Additively-Manufactured Hybrid Rocket Consumable Structure for CubeSat Propulsion

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ADDITIVELY-MANUFACTURED HYBRID ROCKET CONSUMABLE STRUCTURE FOR CUBESAT PROPULSION

by

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Aerospace Engineering

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ABSTRACT

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by

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Three-dimensional, additive printing has emerged as an exciting new technology for the design and manufacture of small spacecraft systems. Using 3-D printed thermoplastic materials, hybrid rocket fuel grains can be printed with nearly any cross-sectional shape, and embedded cavities are easily achieved. Applying this technology to print fuel materials directly into a CubeSat frame results in an efficient, cost-effective alternative to existing CubeSat propulsion systems. Different 3-D printed materials and geometries were evaluated for their performance as propellants and as structural elements. Prototype "thrust columns" with embedded fuel ports were printed from a combination of acrylonitrile butadiene styrene (ABS) and VeroClear, a photopolymer substitute for acrylic. Gaseous oxygen was used as the oxidizer for hot-fire testing of prototype thrusters in ambient and vacuum conditions. Hot-fire testing in ambient and vacuum conditions on nine test articles with a combined total of 25 s burn time demonstrated performance repeatability. Vacuum specific impulse was measured at over 167 s and maximum thrust of individual thrust columns at 9.5 N. The expected $\Delta V$ to be provided by the four thrust columns of the consumable structure is approximately 37 m/s.
With further development and testing, it is expected that the consumable structure has the potential to provide a much-needed propulsive solution within the CubeSat community with further applications for other small satellites.

(114 pages)
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<td>-------------------</td>
<td>--------------------------------------------------</td>
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<tr>
<td>ABS</td>
<td>acrylonitrile butadiene styrene</td>
<td></td>
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<tr>
<td>ADR</td>
<td>active debris removal</td>
<td></td>
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<tr>
<td>CEA</td>
<td>Chemical Equilibrium with Applications</td>
<td></td>
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<tr>
<td>COTS</td>
<td>Commercial-off-the-shelf</td>
<td></td>
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<tr>
<td>ESA</td>
<td>European Space Agency</td>
<td></td>
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<tr>
<td>FDM</td>
<td>fused deposition modeling</td>
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<tr>
<td>GOX</td>
<td>gaseous oxygen</td>
<td></td>
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<tr>
<td>HVPS</td>
<td>high-voltage power supply</td>
<td></td>
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<tr>
<td>IADC</td>
<td>Inter-Agency Space Debris Coordination Committee</td>
<td></td>
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<td>ISRO</td>
<td>Indian Space Research Organisation</td>
<td></td>
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<td>ISS</td>
<td>International Space Station</td>
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<td>LEO</td>
<td>Low earth orbit</td>
<td></td>
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<td>MSFC</td>
<td>Marshall Space Flight Center</td>
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<tr>
<td>N2O</td>
<td>nitrous oxide</td>
<td></td>
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<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
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<tr>
<td>P-POD</td>
<td>Poly Picosatellite Orbital Deployer</td>
<td></td>
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<td>PPU</td>
<td>power processing unit</td>
<td></td>
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<tr>
<td>PT</td>
<td>pressure transducers</td>
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<tr>
<td>RSO</td>
<td>Resident space object</td>
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<tr>
<td>RTV</td>
<td>room-temperature-vulcanizing</td>
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<td>SCAPE</td>
<td>self-contained atmospheric protective ensemble</td>
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<td>U</td>
<td>Unit, CubeSat</td>
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<td>USU</td>
<td>Utah State University</td>
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<td>VI</td>
<td>Virtual Instrument</td>
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NOMENCLATURE

$\Delta V$  change in velocity
$I_{sp}$  specific impulse
$a$  regression rate constant
$n$  regression rate exponent
$G_{ox}$  oxidizer mass flux
$\dot{r}$  regression rate
$\tau$  regression rate coefficient scale factor
$n$  regression rate exponent factor
$P_r$  Prandtl number
$\rho_f$  fuel density
$\Delta h$  change in enthalpy
$h_v$  specific heat of vaporization
$\mu$  dynamic viscosity
$L$  fuel grain port length
$\sigma$  Stefan-Boltzmann constant
$\epsilon$  optical emissivity
$T_0$  combustion flame temperature
$\alpha$  optical absorptivity
$T_f$  surface temperature
$C^*$  characteristic velocity
$O/F$  oxidizer to fuel ratio
$p_i$  internal pressure
$p_o$  external/outer pressure
$r_i$  internal radius
$r_o$  outer radius
$F$  thrust
$\dot{m}_{T,e}$  total exit mass flow rate
$V_e$  nozzle exit velocity
$A_e$  nozzle exit area
$\Theta$ nozzle expansion angle

$P_\infty$ ambient pressure

$P_e$ nozzle exit pressure

$e_0$ initial orbital eccentricity

$a_0$ initial semi-major axis

$n$ mean motion

$I_1$ Bessel function of the first kind and order

$\beta$ ballistic coefficient

$\rho$ atmospheric density

$H$ scale height

$r_i$ initial orbital radius

$r_f$ final orbital radius
CHAPTER 1
INTRODUCTION

With the introduction of the standardized CubeSat concept in 1999, the ease of getting a small satellite project launched to orbit drastically increased. Classified as a nano-satellite, a 1-unit (U) CubeSat has a nominal mass of 1.33 kg and volume of 10 cm$^3$. Several U’s can be combined to make larger spacecraft, as illustrated in Figure 1, although a 6U is the largest flown to date$^1$.

![Figure 1. Multiple U CubeSat configurations](image)

The most prominent allure of CubeSats is their affordability. Compared to a traditional satellite, CubeSats are at least two orders of magnitude less expensive$^2$. Table 1 lists approximate costs of satellites by class. The price gap between classes of satellites is growing with the CubeSat’s popularity as components are in higher demand. Many CubeSat components are now available for commercial-off-the-shelf (COTS) purchase, and as long as the spacecraft adheres to California Polytechnic State University’s CubeSat Standard, available in Appendix A, launch opportunities via rideshare on a

<table>
<thead>
<tr>
<th>Class</th>
<th>Mass (kg)</th>
<th>Cost ($M)</th>
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<tr>
<td>Large Satellite</td>
<td>&gt;1000</td>
<td>&gt;150</td>
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<td>Small Satellite</td>
<td>500-1000</td>
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<td>Minisatellite</td>
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<td>Microsatellite</td>
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<td>Nanosatellite</td>
<td>1-10</td>
<td>0.15-1.5</td>
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<tr>
<td>Picosatellite</td>
<td>&lt;1</td>
<td>&lt;0.15</td>
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larger launch vehicle are readily available. Launch frequency and capacity are on the rise, with providers across the world competing for a share in the space launch market. SpaceX in particular has become a major disruptive force in the market. Elon Musk, the CEO of SpaceX, stated in an interview in May 2018 that their “…Falcon 9 was the most-launched rocket worldwide in 2017. If things go well — big caveat — SpaceX will launch more rockets than any other country in 2018.” SpaceX along with most other launch providers, often fills any remaining payload budget with CubeSats. In February 2017, The Indian Space Research Organisation (ISRO) launched 104 satellites aboard a single PSLV-C37 rocket, 101 of which were CubeSats. Additionally, several dedicated small satellite launchers are in development that will allow CubeSats greater if not complete control over launch date and orbit.

As of June 2018, over 798 CubeSats have been launched or manifested to varying degrees of operational success, as catalogued in Swartwout’s CubeSat Database. This number includes constellations of CubeSats—hundreds of CubeSats working in synchrony—that take the place of a few larger satellites. Because they are relatively inexpensive to build and have regulations to help direct the design process, entities of varying experience levels have launched CubeSats with missions to provide communication, gather scientific data, test new technologies, and teach students a range of hands-on skills. The small satellites are ideal for testing prototype technology, especially that which could pose too high of a risk to the mission success of a conventional satellite. Low-budget satellites and missions, however, come with a host of associated obstacles.

The object of this thesis was to examine a common obstacle for CubeSats: the
propulsion system. As more CubeSats are launched, the need for an effective propulsion system becomes more desperate. Currently, few systems are available, and most are suitable only for attitude control or light station keeping. A more powerful propulsion system is desired to support more advanced maneuvers. A solution was developed in the form of hybrid rocket motors built directly into the CubeSat frame to serve as high-thrust onboard thrusters. In the following chapters, a literature review expands upon the need for CubeSat propulsion and describes available options. Next, background on hybrid rocket mechanics and previously completed work relevant to this thesis is provided. The design process of the hybrid rocket consumable structure is detailed followed by the results of hot-fire testing. Finally, the results are discussed, a systems analysis of the consumable structure is presented, and future work is suggested.
CHAPTER 2
LITERATURE REVIEW

2.1 Overview of Current CubeSat RSO Population

Since the launch of Sputnik in 1957, thousands of objects have been sent into space. While many have since returned to Earth or decomposed upon atmospheric reentry, others are still in orbit as resident space objects (RSO). The sentiment held by many for years was that space is “big,” and any number of satellites could be launched to orbit with no chance of accidental interaction; that assumption has since proven false.

The first confirmed collision in space occurred in 1996 when French satellite Cerise was struck by a catalogued piece of debris. Space debris is any man-made object in space that is not under precise control including inoperable satellites, spent and jettisoned rocket stages, and even paint chips. More collisions have occurred since the Cerise incident, damaging satellites and generating even more debris.

With more satellites being launched every year and new initiatives to advance the state of the art of human space travel, the problems associated with space debris will continue to grow. In a 2014 study, Lewis predicts that by 2023 one in every 10 on-orbit conjunctions will involve a CubeSat, and collisions involving CubeSats could generate up to one fragment of debris for every 4.4 CubeSats on-orbit. Figure 2 shows the number of

![Figure 2. Number of CubeSats launched annually 2000-2017](image-url)
CubeSats launched annually to date. Considering the exponential increase in CubeSats launches since 2014, the number of conjunctions generated today is likely significantly larger. In an attempt to forestall a catastrophic increase in debris, the Inter-Agency Space Debris Coordination Committee (IADC) has imposed a 25-year lifetime limit on satellites. Any satellite in low earth orbit (LEO), is expected to either relocate to a less-populated disposal orbit or fully de-orbit within 25 years of the end of its mission. LEO is heavily populated with satellites, and a satellite must pass through LEO to reach higher orbits, so the likelihood of collisions in LEO is high. The majority of early CubeSats were launched into LEO, but the destination altitudes for many CubeSats is increasing, as shown in Figure 3. Several studies by NASA, ESA, and others have concluded that the growing numbers of large, higher-altitude-orbiting CubeSats are beginning to pose a significant space debris hazard.\textsuperscript{7,9,10} The escalating problem has caught the attention of the President of the United States as well. In a June 2018 meeting of the National Space Council, President Donald Trump signed a directive with the goal of reducing in-space collisions and debris generation.\textsuperscript{11}
Concern over increasing space debris is not sufficient to forestall the increasingly advanced missions that CubeSats aim to accomplish. Complex science missions often require precise orbits, but CubeSat orbits are largely constrained by the orbit of the primary payload aboard the rideshare. The maximum altitude and inclination is dictated by the primary payload, and CubeSats are jettisoned anywhere along the way. The other available option for launching a CubeSat is to have it released from the ISS, which allows for an even smaller range of orbits.

2.2 CubeSat Standard and Propulsion Restrictions

CubeSat affordability is achieved through the size constraints, weight restrictions, cube form factor, and other limitations that allow CubeSats to safely participate in rideshares. Unfortunately, the same standards have limited the capability of these small spacecraft, particularly in regard to propulsion. Before June 2013, the standard restrictions on propulsion included 1) Pyrotechnics shall not be permitted, 2) Total stored chemical energy will not exceed 100 Watt-Hours, and 3) No pressure vessels over 1.2 standard atmosphere shall be permitted. The current standard, available in Appendix A, was revised to omit (3) while adding 3) Propulsion systems shall have at least 3 inhibits to activation, and 4) Any propulsion systems shall be designed, integrated, and tested in accordance with AFSPCMAN 91-710 Volume 3. These specifications have all but excluded traditional chemical propulsion systems from the CubeSat propulsion market; only electric, cold gas, and monopropellant systems could meet the standards or waiver process.

Beyond the restrictions aimed directly at propulsion, the CubeSat shape is restrictive itself. Fitting the tanks, valves, and plumbing required for a liquid bi-
propellant system, for example, in even a 3U is prohibitively challenging. A 3,000 cm$^3$ volume and 4 kg mass are requirements that require creative propulsion system designs. Perhaps of greater concern is the fact that many propulsion systems have large power requirements, which can be challenging to accommodate in the small space and limited budgets that CubeSats often have.

For most CubeSats, in-space maneuvering is limited to basic station keeping and attitude control; collision avoidance is significantly out of scope. Efficient propulsion systems will be required to advance the capabilities of CubeSats to avoid inflicting damage to other spacecraft and meet increasingly more ambitious mission goals.

### 2.3 CubeSat Propulsion Systems

While many conceptual variations on CubeSat propulsion systems are available, few have been flight tested, and most are electric, cold gas, or use toxic monopropellants like hydrazine. Commercially-available propulsion systems for small satellites vary widely in fuel source, power requirements, and dimension, but nearly all systems have low overall thrust or are non-impulsive. Most thrusters for small satellites are rated in milli-Newton or micro-Newton. Large-thrust propulsion systems exist but are often too large by mass or volume for use on a CubeSat. A sample of CubeSat propulsion systems available or in development are compared in Table 2.

