Development of a High Performance Micropropulsion System for CubeSats

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Committee Membership

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Abstract

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Picosatellites are defined as satellites with a mass between 0.1 and 1kg (Miniaturized satellite). Picosatellites are typically designed to work together or function in formations (Miniaturized satellite). A specific type of Picosatellite known as CubeSats were introduced in 1999 and since then have increased in popularity so that there are now over 80 CubeSat programs around the world. CubeSats are defined as cubic units 10cm on each side and no more than 1kg in mass. CubeSats are required to conform to the CubeSat Standard created by California Polytechnic State University and Stanford University and be compatible with Cal Poly's P-POD deployment system (Toorian, 2005). Some CubeSat uses include earth imaging, communications projects and various scientific experiments. CubeSats currently require attitude control and in the future, may require, maintaining a specific orbit, or changing orbit. With this ability many new activities may be possible for CubeSats. These activities could include rendezvous, vehicle inspection, formation flying and de-orbiting. For these activities to be possible, a high performance propulsion system is required. The goal of this thesis is to design and test an affordable, safe, and effective micro-propulsion system for CubeSats.

Keywords: CubeSat, monopropellant, bipropellant, propulsion, Picosatellite, thruster

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Introduction

Due to student interest in having an affordable, viable satellite program, California Polytechnic State University and Stanford University teamed up to develop a program that limits some of the open ended design objectives encountered in satellite design. They limited the scope in such a way as to define the shape, weight, format, mounting, and deployment interface. This is known as the CubeSat program. The CubeSat Standard developed by California Polytechnic State University and Stanford University in 1999 has provided an interface and system for deploying CubeSats as well as a definition of form factor and mass constraints. This CubeSat standard is designed to provide a base upon which students can design and manufacture a satellite to perform a desired research experiment. The standardization of CubeSats resulted in decreasing the overall satellite cost from design and manufacture to launch, as well as the time it takes for the overall design, procurement, and manufacturing. With this program a student entering college may be able to work on a CubeSat project and actually see that satellite built and launched into orbit before graduating.

The CubeSat standard outlines requirements for proper function inside the Poly Picosatellite Orbital Deployer (P-POD). The P-POD structure is a cuboid with the capacity to hold up to three CubeSat units at a time (Figure 1). It consists of slide rails on all four corners that the CubeSats interface with. At one end of the P-POD is a door that opens during CubeSat deployment and at the other end is a slider with a spring that pushes the CubeSat out into orbit at the appropriate time. The slider and slide rails can be seen in Figure 1 on the next page.

Figure 1. CubeSat P-POD satellite deployment system.

In general, CubeSat projects consist of a single 1 kg, 10cm cube. Then the CubeSat is loaded into the P-POD with two other CubeSats with spring plungers separating them from one another. Some CubeSat projects have expanded to “Double Cube” and “Triple Cube” configurations. A “Double Cube” or 2 unit configuration refers to a satellite with dimensions of 10 cm by 10 cm by 20 cm long and 2 kg. It is basically just 2 CubeSats machined into one unit. A “Triple Cube” or 3 unit configuration has a similar definition with dimensions of 10 cm by 10 cm by 30 cm and 3 kg. An attractive feature of the “Double Cube” and “Triple Cube” CubeSats is the fact that a whole one unit may be reserved for a propulsion system and coupled to additional one or two units to form the overall satellite.

The launch vehicles used to deliver the CubeSats into orbit include the EUROROCKET, Japan's M-V-8, and the Russian DNEPER rocket. Because of the size and weight of the CubeSats, they can be affordably added to the payload of the large launch vehicles because in most cases the launch vehicles are not designed around their primary payload so there is usually some mass margin left for the CubeSats to exploit. Since the CubeSats are "hitching a ride" on a launch vehicle and are not the primary payload, the launch time and profile is dictated by the primary payload. Because of this, the CubeSat may not be released at a desired time or in a desired orbit. A high performance propulsion system would allow the CubeSat to attain a more desirable orbit or orientation once it is deployed, where a CubeSat without propulsion will be deployed with an arbitrary orientation and an orbit that may or may not be desirable.

Propulsion System Requirements

The general requirements for the Micropropulsion system will be discussed in the following sections. Since the Micropropulsion system will be designed specifically for CubeSats, all CubeSat related requirements must be met. The design philosophy used throughout this project will be simplification, the use of commercial off the shelf components, and rapid prototyping to support component and system testing rapidly.

CubeSat Compatibility

To ensure compatibility between the CubeSat and P-POD, a CubeSat Standard outlining interface and safety requirements was developed by Cal Poly and Stanford University. The general requirements state that the CubeSat must not present any danger to the P-POD, launch vehicle, or other CubeSats. Pyrotechnic devices which use

explosive charges to perform a specific task such as opening a valve are not allowed inside the CubeSat. In addition, the CubeSat must be designed to operate as smoothly as possible inside the P-POD.

The quantitative and qualitative requirements as outlined by the CubeSat Standard are summarized below in the following tables (Toorian, 2005).

Table 1. CubeSat mass and geometry requirements.

Specification	Value
Dimensions per unit	10cm x 10cm x 10cm
Mass per unit	1kg
Maximum Number of units	3
Center of mass location per unit	Within 2cm of geometric center

Table 2. CubeSat mechanical requirements.

Specification	Value
Rails must be smooth and edges rounded	Minimum radius of 1mm
Minimum length of CubeSat rail in contact with P-POD guide rails	75% (up to 25% of CubeSat rail length can be recessed)
Rails must prevent cold welding, reduce wear and electrically isolate CubeSat from P-POD	Rails must be hard anodized
Separation spring included at designated points	Spring Plungers recommended, custom separation must be approved
Similar thermal expansion to Aluminum 7075-T73 used for main CubeSat structure	Aluminum 7075 or 6061-T6 recommended
Deployables must be constrained within CubeSat	P-POD rails and walls not to contain deployables

Additional requirements that apply to this application include: no electronics active during launch and a remove-before-flight pin to be removed after P-POD integration for CubeSat activation.

Affordability

A very important goal in the creation of the CubeSat program is space access at low cost. In order for University clubs and small companies to participate in the CubeSat program, overall mission costs must be kept to a minimum. For a CubeSat micro-propulsion system to be viable, its cost must also be kept low. Based on cost estimates by Dr. Tomas Svitek the CubeSat propulsion system overall cost per unit should be within \$100k-\$250k (Svitek, 2008-2009). This cost estimate is based on a fully qualified, flight ready assembly including all mechanical components fully assembled and ready to be fueled.

Safety

All ground handling and range safety requirements as outlined in Air Force Space Command Manual 91-710 Volume 4 must be met. Additional multiple redundant systems shall be implemented and part reliability verified.

Redundancy and Parts Reliability

All sealing components in the propulsion system including valves and o-rings shall be redundant as much as possible up to triple redundancy. Since this is a new application with major mass and volume constraints, redundancy may not always be possible. When this is the case, extremely high part reliability is required. This includes testing all components at operating conditions extensively.

