in the fuel storage tube can be caused by heat from the batteries, solar radiation, or the thruster and could result in a pressure increase or an increase in flow rate, deviations would be sent downstream.
27	Fuel storage tube	Temperature	Less	Low temperature in the fuel storage pipe	Heat loss due to radiation; heater failure	Slow flow rate; slow downstream flowrate; solid propellant	Heaters	Experimentation to find fuel viscosity's temperature dependence	Low temperature in the fuel storage tube can be caused by heat loss due to radiation compounded with a heater failure. Experimentation to find the fuel viscosity's dependence on temperature changes would help better define this risk.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
28	Fuel storage tube	Containment	Small failure	Small Leak	Loose fitting; improper tightening; O-ring failure	Run out of fuel early; corrosion of satellite internals; fuel could find its way into thruster and ignite; fuel could cause a short in a circuit board	Board shielding	Follow proper assembly procedures to prevent leaking; leak testing	If a small leak occurs in the fuel storage pipe it would be caused by a connection failure so proper assembly and leak testing are vital. If the leak occurs on the side containing fuel it will corrode satellite internals over time and could ignite if it came into contact with a heated thruster. The fuel is ionic and if it finds its way to a circuit board it may cause a short. The consequence of an O-ring failure will be mild; the satellite will run out of fuel early.
29	Fuel storage tube	Containment	Large failure	Large Leak	Crack; launch vibrations shake a connection loose	No flow; no pressure; downstream no flow; downstream no pressure; downstream reverse flow; short on a circuit board; spark ignition at a circuit board	Board shielding, software logic to close all solenoid valves to mitigate the leak as needed	Consider external connection reinforcement; add logic to shut all solenoid valves in the case of a rapid pressure drop; board shielding	A large containment failure could cause reverse flow from the thruster back to the fuel pipe, which could possibly cause fuel to combust in the fuel transport line. Logic to close all valves in the event of a rapid pressure drop may prevent the flame from progressing. A spark from a circuit board is also a potential ignition source so board shielding will be required.
30	Fuel storage tube	Flow	More	Flow in the fuel transport line too fast	Upstream overpressure or fast flow; leak at thruster feed; temperature increase	Run out of fuel early; overpressure at thruster feed; fast flow at thruster feed	None	Proper pipe assembly procedures must be followed	At this stage an increase in flow will be a cascading effect from upstream, a temperature increase, or a leak at the thruster feed. This will cause an overpressure and/or an increase in flow at the thruster feed and will cause the satellite to run out of fuel before the mission is complete, though still allowing partial mission success.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
31	Fuel Transport line	Flow	Less	Flow too slow in the fuel transport line	Upstream leak/under-pressure/slow flow; temperature drop; flow blockage	Slow flow at the thruster feed; under-pressure at the thruster feed	None	Proper pipe assembly procedures must be followed	Low flow in the fuel transport line will be caused by cascading effects from upstream or a temperature drop and will result in slow flow or low pressure at the thruster feed. These pipes are narrow and the fuel's viscosity is enough that even the smallest flow obstruction will have a significant effect on the flow rate so proper pipe assembly procedures must be followed.
32	Fuel Transport line	Flow	No	No flow in the fuel transport line	Solenoid valve failure; large upstream leak; piston head movement ceases	No flow into the thruster	None	Shut all valves if a rapid pressure drop is detected	No flow in the fuel transport line will result in no thruster feed and will be the result of a large upstream leak or valve failure. Shutting all valves in the event of a rapid pressure drop will prevent reverse flow.
33	Fuel Transport line	Flow	Reverse	Reverse flow in the fuel transport line	Large upstream containment failure	No thruster feed, potential for premature ignition	Check valves	Shut all valves if a rapid pressure drop is detected	Reverse flow will be caused by a large upstream containment failure and could allow heat to propagate upstream causing the propellant to react prematurely. Shutting all valves in the event of a rapid pressure drop will help mitigate the premature reaction by confining it to the fuel transport line.
34	Fuel Transport line	Pressure	More	High pressure in the fuel transport line	Upstream overpressure; flow blockage at the thruster feed; High temp	Over-pressurization at the thruster feed; fast flow at the thruster feed	None	Proper pipe assembly procedures must be followed	High pressure in the fuel transport line will be caused by cascading effects from upstream, an increase in temperature, or a flow obstruction at the thruster. Follow proper pipe assembly procedures to prevent such an obstruction.