Of all systems considered, the highest total thrust is from Aerojet Rocketdyne’s 1.5 N MPS-130 thruster; however, this system is still in development and has an anticipated mass of 2.4-3.5 kg, making it an unlikely candidate for most CubeSats.
Busek’s 0.5 N BGT-X5 has the highest thrust of systems actually available for purchase and has a mass acceptable for use with 2U or larger CubeSats. Overall, while the available electric, cold gas, and monopropellant systems may be suitable for basic station keeping or other small maneuvers, they cannot provide the changes in velocity, $\Delta V$, required for a timely increase of orbital altitude.

This gap in the industry has been recognized by others as well. Since the initiation of this thesis project, multiple aerospace entities have had progressed through development and testing, illustrating just how quickly the industry is advancing to meet demand. Several high $\Delta V$ solid and liquid systems have been tested with success, but obstacles remain to installing these systems in CubeSats. The Aerospace Corporation, for example, designed and tested two small solid rocket propulsion systems that can provide
up to 1400 m/s of $\Delta V$ with thrust vector control, but restartability was an unresolved issue.\textsuperscript{12} A liquid bi-propellant system from Hyperion Technologies has been experimentally shown to provide up to 231 m/s of $\Delta V$ with restart capability, but more testing is required to achieve space-flight readiness.\textsuperscript{13} Raytheon is developing an electrically-ignited solid propellant thruster system in a four-motor configuration similar to the consumable structure of this thesis, and the preliminary tests are promising.\textsuperscript{14}

Some hybrid rocket systems have been experimentally tested as well. Gilmour Space Technologies issued a press release in August 2017 revealing successful preliminary testing of a hybrid rocket CubeSat propulsion system that is estimated to provide over 4 km/s of $\Delta V$ when finished.\textsuperscript{15} Eilers of Utah State University previously developed an acrylonitrile butadiene styrene (ABS) and nitrous oxide (N2O) hybrid thruster sized for a 1U CubeSat that used a thrust-vectoring aerospike nozzle\textsuperscript{16}. Tests results showed the system could produce up to 166 s of specific impulse ($I_{sp}$). While a good proof of concept, several factors prevented the design from being pursued further. In particular, the small aerospike required a complicated assembly and the fuel grain needed to be effectively wrapped around the nozzle to fit, which resulted in poor performance. Additionally, the 1U design did not include the oxidizer or top-pressurant tanks and associated valves and plumbing, which would add at least 2U of additional space.

While these systems are still in the testing phase, many of the companies have the resources to continue work on the projects. It is expected that at least one flight-ready system will be available in the next few years. Furthermore, it is impossible to say that one propulsion system is preferable to another. With the great variability in satellite
missions, goals, and budgets, the more options available, the better.

2.4 Hybrid Rocket Principles

Hybrid rockets possess several qualities that make them a promising chemical propulsion alternative to available systems. In a hybrid rocket, a liquid oxidizer is combined with a solid fuel, and the components are separated until ignition is desired. Thrust is terminated by closing off the oxidizer source. Unlike solids, which are live as soon as the component have been mixed, hybrids are inert until the ignition event occurs while coupled with oxidizer flow. With regard to CubeSat rideshare safety, a hybrid propulsion system is an ideal chemical thruster option. Many hybrid fuels are also highly storable, which is necessary for long-duration satellite missions. Like a liquid system, hybrids can be throttled and restarted. The basic configuration of a hybrid rocket motor is given in Figure 4.

Many different oxidizer and fuel combinations have been investigated for hybrid rockets. Some of the more commonly used fuels are HTPB or paraffin based. Common oxidizers are nitrous oxide, hydrogen peroxide, and oxygen. Most hybrid rocket fuels and oxidizers are considered “green,” meaning that they present less human and environmental hazards. Multiple configurations have exhibited combustion efficiencies of up to 95% of theoretical values. Vacuum $I_{sp}$ for hybrids has been demonstrated as high...
as 380 s, though it is typically in the 230 – 280 s range. The efficiency of the motors is generally comparable to liquid and solid propulsion systems, but combustion instabilities and low regression rates exhibited by most designs have prevented hybrid rockets from becoming mainstream.

2.4.1 Ballistics Modelling

Modeling the internal ballistics of a hybrid rocket is based primarily on the regression rate of the fuel. Marxman and Gilbert developed a relation for regression rate based on oxidizer mass flux, \( G_{ox} \), and constants \( \{a, n\} \) determined empirically based on the given propellants.

\[
\dot{r} = a G_{ox}^n
\]  

(2.4.1)

An enhanced alternative formula that eliminates much of the empirical testing required to obtain the regression constants can be obtained. Applying an energy balance and including terms that account for “wall blowing” results in Equation 2.4.2.

\[
\dot{r} = \frac{0.635 \tau G_{ox}^n (\Delta h)}{n \rho_f P_r^2} \left( \frac{\Delta h}{h_v} \right)^{0.23} \left( \frac{\mu}{L} \right)^{1-n}
\]

(2.4.2)

Where coefficient scale factor, \( \tau \), and the exponent factor, \( n \), (which is different from the constant \( n \) in Equation 2.1) are equal to \( \{0.0592, 0.8\} \), respectively; \( P_r \) is the Prandtl number, \( \rho_f \) is the fuel density, \( \Delta h \) is the change in enthalpy between the combustion zone and the fuel surface, \( h_v \) is the fuel’s specific heat of vaporization, \( \mu \) is the dynamic viscosity of boundary layer products, and \( L \) is the fuel grain port length.
Hybrid rockets do not burn at the surface of the fuel; rather, as the fuel surface ablates from the heat of combustion, the flow pushes the combustion zone away from the wall. The blowing region insulates the fuel surface and decreases the regression rate. This process is illustrated in Figure 5. While the blowing negatively affects fuel regression rate, it also decreases the amount of heat transferred through the fuel grain to the motor case. This is a benefit in spacecraft and other applications.

![Figure 5. Illustration of "wall blowing" effects in hybrid rockets](image)

In equation 2.4.2, radiation and conduction within the fuel grain are considered negligible, which is typically an acceptable assumption. However, work completed by Merkley at Utah State University (USU) has suggested anomalous behavior over time in small scale ABS and gaseous oxygen (GOX) hybrid motors due to radiation heating effects\(^1\). According to Merkley, “the classical Marxman model is not necessarily incorrect, but incomplete for small-scale ABS/GOX propellant combinations”. The model incorporating radiation effects for small scale motors is presented in Equation 2.4.3.
\[
\dot{r} = \frac{0.635\kappa G_{\text{ox}}^n}{n\rho_f \rho_f^2} \left(\frac{\Delta T}{h_v}\right)^{0.23} \left(\frac{\mu}{L}\right)^{1-n} + \frac{\sigma (\epsilon T_0^4 - \alpha T_f^4)}{\rho_f h_v} \tag{2.4.3}
\]

Where \(\sigma\) is the Stefan-Boltzmann constant, \(\epsilon\) is the optical emissivity of the combustion flame, \(T_0\) is the temperature of the combustion flame, \(\alpha\) is the optical absorptivity of the fuel grain, and \(T_f\) is the surface temperature of the fuel grain.

2.5 USU ABS/GOX Hybrid Rockets

The Utah State University (USU) Propulsion Research Lab has developed and extensively tested several variations of a restartable hybrid propulsion system that uses an additively-manufactured ABS fuel grain along with GOX to create thrust. Both the ABS and GOX are safe to handle and present no vapor hazard. Compared to the ever-popular hydrazine, which requires a self-contained atmospheric protective ensemble (SCAPE) suit to handle, managing the hybrid propellants is significantly safer and more affordable. ABS can be stored nearly indefinitely with little to no deterioration, which is ideal for an unknown launch schedule as CubeSat missions might encounter. GOX must be pressurized, but the safety precautions are not as extensive as with most other oxidizers.

Through additive manufacturing, the ABS fuel grain can be printed into any shape with the ignition system and necessary internal cavities built in while maintaining structural integrity. The motors have primarily been printed at USU with fused deposition modeling (FDM) printers using standard density (0.975 g/cm\(^3\)) ABS stock material. Nearly any printer capable of printing ABS can print the motors, which makes manufacturing extremely accessible.

Testing on a scale of sizes has demonstrated that the ABS/GOX rockets can
provide as little as 5 N to greater than 800 N of thrust. Vacuum specific impulse has been demonstrated at over 280 s. Despite its high performance, the material properties of the ABS also provide for excellent thermal management as mentioned in the discussion on wall blowing. ABS melts before vaporizing, channeling the heat generated during combustion almost completely through the nozzle, insulating the unburned fuel, and preventing heat from radiating outwards to the case and beyond\textsuperscript{20}.

The most thoroughly tested motor, referred to as “MicroJoe,” has a diameter of 3.168 cm and length of 6.850 cm. Under ambient conditions with an optimized expansion ratio nozzle, MicroJoe achieves $I_{sp}$ values of 212 s and near 25 N of thrust. Tests completed on the same configuration but under vacuum conditions with a high expansion ratio nozzle at NASA Marshall Space Flight Center showed that the system delivers nearly 30 N of thrust with an $I_{sp}$ of at least 280 s.\textsuperscript{21,22} This performance is comparable to many liquid and solid propulsion systems. Its typical operating chamber pressure range is between 100 and 200 psia. Combustion efficiency was estimated at approximately 95\%, and the regression rate constants were estimated as \{a=6.75*1e-4 (m/s, kg/m2-s), n=0.22\}. A longer motor with otherwise the same configuration produced regression constants of \{a=3.5*1e-4 (m/s, kg/m2-s), n=0.22\}.

Advancing on the design of MicroJoe, Whitmore et. al. hypothesized that instead of using a COTS motor case, a custom nickel case could be additively applied directly onto the ABS fuel.\textsuperscript{23} In addition to possessing desirable burn properties, printed ABS is a very strong and resilient material. ABS has a high tensile strength with an elastic modulus that varies between 2.0 to 2.6 GPa.\textsuperscript{24} Because ABS can be printed in any form or shape, regions of high stress concentration can be printed with additional material
thickness to offset this load concentration. This option does not exist with conventionally manufactured cylindrical pressure vessels. Electroplating the ABS with nickel would further enhance the strength.

A series of prototype ABS fuel grains were designed to test the concept. The prototypes were 3D-printed in two parts; one side with built-in ports for the GOX feed, chamber pressure measurement, and ignition, and the other side with a cavity for an insert-able graphite nozzle and phenolic insulator. The test articles were assembled and glued together using ABS glue before being commercially electroplated with copper and nickel plating. Hydro-burst tests completed on the articles showed that the electroplated ABS design could withstand pressures as high as 2000 psia. Multiple hot-fire tests on one of the test articles were successfully performed with a maximum chamber pressure of approximately 85 psia, providing a mean thrust of over 20 N and over 200 s of $I_{sp}$ in ambient conditions.

A configuration similar to MicroJoe but using an oxidizer of oxygen-enriched air was flown onboard a sounding rocket. Two motors were mounted opposing one another, and four re-starts were achieved.25 This test confirmed that the ABS/GOX hybrid rocket technology is feasible for a space environment. Also tested during the flight was the contamination effects of the rocket plume on optical devices. An experimental setup comprised of photometers and LEDs was mounted near the exhaust of one of the motors. Hard vacuum telemetry data and data collected from ambient and soft vacuum tests on the ground revealed that contamination levels are essentially negligible.26

2.5.1 Arc-Ignition

Motor ignition relies on the patented non-pyrotechnic arc-ignition system
developed at Utah State University by Whitmore, et al.\textsuperscript{27,28} This ignition method was initially developed with 3D-printed ABS, and further study revealed that the method only works with 3D-printed material.\textsuperscript{30} Sample fuel grains were made from a range of extruded and 3D-printed materials, and of the materials tested, only 3D-printed ABS and a printable acrylic substitute from printing company Stratasys called VeroClear sparked sufficiently to initiate ignition.

The basic configuration requires two electrodes to be placed a distance apart and coincident with the top face of the fuel grain. Typically, insulated wires are routed from bullet connector electrodes through channels in the fuel grain to small gaps located on an impingement shelf created within the pre-combustion chamber. Small 3D-printed ABS inserts pin the wires in place within the channels. The wires terminate opposite each other on the shelf, flush with the combustion port surface and exposed to the interior of the combustion chamber. The type of wire leads used is variable; solid and stranded wire of different gauges produce similar sparks. Figure 6 shows a typical design that features a pre-combustion chamber with two impingement shelves intended to trap and mix the pyrolyzed fuel generated electrical spark with injected oxidizer.

![Figure 6. Typical igniter configuration showing channels (left) and complete assembly with bullet connector electrodes (right)](image)

When an electrostatic potential is placed across the electrodes, electricity flows through a pre-existing surface arc-track, resulting in pyrolysis and ignition as soon as
oxidizer flow is initiated. In a nominal ignition sequence, the spark is initiated for a set lead time, the GOX run valve is opened, and the spark is continued for a set lag time. Once combustion has been initiated, it is sustained after the spark is discontinues until the run valve is closed. Typically, the arc-track is pre-set by doping the surface with graphite powder. Once a surface arc-path has been set, graphite doping is no longer required.

The ignition system power processing unit (PPU) is based on the UltraVolt® D-series line of high-voltage power supplies (HVPS). The D-series UltraVolt HVPS units require a 15-volt DC input to provide a 7.5mA current-limited high voltage output up to 1000 V or 6 Watts total output. Higher wattage models have also been used. Depending on the impedance on the arc path between the ignitor electrodes, the dissipated voltage typically varies between 10 and 400 volts. The total energy of ignition is typically less than 3 Joules, and mere milliseconds are required to initiate the spark, providing nearly instantaneous restartability. The ignition system is reusable as long as there is continuous ABS material between the leads.

2.6 Consumable Structure Concept

The use of 3D printing to construct parts for use in a range of applications has become widespread. Considering the strength of the 3D-printed ABS fuel, it is therefore conceivable that the CubeSat frame structure itself could be constructed of the same material. This would allow for hybrid rocket fuel grains to be printed directly into a single structure made of four “thrust columns.” Plating the surfaces with nickel will further strengthen the structure and protect the columns from the space environment. Figure 7 illustrates the concept as a 2U CubeSat with a 1U payload.