Performance

The most common performance parameter used to compare propulsion systems is specific impulse or (I_{sp}). Specific impulse compares the amount of thrust produced by a propulsion system as a function of the mass flow rate through the thrusters. Equation 1 below shows the definition of specific impulse (I_{sp}) in units of seconds, with F being the thrust produced in Newtons, g the acceleration due to gravity in m/s^2 and $\dot{m}$ the mass flow rate of propellant through the thruster in kg/s .

$$I_{sp} = \frac{F}{g \cdot \dot{m}} \quad (1)$$

Some definitions of I_{sp} leave the gravity term out for units of velocity, but the gravity term will be used in this paper because it allows for direct comparisons of propulsion systems using either English or SI units.

Most propulsion systems use either stored pressure energy or a chemical reaction to accelerate a fluid through a nozzle. This causes a change in momentum which produces thrust. The effects of these forces on the motion of the spacecraft will be detailed below. The momentum equation derived from Newton's Second Law and presented in Introduction to Fluid Mechanics is shown below (Fox, 2004). It states that the sum of the forces on the system is equal to the sum of the rate of change of momentum inside the control volume and the net rate of flux of momentum out through the control surface.

$$\overline{F} = \frac{\partial}{\partial t} \int_{CV} \rho \vec{v} dV + \int_{CS} \vec{v} \rho (\vec{v} \cdot \hat{n}) dA + \Delta P \cdot A \quad (2)$$

Starting with equation 2 and assuming a 1-dimensional, incompressible, steady flow, and a constant cross-sectional area control volume, equation 3 is derived (Fox, 2004). Note that the ΔP in equation 2 refers to the difference in atmospheric and fluid exit pressure.

$$F = \dot{m}V_e + (P_e - P_a)A_e \quad (3)$$

Equation 3 relates the total thrust due to both the momentum flow of the fluid through the thruster as well as the thrust produced by the exit pressure of the fluid with $\dot{m}$ being the mass flow rate through the thruster in kg/sec, V_e the exit velocity in m/s, P_e the exit pressure in kPa, P_a the ambient pressure in kPa and A_e the exit cross-sectional area in m^2 .

Overall spacecraft performance is described in terms of a change in velocity for the entire spacecraft over the mission. The term ΔV (said “delta-V”) is used to describe this performance. Mission design analysis is used to calculate the required ΔV for all the spacecraft maneuvers for the mission. The propellant weight required for the mission can be calculated from the required ΔV using the Tsiolkowski equation also known as the basic rocket equation and its corollaries, equation 5 and 6 (Brown, 1996). This equation written in its final form (equation 5) relates the propellant weight to the ΔV in m/s, specific impulse (I_{sp}) in seconds and the final vehicle weight (W_f) in kg.

$$\Delta V = g * I_{sp} * \ln \frac{W_i}{W_f} \quad (4)$$

$$W_p = W_f \left[\exp \left(\frac{\Delta V}{g * I_{sp}} \right) - 1 \right] \quad (5)$$

From the previous equations, with a desired ΔV and initial or final spacecraft weight, the required propellant weight for a given specific impulse can be calculated. Since CubeSats have a defined maximum allowable mass and the ΔV is calculated from the mission requirements, a propellant weight can be calculated for a given specific impulse. In order to take a modular approach to this project, it was decided to allocate a full 1 kg, 10 cm cube unit for propulsion integrated into a 3-unit configuration or “Triple Cube” CubeSat. An example of a “Triple Cube” is the CAN X2 built by the University of Toronto Space Lab shown below in Figure 2.

Figure 2. CAN-X2 triple CubeSat.

From mission analysis calculated by Dr. Tomas Svitek of Stellar Exploration Inc., the micro-propulsion system shall produce a $\Delta V = 500$ m/sec in a 3-unit configuration for substantial orbit maintenance and maneuvers to be possible. With a $\Delta V = 500$ m/sec the propulsion system will be capable of changing a circular 300 km orbit to an elliptical

orbit with minimum radius of 300 km and maximum radius of 2400 km. This will be the design target for the propulsion system.

Table 3 below shows the required propellant weight for a given specific impulse for the desired ΔV of 500 m/s and the maximum allowable 3-unit CubeSat mass of 3 kg. Since each CubeSat unit can weigh a maximum of 1 kg, the propellant mass percentage of the propulsion system unit is also included in the table. Note that propulsion system propellant mass fraction is defined as the mass of propellant divided by the total mass of the propulsion system with propellant.

Table 3. Required propellant mass for desired performance.

Propellant Mass (kg)	Specific Impulse, I_{sp} (s)	Propulsion System Propellant Mass Fraction
1.01	175	1.01
0.87	200	0.87
0.76	225	0.76
0.68	250	0.68
0.61	275	0.61
0.56	300	0.56
0.51	325	0.51
0.47	350	0.47
0.44	375	0.44

As shown by the table above, any I_{sp} below 175 s will require over 100% of the propulsion system unit allowable mass to be propellant, which is not possible. A more likely specific impulse requirement would be in the range of 220 s or larger to allow for the mass of the thrusters, valves, and storage tank to be included in the 1 kg maximum mass requirement.

Typical cold gas systems which use inert gas stored at high pressure can achieve specific impulses between 50 to 120 s which is not enough to meet the 500 m/s design goal since the required propellant mass would be greater than the maximum allowable mass (Sutton, 2001). Well designed monopropellant Hydrazine engines have achieved an I_{sp} of around 220-230 s, which would require approximately 75% of the allowable 1 unit mass, leaving approximately 250 g for the tank, valves, catalysts and thrusters. Bipropellant propulsion systems typically produce specific impulses between 230 and 325 s (Sutton, 2001). Shown below in Figure 3 is the effect of I_{sp} and dry mass fraction, on the overall spacecraft performance (Brown, 1996). The dry mass fraction is defined as the empty spacecraft mass divided by the fueled spacecraft mass.

Figure 3. Performance vs. Dry Mass Fraction for various I_{sp} values.

If the objectives of the mission can be accomplished with a 2-unit CubeSat or “Double Cube” with one of those units being the Micro Propulsion system, a much larger Delta V is possible. However, for this project only a 3-unit CubeSat will be considered.

To estimate proper thrust requirements, the thrust to weight ratio of past spacecraft with similar mission objectives was studied. Since the thrust to weight ratio is a dimensionless quantity, the overall spacecraft mass is not important for the analysis to be valid. This process was described in “Space Propulsion Analysis and Design” (Humble, 1995). A general trend in past spacecraft includes a thrust to weight ratio of at least 0.2 for missions requiring orbital transfer. For missions only requiring orbit maintenance or attitude control, a thrust to weight ratio of less than 0.1 is recommended (Humble, 1995). The mission requirements for the CubeSat propulsion may include both orbit transfer and orbit maintenance, a thrust to weight ratio of 0.25-0.35 will be considered to allow for orbit transfer maneuvers. The mass of the 3-unit CubeSat is defined as 3 kg, following the past trends of spacecraft, a total thrust level of around 7.4-10.3 Newtons is recommended. For this project, the thrusters will be designed to fall within these values and adjusted if empirical testing deems necessary. This requirement puts more importance on valve response and the control system to act quickly to pulse the thrusters for orbital maintenance and attitude control since the thrusters may be oversized for these types of maneuvers. This is necessary to ensure proper thruster performance for orbital transfer capability.

Propulsion System Comparison

Two types of propulsion systems will be considered and evaluated to choose the most appropriate system. These two systems include the monopropellant and the bipropellant propulsion systems. From this, further design studies will be conducted to choose the propellant and other details required to fully define the propulsion system with components.