35	Fuel Transport line	Pressure	Less	Low pressure in the fuel transport line	Upstream leak/under-pressure; piston head ceases movement; low temperature	Under-pressurization at thruster feed; low flow at thruster feed	None	Follow proper assembly procedures to prevent leaking; leak-proof testing	A low pressure in the fuel transport line will be the result of cascading effects from upstream or a drop in temperature. The deviation will make the thruster perform less efficiently.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
36	Fuel Transport line	Pressure	No	No pressure in the fuel transport line	Large upstream containment failure	No pressure at thruster feed; no flow at thruster feed, reverse flow from thruster	None	Shut all valves if a rapid pressure drop is detected	No pressure in the fuel transport line will be the result of a large containment failure and could cause reverse flow from the thruster. Check valves prevent reverse flow, but it could possibly be mitigated more quickly by shutting all valves if a rapid pressure drop is detected.
37	Fuel Transport line	Temperature	More	High temperature in the fuel transport line	Batteries, solar radiation, radiation from thruster combustion	Increase in flow rate, temperature, and pressure at the thruster feed	None	Thermal coatings; experimentation to find fuel viscosity's temperature dependence	Temperature increases become more likely because this line is close to the thruster. A temperature increase here can cause an increase in flow rate or pressure, deviations that will also be felt at the thruster feed.
38	Fuel Transport line	Temperature	Less	Low temperature in the fuel transport line	Heat loss due to radiation	Slow flow rate; slow flow rate at thruster feed	None	Experimentation to find fuel viscosity's temperature dependence	A temperature drop in the fuel transport line could cause a drop in pressure and a drop in flow rate, which will also be realized at the thruster feed.
39	Fuel Transport line	Containment	Small failure	Small leak in the fuel transport line	Loose fitting; improper tightening; valve failure; crack	Corrosion of satellite internals; fuel could find it's way into thruster and ignite; short in a circuit board	Board shielding	Follow proper assembly procedures to prevent leaking; leak testing; close solenoid valves in case of a leak to mitigate	If a small leak occurs in the fuel transport line it would be caused by a connection failure, so proper assembly and leak testing are vital. If fuel escapes into the satellite it could corrode satellite internals with enough time or ignite if it comes into contact with a heated thruster. This is unlikely with a small failure.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
40	Fuel Transport line	Containment	Large failure	Large leak in the fuel transport line	Launch vibrations shake a connection loose; debris shears off a pipe	No flow; no flow at thruster feed; potential for combustion; corrosion of satellite internals; short in circuit boards; spark ignition from circuit board	Board shielding, logic to close valves	Close all valves in the event of a rapid pressure drop; reinforce connections	A large containment failure could cause reverse flow from the thruster back into the fuel pipe. This could possibly cause fuel to combust in the transport line. Programming the flight computer to close all solenoid valves in the event of a rapid pressure drop (in addition to check valves) will prevent the reaction from propagating all the way back to fuel storage. A spark from a circuit board is also a potential ignition source, so board shielding is required.
41	Thruster Feed	Flow	More	Thruster feed flow rate too high	Upstream overpressure; upstream fast flow; increase in temperature	Inefficient performance; run out of fuel early	None	None	If the thruster feed is too fast it is either the result of cascading effects from upstream deviations or a temperature increase. If the feed is too fast the thruster will still work, but not as efficiently.
42	Thruster Feed	Flow	Less	Thruster feed flow rate too low	Upstream leak/under-pressure	Inefficient performance	None	None	If the thruster feed is too slow it is either the result of cascading effects form upstream deviations or a temperature decrease. If the flow into the thruster is too slow it will still work, but not as efficiently.
43	Thruster Feed	Flow	No	No flow at the thruster feed	Solenoid valve failure; large upstream leak; upstream no flow	Thruster non operational	None	None	If the flow at the thruster feed stops it will be because of a large upstream leak or a valve failure. As stated above adding more valves decreases the likelihood of reverse flow, but increases the likelihood of valve failure, stopping flow.