The weight of space vehicles is one of the most important design constraints. For
small satellite programs requiring precision orbit station keeping and attitude control, a substantial portion of the payload mass fraction must be allocated to the propulsion system. The integrated fuel grains will provide structural support in addition to maneuvering capability, thereby reducing dry mass and launch costs. The mass margin created could then be allocated to the payload. Additionally, "all-additive" designs reduce component fabrication and procurement cycle time and significantly reduce overall system complexity. Nickel coated plastics have demonstrated significant structural load bearing capabilities and a high strength to weight ratio. These high strength and low weight properties can be utilized in rocket design to safely transport payloads while lowering inert mass, making spaceflight more economical.

At the end of the CubeSat mission, structural support is no longer required, so the fuel grains can be burned to their limit, consuming the structure for added thrust to place the CubeSat in a disposal orbit as the remaining propellant allows. The lower the final disposal altitude, the quicker the CubeSat will reenter the atmosphere. Using a propulsion system to deorbit a CubeSat provides an advantage over other existing space debris
remediation and mitigation methods in that the system can be used during the mission to provide required propulsion, and at the end of the CubeSat’s life, the same system can be used to deorbit the spacecraft as a form of active debris removal (ADR).
CHAPTER III

THRUSTER DESIGN

The ABS/GOX hybrid work completed at USU as described in the previous chapter was adapted and simplified to a consumable structure system. This novel concept has been developed for a propulsion module in a 2U CubeSat form factor to support at least a 1U volume payload for an expected maximum total satellite mass of 5 kg, which is the maximum reported 3U CubeSat mass.\(^3\) Based on these top-level specifications, the oxidizer tank, fuel grains, and all remaining components were optimally sized.

3.1 Fuel and Oxidizer

The design of the integrated ABS fuel grain structure was significantly driven by the available oxidizer tank. Few high pressure GOX tanks are available that fit within a 2U space while leaving room for four thrust columns. A lightweight composite tank was procured from HyPerComp Engineering Inc. that measures 4.45 cm in diameter, 17 cm in height, and weighs under 0.5 kg dry. The tank is designed to accommodate up to 4500 psig (31,000 kPa) operating pressure and has an internal volume of approximately 190 cm\(^2\). The optimal oxidizer to fuel ratio (\(O/F\)) for ABS/GOX combustion was determined using NASA’s Chemical Equilibrium with Applications (CEA) program and examining characteristic velocity, \(C^*\), which is a measure of combustion efficiency that is
independent of nozzle geometry. As shown in Figure 8, maximum $C^*$ is achieved when
the mixture ratio is approximately 1.5. While the actual $O/F$ shifts during motor
operation, the design was completed based on the optimal value. Based on a maximum

![Figure 8. $O/F$ versus $C^*$ at varying chamber pressures](image)

GOX mass of 0.077 kg, the total fuel required for complete consumption under optimal
operating conditions is 0.051 kg. Achieving 100% utilization of both oxidizer and fuel is,
however, improbable. Fuel utilization of 80% was assumed, and the total required mass
of fuel was calculated as 0.062 kg. Each fuel grain of the quad thrust column structure is
required to have a mass of at least 0.015 kg. However, a more conservative fill pressure
of 3,000 psi was assumed for the GOX tank, so even less fuel is required.

Using the required mass, the dimensions of the fuel grain were determined. The
standard density of 3D printed ABS is 0.975 g/cm$^3$, which was used to calculate the
require volume of ABS. Considering the available 2U space of 10 cm x 10 cm x 20 cm and allocating margin for a motor case in the radial direction and fittings and plumbing in the axial direction, the fuel grain diameter was set at 1.5 cm and the length at 10 cm.

3.2 Nozzles

Both low and high expansion ratio conical nozzles were designed for testing in ambient and vacuum conditions, respectively. A 45-degree conical convergent section was used for each. Initially, the throat diameter for both nozzles was 0.114” to produce a choking exit mass flow rate of 5 g/s, a mass flow rate typical of other small scale ABS/GOX motors. Ignition issues to be described in the results section prompted a re-design to a throat diameter of 0.0825” for a choking exit mass flow rate of just over 2 g/s. The expansion ratio for the ambient nozzle is approximately 1.9:1. The expansion ratio for the vacuum nozzle is approximately 16.33:1; this was determined by the widest exit diameter achievable within the existing geometry. The exact ratio is measured on the actual machined nozzle before testing. Multiple copies of the nozzles were made from graphite as needed.

Typically phenolic insulation is used around the graphite, since the nozzle experiences the highest temperatures in the motor. To reduce mass, an insulator was not used in this design. It was anticipated that at this small scale and low mass flow rates, foregoing the insulation would be acceptable.

3.3 Performance Modeling

Using the ballistics modeling previously described, a hybrid rocket performance model was developed to estimate the expected performance of the system. The model
simulates the four, integrated thrusters burning until the port diameter is equal to the outer diameter. CEA data collected on ABS/GOX is interpolated at each time step, and regression rate is integrated along with chamber pressure and the consumed masses of fuel and oxidizer. The most accurate predictions were obtained by using Equation 2.1 and the regression rate from the long MicroJoe.

3.4 Injector

Sizing the injector orifice diameter required the optimization of multiple parameters. The driving value was the oxidizer injector mass flow rate, which was desired to be a constant 2 g/s per motor for a total of 8 g/s supplied by the GOX tank for the four thrusters. This mass flow rate would provide approximately 9 s of GOX flow. Varying the injector pressure affects both the chamber pressure and mass flow rate, so the predicted chamber pressure was monitored to ensure it remained greater than approximately 100 psi. Injector orifice diameter, total initial mass flow rate, and injector pressure were iterated within the performance model until a 2 g/s choked oxidizer mass flow rate was obtained with a minimum chamber pressure greater than 100 psi. This occurred with an orifice diameter of 0.0292” and injection pressure of 265 psi. A 1/8” NPT brass plug was procured, and the orifice was machined through it.

3.5 Motor Case

While ABS has strong material properties, the additive manufacturing process makes the material porous, and a motor case is required to sustain the chamber pressures generated by the hybrid thruster. Traditional COTS motor cases are heavy, cylindrical, and only available in specific sizes. Two different nickel-plated base materials for a
custom motor case alternative were considered for use with the consumable structure: ABS and VeroClear.

Having the motors electroplated commercially introduces significant production time and financial strain. A new process for in-house electroplating was discovered that is cost-effective and can be completed in less than 48 hours. The process has been successfully applied to both ABS and VeroClear to date, and photos of the process as applied to ABS are given in Figure 9. The essential steps in the process are:

1. Thoroughly seal any fittings and joint on the specimen. Cover any portions that are not to be plated (tape is sufficient for this step).
2. Smooth the surface of the specimen. This step is most important with 3D-printed ABS to reduce porosity at the surface.
3. Coat every surface to be plated with MG Chemical’s SuperShield Nickel Conductive Paint.
4. Electroplate using copper plating solution purchased from Caswell Plating

Figure 9. Plating process specimen from left to right: Smoothed 3D-printed ABS; nickel painted; copper electroplated; nickel electroplated
5. Electroplate using nickel plating solution from Caswell

The process is ultimately still in development. Further work is underway to determine an effective method for controlling the thickness and evenness of the paint and plating. However, initial hydro-static burst tests of a nickel-plated ABS specimen yielded an estimated elastic modulus of 8.10 GPa and Poisson’s ratio of 0.40. This is significantly better than ABS alone and comparable to the results of hydro-static testing of identical commercially-plated specimens. A similar outcome is expected for nickel-plated VeroClear. Regardless of base material, final design would be plated with aluminum to conform to the CubeSat standard.

3.5.1 Comparison of Base Materials

The first motor case considered was 3D-printed ABS. This material is easy and affordable to obtain, and it is widely used in many types of FDM 3D printers. As described previously, printed ABS has an elastic modulus that varies between 2.0 to 2.6 GPa – that number increases to 8.10 when nickel plated. The finished material is relatively malleable, and most flaws can be corrected with acetone or ABS glue. Tests completed at USU on electroplated ABS motors proved that a nickel-plated ABS motor can withstand normal hot-fire operation. Those tests were on 3D-printed ABS motors with a traditional cylindrical shape, but a CubeSat frame must be shaped according to the standard. Since the ABS must be 3D printed, it will be porous throughout, and the external surfaces will form the shape of the pressure vessel. Angular corners are not ideal for holding high pressure.

The second motor case considered was VeroClear. Produced by Stratasys, VeroClear is designed to be a substitute for acrylic (Poly(methyl methacrylate), PMMA)
that can be additively printed. Its modulus of elasticity ranges from 2.0 to 3.0 GPa. It is printed using Stratasys’ PolyJet technology that works by adding layers of the liquid photopolymer and instantly curing them with UV light. Unlike ABS, the printed VeroClear is non-porous, and can hold pressure without the addition of electroplating. The finished material is, however, quite brittle and can be difficult to work with in post processing. If a part cracks or chips, it is nearly impossible to correct. Hot-fire tests have been conducted on VeroClear, and while it does burn, its performance is inferior to ABS. While the VeroClear could function as a case on its own, the addition of electroplating is expected to add strength and protect the plastic from damage.

3.5.2 Final Design

Ultimately, VeroClear was selected as the case material for the consumable CubeSat structure for its solidity. Electroplated VeroClear is superior to electroplated ABS as it allows for each of the four thrust columns to function as individual pressure vessels. Using ABS would require each thrust column to be printed separately, electroplated once, assembled, and electroplated together. A cylindrical core of ABS will still be used for the primary fuel, but as the ABS burns away, the motor case can also be consumed. This is ideal for deorbiting CubeSats when optimal performance is no longer vital and burning through the structure is acceptable.

The nozzle interface was designed to accommodate interchangeable high and low expansion ratio conical nozzles. The cap is printed with a port that is threaded for the injector insert and GOX feed fitting. Grooves at the injector end of the case line up with grooves in the fuel grain for the ignition leads. A port printed into the hypotenuse side allows a length of 1/16” tubing to be inserted to measure chamber pressure.
3.5.3 Structural Analysis and Hydrostatic Testing

The case must withstand the loads experienced during hot-fire operation. A thick-walled pressure vessel analysis was used assuming a cylindrical vessel with a wall thickness equal to the thinnest wall of the triangular vessel, equal to 0.1576”. The internal radius is 0.2931”. The ultimate tensile strength of VeroClear is reported by Stratasys as 7250-9450 psi, and the lowest value was used for this analysis. Axial, circumferential, and radial stresses were calculated as follows:

\[
\sigma_\theta = \frac{(p_i r_i^2 - p_o r_o^2)}{(r_o^2 - r_i^2)} \quad (3.5.3.1)
\]

\[
\sigma_c = \frac{[(p_i r_i^2 - p_o r_o^2) / (r_o^2 - r_i^2)] - [r_i^2 r_o^2 (p_o - p_i) / (r_o^2 - r_i^2)]]}{ \quad (3.5.3.2)}
\]

\[
\sigma_r = \frac{[(p_i r_i^2 - p_o r_o^2) / (r_o^2 - r_i^2)] + [r_i^2 r_o^2 (p_o - p_i) / (r_o^2 - r_i^2)]]}{ \quad (3.5.3.3)}
\]

where \(p_i\) is the internal pressure, \(p_o\) is the external/outer pressure, \(r_i\) is the internal radius, and \(r_o\) is the outer radius. For circumferential stress, the radius of interest is \(r = r_i\), where stress is at a maximum. For radial stress, the radius of interest is \(r = r_o\). During vacuum testing, pressure in the vacuum chamber will be nearly equal to zero. From tests completed on this test article in ambient conditions with varying injection pressures, the maximum expected internal pressure will be 250 psig. With these assumptions, the minimum factor of safety calculated is approximately 16.0.
To further confirm that the VeroClear case can withstand the pressures experienced during hot-fire testing, a hydro-static burst pressure test was executed on the test article. A pressure transducer was fitted to the pressure port, and a hand-operated hydraulic pump was affixed to the GOX feed fitting. The water pressure was increased until the case burst. A plot of the recorded pressure data is given in Figure 10.

![Figure 10. Burst test profile of VeroClear case specimen](image)

As shown in the figure, the initial jump to approximately 200 psi is the facility water pressure. The orange line illustrates the intended pressure application profile. Due to the nature of the hand operated pump, each spike in the raw data in blue is when pressure was applied, but the pressure tended to bleed back down. The test article burst soon after a pressure of approximately 1500 psi was applied. Limited visual confirmation of at least 870 psi was made on an analog pressure gauge. The maximum pressure recorded was 1652 psi. Comparing this to the expected maximum internal pressure of 250 psi gives an optimistic factor of safety of 6.6.

Other expected loads on the case would be experienced during launch and ejection. CubeSats are installed in a device called a Poly Picosatellite Orbital Deployer
(P-POD) for launch. The device can accommodate three 1U CubSats or one 3U. The satellites are ejected with a spring mechanism that interfaces with one of the 10 cm² sides, sliding along rails until free of the device. Further detail is provided in the CubeSat standard in Appendix A. Launch loads are transferred to the CubeSat through the P-Pod at the rails and spring interface. If multiple CubeSats are installed, additional loading is transferred from the surrounding satellites. Actual launch loads vary by launch vehicle and can be difficult to characterize. Finite element analysis (FEA) programs can be used to estimate the response to static, sinusoidal vibration, and shock loads generated by the vehicle. Studies by Herrera-Arroyave et. al. and Fagerudd provide details on how such a simulation is completed. However, any calculations would be speculative until the material properties of nickel plating on VeroClear have been determined.