Monopropellant

A monopropellant system uses a single fluid that is injected through a catalyst bed or an electrically heated combustion chamber, where it decomposes generating hot, high velocity gas. The hot gas flows through a converging diverging nozzle producing thrust (Brown, 1996). A compatible fluid for use in a monopropellant system must be a slightly unstable chemical that decomposes exothermically (Brown, 1996). According to Charles D. Brown only three monopropellants have ever been used on flight vehicles including: hydrazine, hydrogen peroxide, and propyl nitrate. Propyl nitrate use was discontinued due to shock sensitivity. Hydrogen peroxide was used considerably starting with the German V2 rockets however; slow decomposition during storage problems has discontinued its mainstream use (Brown, 1996). Hydrazine was experimented with and found to have difficulty in initial ignition. This problem was solved by the Shell Development Company and Jet Propulsion Laboratory developing the Shell 405 iridium pellet catalyst bed in 1962, allowing for almost unlimited spontaneous restart capability. Due to the high performance, spontaneous restart capability, clean exhaust, low flame temperature, and the relative stability of hydrazine made it the chosen propellant for monopropellant systems and is the only monopropellant in mainstream use today (Brown, 1996). Since

hydrazine is used exclusively in current monopropellant systems, its use will be assumed for the trade studies in this paper.

The gas produced after Hydrazine decomposition is in the 2200°F range which means that high-temperature alloys should be used for the nozzle, but a cooling system is not generally required (Brown, 1996). Monopropellant Hydrazine thrusters have achieved thrust levels between 0.1 and 600 lbs and many systems are in use today.

Bipropellant

A bipropellant system uses two fluids stored separately consisting of a fuel and an oxidizer. A bipropellant system can be pressure or pump fed. The oxidizer and fuel are fed in their liquid state through the injector (Brown, 1996). Combustion takes place in the combustion chamber as the liquids mix producing a hot, high velocity gas that flows through the converging diverging nozzle producing thrust (Brown, 1996). The role of the injector is to promote stable, efficient combustion in the combustion chamber without overheating the chamber (Brown, 1996). The injector design in a bipropellant system is the most important component relating to stability and performance (Brown, 1996). Typical exhaust temperatures are in the range of 6000°F and velocities around 2000 ft/s (Brown, 1996). These high temperatures require cooling of the combustion chamber and nozzle. Some cooling methods used include: film cooling which washes liquid propellant down the combustion chamber wall cooling the wall by transferring heat from the wall for fuel vaporization and ablative cooling which requires the use of special materials for the chamber wall that are evaporated and eroded away during firing (Brown, 1996).

The fuel most commonly used in a bi-propellant system today is a mixture of Hydrazine and monomethylhydrazine (MMH) (other than cryogenics as used in the Space Shuttle) (Brown, 1996). Nitrogen Tetroxide is used with Hydrazine based fuels and is hypergolic, meaning it ignites spontaneously on contact (Brown, 1996). This property allows for pulsing of the thruster. The exhaust gases of this bipropellant system are generally compatible with stainless steel, aluminum, and Teflon. Since the Hydrazine and MMH mixture with Nitrogen Tetroxide oxidizer is the most common fuel-oxidizer pair in a bipropellant system, its use will be assumed in the bipropellant system for the system comparison in this paper (Brown, 1996). Table 4 shown on the next page outlines advantages and disadvantages between monopropellant and bipropellant propulsion systems.

Table 4. Advantages and disadvantages of monopropellant and bipropellant systems.

Monopropellant Propulsion System	
Advantages	Disadvantages
Reliable	Catalyst bed miniaturization complexity
Demonstrated required I_{sp}	Cold start damage to and Flow erosion of catalyst bed
Pulsing capable	
Rapid start/stop capable	
High I_{sp}	
Part availability	
Propellant availability	
Compatible with CubeSat material requirements (Aluminum)	
Bipropellant Propulsion System	
Advantages	Disadvantages
Increased I_{sp} over monopropellant	Most complex propulsion system (Injector design etc.)
No catalyst bed required	Most expensive propulsion system
	Larger mass due to plumbing and storage of fuel and oxidizer
	Miniaturization very complex
	Oxidizer compatibility issues with tank/bladder material
	Cooling system/systems required
	Decreased volumetric efficiency from fuel oxidizer storage
	Increased number of failure modes due to complexity

The monopropellant propulsion system is the better choice in this case since it has demonstrated the performance required while being the least complex and the overall less expensive system. The catalyst bed degradation problem can be solved by redesigning the catalyst or by using electrothermal thrusters which will be considered next.

Electrothermal Thrusters

The I_{sp} of a hydrazine monopropellant system can be greatly increased if the temperature of the decomposition products is increased. One way of achieving this includes the use of electrothermal thrusters over conventional monopropellant system thrusters (Brown, 1996). This method should be considered when adequate time before firing is allowed to condition the thrusters, steady-state burns are required, low thrust levels are required and substantial spacecraft power is available for a short time (Brown, 1996).

The electrothermal thrusters use electric energy to both cause decomposition and heat decomposition products (Brown, 1996). The hydrazine is injected into the combustion chamber and decomposes over a platinum thermal capacitance screen pack that has been heated to approximately 1300°F prior to firing (Brown, 1996). The decomposition products then enter a vortex heat exchanger tangentially and are heated further to approximately 3500°F (Brown, 1996). This can increase the I_{sp} up to 28% (Brown, 1996). A commercial thruster, Intelsat V, produces a thrust of 0.1 lbs. and requires a maximum of 414 W. The main advantage of this type of thruster over the conventional monopropellant system is the fact that a catalyst bed is no longer required which eliminates the problem of catalyst bed degradation. Table 5 shown on the next page summarizes the advantages and disadvantages of using an Electrothermal Thruster over the conventional thruster used in a monopropellant system

Table 5. Advantages and disadvantages of electrothermal thrusters.

Electrothermal Thrusters	
Advantages	Disadvantages
Increased Isp over conventional monopropellant system	Time needed before firing to heat components
No Catalyst bed required	Large electrical power required
	Added complexity to conventional monopropellant system
	Higher cost over conventional monopropellant system
	More mass over conventional monopropellant system
	Arcjet thrusters require time to start arc, short pulses not practical

The Electrothermal thruster is not a viable option for CubeSat propulsion due to the added mass and the large electrical power required for heating. In addition, the extra time needed before firing the thruster to heat the chamber would not be acceptable for attitude control thrusters.

The Arcjet thruster also uses electrical energy to heat the decomposition products (Brown, 1996). The hydrazine is decomposed over an iridium catalyst bed similar to a typical monopropellant propulsion system (Brown, 1996). A gas manifold directs the hot decomposition products into the electrode region while inducing a rotary motion (Brown, 1996). The products pass through the throat of the thruster where they are heated and accelerated producing thrust (Brown, 1996). Similar decomposition product temperatures to the electrothermal thrusters as well as power requirements are expected. The Arcjet thruster acts as a conventional monopropellant thruster with the arc turned off. To start the arc requires a process which includes firing the thruster in conventional form for approximately 20 s to ensure propellant flow through the arc region of the thruster, then the arc is started and steady-state temperature and performance are reached. This process

clearly shows that Arcjet thrusters are used in longer, steady-state burns and not necessarily for short burns used in attitude control. For the same reasons listed above for the Electrothermal Thrusters, Arcjet thrusters are not a viable option for use in CubeSats. However, if the mass requirements were relaxed slightly the Electrothermal or Arcjet thrusters may be an attractive option for missions requiring longer steady state burns (Brown, 1996).