Table C.1. Risk Assessment (Cont.)

Entry Number	Unit	Parameter	Guide Word	Deviation	Causes	Effects	Mitigating Components	Actions Required	Summary
44	Thruster Feed	Flow	Reverse	Reverse flow at thruster feed	Large upstream containment failure	Fuel deflagration	Check valve	Close all valves in the event of a rapid pressure drop	Reverse flow at the thruster feed can be caused by a large upstream containment failure. If reverse flow occurs here it will result in premature reaction propagating up the fuel transport pipe, potentially entering and igniting the fuel storage tube. Closing all valves at the detection of a rapid pressure drop will at least contain the flame propagation to the fuel transport line.
45	Thruster Feed	Pressure	More	High pressure at the thruster feed	Upstream over-pressurization; temperature increase	Inefficient thruster performance	None	Follow proper pipe assembly procedure to prevent leaks; leak testing	If the pressure at the thruster feed is too high the thruster will not perform efficiently.
46	Thruster Feed	Pressure	Less	Low pressure at the thruster feed	Upstream leak/under-pressurization, temperature decrease	Inefficient thruster performance	None	None	If the pressure at the thruster feed is too low the thruster will not perform efficiently.
47	Thruster Feed	Pressure	No	No pressure at the thruster feed	Large upstream containment failure; valve failure	Thruster inoperable	None	Reinforce external connections	If the thruster feed stops the thruster will stop working. One possible cause of this is a valve failing closed, adding more valves will prevent reverse flow but will increase the likelihood of valve failure affecting the thruster feed.
48	Thruster Feed	Temperature	More	High temperature at the thruster feed	Batteries, solar radiation, radiation from thruster combustion, higher voltage than expected	Inefficient performance;	Voltage regulator	Cut power if overheating indicating by thermal sensors	A high temperature in the thruster feed will result in inefficient performance. A voltage regulator is in place to prevent heat caused by overvoltage and procedures to cut power from select components can be added to the flight computer to lower temperature.
49	Thruster Feed	Temperature	Less	Low temperature at the thruster feed	Heat loss due to radiation, lower voltage than expected	Inefficient performance	Heaters	None	If the temperature of the thruster feed drops it will result in inefficient thruster performance.

APPENDIX D.

M-SAT TEAM COMPOSITION SURVEY

The author sent a survey to the M-SAT team to determine team composition and motivation for participating in M-SAT activities. Thirty-four students participated from all three projects. Select results follow:

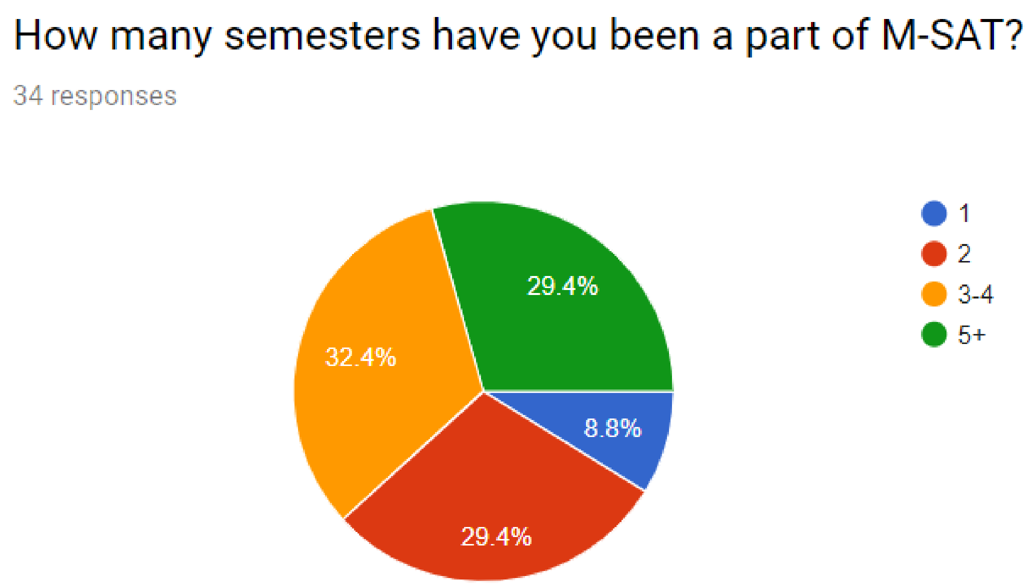

Figure D.1. Duration of Team Activity.

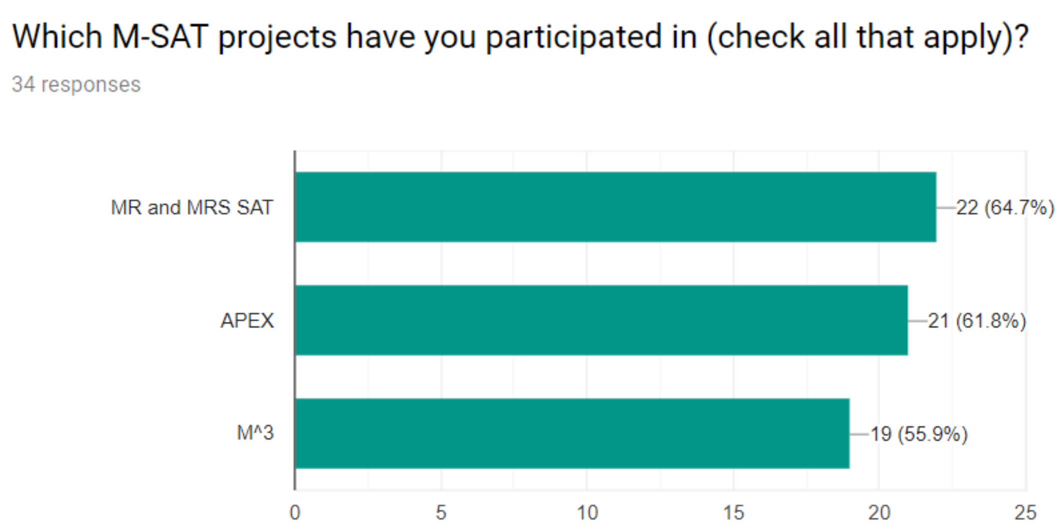

Figure D.2. Projects Participated.