3.6 Igniter

While the basic technology for the ignitor had been developed previously at USU, the preferred implementation using bullet connectors was designed for use in larger, cylindrical motors with reusable motor caps. The compact triangular geometry of the consumable structure fuel grains required a modified design. Simple wires feed into grooves that pass through the fuel grain and motor case and up through ports in the motor.

![Figure 11. CAD model (left) and 3D printed ABS test article (right) of the Ignitor](image-url)
cap where they are connected to the HVPS. The lead wires are held in place in the fuel grain with ABS glue and small 3D-printed ABS inserts. Little room was available to place the lead ends directly across from one another; instead, they enter the pre-combustion chamber at an angle and are flush with the chamber wall and shelf. The shortest gap distance between the leads is approximately 0.33”, but the spark tends to take a slightly different path along the fuel in each different fuel grain. Occasionally, the spark was observed to travel along the long arc around the shelf. The depth of the pre-combustion chamber is 0.5”. The ignitor section is shown in Figure 11.

3.7 Final Design Summary

Simplicity was paramount in the design of each of the preceding components. The system will use 3D-printed or COTS parts wherever possible and assembled in as few parts as possible to reduce weight and complexity. Figure 12 and Figure 13 show the final design of the consumable structure and a single thrust column respectively. Table 3 lists the specifications for the individual thrust column. Approximated specifications of the

Figure 12. CAD model of consumable structure concept
combined consumable structure will be discussed in the systems analysis presented in Chapter 7.

The fuel grains were printed at USU with a Fortus 250MC FDM printer using black ABS stock material. The VeroClear cases and caps were printed at USU with an Objet 260 Connex3 PolyJet printer. All of the additively manufactured components will serve as fuel for the thrusters. Each thrust column test article was assembled as shown in Figure 14. High temperature, room-temperature-vulcanizing (RTV) silicone was used to seal the nozzle to the fuel grain and the cap to the case. A cyanoacrylate adhesive (super glue) was used around the edges of the cap and the fittings to further secure the assembly.
A nozzle retainer plate was used on early tests to prevent the nozzle from ejecting if it melted through the VeroClear case. This was a potential concern since no insulator was used, but no deformation was observed after testing, so the retainer was discontinued.

All the thrusters tested in ambient and vacuum conditions were identical with the exception of the nozzle. Slight variations may also have been created during assembly like the amount of sealant used and the depth of the fitting threads.
CHAPTER IV

TESTING

Static hot-fire testing was completed in both ambient conditions and within a vacuum chamber. Mass flow rates in both conditions were measured using Venturi meters with identical geometries; the inlet port diameter was 0.281 in, and the throat diameter was 0.125 in. As many parameters as possible were unchanged between testing conditions; variations are outlined in Table 4 and in the following sections.

Table 4. Description of test articles

<table>
<thead>
<tr>
<th>Name</th>
<th>Test Number</th>
<th>Test Duration (s)</th>
<th>GOX Feed Pressure (psia)</th>
<th>Fuel Port Diameter (in)</th>
<th>Pressure Port Configuration</th>
<th>Testing Condition</th>
</tr>
</thead>
<tbody>
<tr>
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<td>1</td>
<td>1</td>
<td>278</td>
<td>0.197</td>
<td>Flat</td>
<td>Ambient</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td></td>
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<td>4</td>
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<tr>
<td>B</td>
<td>1</td>
<td>2</td>
<td>463</td>
<td>0.118</td>
<td>Flat</td>
<td>Ambient</td>
</tr>
<tr>
<td></td>
<td>2</td>
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<td>3</td>
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<td>267</td>
</tr>
<tr>
<td>D</td>
<td>1</td>
<td>1</td>
<td>398</td>
<td>0.586</td>
<td>Raised</td>
<td>Ambient</td>
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<td>3</td>
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<td>1</td>
</tr>
<tr>
<td>E</td>
<td>1</td>
<td>0.5</td>
<td>247</td>
<td>0.586</td>
<td>Raised</td>
<td>Ambient</td>
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<td>2</td>
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<td>0.5</td>
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<td>3</td>
<td></td>
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<td></td>
<td>1</td>
</tr>
<tr>
<td>F</td>
<td>1</td>
<td>0.5</td>
<td>408</td>
<td>0.586</td>
<td>Raised</td>
<td>Ambient</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>1</td>
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<tr>
<td></td>
<td>3</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>0.5</td>
</tr>
</tbody>
</table>
4.1 Ambient Conditions: USU

Initial tests were completed at the USU Propulsion Lab facilities in ambient conditions. The average ambient pressure in Logan, Utah, at an elevation of approximately 4600 ft.) altitude, is approximately 12.4 psia. A graphite nozzle was optimally designed for the altitude with a forty-five-degree conical convergent section, four-degree conical divergent section, and expansion ratio of approximately 1.91.

4.1.1 Experimental Setup and Procedures

Cold-flow and hot-fire data were gathered from the test stand pictured in Figure 15 that includes a refillable GOX tank, GOX injection feed pressure regulator, Venturi flow meter, GOX solenoid run valve, pressure transducers (PT), and load cell connected to a stand supported by flexures. The stand interfaces with a data acquisition instrumentation deck that collects data from the PTs and load cell using a National Instruments LabVIEW Virtual Instrument (VI) to calculate mass flow rate, chamber
pressure, and thrust. High voltage lead wires extend from the HVPS to the thrust column arc-ignition electrodes. An UltraVolt 1AA24-P30, which requires a 24V input, was used to supply up to 1000V (30W) for ignition.

The variable parameter for each test is GOX injection feed pressure, which dictates the chamber pressure and thrust output. Tests were completed at a range of injection pressures to verify the minimum pressure for combustion; this is of great importance onboard a CubeSat where the GOX supply is limited. After regulating the injection pressure, test duration is set on the control VI. Individual test durations ranged from 0.5 to 2 seconds. Altogether, six test articles were hot-fired in ambient conditions for a total of nineteen tests.

![Figure 15. Article “A” installed on test stand at USU](image)

4.2 Vacuum Conditions: MSFC

Vacuum condition tests were completed at NASA Marshall Space Flight Center (MSFC) in Huntsville, Alabama. A graphite nozzle was optimally designed for full vacuum with a forty-five-degree conical convergent section, fourteen-degree conical divergent section, and expansion ratio of approximately 16.33.

4.2.1 Experimental Setup and Procedures

The test stand used for the vacuum tests at MSFC is similar to that used for
ambient testing at USU. Inside the altitude chamber is the thrust stand with flexures, load cell, and tubing for the GOX feed line and chamber pressure measurement. The high voltage leads and load cell wiring are fed through a hermetic seal in the chamber. GOX is supplied by an external tank regulated to the desired feed pressure; this regulator exhibits a significant amount of creep, so it is difficult to maintain a steady feed pressure. The chamber also features a high-pressure gaseous nitrogen purge fed by the facility gas supply to extinguish uncontrolled burning in an emergency. The set-up including altitude chamber, GOX supply, and gas flow control panel is shown in Figure 16.

The vacuum pump is located in a separate room and connects to the chamber through a series of tubing that interfaces with a port located just behind the plume. The amount of vacuum pulled on the altitude chamber can be moderated with a hand-operated valve just outside the altitude chamber, but the vacuum pump capability is limited. Because the vacuum chamber internal volume is only approximately 4.5 ft³, the pressure output from the plume during testing may be greater than the pump can quickly remove, causing difficulties in maintaining the exact desired altitude chamber pressure. Another concern due to the altitude chamber size is thermal management. Very little heat is transferred to the thruster case during operation, but the plume impinges directly on the
vacuum chamber door. In an effort to divert that heat, a hydro heat exchanger is installed on the internal side of the chamber door, and cool water is constantly cycled during hot-fire testing.

A power- and instrumentation-patch panel connects the instrumentation in the test cell to the data acquisition instrumentation deck in the control room. The vacuum condition test setup is considerably more complicated than for ambient tests, so a VI that interfaces with the NASA gas flow system controls the run sequence, and a separate VI that interfaces with the USU-built instrumentation deck logs data. A live-feed camera is installed at one of the two viewing ports in the chamber; the video is also logged for post-test review. A battery-powered back-up camera of lower fidelity is installed in the second viewing port. Plumbing and instrumentation diagrams along with wiring diagrams of the system are available in Appendix B.
CHAPTER V
RESULTS AND DISCUSSION

The raw data collected from the test instrumentation setup was processed using a combination of MATLAB and LabView programs. A method developed to further filter and reconstruct the chamber pressure curve is discussed. Additional performance parameters are calculated from the collected data. The results from both ambient and vacuum testing are presented.

5.1 Reconstruction of Attenuated Chamber Pressure Signal

Sensing the chamber pressure on the consumable structure thrust column prototype presented several significant challenges. First, the extreme temperatures and significant strains in the printed motor case prevented direct in-situ mounting of the chamber pressure sensor. A less complex installation solution was to 3D print a very small pressure port at the mid-length of the thrust column and transmit pressure from the port to a pressure transducer using a significant length of pneumatic transmission tubing. This installation allows the transducer to be mounted in a controlled environment, allows for high frequency measurement, and virtually eliminates any sensing errors due to motor-case strain.

The mid-length location of the chamber pressure port raises new challenges, however. The close-coupled installation results in potential structural weakening of chamber walls, overheating of the pressure sensing element, and resonance within the measurement port. Unfortunately, within the pneumatic installation acoustical distortion due to friction, acoustical resonance and latency, and wave reflections will compromise the fidelity of the sensed pressure measurement and induce considerable measurement
latency. Soot generated during combustion is easily introduced to the tubing, making the port susceptible to plugging, and further decreasing the response fidelity.

For this study, an adaptation of Weiner filtering\(^\text{35}\) as developed by Whitmore et al (2010) was used to reconstruct the attenuated pressure signal\(^\text{36}\). The method, based on optimal deconvolution theory, amplifies attenuated pressure signals while rejecting additive noise. The method has been previously validated using laboratory-derived data and then applied to reconstructing transient chamber pressure for a small-scale solid rocket motor. A second order model of the sensor response dynamics seeds the deconvolution method. The parameters of this model can be calculated either analytically using the known sensor geometry, as modeled by Whitmore (2009), or calculated empirically using one of several curve fitting methods\(^\text{37}\). The second order model allows the discrete spectrum of a simple transfer function to be calculated and then inverted to reconstruct the high-fidelity pressure signal. Noise amplification problems are overcome using methods of optimal filtering theory\(^\text{38}\). When properly tuned, the derived method amplifies the attenuated pressure signals while rejecting additive noise. The chamber pressure curves for both the raw data and deconvolved data are shown in sub-figure “e” of each set of time histories.

5.2 Calculated Parameters

Chamber pressure is used to check thrust measurements from the load cell by using the general thrust equation with a correction for the conical nozzle losses:

\[
F = \lambda \dot{m}_T V_e + A_{ce}(P_e - P_\infty)
\]  
\[
\lambda = \frac{1}{2} (1 + \cos \theta)
\]
Where $F$ is thrust, $\dot{m}_{T,e}$ is the total exit mass flow rate, $V_e$ is the nozzle exit velocity, $A_e$ is the nozzle exit area, $\Theta$ is the nozzle expansion angle, and $P_\infty$ is the ambient pressure. $P_e$ is the nozzle exit pressure calculated from the measured chamber pressure using isentropic flow equations with the mixture stagnation temperature and ratio of specific heats calculated using CEA. Both the thrust directly measured with the load cell and the thrust calculated from chamber pressure were used to calculate specific impulse and total accumulated impulse. Oxidizer mass flow rate is calculated from the absolute pressure measurement at the Venturi inlet and differential pressure measurement between the inlet and throat. Total mass flow rate is calculated from the choking mass flow equation at the nozzle throat, and fuel mass flow rate is calculated as the difference of total and oxidizer mass flow rate. Multiplied by the data collection time step, the mass flow rates yield total consumed mass. The mixture O/F is calculated by dividing the oxidizer by the fuel mass flow rate. Total accumulated ignition energy is the power output from the HVPS integrated over the duration of the ignition sequence, which typically ranges from 0.5 to 2 seconds.

5.3 Ambient Test Results

Prototype test articles “A” and “B” were primarily used to obtain the optimized thrust column geometry. Both test articles had an original VeroClear case design with a chamber pressure port equal to the outer diameter of the pressure tubing. This allowed the VeroClear case to burn out around the tubing quickly, and the port was reduced with a raised extension added at the full diameter to extend burn lifetime.

While article “D” was successfully ignited, the chamber pressure port burned out quickly into the first test. Article “E” failed at the GOX fitting threads during its last test,
which is believed to be due to loose threads from a manufacture and assembly mistake. The chamber pressure port was observed to be plugged during the first test on “E,” and was unable to be un-plugged, so no chamber pressure data was collected. Article “F” was not tested to failure. Article “G” failed at the chamber pressure port near the end of the last test. Figure 17 shows an example hot-fire test of article “G”.

Data time histories for two successive tests on article “G”, which are

![Figure 17. Hot-fire test of article "G"](image)

![Figure 18. Representative time histories for test 2 on article “G”](image)
representative of the nominal tests completed on articles “E” through “G”, are given in Figure 18 and Figure 19. In total, three tests were completely nominal. The average GOX injection pressure for these tests was 416.3 psi. The mean parameters are presented in Table 5 with a Student-T 95% confidence interval.