The monopropellant system using a conventional catalyst bed decomposition thruster is the most appropriate propulsion system for CubeSat propulsion.

Hydrogen Peroxide vs. Hydrazine

Hydrogen Peroxide (H_2O_2) has had success as a monopropellant in the past and has received more attention presently due to its environmentally friendly decomposition products and less stringent handling requirements. Hydrogen Peroxide decomposes into water and oxygen making it a non-toxic propellant. Problems in the past with slow decomposition during long storage periods have seen it replaced by Hydrazine (Brown, 1996). Another reason for Hydrogen Peroxide's replacement is the fact that Hydrazine thrusters have been developed to produce a specific impulse on the high end of approximately 220-230 s while Hydrogen Peroxide is on the order of 150 s (Micci, 2000). The fact that Hydrogen Peroxide is non toxic is attractive from a cost standpoint also because the ground handling and storage may require less infrastructure and equipment than that of Hydrazine. However, care needs to be taken in the storage of Hydrogen Peroxide because it will decompose when it comes into contact with a catalyst and almost any organic material can act as a catalyst. Since Hydrazine systems have been the most common monopropellant propulsion systems in use, the testing and launch site range

safety personnel have the processes and infrastructure needed for the handling of Hydrazine. Even though Hydrogen Peroxide may be a “greener” propellant it may require more effort from a range safety standpoint to handle it since it has not been extensively used.

Because the estimated required specific impulse for this application is on the order of 220 s as shown in Table 1, the lower specific impulse of Hydrogen Peroxide eliminates it from consideration in this application. If the required ΔV requirements decreased, Hydrogen Peroxide as a propellant may be acceptable. This may decrease the overall mission cost due to possibly less stringent handling requirements as compared with Hydrazine.

For this application a monopropellant propulsion system using Hydrazine propellant gives the best chance of project success.

Propellant Safety

With the use of Hydrazine for the propellant in the micro propulsion system, come safety concerns. Since Hydrazine propulsion technology is mature, many guidelines and procedures have been established to ensure safe handling and use of Hydrazine as a propellant. These guidelines and procedures must be followed closely to ensure personnel safety.

Storage and Ground Handling

Anhydrous Hydrazine can be handled safely with strict adherence to safety requirements. It is preferred to store Hydrazine outside in a free standing storage container separate from other buildings (Company). If the Hydrazine is stored inside it

should be stored in a standard flammable liquid storage room or cabinet (Company). The storage tanks should be located in a diked area, meaning any spilled Hydrazine will not be able to escape its designated storage area (Company). Hydrazine must be stored away from heat, sparks, open flame, and oxidizers (Company). All Hydrazine storage containers and cabinets must be clean and must not be contaminated (Company). Hydrazine may be stored in DOT approved containers in which it had been shipped (Company). Since Hydrazine is Hygroscopic meaning it absorbs moisture from the air, any open containers must be purged with nitrogen to prevent exposure to moisture (Company). For this application small amounts of Hydrazine will need to be handled and stored. An open hydrazine storage container is not expected to be needed and the hydrazine will probably be stored in its DOT approved shipping container (Company). Since hydrazine is stable it can be stored for long periods of time without decomposition (Company). The storage container should always remain securely closed and should be carefully vented when opened (Company). Anhydrous Hydrazine is insensitive to shock and friction (Company).

Since Hydrazine is toxic and a suspected carcinogen, special care must be taken during handling to avoid exposure to the skin and eyes. In the event of exposure to eyes or skin, flushing with water for at least 15 minutes is recommended. Ingestion of Hydrazine is also very dangerous and immediate medical attention is required. In order to avoid ingestion or inhalation exposure the National Institute of Occupational Safety and Health (NIOSH) has published recommendations for proper respirators. These respirators were selected based on their assigned protection factor. The estimated concentration of a contaminant is divided by its exposure limit to attain a hazard ratio. The assigned

protection factor must be greater than the hazard ratio. For this application NIOSH recommends a respirator with an assigned protection factor of at least 10,000. Only respirators with a positive pressure fresh air supply meet this requirement (Stellman). In order to avoid exposure to the skin and eyes, OSHA recommends anyone near hydrazine should wear an acid resistant coverall suit with boots, gloves, goggles, a respirator and a face shield.

In the case of a spill, dilute spilled Hydrazine to 40% Hydrazine 60% water solution. Hydrazine can be disposed of after dilution and subsequent neutralization using dilute sulfuric acid. This solution may be drained in the sewer with abundant water (Organization, 1991). Foam may be applied to slow vaporization and small spills may be absorbed using sand (Organization, 1991). Any personnel attempting to clean up a spill must ensure personal protection including a self contained positive pressure respirator as described in the preceding paragraph (Organization, 1991).

Thruster Configuration

Many different thruster configurations have been proposed and used on spacecraft. These configurations range from 24 thrusters used on spacecraft with deep space missions to 4 thrusters on small satellites (Svitek, 2008-2009). The larger number of thrusters is used when maneuvers in any of the 6 degrees of freedom are required instantly. As this requirement is relaxed, thrusters may be removed down to the simplest 4 thruster configuration. The four thruster configuration uses thrusters that are canted outward to allow for rotation and translation on all three axes but lose efficiency because the canted angle of the thrusters forces them to act against each other, wasting propellant

(Svitek, 2008-2009). For this application, a four thruster configuration will be used because it is the simplest configuration and will still allow for translation and rotation in all three principal axes. This configuration will provide useful maneuvers especially orbit maintenance and transfer. In addition, the propulsion system can be modified in the future for more maneuverability as mission requirements dictate by incorporating more thrusters when mass and geometric requirements allow.

Required Thrust

The thrust required for any spacecraft is determined by the mission requirements. Since this is a new application, the required thrust magnitude and duration is not well defined because it is not known what maneuvers will be required of the micro propulsion system. As discussed in the performance section of this paper, the thrust to weight ratio of previous spacecrafts have been studied and a general thrust to weight ratio defined for desired maneuvers. From this analysis a required total thrust level of 7 - 10 N was derived. Since a four thruster configuration will be used, a corresponding thrust of between 1.8 - 2.6 N per thruster will be assumed in the thruster design. With the desired thrust and assumed I_{sp} , the mass flow rate can be calculated using Equation 1 in the performance section of this paper. From this the corresponding thruster combustion chamber pressure, combustion chamber dimensions, and converging- diverging nozzle shape and dimensions can be derived.

Thruster Combustion Chamber Pressure

The thruster combustion chamber pressure is related to the desired thrust and the combustion chamber dimensions including the nozzle throat diameter. The propellant I_{sp} is also affected by the thruster combustion chamber pressure for a hydrazine system.

Hydrazine decomposes into ammonia and nitrogen in an exothermic reaction. The ammonia then breaks down into hydrogen and nitrogen endothermically (Brown, 1996). Since this reaction is endothermic the overall specific impulse is reduced compared with a decomposition of Hydrazine into ammonia and nitrogen only. Ammonia dissociation can be reduced with one of the affecting factors being the thruster combustion chamber pressure (Jurgen Mueller). Because of the large number of coupled variables involved in thruster design including the thruster combustion chamber pressure, historical data on commercial thrusters was collected to use as a starting point.