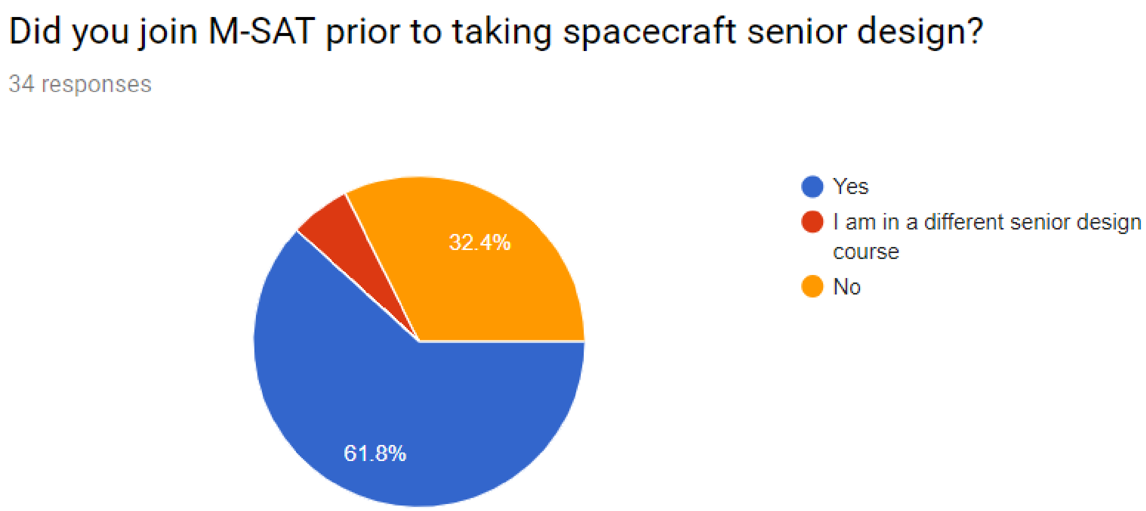

Figure D.3. Students That Join M-SAT as Seniors.

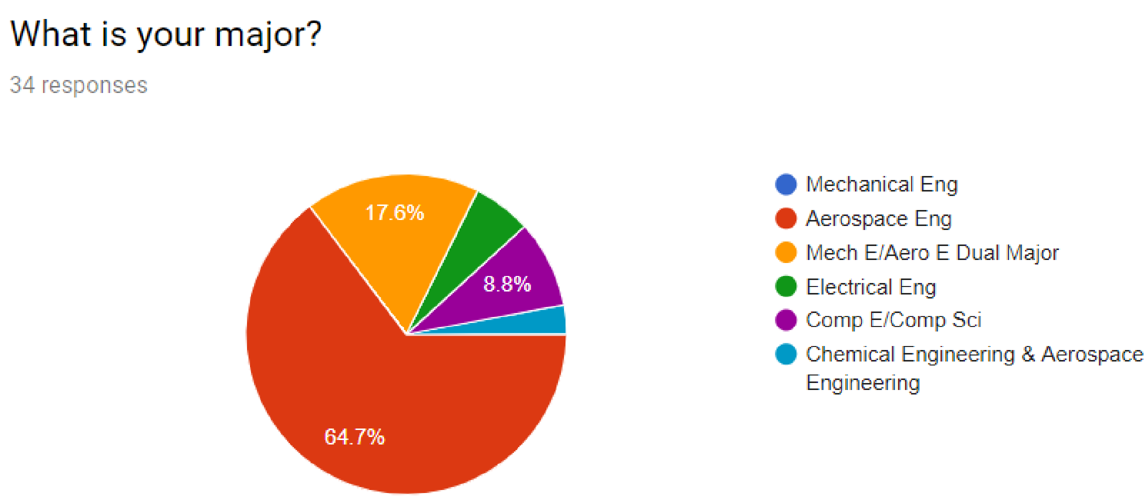

Figure D.4. M-SAT Team Composition: Majors.