Table 5. Summary of ambient test data with 95% confidence interval

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Mean</th>
<th>+/-</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (N)</td>
<td>10.94</td>
<td>2.35</td>
</tr>
<tr>
<td>$I_{sp}$ (s)</td>
<td>161.67</td>
<td>36.71</td>
</tr>
<tr>
<td>Total Impulse (N-s)</td>
<td>8.74</td>
<td>1.92</td>
</tr>
</tbody>
</table>

A nozzle throat of 0.114 inches was used with the developmental test articles, and the nozzle had to be deliberately plugged with a piece of tape in order to boost the transient chamber pressure high enough (approximately 28 psi) to initiate combustion at the 265 psi design injection pressure. An injection pressure of over 400 psi was required.
for ignition without a plug. The initial nozzle throat diameter used for the remaining test articles was 0.096 inches, which initially eliminated the need for a nozzle plug. However, nozzle throat and exit diameters were measured after each test, and some erosion at the throat was noted. Even relatively minimum erosion on the order of 0.001 inches was sufficient to require an increased injection pressure for continued successful combustion without a nozzle plug. Using a metallic nozzle material rather than graphite would alleviate this issue and increase design fidelity between thrusters.

While the ABS fuel core was expected to burn in the bowed shape typical of hybrids, as shown in Figure 21, the asymmetry was accentuated around the chamber pressure port. As material around the port support is consumed, the port tends to open enough that the tubing is ejected from the thruster. Where the ABS fuel was completely consumed in patches, as seen in Figure 20, all test articles clearly showed that the

![Figure 21. Bowed fuel grain in "F"](image)

![Figure 20. Photo of partially consumed VeroClear case on "B"](image)
VeroClear case material begins to burn without failure of the case until the pressure line is ejected. In this manner, it is demonstrated that the structure is successfully used as fuel.

5.4 Vacuum Test Results

Tests were attempted at a range of injection pressures, but although the transient chamber pressure was well above the minimum 28 psi, and all systems appeared nominal, ignition was not achieved during the initial test campaign. Troubleshooting and a second testing campaign revealed that the ignition system experiences anomalies during the ignition sequence when conducted in full vacuum, as detailed in a later section. To characterize the minimum ambient pressure allowable for ignition without corona discharge, testing was completed under soft vacuum of varying ambient pressures from 6.16 to 3.15 psia as shown in Table 6.

Table 6. Ambient pressure of soft vacuum hot-fire tests

<table>
<thead>
<tr>
<th>Test Article/Test #</th>
<th>Ambient Pressure (psia)</th>
</tr>
</thead>
<tbody>
<tr>
<td>“H”/1</td>
<td>5.3</td>
</tr>
<tr>
<td>“J”/1</td>
<td>6.16</td>
</tr>
<tr>
<td>“J”/2</td>
<td>4.53</td>
</tr>
<tr>
<td>“J”/3</td>
<td>3.26</td>
</tr>
<tr>
<td>“J”/4</td>
<td>3.15</td>
</tr>
<tr>
<td>“K”/1</td>
<td>3.20</td>
</tr>
</tbody>
</table>
Unfortunately, “H” failed at the seal between cap and case almost immediately upon ignition, so accurate data was not recorded. The failure is attributed to the extended sparking on the test article while attempting to characterize the cause of the ignition anomalies, which likely burned out large portions of the ignition section. Figure 22 shows “J”; the sparks are from the plume impinging on the door and being captured by the vacuum orifice. Article “K” also failed immediately during its first hot-fire test; it appeared to have failed at the seal between cap and case. Data time histories for two

*Figure 22. Hot fire test of “J” under soft vacuum*

*Figure 23. Representative time histories for test 1 on article “J”*
successive tests on article “J”, which are representative of all the nominal tests completed in vacuum conditions are given in Figure 23 and Figure 24. Overlaid chamber pressure data from all three nominal tests completed on “J” is shown in Figure 25. In total, three
tests were completely nominal. The mean parameters are presented in Table 7 with a Student-T 95% confidence interval.

Table 7. Summary of vacuum test data with 95% confidence interval

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Mean</th>
<th>+/-</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (N)</td>
<td>9.50</td>
<td>0.98</td>
</tr>
<tr>
<td>$I_{sp}$ (s)</td>
<td>167.05</td>
<td>13.90</td>
</tr>
<tr>
<td>Total Impulse (N-s)</td>
<td>9.54</td>
<td>0.77</td>
</tr>
</tbody>
</table>

The average GOX injection pressure for these tests was 293 psi. Note that this is approximately 123 psi less than the ambient test results, so the vacuum data results are not directly comparable. Increasing injection pressure—and therefore chamber pressure—increases thrust, specific impulse, and total impulse. Considering that the vacuum specific and total impulses are higher than the ambient values even at a lower injection pressure indicates that the performance is greatly improved under vacuum conditions as expected.

5.5 Arc-Ignition Anomalies in Vacuum

While previous ABS/GOX thrusters have been tested under nearly identical conditions, the consumable structure thrust column exhibited anomalous behavior during vacuum testing. By happenstance, blue colored LED lights were on hand during construction of the vacuum chamber thrust stand and were used to illuminate the sealed chamber. Blue glowing characteristic of corona activity, shown in Figure 26.
was observed during initial tests, but because the chamber lighting was blue, the additional glow was attributed to the ignition spark reflecting through the clear plastic case. During ignition, a high voltage signal is sent through the positive electrode in the thrust column ignition section. Under normal circumstances, the high voltage spark arcs along the ABS shelf to the negative electrode, pyrolizing ABS fuel in the process. No ignition issues were observed during ambient testing or during routine spark testing under vacuum. The same HVPS model is used for all tests, and the output voltage and current curves are typically very repeatable, especially for tests in succession. Interestingly, the apparent corona only occurs when the GOX run valve is opened under full vacuum. In fact, it appears to correlate directly with the advent of GOX flow. Plots of the HVPS output voltage and current from a nominal spark-only test and a failed vacuum hot-fire test exhibiting corona discharge conducted in succession under identical full vacuum conditions are presented in Figure 28. Both curves represent an ignition sequence with a
1 second lead and lag time. The “nominal” curve shows that with no extraneous influences, the output voltage and current required to sustain a spark along the fuel grain surface is essentially constant. As is evident from the “corona” curve, however, as soon as the GOX run valve is opened, the voltage required to sustain the arc increases until the path opens in corona. The output current curves show that the current tails off quickly once corona discharge has begun.

In comparison, Figure 29 shows plots of the HVPS output voltage and current from a nominal ambient test and a successful vacuum test under partial (ambient pressure of 4.5 psi) vacuum conditions. The “ambient” and “vacuum” curves are aligned by the instant that the GOX run valve was opened for each test—the ignition sequence for the “ambient” test used a spark lead time longer than that use for the “vacuum” test by 0.5 seconds.

![Figure 29. HVPS output for ignition in ambient and partial vacuum conditions](image)

Multiple hypotheses are being considered to explain the cause of the ignition anomalies. It is speculated that the impingement of pressurized GOX on the electrodes induces a change in localized pressure, changing the physical conditions necessary for a nominal spark across the fuel. One hypothesis is that the arc-ignition process follows a Paschen’s law like curve. According to Paschen’s law, the breakdown voltage required
for the spark to arc through the surrounding gas rather than across the surface of the ABS fuel decreases with ambient pressure to a point, and then begins to increase. A Paschen’s law curve for air is given in Figure 30; the actual gas surrounding the electrodes during operation in vacuum is more complex than air, but the shape of the curve is universal. When the spark travels through the gas or across another surface, no ABS fuel is pyrolized. Without fuel present when the GOX valve opens, ignition cannot occur. For this thrust column design, it appears that reducing the ambient pressure to full vacuum may place the breakdown voltage to the left of the critical minimum point on Paschen’s curve, where breakdown voltage is higher. At this point, as illustrated in the “nominal” curve of Figure 28, sparking resulting in successful pyrolysis occurs. When the GOX valve is opened in the same full vacuum environment, however, the localized pressure appears to be increased to the point that the ambient pressure falls at a point on Paschen’s curve where the breakdown voltage is lower; the spark arcs through the gas, corona appears to be visible, and pyrolysis—and ignition—is not achieved. Increasing the ambient pressure slightly to partial vacuum seems to move the breakdown voltage far enough to the right of the Paschen’s curve minimum that the spark again arcs across the fuel surface, resulting in successful ignition. Interestingly, the peaks in output voltage

![Paschen's law curve for air](image-url)
after the GOX valve is opened as shown in successful tests in Figure 29 suggests that localized plasma creation may still be occurring. If this is the case, it does not appear to be enough to impact ignition. It may be that the spark arcs through the GOX momentarily but returns to the ABS surface. Under full vacuum, the conditions appear to be such that the spark path is irreparably displaced.

In an attempt to mitigate the anomalies and achieve ignition in full vacuum assuming corona discharge hypothesis was considered, and efforts were made to reduce the potential for corona. The fixed viewing ports of the vacuum chamber prevent a clear view of the corona, so determining the correct solution is challenging. The entire test article was covered with aluminum foil, as shown in Figure 33, in an attempt to shield the VeroClear case and further round out edges. Testing was attempted, but corona discharge was still observed, and ignition was not achieved at full vacuum.

Another hypothesis is that in full vacuum, the pyrolized fuel particles are essentially “blown” away when the GOX valve is opened, extinguishing the spark and creating an open circuit. As Figure 28 shows, when the valve is opened, voltage begins to increase and current begins to decrease. This indicates that the resistance of the spark path is increasing. Normally, the spark must travel across the continuous fuel surface.
Small, cyclical peaks in the voltage likely signify fuel breaking down, the spark path momentarily being displaced, and the spark path being regained along other solid particles. Particles being transported away along the path disrupts that continuous path, forcing the HVPS to supply greater voltage to maintain the path. In full vacuum, it appears that the particles are blown away quicker than the spark can find a new path. The resistance of the discontinuous path increases until the HVPS cannot supply any current, and the circuit opens. In successful ambient and partial vacuum tests, as shown in Figure 29, the voltage increases when the valve opens, indicating that some of the path may have been carried away, but the voltage returns to a nominal level when a path along the fuel surface is regained. The magnitude of the problem appears to be a function of ambient pressure, perhaps because under higher vacuum pressure, the particles are carried away both by vacuum and the impinging GOX.

![Figure 32. Mean ignition voltage versus ambient pressure](image-url)
Other correlations were examined by plotting mean ignition voltage and ignition energy against ambient pressure for all tests, successful and not. Figure 31 shows mean voltage for each test. There appears to be a mild correlation between ambient pressure and mean voltage. As ambient pressure decreases, the observed spark voltage generally increases, which supports the previous hypotheses explaining the cause of the ignition anomalies in full vacuum. There is even less of an apparent trend in the total ignition energy, as shown in Figure 32. While the total energy was higher for many of the failed tests in full vacuum compared to successful tests in partial vacuum and ambient pressure, there were several exceptions in all test conditions. Data was also compared between test articles and test order on each article, but no correlations were found. The only firm conclusion is that this hybrid motor design exhibits ignition failure in full vacuum conditions.
5.6 Discussion

Overall, three factors contributed to the performance of the thrust column. One, the length of the fuel grain, dictated by the available space in the CubeSat form factor, exposes a greater fuel area to the combustion zone than is optimal for the design oxidizer mass flow rate. The resulting higher fuel flux causes fuel-rich operation, which reduces combustion efficiency. Two, the nozzle geometries used for both ambient and vacuum testing were not ideal. Three, variation due to imperfect manufacture and assembly led to several testing failures, reducing the demonstrable burn time, total impulse, and data sample size.

The mean vacuum $I_{sp}$ of 167.05 is reasonable for this hybrid rocket design. In comparison, hydrazine has an ideal $I_{sp}$ of approximately of 199 s. While the thrust column’s specific impulse is less, the benefits of the “green” ABS/GOX propellants and simple design justify the lesser performance. Additionally, from the data and test videos, the nozzle used for vacuum testing appears to be under-expanded even in the partial vacuum. The expansion ratio of 16.33 was the maximum value achievable in the design geometry, so increasing the ratio for true vacuum operation would require extensive redesign. An optimal nozzle would, however, boost performance.

Further interpretation of the observed ignition anomalies in full vacuum conditions needs to be considered. Covering the test column in aluminum foil may have reduced any potential corona effects, but it did not completely solve the problem. Electroplating the entire thrust column and eventually the full consumable structure with nickel is expected to further prevent any potential corona during vacuum ignition. If anodizing is unsuccessful, further design considerations on electrode gap length, thrust
column exterior shape, and injector impingement mass flow rates and angle may be necessary.

Repeatability between individual tests was better than expected. Issues in full vacuum aside, the igniter design for the thrust columns sparked nearly every time. Tests completed on “E”, “F”, and “G” were executed in succession with no intervention with the igniter; the articles were tested in four, five, and three pulses respectively. This level of reliability is essential for the consumable structure where intervention on orbit is impossible. Yet some variability in manufacturing and assembly needs to be resolved. While every effort was made to ensure fidelity between the two test setups, inevitably there were some differences in equipment. The vacuum results were remarkably similar between tests and test articles, even with “H” and “K” that experienced failures midway through testing. The ambient data was noisy and varied somewhat even between successive tests. A different lab technician printed each batch of the additively manufactured components, and slight variations were noticeable. Further human errors led to some variability in assembly.

The amount of burn lifetime available for each test article needs to be improved. The maximum burn time achieved in testing was approximately 5 s. After this time, however, there was clearly residual ABS fuel and VeroClear. Relocating the chamber pressure port to the GOX inlet end of the thrust column would reduce the chances for a chamber pressure burn through, increase available burn time for testing, and allow for better statistical results.
CHAPTER VII
SYSTEMS ANALYSIS OF CONSUMABLE STRUCTURE

The thrust column hot-fire test results can be extrapolated to the overall performance of the quad column consumable structure. Expected overall performance will be discussed along with other necessary requirements for the propulsive structure’s in situ operation. Hot-fire and structural testing of the complete structure will be required to fully characterize the system’s performance.

6.1 Mass Estimates

Calculating the achievable delta-V is dependent on an estimate of mass. Based on the above results, the masses of the thrust columns are fixed, and the mass of the GOX tank is known. The remaining dry mass components were estimated for the consumable structure. Assuming a blow-down feed system, an orifice will be used in lieu of a regulator. The run valve and orifice masses were estimated based on available miniature solenoid valves. Stainless steel tubing and required fittings along with additional joining structural elements were mapped in SolidWorks as shown in Figure 12. Note that a flight-ready structure will be electroplated. Mass estimates are given in Table 8.