A survey of commercial, off-the-shelf thrusters in the range of 0.5 – 3 N was conducted. The specifications for each were collected including nozzle diameter, and inlet or chamber pressure (Monopropellant Hydrazine Thrusters, 2003). The chamber pressure for each ranged from 550 – 2250 kPa. Over this pressure range, a table was constructed comparing chamber pressure to the theoretical nozzle throat diameter required for a given thrust. From the required thrust, the minimum mass flow rate of propellant was calculated and used to calculate the nozzle throat diameter. The results are recorded in Table 6 on the next page and show that the required theoretical nozzle throat diameter is on the order of 0.9-2.1 mm.

Table 6. Chamber pressure and corresponding throat diameter.

Thrust = 1.84N		Thrust = 2.58N	
Chamber Pressure (kPa)	Throat Diameter (mm)	Chamber Pressure (kPa)	Throat Diameter (mm)
550	1.82	550	2.15
650	1.67	650	1.98
750	1.56	750	1.84
850	1.46	850	1.73
950	1.38	950	1.64
1050	1.31	1050	1.56
1150	1.26	1150	1.49
1250	1.21	1250	1.43
1350	1.16	1350	1.37
1450	1.12	1450	1.32
1550	1.08	1550	1.28
1650	1.05	1650	1.24
1750	1.02	1750	1.21
1850	0.99	1850	1.17
1950	0.96	1950	1.14
2050	0.94	2050	1.11
2150	0.92	2150	1.09
2250	0.90	2250	1.06

Table 6 demonstrates that as the chamber pressure is increased the throat diameter for a given thrust decreases. This means that a smaller engine both in mass and volume can be used for a system with a higher chamber pressure. However, to get a higher chamber pressure, the tank storage pressure must be increased which then increases the storage tank mass. In the range of nozzle throat diameters listed in the table above, as the nozzle throat diameter decreases, the complexity and cost of machining the nozzles increase rapidly. Also, because of the small size of the nozzle and combustion chamber, the mass savings due to increased pressure allowing for an overall smaller engine is not significant. Therefore the manufacturability of the nozzle and combustion chamber will drive the design of the nozzle and dictate the chamber pressure accordingly. Five hundred

and fifty kPa was chosen based on the manufacturability of the nozzle and combustion chamber and the fact that the lower pressure will allow for a much lighter storage tank design. The chamber pressure of 550 kPa corresponds to a throat diameter in the range of 1.8 – 2.1 mm. This may be modified if needed after empirical testing of the thruster. The throat diameter used in the thruster design will be 2.0 mm.

Catalyst Bed

Historically, the catalyst used in most commercial spacecraft applications is the SHELL 405 iridium coated ceramic catalyst (Monopropellant Rocket, 2009). These catalyst pellets are held in place with retaining screens on the entrance and outlet sides of the chamber. A ceramic pellet with a catalyst coating is also common in automotive and industrial pollution control applications. The ceramic acts as the structure for the platinum or iridium based catalyst. A common catalyst design challenge is reducing the erosion of the catalyst material. This occurs due to the high pressure and temperature of the entering propellant “washing” the catalyst coating or plating from the support structure. Armadillo Aerospace has conducted experiments with stainless steel wire with platinum plating for use as a catalyst as well. This came with poor results due to the platinum being “washed off” of the wire from the flow of propellant (News Archive, 2003). Don Platt of Micro Aerospace Solutions has experimented with platinum metal “gauze” for use as a catalyst with good results. Because of the difficulty in obtaining Shell 405 catalyst which is now owned and produced by Aerojet, a platinum wire or platinum coated wire or mesh catalyst may be the right choice for this application.

A platinum wire mesh disc with a platinum/iridium ring support welded around the circumference was designed with the intention of “stacking” as many of these catalyst washers inside the combustion chamber as required (Edney, 2009). Graham Edney of Santa Barbara aided in the design of the catalyst structure and came up with the modular catalyst design approach. This design allows for adding or subtracting catalyst surface area in order to use the right amount of catalyst for complete decomposition of hydrazine. In order to hold the screens in place, the combustion chamber will be machined in a way that the catalyst screens will be held by an internal machined shoulder.

Figure 4. Nozzle section view with catalyst screen concept.

Another catalyst iteration uses the same structure shown in Figure 4 with the addition of a bolt through the center of the “screens” with a nut attached to hold the entire unit together. This was another design idea provided by Graham Edney. This results in both keeping the assembly together during impact from the propellant stream as well as making the loading of the catalyst into the thruster assembly easier by loading the entire

catalyst at once rather than each individual screen at a time. Because of the small size of the rings and very fine mesh size, a laser welder was required to manufacture the catalyst. Graham Edney formed the rings and cut the wire mesh to the appropriate size by hand and laser welded each individual screen assembly. The assembled catalyst is shown in Figure 5. Hot fire testing will be required to verify the proper function and structural integrity of the catalyst. Also the correct amount of catalyst surface area can be attained empirically by a series of hot fire tests.

Figure 5. Assembled catalyst shown next to the valve stud (left) and partially loaded into combustion chamber (right).

Storage Pressure

The main factor driving the storage pressure is the chamber pressure required for a specific thrust (Humble, 1995). The thrust is proportional to the mass flow rate of the propellant which is proportional to the chamber pressure and chamber throat area (Humble, 1995). Based on this, the throat area can be reduced by increasing the chamber pressure for a given mass flow rate (Humble, 1995). And if the chamber dimensions can be reduced the entire engine's dimensions can be reduced thus reducing the overall mass

of the engine (Humble, 1995). However, as chamber pressure increases the storage pressure increases leading to greater storage tank mass (Humble, 1995).

The hydrazine propellant is pressurized with a small volume of gas at the required storage pressure. Since the designed operating pressure is at 550 kPa, the initial storage pressure must be larger in order to have a longer operating time. The theoretical minimum tank absolute pressure for thruster operation is equal to the pressure losses through the valves, catalyst bed, and the nozzle throat. Based on historical data about 75-80% of the propellant is used over the lifetime of the mission (Svitek, 2008-2009). Dr. Tomas Svitek proposed calculating a storage pressure by assuming that 20% of the storage pressure is equal to the pressure losses through the system and multiplying that value by five to get the storage pressure (Svitek, 2008-2009). This is referred to as the blow down ratio and is defined by the ratio of initial pressurant gas pressure over the final pressurant gas pressure. Charles D. Brown, the author of “Spacecraft Propulsion”, states that blow down ratios of 3-4 are in common use and ratios of up to 6 have been flown (Brown, 1996). He also provided a function for the decrease in I_{sp} over the blow down period, which predicted an I_{sp} decrease of 2.5% for a blow down ratio of 5 (Brown, 1996). It was assumed that approximately 220 kPa would be required for thruster operation which led to an initial storage pressure of approximately 1100 kPa (Svitek, 2008-2009). For this project 1100 kPa will be assumed for the initial storage tank pressure.

Thruster Valve Design and Integration

The thruster valve requirements are based on the pressure and mass flow rate of the thrusters. The maximum pressure rating of the thruster valves should be at least equal

to the storage pressure of the propellant tank. The flow rate through the valve must be greater than or equal to the minimum flow through the thruster. The valve must not allow any propellant leakage over the entire pressure range. The entire propulsion system will be designed around the valve, catalyst bed and nozzle. Off the shelf components or modified off the shelf components will be used if possible in order to narrow the scope of this project and begin testing a viable propulsion system.