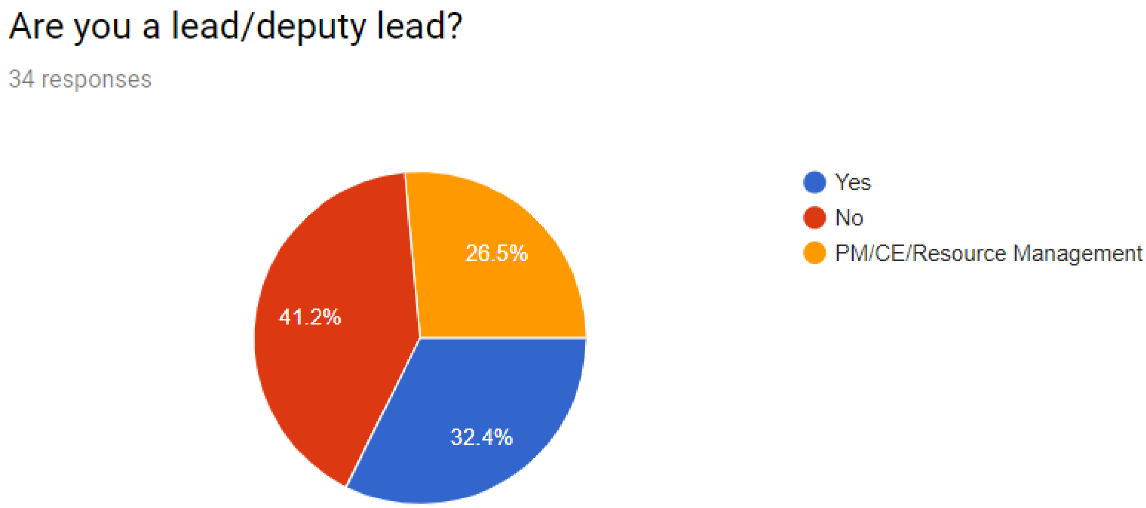

Figure D.5. Students in Leadership Roles.

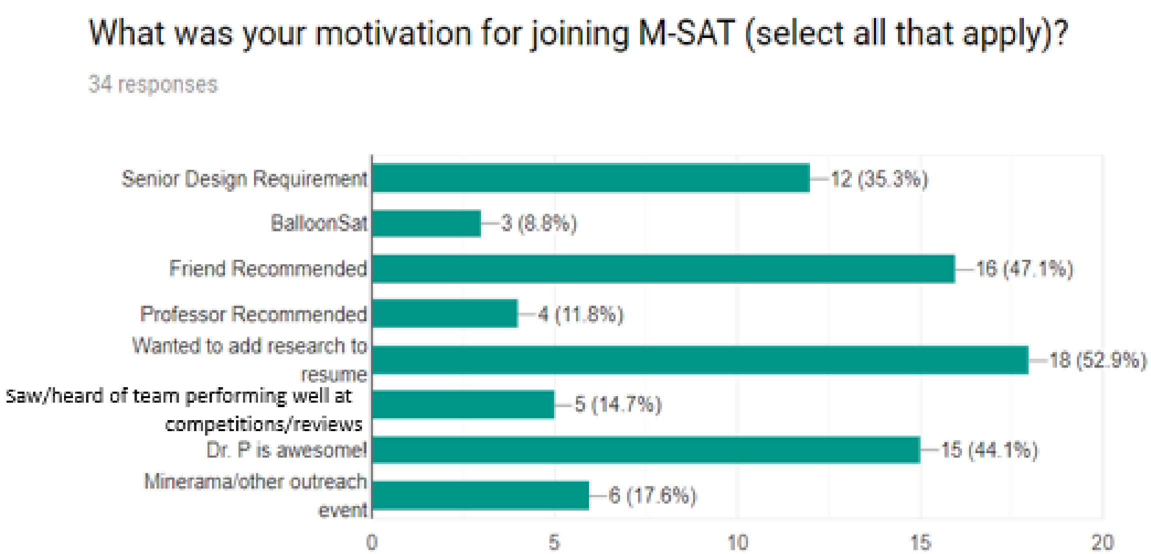

Figure D.6. Motivation for Joining M-SAT Research Team.

Are you considering grad school (at Missouri S&T or elsewhere)?

34 responses

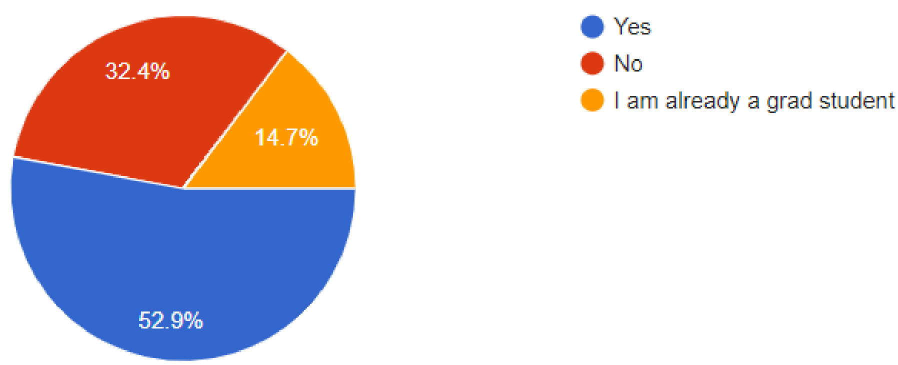

Figure D.7. Graduate School Interest.