<table>
<thead>
<tr>
<th>Component</th>
<th>Mass (g)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust Columns (x4)</td>
<td>537</td>
</tr>
<tr>
<td>GOX (3000 psi)</td>
<td>54</td>
</tr>
<tr>
<td>GOX Tank (dry)</td>
<td>472</td>
</tr>
<tr>
<td>Run Valve</td>
<td>45</td>
</tr>
<tr>
<td>Plumbing and Fittings</td>
<td>64</td>
</tr>
<tr>
<td>Additional Structure</td>
<td>125</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>1,297</strong></td>
</tr>
</tbody>
</table>

At under 1.5 kg, the propulsion unit wet mass is well under the allotted 2U maximum mass of 2.66 kg. This estimate does not include power, command, or control
elements, which are described in a later section. The limiting factor in the propulsion system design is available volume, and it does occupy most of the 2U space. Assuming a 3U CubeSat, at least 2.5 kg are available for payload, power, and computing—3U masses as high as 5 kg have been approved through the waiver process. Some components could be placed within the propulsion 2U.

6.2 Expected Overall Performance

Based on the reported vacuum test results, it is expected that on average, a quad thrust column consumable structure could supply at minimum 4 s of burn time for each thrust column. Over a 4 s burn time with all four thrust columns in operation, the system can supply 38 N of average thrust and 152 N-s of total impulse or greater. Since none of the test articles were tested until the ABS fuel was completely exhausted, let alone the VeroClear case, it is anticipated that the true values are much higher. The thrust columns do not have to be operated in synchrony, so individual columns could be burned as desired to control the thrust. Additionally, the system can be operated as a cold gas system if lower thrust levels are desired.

The mean ignition energy for an individual pulse in vacuum was 8.9 J. Ignition energy in ambient conditions was observed to be as low as less than 1 J. The increased ignition energy under vacuum conditions may be an artifact of the ignition anomalies. While successful operation with only a VeroClear case was confirmed, the addition of electroplating would further strengthen the structure, protect the thrust columns from the space environment, and potentially mitigate ignition anomalies in full vacuum.

In the presented results, each thruster had access to an essentially unlimited supply of GOX for the reported tests. Future tests will need to be conducted using the
flight-weight GOX tank with a blow-down feed system to characterize the effects on performance as injection pressure drops. From the results above, it is expected that a blow-down system would present challenges with the current nozzle configuration, as the throat would likely begin to erode at higher operating pressures, rendering it ineffective once the pressure began to fall as the tank emptied. Once the nozzle erosion has been addressed, the performance from a blow-down system would be desirable for the type of maneuvers expected. Higher feed pressures early in the lifetime correlate to higher thrust and specific impulse, yielding greater $\Delta V$ for controlled maneuvers. Once the satellite has been pointed, the remaining burn lifetime could be used effectively for deorbit.

The amount of $\Delta V$ that the propulsion system will be able to provide is dependent on the useable ABS and GOX. With 100% ABS fuel utilization, for a 5 kg, 3U CubeSat, the system can provide at least 36.99 m/s of $\Delta V$ as calculated from Equation 6.2.1. For a 4 kg CubeSat, the $\Delta V$ increases to 46.37 m/s. If a larger GOX tank could be procured and thrust column specific impulse increased, the available $\Delta V$ would be much higher.

$$\Delta V = I_{sp} g_0 \ln \left( \frac{m_{wet}}{m_{dry}} \right)$$  \hspace{1cm} (6.2.1)

6.3 Potential Orbital Maneuvers

With the available $\Delta V$, the satellite would be able to complete a range of orbital maneuvers including orbit raising, phasing, or rendezvous. The shortest hot-fire test of the thrust columns was 0.5 s. While shorter pulses are possible, precise attitude control would be better achieved using reaction wheels. While not as efficient as other available systems, the thrust that the consumable structure can provide is orders of magnitude greater. With a high-thrust system, maneuvers can be completed quicker.
6.3.1 Power, Command, and Control Requirements

The limited size and budget of CubeSats prohibits excessive redundancy of components including those for command, control, and power necessary to execute maneuvers. It is assumed that a power supply will be required for the payload and/or communications. A computer is necessary as well. Ideally, both power supply and computer would be used for powering, commanding, and controlling the propulsion system in addition to supporting the payload.

The power required to operate the consumable structure thrust columns is limited to the PS necessary for ignition and a solenoid run valve. Both the HVPS and run valve have been operated with as little as a 15 V supply. Alternative HVPS and solenoid valves may need to be investigated that operate with the same supply as the payload. The routines required to command operation of this system are not particularly complex and can be controlled from the same computer as the payload operation. The nozzles used for the hot-fire testing presented in this thesis were not equipped for any amount of thrust vectoring. Controlling even the amount of thrust provided by one column is challenging, and throttling isn’t an option in a system of this size.

Regarding the three inhibits to ignition required by the CubeSat standard, none of the thrust columns can be ignited until at minimum the command is given to ignite, high voltage is supplied to the igniter leads, and the GOX flow is initiated. During static ground tests, the GOX bottle must be opened, pressure regulated, HVPS powered, spark command given, and GOX run valve open command given for ignition to occur.

6.4 Lifetime Decay Comparison

While it is desired that the system be able to support a range of in-space
maneuvers, the ability to deorbit the 3U CubeSat from LEO is a good benchmark. Deorbiting involves moving the CubeSat to an orbital altitude at which atmospheric drag will quickly cause the satellite to lose altitude and disintegrate upon atmospheric reentry in order to avoid potential collisions with other spacecraft or debris. A 5 kg, 3U CubeSat orbiting above approximately 500 km altitude is at risk of not meeting the international 25-year lifetime limitation guideline without any active deorbiting. Historically, most CubeSats have been placed in orbit at that altitude or lower. With trust in the satellites functionality and launch opportunities increasing, however, their altitudes are getting higher—even outside of earth orbit. The higher the orbital altitude, the longer it takes the orbit to decay.

Various models for predicting deorbit time are available, so a comparison was performed between several models in order to select one that provides the best compromise between computation time and predicted accuracy. Ultimately, an analytical lifetime estimate as presented in Equations 7.2.1 was selected.

$$\tau = \frac{e_t^2}{2B} \left( 1 - \frac{11}{6} e_t + \frac{29}{16} e_t^2 + \frac{7 H}{8 a_t} \right)$$

$$B = n \beta I_1 e_t a_t \exp \left( -e_t \left[ 1 + \frac{a_t}{H} \right] \right)$$ (7.2.1)

Where $e_0$ and $a_0$ are the initial orbital eccentricity and semi-major axis respectively, $n$ is the mean motion, and $I_1$ is the Bessel function of the first kind and order. The average cross-sectional area of the CubeSat was estimated to be 350 cm$^2$ and used to calculate $\beta$, the ballistic coefficient. Atmospheric density, $\rho$, and scale height, $H$, were obtained from the 1976 Standard Atmosphere data.

Moving the CubeSat to a circular disposal orbit can be achieved with a Hohmann
transfer, which requires two, impulsive thruster burns. The amount of $\Delta V$ required to complete such a maneuver from a range of initial orbit altitudes to a range of disposal orbit altitudes was calculated by summing $\Delta V_1$ and $\Delta V_2$ from Equations 7.2.2.

$$
\Delta V_1 = \sqrt{\frac{2 \mu}{r_i}} - \mu/a_H - \sqrt{\mu/r_i}
$$

$$
\Delta V_2 = \sqrt{\mu/r_f} - \sqrt{\frac{2 \mu}{r_f}} - \mu/a_H
$$

(7.2.2)

Figure 33 shows the $\Delta V$ required to move a CubeSat from its initial orbit to a disposal orbit against time to completed deorbit. 50 m/s would deorbit a CubeSat at a 700 km altitude that would currently exceed the lifetime guideline. The consumable structure available $\Delta V$ of over 46 m/s will be able to deorbit from a slightly lower altitude. Any lesser amount of available $\Delta V$ would also still be useful for deorbiting in less time than is possible from atmospheric drag alone.

![Graph](image)

*Figure 34. Required $\Delta V$ for Desired Time to Disposal for a 5 kg CubeSat*

6.5 Space Environment Considerations

Aside from atmospheric drag, other aspects of the space environment must be considered. Periods of high sun activity can cause quicker orbital decay than that predicted above. Temperatures experience in space vary widely in range. Experimental
tests have been done on frozen motors. The motors ignite despite the cold fuel, although not always on first try.

Printed ABS possesses unique self-cooling properties that are beneficial for thermal management. When ABS pyrolyzes, an amorphous fluid-like layer is formed before vaporizing, and this layer allows for significant film cooling. Because of this self-cooling property and the low thermal conductivity of ABS, virtually all the heat of combustion is imparted to the exhaust plume, and the fuel grain exterior does not heat up during the rocket burn. Not only does this property improve combustion efficiency, it also keeps the external surface of the fuel material cool; this is important for a satellite where many heat sensitive components are packed tightly together, and thermal management is complicated. Solar panels could be safely installed on the exterior of the propulsion unit.

Exhaust plume contamination of the payload sensors is another concern in a vacuum. As mentioned previously, the plume contamination has been studied by Brewer. Some of the results were inconclusive, but the contamination appeared to be somewhat increased in space. The effects on the payload or solar panels would be dependent on the use of the consumable structure. It is unclear whether build-up would occur quicker with frequent short pulses or infrequent long duration pulses. Occasionally fragments of the rocket components are also expelled from the nozzle, particularly pieces of a decomposing injector or eroding nozzle. While unlikely, such fragments do have the potential to damage the satellite.

Perhaps the greatest concern illuminated by this project is the prospect of repeated unsuccessful ignition in full vacuum. It is suggested that a study be completed on the
effects of igniter geometry including but not limited to electrode gap distance, shelf depth and width, injector diameter and impingement angle, GOX injection pressure, and ambient pressure. Until the anomalies are better characterized, the operation of the ignition system cannot be guaranteed.
CHAPTER VIII
CONCLUSION

Effective propulsion system options for CubeSats are essential. The number of small satellite missions launched each year is increasing, but the primary allure of CubeSats—affordability—has consequences. Limited rideshare opportunities mean CubeSats may end up in non-optimal orbits where their missions may not be completed, and they risk becoming dangerous space debris. In order to provide maneuvering options for CubeSats including attitude control, station keeping, orbit raising, and deorbiting, an ABS/GOX hybrid propulsion system has been designed that will be partially contained in the spacecraft structure. The safe, reliable, restartable propulsion system is desirable for small spacecraft that must participate in rideshares to reach orbit. The proposed partially-consumable structure will increase the safety and efficiency of small spacecraft propulsion compared to currently available hydrazine systems. This is significant for small spacecraft endeavors because rideshares are more likely to accept spacecraft that contain non-hazardous propulsive fuels that will not accidentally ignite.

As a drop-in propulsion unit, this system has the potential to provide great benefit to the CubeSat community. The partially-consumable ABS and VeroClear structure can replace a traditional CubeSat frame, thereby decreasing overall mass and launch costs. The entire safe, restartable, non-hazardous system is contained in the 2U volume, which can be attached to a 1U payload. The 3D-printed VeroClear structure of the 2U section will be comprised of four hybrid rocket thrust columns with 3D printed ABS fuel cores. Additively manufacturing the thrust columns allows the spacecraft structure to be printed out of fuel, thereby reducing structural dry mass. Nickel plating the exterior increases the strength and survivability.
The system will provide approximately 37 m/s of ΔV or greater. This is sufficient impulsive propulsion potential to allow CubeSats to quickly alter their positions in space. Multiple hot-fire tests of thrust columns in ambient and partial vacuum conditions have been achieved, and the preliminary results show consistency. The demonstrated vacuum $I_{sp}$ is over 167 s, and the mean thrust for the structure is 38 N. The overall mass of the system is less than 1.5 kg. While simplifying the hybrid rocket concept has many benefits, it introduced some variation in the performance of individual thrust columns. Further work is suggested before the structure will be ready for flight.

As more CubeSats are launched, the hazard to other spacecraft from space debris will become critical. A hybrid rocket propulsion system that has high efficiency while remaining non-hazardous to ride shares will allow CubeSats to perform more complex missions. The proposed hybrid rocket propulsion system will help ensure that CubeSat missions continue having the opportunity to launch for many years more.
REFERENCES


APPENDICES
APPENDIX A

CUBESAT STANDARD

Begins on next page.
CubeSat Design Specification

(CDS)
REV 13
### CHANGE HISTORY LOG

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<td>Riki Munakata</td>
<td>Format, Design specification and Mk.III P-POD compatibility update.</td>
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<td>Added applicable documents section. Removed restrictions on propulsion, added guidance for propulsion systems and hazardous materials. Added magnetic field restrictions and suggestions. Cleared Section 3.2. Added custom spring plunger specs and recommendation. Extended restrictions on inhibits. Added links to outside resources. Cleared Section 4.</td>
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<td>Total Mass Loss</td>
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Applicable Documents

The following documents form a part of this document to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this document, the contents of this document shall take precedence.