The valve chosen for this application is an ASCO micro-solenoid valve (Figure 6). The valve is rated to 760 kPa but will be modified in order to accept 1100 kPa if possible. The input and outlet ports use a #10-32 thread and contain a 0.635 mm diameter orifice. The valve has a diameter of approximately 19mm and is constructed of steel and polymers.

Figure 6. ASCO miniature solenoid valve.

The valve will be placed between the tank and thruster and mounted to the tank cap, via a threaded steel stud threaded into the cap with an o-ring seal and a 1.17 mm orifice. This is

the best off-the-shelf valve in respect to mass and meeting the requirements found up to this point.

Nozzle Design

The converging-diverging nozzle in the engine has four main sections that include the combustion chamber, converging section, the throat, and the diverging exit section. The throat diameter is determined by the mass flow rate of the propellant and the combustion chamber pressure and is the first of the three main sections to be designed. As discussed in the feed pressure section, the required throat diameter is 2 mm as calculated for an ideal case. From this, the combustion chamber and nozzle dimensions can be estimated. As stated in Space Propulsion Analysis and Design, the combustion chamber design is largely based on empirical data (Humble, 1995). To find appropriate combustion chamber dimensions, gas dynamics as well as empirical data will be combined. The first step was to find a theoretical combustion chamber area.

In order to determine appropriate combustion chamber dimensions the chamber contraction ratio needs to be evaluated. The contraction ratio is defined as the area ratio of the combustion chamber and the throat. This ratio is used in calculating the appropriate combustion chamber cross sectional area. From the appropriate cross sectional area, and assuming a cylindrical cross section, the combustion chamber diameter can be calculated.

Assuming isentropic, one-dimensional flow of an ideal gas, the continuity equation relates the density, cross-sectional area, and velocities of a flow at axial locations inside the combustion chamber. From this relationship, an area ratio between the two locations can be derived and the velocities at the two locations can be written in

terms of Mach number. By assuming one location is the nozzle throat where the velocity is known results in Equation 7. The velocity is known at this location assuming choked flow. This allows for calculation of the chamber contraction ratio. Equation 7 below is the final form of the chamber to throat area ratio.

$$\frac{A_c}{A_t} = \frac{1}{M} \left[\left(\frac{2}{\gamma+1} \right) \left(1 + \frac{\gamma-1}{2} \right) M^2 \right]^{\frac{\gamma+1}{2(\gamma-1)}} \quad (7)$$

With an assumed gas velocity inside the combustion chamber, the chamber area to throat area ratio was calculated. The combustion chamber velocities are typically Mach 0.2-0.4. A conservative approach is to assume a Mach number of 0.1 for this analysis (Humble, 1995). A small chamber Mach number corresponds to a larger chamber volume which means a longer residence time for the chemical reaction to occur within the combustion chamber (Humble, 1995). Assuming a Mach number of 0.1, the chamber contraction ratio was calculated to be approximately 6. This result leads to a chamber diameter of 5mm. In order to specify a combustion chamber length, a survey of length to diameter ratios of previous engines was created to use as a starting point. From the historical data gathered by the authors of Space Propulsion Analysis and Design, common combustion chamber length to diameter ratios range from 0.5-2.5, which means the ratio, may be driven by many factors not including combustion performance such as volume constraints. A length to diameter ratio of 1.5 was chosen for this project. It was decided to add a small margin to the diameter for ease of manufacturing both the combustion chamber and the catalyst itself. This lead to combustion chamber dimensions of 6 mm in diameter and 9 mm in length.

For the nozzle design, an appropriate expansion ratio must be determined as well as the overall shape. The theoretical I_{sp} of the thruster is a function of the propellant properties as well as the chamber pressure, ambient pressure, and exit pressure which is a function of the nozzle expansion ratio. Assuming an ambient pressure of 0 kPa and a nozzle efficiency of 90%, the theoretical I_{sp} as a function of expansion ratio was plotted in order to pick an appropriate expansion ratio (Figure 7).

Figure 7. Plot of theoretical I_{sp} vs. Nozzle Expansion Ratio for a chamber pressure of 550 kPa.

Ideally the exit pressure at the end of the nozzle should equal the ambient pressure, but in this application that would require a nozzle of infinite length, since the exhaust exit is at zero absolute pressure. Also the length and mass constraints may

override a longer nozzle if the gain in performance is not great. As shown by Figure 7, the I_{sp} increases greatly for increasing expansion ratio for small expansion ratio values then starts to level off for larger expansion ratios. A table of the change in I_{sp} divided by the change in expansion ratio was also constructed and an expansion ratio of 15 was chosen based on good performance without an excessive nozzle length and diameter. This expansion ratio leads to a nozzle exit diameter of approximately 7 mm and a length of 13 mm.

The nozzle shape is driven by the overall reduction of exhaust gas velocity not normal to the nozzle exit plane. These velocities do not contribute to axial thrust which is considered an energy loss. Many nozzle shapes have been proposed and studied from a general conical shape to an Aerospike and bell mouthed. For this application a conical 15° half angle nozzle is chosen based on the manufacturability and acceptable performance. According to Charles E. Rodgers, a gain in performance of about 3.5-8% was shown in solid rocket nozzle design experiments using a bell-shaped throat and nozzle exit versus a straight cut conical nozzle (Rodgers, 2004). This is a significant gain in performance, however because of the size of the nozzle and the required development of the overall system, the simpler straight cut conical nozzle will be used.

Tank Material and Fabrication

The storage pressure of the hydrazine will be the main driver of the tank design and thickness. Other considerations to take into account in the tank design include the launch loads and the proper function between the propulsion module and the P-POD which may be affected by deflections of the tank. Special care is taken to ensure no

leakage of Hydrazine from the tank which means all mounting and sealing interfaces must have minimal deflections. According to the CubeSat Standard, the material used in the propulsion system module must have a thermal expansion coefficient comparable to 7075 or 6061 Aluminum. The recommended material is either 7075-T6 or 6061-T6 Aluminum. This fits with the material compatibility of Hydrazine which is generally Aluminum, Nickel and Stainless Steel (Riegel, 2003). The benefits of using 7075-T6 over 6061-T6 include a greater yield and ultimate strength, however 7075 is not weldable (Miller, 2008-2009). For this design a thin walled pressure tank of either cubic or cylindrical cross-section was studied. If 7075 Aluminum was to be used the tank must be machined or cast then attached to the CubeSat unit with some type of mechanical hardware since welding is not possible. The tank may have a mounting flange machined or cast into it for mounting to the CubeSat 3-unit structure.

A drawback of using a cylindrical or spherical tank to store propellant is the volume loss over a cubic tank. Cylindrical or Spherical tanks will have a mass savings over a cubic tank due to the large overall wall thickness and large internal radii to reduce stress concentrations required by the cubic tank. Since this application is both mass and volume critical, design studies of several different tank and cap shapes have been conducted and iterations performed to arrive at the maximum allowable propulsion system weight of 1 kg with the lowest dry mass fraction.