Expected graduation date

31 responses

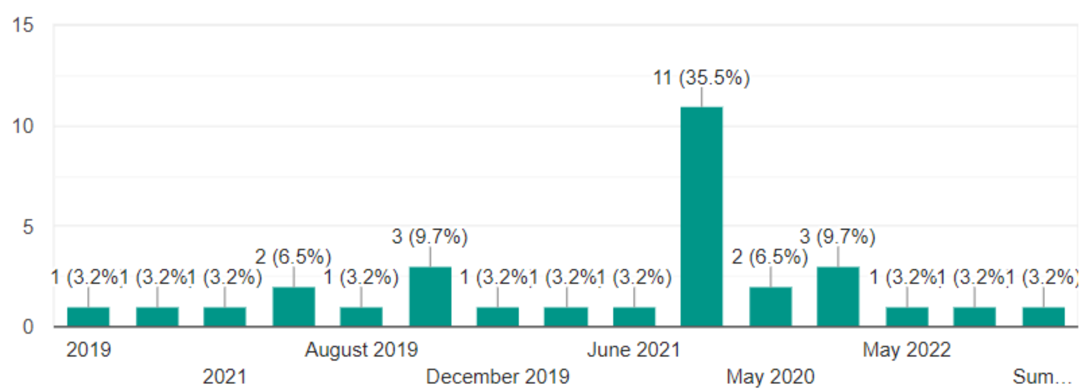

Figure D.8. Expected Graduation Dates.

BIBLIOGRAPHY

1. Dunbar, Brain. “Small Satellite Missions.” NASA. Ed. Sarah Loff., 20 Oct. 2016. Web. 09 Feb. 2017. <https://www.nasa.gov/mission pages/smallsats>. Accessed 07 Feb. 2017.

2. Williams, Kimberly. “NASA Awards Contract for Series of CubeSat Technology Missions.” NASA. Ed. Loura Hall., 10 Jan. 2017. Web. 21 Feb. 2018. <https://www.nasa.gov/directorates/spacetech/small spacecraft/feature/CubeSat Technology Missions>. Accessed 07 Feb. 2018.

3. Gustafson, Charles L., and Siegfried W. Janson. "Think Big, Fly Small." The Aerospace Corporation. *Crosslink Magazine*, 1 Sept. 2014. Accessed 07 Feb. 2018.

4. “What Is a CubeSat?” *Canadian Space Agency*, Government of Canada, 27 Nov. 2018, http://www.asc-csa.gc.ca/eng/satellites/cubesat/what-is-a-cubesat.asp. Accessed 11 Mar. 2019.

5. Parker, Khary I. State-of-the-Art for Small Satellite Propulsion Systems. 2nd Planetary CubeSat Science Institute, 26 Sept. 2017, Greenbelt, MD, NASA/Goddard Space Flight Center.

6. Polzin, Kurt. Electric Propulsion at NASA-Marshall Flight Center. Graduate Seminar Series. 19 Mar. 2018, Missouri University of Science and Technology, Rolla, MO, Toomey Hall.

7. Smith-Strickland, Kiona., “Coming Soon: Swarms of Space Robots.” *Air & Space Magazine*, 6 May 2016, www.airspacemag.com/daily-planet/coming-soon-swarms-space-robots-180959020/. Accessed 07 Feb. 2018.

8. Patel, Shivani. et al. 2018. “Helping Students Reach Space: University Nanosatellite Program Challenges and Lessons Learned,” *Proceedings of the AIAA/USU Conference on Small Satellites*, Session 8: Educational Programs II. https://digitalcommons.usu.edu/smallsat/2018/all2018/464/. Accessed 10 Mar. 2019.

9. Stewart, Abbie Marie. *A Guide to the Establishment of a University Satellite Program*. Master’s thesis, Missouri University of Science and Technology, 2007.

10. Dahm, Werner J. A. *Technology Horizons: A Vision for Air Force Science & Technology During 2010-2030*. Office of the Chief Scientist of the U.S. Air Force, 2010, p. 73, *Technology Horizons: A Vision for Air Force Science & Technology During 2010-2030*.