LSP Program Level P-POD and CubeSat Requirements Document (LSP-REQ-317.01)

General Environmental Verification Standard for GSFC Flight Programs and Projects (GSFC-STD-7000)

Military Standard Test Requirements for Launch, Upper-stage, and Space Vehicles (MIL-STD-1540)


Metallic Material Properties (MIL-HDBK-5)

Standard Materials and Processes Requirements for Spacecraft (NASA-STD-6016)

NASA Procedural Requirements for Limiting Orbital Debris (NPR 8715.6)
Introduction

1.1 Overview
Started in 1999, the CubeSat Project began as a collaborative effort between Prof. Jordi Puig-Suari at California Polytechnic State University (Cal Poly), San Luis Obispo, and Prof. Bob Twiggs at Stanford University’s Space Systems Development Laboratory (SSDL). The purpose of the project is to provide a standard for design of picosatellites to reduce cost and development time, increase accessibility to space, and sustain frequent launches. Presently, the CubeSat Project is an international collaboration of over 100 universities, high schools, and private firms developing picosatellites containing scientific, private, and government payloads. A CubeSat is a 10 cm cube with a mass of up to 1.33 kg. Developers benefit from the sharing of information within the community. If you are planning to start a CubeSat project, please contact Cal Poly. Visit the CubeSat website at http://cubesat.org for more information.

Figure 1: Six CubeSats and their deployment systems.

1.2 Purpose
The primary mission of the CubeSat Program is to provide access to space for small payloads. The primary responsibility of Cal Poly, as the developer of the Poly Picosatellite Orbital Deployer (P-POD), is to ensure the safety of the CubeSat and protect the Launch vehicle (LV), primary payload, and other CubeSats. CubeSat developers should play an active role in ensuring the safety and success of CubeSat missions by implementing good engineering practice, testing, and verification of their systems. Failures of CubeSats, the P-POD, or interface hardware can damage the LV or a primary payload and put the entire CubeSat Program in jeopardy. As part of the CubeSat Community, all participants have an obligation to ensure safe operation of their systems and to meet the design and minimum testing requirements outlined in this document. Requirements in this document may be superseded by launch provider requirements.

1.3 Waiver Process
Developers will fill out a "Deviation Waiver Approval Request (DAR)" (see appendix A) if their CubeSat is in violation of any requirements in sections 2 or 3. The waiver process is intended to be quick and easy. The intent is to help facilitate communication and explicit documentation
between CubeSat developers, P-POD integrators, range safety personnel, and launch vehicle providers. This will help to better identify and address any issues that may arise prior to integration and launch. The DAR can be found at http://www.cubesat.org/ and waiver requests should be sent to standards@cubesat.org.

Upon completion of the DAR, the P-POD Integrator will review the request, resolve any questions, and determine if there are any additional tests, analyses or costs to support the waiver. If so, the Developer, with inputs from the P-POD Integrator, will write a test plan and perform the tests before the waiver is conditionally accepted by the P-POD Integrator. Waivers can only be conditionally accepted by the P-POD Integrator until a launch has been identified for the CubeSat. Once a launch has been identified, the waiver becomes mission specific and passes to the launch vehicle Mission Manager for review. The launch vehicle Mission Manager has the final say on acceptance of the waiver, and the Mission Manager may require more corrections and/or testing to be performed before approving the waiver. Developers should realize that each waiver submitted reduces the chances of finding a suitable launch opportunity.

Figure 2: CubeSat Standard Deviation Waiver Process Flow Diagram
2. Poly Picosatellite Orbital Deployer

2.1 Interface
The Poly Picosatellite Orbital Deployer (P-POD) is Cal Poly's standardized CubeSat deployment system. It is capable of carrying three standard CubeSats and serves as the interface between the CubeSats and L.V. The P-POD is a rectangular box with a door and a spring mechanism. Once the release mechanism of the P-POD is actuated by a deployment signal sent from the L.V., a set of torsion springs at the door hinge force the door open and the CubeSats are deployed by the main spring gliding on its rails and the P-POD's rails (P-POD rails are shown in Figure 3b). The P-POD is made up of anodized aluminum. CubeSats slide along a series of rails during ejection into orbit. CubeSats will be compatible with the P-POD to ensure safety and success of the mission by meeting the requirements outlined in this document. The P-POD is backward compatible, and any CubeSat developed within the design specification of CDS rev. 9 and later will not have compatibility issues. Developers are encouraged to design to the most current CDS to take full advantage of the P-POD features.

![Figure 3a and 3b: Poly Picosatellite Orbital Deployer (P-POD) and cross section](image)

3. CubeSat Specification

3.1 General Requirements
3.1.1 CubeSats which incorporate any deviation from the CDS will submit a DAR and adhere to the waiver process (see Section 1.3 and Appendix A).
3.1.2 All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created.
3.1.3 No pyrotechnics shall be permitted.
3.1.4 Any propulsion systems shall be designed, integrated, and tested in accordance with AFSPCMAN 91-710 Volume 3.
3.1.5 Propulsion systems shall have at least 3 inhibits to activation.
3.1.6 Total stored chemical energy will not exceed 100 Watt-Hours.
3.1.6.1 Note: Higher capacities may be permitted, but could potentially limit launch opportunities.

3.1.7 CubeSat hazardous materials shall conform to AFSPCMAN 91-710, Volume 3.

3.1.8 CubeSat materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft during integration, testing, and launch. A list of NASA approved low out-gassing materials can be found at: http://outgassing.nasa.gov

3.1.8.1 CubeSats materials shall have a Total Mass Loss (TML) ≤ 1.0 %

3.1.8.2 CubeSat materials shall have a Collected Volatile Condensable Material (CVCM) ≤ 0.1 %

3.1.9 The latest revision of the CubeSat Design Specification will be the official version which all CubeSat developers will adhere to. The latest revision is available at http://www.cubesat.org.

3.1.9.1 Cal Poly will send updates to the CubeSat mailing list upon any changes to the specification. You can sign-up for the CubeSat mailing list here: www.cubesat.org/index.php/about-us/how-to-join

3.1.10 Note: Some launch vehicles hold requirements on magnetic field strength. Additionally, strong magnets can interfere with the separation between CubeSat spacecraft in the same P-POD. As a general guideline, it is advised to limit magnetic field outside the CubeSat static envelope to 0.5 Gauss above Earth’s magnetic field.

3.1.11 The CubeSat shall be designed to accommodate ascent venting per ventable volume/area < 2000 inches.

3.2 CubeSat Mechanical Requirements

CubeSats are cube shaped picosatellites with dimensions and features outlined in the CubeSat Specification Drawing (Appendix B). The P-POD coordinate system is shown below in Figure 4 for reference. General features of all CubeSats include:

Figure 4: P-POD Coordinate System
3.2.1 The CubeSat shall use the coordinate system as defined in Appendix B for the appropriate size. The CubeSat coordinate system will match the P-POD coordinate system while integrated into the P-POD. The origin of the CubeSat coordinate system is located at the geometric center of the CubeSat.

3.2.1.1 The CubeSat configuration and physical dimensions shall be per the appropriate section of Appendix B.

3.2.1.2 The extra volume available for 3U+ CubeSats is shown in Figure 6.

3.2.2 The -Z face of the CubeSat will be inserted first into the P-POD.

3.2.3 No components on the green and yellow shaded sides shall exceed 6.5 mm normal to the surface.

3.2.3.1 When completing a CubeSat Acceptance Checklist (CAC), protrusions will be measured from the plane of the rails.

3.2.4 Deployables shall be constrained by the CubeSat, not the P-POD.

3.2.5 Rails shall have a minimum width of 8.5mm.

3.2.6 Rails will have a surface roughness less than 1.6 μm.

3.2.7 The edges of the rails will be rounded to a radius of at least 1 mm.

3.2.8 The ends of the rails on the +/- Z face shall have a minimum surface area of 6.5 mm x 6.5 mm contact area for neighboring CubeSat rails (as per Figure 6).

3.2.9 At least 75% of the rail will be in contact with the P-POD rails. 25% of the rails may be recessed and no part of the rails will exceed the specification.

3.2.10 The maximum mass of a 1U CubeSat shall be 1.33 kg.

3.2.10.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.11 The maximum mass of a 1.5U CubeSat shall be 2.00 kg.

3.2.11.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.12 The maximum mass of a 2U CubeSat shall be 2.66 kg.

3.2.12.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.13 The maximum mass of a 3U CubeSat shall be 4.00 kg.

3.2.13.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.14 The CubeSat center of gravity shall be located within 2 cm from its geometric center in the X and Y direction.

3.2.14.1 The 1U CubeSat center of gravity shall be located within 2 cm from its geometric center in the Z direction.

3.2.14.2 The 1.5U CubeSat center of gravity shall be located within 3 cm from its geometric center in the Z direction.

3.2.14.3 The 2U CubeSat center of gravity shall be located within 4.5 cm from its geometric center in the Z direction.

3.2.14.4 3U and 3U+ CubeSats’ center of gravity shall be located within 7 cm from its geometric center in the Z direction.

3.2.15 Aluminum 7075, 6061, 5005, and/or 5052 will be used for both the main CubeSat structure and the rails.

3.2.15.1 If other materials are used the developer will submit a DAR and adhere to the waiver process.

3.2.16 The CubeSat rails and standoff, which contact the P-POD rails and adjacent CubeSat standoffs, shall be hard anodized aluminum to prevent any cold welding within the P-POD.
3.2.17 The 1U, 1.5U, and 2U CubeSats shall use separation springs to ensure adequate separation.

3.2.17.1 Note: Recommended separation spring specifications are shown below in Table 1. These are a custom part available through Cal Poly. Contact cubesat@calpoly.edu in order to obtain these separation springs.

3.2.17.2 The compressed separation springs shall be at or below the level of the standoff.

3.2.17.3 The 1U, 1.5U, and 2U CubeSat separation spring will be centered on the end of the standoff on the CubeSat's -Z face as per Figure 7.

3.2.17.4 Separation springs are not required for 3U CubeSats.

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<td>End Force Initial/Final</td>
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<td>Thread Pitch</td>
<td>8-36 UNP-2B</td>
</tr>
</tbody>
</table>

Figure 5: Custom Spec Spring Plunger (Separation Spring)
Figure 6: 3U+ Extra Volume ("Tuna Can")

Figure 7: Deployment Switches and Separation Spring Locations
3.3 **Electrical Requirements**

Electronic systems will be designed with the following safety features.

3.3.1 The CubeSat power system shall be at a power off state to prevent CubeSat from activating any powered functions while integrated in the P-POD from the time of delivery to the LV through on-orbit deployment. CubeSat powered function include the variety of subsystems such as Command and Data Handling (C&DH), RF Communication, Attitude Determine and Control (ADC), deployable mechanism actuation. CubeSat power systems include all battery assemblies, solar cells, and coin cell batteries.

3.3.2 The CubeSat shall have, at a minimum, one deployment switch on a rail standoff, per Figure 7.

3.3.3 In the actuated state, the CubeSat deployment switch shall electrically disconnect the power system from the powered functions; this includes real time clocks (RTC).

3.3.4 The deployment switch shall be in the actuated state at all times while integrated in the P-POD.

3.3.4.1 In the actuated state, the CubeSat deployment switch will be at or below the level of the standoff.

3.3.5 If the CubeSat deployment switch toggles from the actuated state and back, the transmission and deployable timers shall reset to F-0.

3.3.6 The RBF pin and all CubeSat umbilical connectors shall be within the designated Access Port locations, green shaded areas shown in Appendix B.

3.3.6.1 Note: All diagnostics and battery charging within the P-POD will be done while the deployment switch is depressed.

3.3.7 The CubeSat shall include an RBF pin.

3.3.7.1 The RBF pin shall cut all power to the satellite once it is inserted into the satellite.

3.3.7.2 The RBF pin shall be removed from the CubeSat after integration into the P-POD.

3.3.7.3 The RBF pin shall protrude no more than 6.5 mm from the rails when it is fully inserted into the satellite.

3.3.8 CubeSats shall incorporate battery circuit protection for charging/discharging to avoid unbalanced cell conditions.

3.3.9 The CubeSat shall be designed to meet at least one of the following requirements to prohibit inadvertent radio frequency (RF) transmission. The use of three independent inhibits is highly recommended and can reduce required documentation and analysis.

An inhibit is a physical device between a power source and a hazard. A timer is not considered an independent inhibit.

3.3.9.1 The CubeSat will have one RF inhibit and RF power output of no greater than 1.5W at the transmitting antenna’s RF input.

3.3.9.2 The CubeSat will have two independent RF inhibits
3.4 Operational Requirements

CubeSats will meet certain requirements pertaining to integration and operation to meet legal obligations and ensure safety of other CubeSats.

3.4.1 Operators will obtain and provide documentation of proper licenses for use of radio frequencies.

3.4.1.1 For amateur frequency use, this requires proof of frequency coordination by the International Amateur Radio Union (IARU). Applications can be found at www.iaru.org.

3.4.2 CubeSats will comply with their country’s radio license agreements and restrictions.

3.4.3 CubeSats mission design and hardware shall be in accordance with NPR 8715.6 to limit orbital debris.

3.4.3.1 Any CubeSat component shall re-enter with energy less than 15 Joules.

3.4.3.2 Developers will obtain and provide documentation of approval of an orbital debris mitigation plan from the FCC (or NOAA if imager is present).

3.4.3.2.1 Note: To view FCC amateur radio regulations, go to http://www.arrl.org/part-97-amateur-radio

3.4.3.3 Note: Analysis can be conducted to satisfy the above with NASA DAS, available at http://orbitaldebris.jsc.nasa.gov/mitigate/das.html

3.4.4 All deployables such as booms, antennas, and solar panels shall wait to deploy a minimum of 30 minutes after the CubeSat's deployment switch(es) are activated from P-POD ejection.

3.4.5 No CubeSats shall generate or transmit any signal from the time of integration into the P-POD through 45 minutes after on-orbit deployment from the P-POD. However, the CubeSat can be powered on following deployment from the P-POD.

3.4.6 Private entities (non-U.S. Government) under the jurisdiction or control of the United States who propose to operate a remote sensing space system (satellite) may need to have a license as required by U.S. law. For more information visit http://www.nesdis.noaa.gov/CRSRA/licenseHome.html. Click on the Application Process link under the Applying for a License drop down section to begin the process.