Tank Design Concepts

A brainstorming session was conducted at Stellar Exploration, San Luis Obispo and Maglio Inc. on December 15-18, 2008 in order to discuss the storage tank shape and

dimensions along with the layout and initial sizing of the propulsion system components. Present at the meeting include: Dr. Tomas Svitek, Raymond “Bud” Frazee, Chris Maglio, Brian Miller, Alex Diaz, Brian Riskas, and Chris Biddy.

The content of the meeting included a discussion on CNC machining the propellant tank and cap with a cubic cross-section using as much of the 10 cm by 10 cm by 10 cm CubeSat volume constraint as possible and using 7075-T6 aluminum for its superior material strength properties and preferred thermal expansion. Also discussed was a welded sheet metal tank design with welded hypodermic needle tubing to deliver the propellant to the catalyst bed and nozzle. Some concerns were raised with the welded structure which included micro cracks at the weld surface due to both thermal and mechanical cycling. Also as the large stress concentrations in the storage tank at the weld bead, and larger costs associated with weld bead x-raying and laser welding manufacturing led to concerns (Miller, 2008-2009).

It was agreed upon that a cubic tank structure with the CubeSat interface and rails for the P-POD integration, using 7075-T6 Aluminum and CNC machining was the proper solution considering the time, size, and cost constraints. Also, all components are to be mounted to the tank cap including the thruster assembly, fill and drain valves, and any ports required for pressure or temperature measuring devices. This is necessary due to the minimum thickness required by the threads for mounting these components and the greater wall thickness of the cap over the tank.

After the initial tank geometry and configuration discussion, valve design and integration was discussed. It was decided that a custom valve integrated into the cap and

placed inside the tank would maximize propellant volume as well as minimize valve mass because the tank cap would also act as the valve case. In order to narrow the scope of the project to fit the time and cost constraints, it was decided that an off the shelf miniature solenoid valve should be used with the remaining propulsion system designed around the chosen valve. This would allow for the project focus to remain on a system design and integration without the redesign of each individual component.

From this brainstorming session a cubic prototype tank was designed and modeled using SolidWorks and the mass properties and volume capacity were studied. For this initial prototype a 10 cm by 10 cm by 8.5 cm tank was constructed and COSMOS analysis was used to help determine the dimensions required in order to ensure a desired factor of safety was met concerning both stress and deflections. COSMOS is an analysis tool available through SolidWorks which calculates and displays stresses and strains. It was used to compare different design iterations. After analysis of this initial design, it was shown that the maximum allowable weight of 1 kg for the propulsion module may be exceeded without even taking up the entire tank volume with propellant. From this result it was decided that both a cylindrical and cubic tank would be designed in parallel and compared.

Proposed Designs Comparison and Evaluation

Two prototype assemblies were created, one with a cubic tank and cap and another with a cylindrical tank and cap. Both initial designs consisted of a tank and matching cap with the thrusters mounted to the cap and facing normal to the cap. The valves would be threaded onto stainless steel studs with o-ring seals on the inlet end and the thruster assembly would be threaded into the valve in the outlet side of the valve. The

catalyst bed and nozzle components will be machined separately and then laser welded together to seal the catalyst bed inside. Laser welding is required because of the small size of the components as well as the mass savings of a welded connection over a threaded connection.

High pressure Schrader valves were chosen for use as the drain and fill valves due to their widespread use, reliability and low mass. The Schrader valves used in this design come from a FOX nitrogen filled damper for off-road race vehicles and are rated at 1380 kPa maximum operating pressure (Products, 2009). They also incorporate an o-ring seal which is necessary for proper sealing.

Figure 8. FOX Schrader valve and cap with o-ring rated at 1380 kPa operating.

From the fixed dimensions of the valve and thruster assembly and the constraints described in the CubeSat standard a cubic tank and cap was designed while trying to minimize tank mass and hold an initial pressure of 1100 kPa. The cubic tank assembly is shown in Figure 9 below.

Figure 9. Cubic tank and cap prototype micropropulsion system assembly.

This design has a large internal volume capacity, but due to the large internal radii at the corners in order to reduce stress concentrations has a larger mass than a cylindrical tank. As per the CubeSat standard the mounting flanges of the tank and cap protrudes 6mm from the four faces to allow access to the fasteners. This still allows for a 0.5 mm clearance with the P-POD walls while loaded in the P-POD. This design, while slightly less mass efficient, easily incorporates guide rails and utilizes the extra mass contained in the four vertical corners to satisfy the rail requirements as stated in the CubeSat standard.

A cylindrical tank was designed in order to reduce the dry mass required by the cubic tank while attempting to achieve internal volume capacity at or near that of the cubic tank design. Shown in Figure 10 is the cylindrical tank assembly with integrated guide rails as required by the CubeSat Standard.

Figure 10. Cylindrical tank and cap prototype micropropulsion system assembly.

The cylindrical tank shown above reduces the overall mass of the assembly but at the expense of a large amount of fuel. As can be seen in Figure 10, a large amount of volume is wasted in the open space in the corners. While the wall thickness for this design is drastically reduced when compared to the wall thickness required for the cubic tank, too much propellant volume is lost.

Because of the overall length of the thruster assembly, the cubic and cylindrical tanks shown in Figure 9 and 10 are shorter than expected in order to meet the required maximum length. A second iteration of the cylindrical tank design includes a longer, narrower tank in order to mount the valves along the axis of the tank at the four corners of the assembly allowing for more tank height (Figure 11). This prototype was created in order to investigate if more internal volume can be gained while still achieving the lower mass benefits of having a cylindrical tank design.

Figure 11. Cylindrical tank with vertically mounted thruster prototype Micropropulsion system assembly.

A table was created listing propulsion assembly dry mass and internal volume, along with an estimated overall spacecraft performance to compare the three concepts (Table 7).

Table 7. Tank geometry, mass, volume and corresponding delta V.

Tank Shape and Description	Tank Dimensions (mm)	Volume Capacity (ml)	Assembly Mass (g)	Propellant Mass (g)	Fueled System Mass (kg)	Dry Mass Fraction (%)	Delta V* (m/s)
Cubic Tank and Cap with thruster mounted to cap	100 X 100 X 55	475.00	433.00	522.50	955.50	0.45	419.86

Cylindrical Tank and Cap with thrusters mounted horizontally	97 O.D. X 77.85	443.00	345.00	487.30	832.30	0.41	407.48
Cylindrical Tank and Cap with thrusters mounted vertically	89 O.D. X 80	457.00	369.00	502.70	871.70	0.42	415.32

*Assumes 220 second Isp

As shown from Table 7, the theoretical performance of the cubic tank is the largest of the three designs even though it has the highest unit dry mass fraction. The reason for this is because the cubic tank can hold much more propellant mass than the other two and when evaluated as a 3-unit CubeSat configuration the extra mass of propellant is of more importance than the greater mass of the tank. The cubic tank design is also beneficial from a manufacturability standpoint. The cubic tank shape is more easily manufactured with a CNC mill than the cylindrical tanks because of the undercuts required for the mounting flanges.

Micropropulsion Design Details

Since the cubic tank and cap design produced the largest theoretical performance it was chosen to pursue further with the construction of a prototype. The SolidWorks model was converted to a step file and sent to Maglio Inc. in San Luis Obispo for machining. Figure 12 shows the Aluminum 7075-T6 raw material next to a finished tank and cap.