11. *NASA Technology Roadmaps TA 2: In-Space Propulsion Technologies*. NASA, 2015, *NASA Technology Roadmaps TA 2: In-Space Propulsion Technologies*.

12. M. Martinez-Sanchez and J. E. Pollard. "Spacecraft Electric Propulsion-An Overview", *Journal of Propulsion and Power*, Vol. 14, No. 5 (1998). https://www.researchgate.net/publication/245434828_Spacecraft_Electric_Propulsion An_Overview. Accessed 10 Mar. 2019.

13. Lemmer, Kristina. "Propulsion for CubeSats." *Acta Astronautica*, Vol. 134, Feb. 2017, pp. 231–243.

14. Palmer, Kristoffer, et al. *In-Orbit Demonstration of a MEMS-Based Micropropulsion System for Cubesats*. SSC Nanospace briefing.

15. J. Kolbeck, J. Lukas, G. Teel, M. Keidar, E. Hanlon, J. Pittman, M. Lange, J. Kang, μCAT Micro-Propulsion Solution for Autonomous Mobile On-Orbit Diagnostic System, in: Proceedings of the 30th Annu. AIAA/USU Conference Small Satell., Logan, UT: SSC16–V–7, 2016.

16. F. Rysanek, J.W. Hartmann, J. Schein, R. Binder, Micro Vacuum Arc Thruster Design for a CubeSat Class Satellite, in: Proceedings of the 16th Annu. USU Conference Small Satell: SSC02-I-2, 2002.

17. NASA: National Space Science Data Center, ⟨http://nssdc.gsfc.nasa.gov/nmc/ spacecraftDisplay.do?Id=2015-025J⟩ Accessed 10 Mar. 2019.

18. D. Krejci, F. Mier-Hicks, C. Fucetola, P. Lozano, A.H. Schouten, F. Martel, Design and Characterization of a Scalable ion Electrospray Propulsion System, in: Proceedings of the 34th International Electr. Propuls. Conference, Kobe, Japan: IEPC-2015–2149, 2015.

19. Busek Co. Inc. *BIT-3 RF Ion Thruster*. *BIT-3 RF Ion Thruster,* Busek Space Propulsion and Systems, 2018.

20. Withrow-Maser, Shannah, and Matthew Klosterman. "AIAA Region V Student Paper Conference." *Multi-Mode Micropropulsion Systems Enabling Swarm Technology*, 2018.

21. Myers, Roger M, Oleson, Steven R, Curran, Francis M, Schneider, Steven J. "30th Joint Propulsion Conference." AIAA-94-2997, NASA Technical Memorandum 106701, ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19950005075.pdf. Accessed 11 Mar. 2019.

22. Parker, Khary I. "State-of-the-Art for Small Satellite Propulsion Systems." 2nd Planetary CubeSat Science Institute, 26 Sept. 2017, Greenbelt, MD, NASA/Goddard Space Flight Center.

23. Berg, Steven Paul. *Development of Ionic Liquid Multi-Mode Spacecraft Micropropulsion Systems*. Ph.D. dissertation, Missouri University of Science and Technology, 2015.

24. Mundahl, Alex J. *Characterization of Ionic Liquid Monopropellants for a Multi-Mode Propulsion System*. Master's thesis, Missouri University of Science and Technology, 2018.

25. Taylor, Andrew Paul. *Development of a CFD Model of the Catalytic Combustion of a Microtube Multi-Mode Propulsion System*. Master's thesis, Missouri University of Science and Technology, 2018.

26. Morton, Bradyn, and Shannah Withrow-Maser. "Small Satellite Conference." *On-Orbit CubeSat Performance Validation of a Multi-Mode Micropropulsion System*, 2017.

27. *316 Stainless Steel Double Ended Miniature Sample Cylinder, 50 cm3, 1000 Psig (68.9 Bar): SS-4CD-TW-50*, Swagelok Product Catalog. Accessed 10 Mar. 2019.

28. *IEP Series: IEPA1221241H*. *IEP Series: IEPA1221241H*, The Lee Company Product Catalog. Accessed 10 Mar. 2019.

29. *Poppet Check Valves: SS-CHS2-1* . *Poppet Check Valves: SS-CHS2-1* , Swagelok Product Sheet. Accessed 10 Mar. 2019.

30. *Vent Relief Valve High Pressure*. *Vent Relief Valve High Pressure*, Generant Product Sheet. Accessed 10 Mar. 2019.

31. *High-Precision 1-Wire Digital Thermometer: DS18S20*. *High-Precision 1-Wire Digital Thermometer: DS18S20*, Maxim Integrated Product Sheet. Accessed 10 Mar. 2019.

32. *Miniature Pressure Transducer: EPRB-1*. *Miniature Pressure Transducer: EPRB-1,* TE Connectivty Product Sheet. Accessed 10 Mar. 2019.