3.4.7 Cal Poly will conduct a minimum of one fit check in which developer hardware will be inspected and integrated into the P-POD or TestPOD. A final fit check will be conducted prior to launch. The CubeSat Acceptance Checklist (CAC) will be used to verify compliance of the specification (Found in the appendix of this document or online at http://cubesat.org/index.php/documents/developers).
4. Testing Requirements

Testing will be performed to meet all launch provider requirements as well as any additional testing requirements deemed necessary to ensure the safety of the CubeSats, P-POD, and the primary mission. If the launch vehicle environment is unknown, The General Environmental Verification Standard (GEVS, GSFC-STD-7000) and MIL-STD-1540 can be used to derive testing requirements. GSFC-STD-7000 and MIL-STD-1540 are useful references when defining testing environments and requirements, however the test levels defined in GSFC-STD-7000 and MIL-STD-1540 are not guaranteed to encompass or satisfy all LV testing environments. Test requirements and levels that are not generated by the launch provider or P-POD Integrator are considered to be unofficial. The launch provider testing requirements will supersede testing environments from any other source. The P-POD will be tested in a similar fashion to ensure the safety and workmanship before integration with the CubeSats. At the very minimum, all CubeSats will undergo the following tests.

4.1 Random Vibration

Random vibration testing shall be performed as defined by the launch provider.

4.2 Thermal Vacuum Bakeout

Thermal vacuum bakeout shall be performed to ensure proper outgassing of components. The test specification will be outlined by the launch provider.

4.3 Shock Testing

Shock testing shall be performed as defined by the launch provider.

4.4 Visual Inspection

Visual inspection of the CubeSat and measurement of critical areas will be performed per the appropriate CAC (Appendix C).

4.5 CubeSat Testing Philosophy

The CubeSat shall be subjected to either a qualification or protoflight testing as defined in the CubeSat Testing Flow Diagram, shown in Figure 88. The test levels and durations will be supplied by the launch provider or P-POD integrator.

4.5.1 Qualification

Qualification testing is performed on an engineering unit hardware that is identical to the flight model CubeSat. Qualification levels will be determined by the launch vehicle provider or P-POD integrator. Both MIL-STD-1540 and LSP-REQ-317.01 are used as guides in determining testing levels. The flight model will then be tested to Acceptance levels in a TestPOD then integrated into the flight P-POD for a final acceptance/workmanship random vibration test. Additional testing may be required if modifications or changes are made to the CubeSats after qualification testing.
4.5.2 Protoflight
Protoflight testing is performed on the flight model CubeSat. Protoflight levels will be determined by the launch vehicle provider or P-POD integrator. Both MIL-STD-1540 and LSP-REQ-317.01 are used as guides in determining testing levels. The flight model will be tested to Protoflight levels in a TestPOD then integrated into the flight P-POD for a final acceptance/workmanship random vibration test. The flight CubeSat SHALL NOT be disassembled or modified after protoflight testing. Disassembly of hardware after protoflight testing will require the developer to submit a DAR and adhere to the waiver process prior to disassembly. Additional testing will be required if modifications or changes are made to the CubeSats after protoflight testing.

4.5.3 Acceptance
After delivery and integration of the CubeSat into the P-POD, additional testing will be performed with the integrated system. This test ensures proper integration of the CubeSat into the P-POD. Additionally, any unknown, harmful interactions between CubeSats may be discovered during acceptance testing. The P-POD Integrator will coordinate and perform acceptance testing. Acceptance levels will be determined by the launch vehicle provider or P-POD integrator. Both MIL-STD-1540 and LSP-REQ-317.01 are used as guides in determining testing levels. The P-POD SHALL NOT be deintegrated at this point. If a CubeSat failure is discovered, a decision to deintegrate the P-POD will be made by the developers, in that P-POD, and the P-POD Integrator based on safety concerns. The developer is responsible for any additional testing required due to corrective modifications to deintegrated P-PODs and CubeSats.

### CubeSat Qualification / Acceptance Test Flow

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<thead>
<tr>
<th>Hardware</th>
<th>Qualification</th>
<th>ProtoFlight</th>
<th>Acceptance</th>
<th>Flight</th>
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<td>Vibration Testing</td>
<td>Shock Testing</td>
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<td>Information</td>
<td>Cubesat Flight Unit</td>
<td>Vibration Testing</td>
<td>Shock Testing</td>
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<td>P-POD Flight System</td>
<td>Vibration</td>
<td>Flight</td>
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</tbody>
</table>

Figure 8: CubeSat General Testing Flow Diagram
5. Contacts

**Cal Poly - San Luis Obispo**
Prof. Jordi Puig-Suari
Aerospace Engineering Dept.
(805) 756-5087
(805) 756-2376 fax
jpuigsua@calpoly.edu

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**Cal Poly Program Manager**
Roland Coelho
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(805) 756-5165 fax
rcelho@calpoly.edu

**Cal Poly Student Contacts**
(805) 756-5087
(805) 756-5165 fax
cubesat@gmail.com
Appendix A:
Waiver Form
<table>
<thead>
<tr>
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<th>2. DAR NUMBER:</th>
<th>3. DATE:</th>
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<td>Testing</td>
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<th>2. DAR NO.</th>
<th>3. DATE:</th>
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</table>

16. CONTINUATION (indicate item or block number):
Appendix B:
1U, 1.5U, 2U, 3U, and 3U+
CubeSat Specification Drawing
Section 1
1U CubeSat Design Specification Drawing
Section 2
1.5U CubeSat Design Specification Drawing
Section 3
2U CubeSat Design Specification Drawing
Section 4
3U CubeSat Design Specification Drawing
Section 5
3U+ CubeSat Design Specification Drawing
Appendix C:
1U, 1.5U, 2U, 3U, and 3U+
CubeSat Acceptance Checklist
Section 1
1U CubeSat Acceptance Checklist
# 1U CubeSat Acceptance Checklist

<table>
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<tr>
<th>Mass (≤ 1.33 kg)</th>
<th>RBF Pin (≤6.5mm)</th>
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<tbody>
<tr>
<td>Spring Plungers (Depressed)</td>
<td>Functional Y / N</td>
</tr>
<tr>
<td>Deployment Switches (Depressed)</td>
<td>Functional Y / N</td>
</tr>
</tbody>
</table>

Mark the diagram showing the locations of the RBF pin, connectors, deployables, and any envelope violations.

**List Item** | **As Measured** | **Required**
---|---|---
**Width [x-y]** | Side 1 (-Y) | Side 2 (-X) | Side 3 (+Y) | Side 4 (+X) | 100.0 ± 0.1mm
- +Z | | | | | 100.0 ± 0.1mm
- Middle | | | | | 100.0 ± 0.1mm
- -Z | | | | | 100.0 ± 0.1mm
**Height [x-y]** | Rail 1 (+X, +Y) | Rail 2 (-X, -Y) | Rail 3 (-X, +Y) | Rail 4 (+X, +Y) | 113.5 ± 0.1mm
- +Z Standoffs | | | | | ≥ 6.5mm
- -Z Standoffs | | | | | ≥ 6.5mm
**Protrusions** | Side 1 (-Y) | Side 2 (-X) | Side 3 (+Y) | Side 4 (+X) | Side 5 (-Z) | Side 6 (+Z) | ≤ 6.5mm

Authorized By:
IT #1: 
IT #2: 
Passed: Y / N
Section 2
1.5U CubeSat Acceptance Checklist
1.5U CubeSat Acceptance Checklist

Project: Date/Time: Engineers:
Organization: Location:
Satellite Name: Satellite S/N: Revision Date: 02/20/2014

<table>
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<th>Mass (&lt; 2.00 kg)</th>
<th>RBF Pin (≤6.5mm)</th>
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<tbody>
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<td>Spring Plungers</td>
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<td>Flush with Standoff Y / N</td>
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<tr>
<td>Deployment Switches</td>
<td>Functional Y / N</td>
</tr>
<tr>
<td>(Depressed)</td>
<td>Flush with Standoff Y / N</td>
</tr>
<tr>
<td>Rails Anodized</td>
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</tr>
<tr>
<td>Deployables Constrained</td>
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</table>

Mark on the diagram the locations of the RBF pin, connectors, deployables, and any envelope violations.

Authorized By:
IT #1: ________
IT #2: ________
Passed: Y / N

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<th>List Item</th>
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<td></td>
</tr>
<tr>
<td>+Z</td>
<td></td>
<td>100.0 ± 0.1mm</td>
</tr>
<tr>
<td>Middle</td>
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<td>100.0 ± 0.1mm</td>
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<tr>
<td>Height [x-y]</td>
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<tr>
<td>+Z Standoffs</td>
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<td>≥ 6.5mm</td>
</tr>
<tr>
<td>-Z Standoffs</td>
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<td>≥ 6.5mm</td>
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<tr>
<td>Protrusions</td>
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<td>100.0 ± 0.1mm</td>
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<tr>
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<td>Height [x-y]</td>
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<td>-Z Standoffs</td>
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Section 3
2U CubeSat Acceptance Checklist
# 2U CubeSat Acceptance Checklist

**Project:**

**Date/Time:**

**Organization:**

**Location:**

**Satellite Name:**

**Satellite S/N:**

**Revision Date:** 02/20/2014

## Mass (< 2.66 kg)

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<th>Spring Plungers (Depressed)</th>
<th>RBF Pin (≤ 6.5 mm)</th>
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<tr>
<td>Flush with Standoff Y / N</td>
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## Deployment Switches (Depressed)

<table>
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<th>Deployables Constrained</th>
</tr>
</thead>
<tbody>
<tr>
<td>Y / N</td>
<td>Y / N</td>
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Mark on the diagram the locations of the RBF pin, connectors, deployables, and any envelope violations.

---

**List Item** | **As Measured** | **Required** |
---|---|---|
**Width [x-y]** | | |
- **Side 1 (-Y)** | | |
  - **+Z** | | 100.0 ± 0.1 mm |
  - **Middle** | | |
  - **-Z** | | |
- **Side 2 (-X)** | | |
- **Side 3 (+Y)** | | |
- **Side 4 (+X)** | | |
**Height [x-y]** | | |
- **Rail 1 (+X, -Y)** | | 227.0 ± 0.2 mm |
- **Rail 2 (-X, -Y)** | | |
- **Rail 3 (-X, +Y)** | | |
- **Rail 4 (+X, +Y)** | | |
**+Z Standoffs** | | ≥ 6.5 mm |
- **length x width** | | |
**-Z Standoffs** | | ≥ 6.5 mm |
- **length x width** | | |
**Protrusions** | | ≤ 6.5 mm |
- **Side 1 (-Y)** | | |
- **Side 2 (-X)** | | |
- **Side 3 (+Y)** | | |
- **Side 4 (+X)** | | |
- **Side 5 (-Z)** | | |
- **Side 6 (+Z)** | | |

**Authorized By:**

- IT #1: ________
- IT #2: ________
- Passed: Y / N
Section 4
3U CubeSat Acceptance Checklist
3U CubeSat Acceptance Checklist

Mass (< 4.00 kg) Functional Y / N
Spring Plungers (Depressed) RBF Pin (≤ 6.5mm)
Deployment Switches (Depressed) Y / N
Rails Anodized
Deployables Constrained

Mark on the diagram the locations of the RBF pin, connectors, deployables, and any envelope violations.

Authorized By:
IT #1: ________
IT #2: ________
Passed: Y / N

<table>
<thead>
<tr>
<th>List Item</th>
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<tbody>
<tr>
<td>Width [x-y]</td>
<td>Side 1 (-Y)</td>
<td>Side 2 (-X) Side 3 (+Y) Side 4 (+X)</td>
</tr>
<tr>
<td></td>
<td>+Z</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Middle</td>
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</tr>
<tr>
<td></td>
<td>-Z</td>
<td></td>
</tr>
<tr>
<td>Height [x-y]</td>
<td>Rail 1 (+X, -Y)</td>
<td>Rail 2 (-X, -Y)</td>
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</tr>
<tr>
<td>+Z Standoffs</td>
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<tr>
<td>-Z Standoffs</td>
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<tr>
<td>Protrusions</td>
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<td>Side 2 (-X) Side 3 (+Y) Side 4 (+X) Side 5 (-Z) Side 6 (+Z)</td>
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Section 5
3U+ CubeSat Acceptance Checklist
3U+ CubeSat Acceptance Checklist

Project: ____________________________ Date/Time: ____________________________
Organization: __________________ Location: ____________________________
Satellite Name: __________________ Satellite S/N: ____________________________

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<tr>
<th>Mass (&lt; 4.00 kg)</th>
<th>RBF Pin (≤ 6.5mm)</th>
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<tr>
<td>Spring Plungers</td>
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<tr>
<td>(Depressed)</td>
<td>Flush with Standoff Y / N</td>
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<tr>
<td>Deployment Switches</td>
<td>Functional Y / N</td>
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<tr>
<td>(Depressed)</td>
<td>Flush with Standoff Y / N</td>
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<tr>
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<td>Y / N</td>
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<tr>
<td>Deployables Constrained</td>
<td>Y / N</td>
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Mark on the diagram the locations of the RBF pin, connectors, deployables, 3U+ Protrusion, and any envelope violations.

Authorized By:
IT #1: ____________
IT #2: ____________
Passed: Y / N

3U+ Volume

Length (Z): ______ ≤ 36mm
Diameter: ______ ≤ 64mm
3U+ Centered: Y / N

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<td>Side 2 (-X)</td>
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<td>+Z Standoffs</td>
<td>Length x Width</td>
<td>≤ 6.5mm</td>
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<tr>
<td>-Z Standoffs</td>
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<td>Side 6 (+Z)</td>
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APPENDIX B

MSFC TEST SETUP SCHEMATIC

Figure 35. Test stand P&ID
Figure 36. Wiring diagram – data acquisition and ignition
Figure 37. Wiring diagram – data acquisition and control interface