33. Chang, J. B., and R. W. Seibold. *Operational Guidelines for Spacecraft Pressure Vessels.* The Aerospace Corporation, 2005, *Operational Guidelines for Spacecraft Pressure Vessels*.

34. Fizell, Zachary. "AIAA Region V Student Paper Conference." *Adaptation of a University Satellite Design for Military and Industry Applications*, 2019.

35. Savovic, Zoran. "Compressible Gas Flow." *Flow in Pipe - Bernoulli Equation*, Pipe Flow Calculations, www.pipeflowcalculations.com/pipe-valve-fitting-flow/compressible-gas-flow.xhtml. Accessed 10 Mar. 2019.

36. “What Is Leadership?.” *David Marquet*, 18 Jan. 2018, www.davidmarquet.com/our-story/. Accessed 10 Mar. 2019.

37. Yuri Gerasimov, Rich Kelly, and Vitor A. Cardoso. "Pressure Transients in Propulsion System Feedlines", 53rd AIAA/SAE/ASEE Joint Propulsion Conference, AIAA Propulsion and Energy Forum, (AIAA 2017-5005

38. Krusharev, I. *Micro-Thruster Development*. Master’s thesis, Delft University of Technology, 2015.

39. Morton, Bradyn William. “Quantifying On-Orbit Performance of CubeSat Micropropulsion Systems by Observing Orbital Element Variations.” *Missouri University of Science and Technology*, 2018.

40. Thomas Moore, “Magnetospheric Multiscale Code 461.” NASA Godard Space Flight Center, NASA, mms.gsfc.nasa.gov/spacecraft.html. Accessed 07 Feb. 2018.

41. Kramer, Herbert J. TDX (TanDEM-X: TerraSAR-X Add-on for Digital Elevation Measurement). EoPortal Directory - Satellite Missions, directory.eoportal.org/web/eoportal/satellite-missions/t/tandem-x. Accessed 07 Feb. 2018.

42. Tooley, Craig. “NASAs Magnetospheric Multiscale (MMS) Mission.” 2015 University of North Dakota Space Studies Colloquium. Accessed 07 Feb. 2018.

43. Davis, Lorrie A, and Lucien Filip. “How Long Does It Take to Develop and Launch Government Satellite Systems? .” 12 Mar. 2015, www.iceaaonline.com/ready/wp-content/uploads/2014/03/Davis-Satellite-ICEAASoCal-090915.pdf. Accessed 07 Feb. 2018.

44. Withrow, Shannah. “M-SAT Composition”. Questionnaire. 6 March 2019.

45. Welsch, Carol. “Systems Engineering at Northrop Grumman Innovation Systems-Albuquerque.” July 2018.

46. Hoffman, Gerald. “Systems Engineering Tools.” SYS ENG 6105: Complex Engineering Systems Project Management. SYS ENG 6105: Complex Engineering Systems Project Management, Feb. 2018, Rolla, MO, Missouri University of Science and Technology.

47. Blau, Patrick. “MarCO – Mars Cube One.” *SPACEFLIGHT101.com*, Insight Spaceflight 101, 4 Apr. 2019, spaceflight101.com/insight/marco/. Accessed 04 April. 2019.

VITA

Shannah Nichole Withrow (Maser) was born in 1994, and grew up in Odessa, Missouri. Shannah started attending Missouri University of Science and Technology after graduating from Summit Christian Academy in 2012. As an undergraduate, Shannah was president of the Miners in Space Microgravity Research team, actively participated in Christian Campus Fellowship, played and coached intramural soccer, was a member of the Sigma Gamma Tau Aerospace Honor Society and Kappa Mu Epsilon Mathematics Honor Society, and joined the M-SAT satellite research team her last three semesters. As part of the M-SAT team, she was active on the Guidance, Navigation, and Control, Communications, and Integration subsystems. In January of 2016, Shannah became the Program Manager of the M-SAT Advanced Propulsion Experiment (APEX) mission. She also participated in two summer internships, 2014 and 2015, at NASA Ames Research Center. In December of 2016, Shannah received her B.S. in Aerospace Engineering at Missouri S&T. She then returned to Missouri S&T to pursue her M.S. in Aerospace Engineering with an emphasis in Systems Engineering. She continued in her role of Program Manager, as well as, becoming the propulsion subsystem lead for APEX and the primary graduate student mentor (pseudo-PI) for M-SAT's Multi-Mode Micropropulsion (M^3) mission. Shannah completed a summer internship at Orbital ATK (now Northrop Grumman Innovation Systems) and an additional eight months in the Pathways Program at NASA Ames before receiving her Master of Science in Aerospace Engineering in May of 2019 from Missouri S&T.