Figure 12. Tank and cap shown with respective raw materials before machining

To seal the tank and cap interface, a Parker EDPM o-ring was specified and a corresponding channel was machined into the cap. The channel with o-ring on the underside of the cap is shown in Figure 13.

Figure 13. Underside of cap with o-ring installed.

In order to properly align the mounting holes between the tank and cap, four alignment tabs were machined on the underside of the cap. These tabs as shown in Figure

14 in the circle contact the tank walls and align the cap such that the mounting hardware will properly engage all the threads.

Figure 14. Cap alignment tabs shown on the underside of the cap.

To carry propellant from the tank to the valve inlet, stainless steel studs with laser cut holes through the center were used. The studs shown in Figure 15 act as both a mounting interface to the solenoid valve as well as propellant feed lines. An EPDM o-ring was used to seal the cap and is shown in Figure 15.

Figure 15. Stainless steel stud with EPDM o-ring and laser cut through hole.

The stainless steel stud is seated into the underside of the cap with the head flush with the propellant channel as shown below in Figure 16 on the left. The stud threads

through the mounting boss on the cap to seat the o-ring. The external threads protruding from the mounting boss on the top side of the cap shown in Figure 16 on the right are engaged by the solenoid valve.

Figure 16. Stainless steel stud mounted into cap.

The solenoid valve is threaded onto the stud until it seats against the mounting boss as shown below in Figure 17.

Figure 17. Solenoid valve mounted to stud and seated against mounting boss face.

The fill and drain valves thread into the top side on the cap and also utilize an o-ring for sealing.

Figure 18. Fill and drain valve shown installed in the tank cap.

At any stage after the stainless steel stud assembly and cap o-ring installation, the cap may be bolted onto the tank with the appropriate hardware. The assembled propulsion system is shown below in Figure 19.

Figure 19. Assembled Micropropulsion system.

Tension Joint Details and Analysis

As described above, small metric cap screws will be used for the connection between the tank and cap. Based on an initial estimate, twelve M3-0.5 with property class 12.9 screws would be required to join the cap and tank while providing enough compressive load to the sealing o-ring for no leakage of propellant at the joint. This would be achieved by tapping the through holes on the cap and routing the cap screws through the tank flange and into the threaded cap at the twelve points on the tank and cap. The four corners will use M2-0.4 property class 12.9 screws to take some of the load off of the twelve main bolts giving a larger factor of safety for the threaded connection. This will be achieved by using threaded blind holes 4 mm deep into the tank and 2mm diameter through holes on the cap at the four corners. Figure 20 shows a detailed view of the tank and cap fastener orientation.

Figure 20. Details of tank and cap fastener orientation.

A detailed analysis of the joined connection was performed with bolt stresses and joint loads evaluated. Figure 21 on the next page shows the factor of safety for yield of the bolts as well as the factor of safety for the joint connection. This is based on assuming that the twelve M3 bolts will take the entire load due to the pressure in the tank and that they have a yield strength of 1100 MPa as published (Shigley, 2003).

Figure 21. Bolted joint and bolt yield factor of safety as a function of tank storage pressure.

As shown from the figure above the factor of safety for bolt yield and joint separation are approximately 1.4 and 6.0 respectively for an operating pressure of 1100 MPa (Shigley, 2003). Also shown by the figure is the fact that acceptable joint separation and yield factors of safety are met throughout a large range of tank pressures. This

ensures a safe joint even in the event of over pressurization of the tank due to operator error of elevated temperature of the propulsion system.

Propellant Management

Propellant Management Devices or PMDs are used to separate the pressurant gas from the liquid propellant. These devices range from bladders that physically separate the gas and liquid to vanes and screens that use the surface tension of the liquid propellant to separate it from the gas. Also used as a PMD are troughs and traps which use hydrostatics to keep only liquid entering into the thruster. Bladders are almost exclusively used in bi-propellant applications where the fuel oxidizer ratio must be carefully controlled and free of pressurant gas. The penalty of using a bladder is complexity and extra mass. Most monopropellant propulsion systems today use some type of surface tension device utilizing screens, vanes or both to keep only liquid propellant entering the thruster (Svitek, 2008-2009). These types of devices are effective in zero g environments but attention must also be paid while the propulsion system is on the ground during fueling and loading. The propellant management device will be left for future work.

Testing

The design validation plan is summarized in Table 8 on the next page. The tests include proper o-ring and fastener fits and operations, proper valve function, proper P-POD operation, maximum pressure tests, and nozzle function tests. The purpose of the design validation plan is to ensure that the Micropropulsion system will be safe and reliable for hot fire tests to be performed in the future.

Table 8. Micropropulsion test plan.

Test	Description	Requirements
Initial Fit Test	Assemble Propulsion system, check that valves thread onto studs and seat against flange, check o-ring fitment in groove, threaded fasteners engage threads and seat tank and cap together	Visual Inspection
Initial Pressurization	Fill tank with water and pressurize with Nitrogen to 800 kPa	Visual Inspection (No visible leaks)
Initial Valve function check	Pressurize to 800 kPa with water and Nitrogen and run valve test	Verify valve function at 1Hz pulses under pressure
P-POD integration	Load Propulsion system into P-POD, check for binding, interferences (unpressurized)	Visual Inspection
Tank Leak/Burst Test	Incrementally increase pressure until tank leakage is detected or until mechanical failure	Record Pressure at Visually detected leakage or failure
Nozzle Function Test	Perform Cold Gas thrust measurements with Nitrogen starting at 800 kPa	Compare actual thrust to theoretical to verify nozzle function

After machining was completed the micro-propulsion system was assembled and checked for proper fitment and operation. Next the valves were tested with incremental increases in pressure until they failed to operate. The Micropropulsion system was then proof tested to the design operating pressure of 1100 kPa. Following these initial tests the system was tested to failure. This test identified the structural design factor of safety as

well as the failure mode. By identifying the failure mode, improvements can be tailored to it in order to increase the structural factor of safety.

A hot fire test will be conducted in the future but is out of the scope of this paper. The main goals of the hot fire test are to find the minimum tank pressure for proper thruster operation and to verify catalyst structural integrity after many thrust cycles. Qualification testing will also be conducted after initial hot fire tests which include random vibration testing as well as long term propellant exposure testing and temperature cycle testing.

Test Results

The initial fit test was successful for all components. All o-rings seated properly and all mechanical interfaces acted as designed. Some valve stud trimming or shimming may be necessary for the valves to seat against the mounting boss with the proper orientation. Shown in Figure 22 is the correct orientation of the four thruster valves.

Figure 22. Assembled Micropropulsion system.

Each valve was designed to seat against the mounting boss parallel with the edge of the tank forming a square pattern as it was threaded onto the valve mounting stud. However, during valve tightening a couple valves did not seat against the boss until they were rotated past the position parallel to the tank edge. This causes the center of gravity to move outward from the center of the tank towards the misaligned valve which is not desirable. Also for a more extreme case the valve can be rotated past the tank edge, “overhanging” the cap which can cause interference with the P-POD. This is most likely caused by the allowable length tolerance in the valve stud and can be solved by some light machining on the studs to ensure consistent lengths among batches or by adding small shims between the valve and boss.

Figure 23. Initial fit test.

After the initial fit test was completed, the initial tank pressurization and valve function tests were conducted. The tank was filled with approximately 250 ml of water and pressurized to 800 kPa using Nitrogen. No leaks were detected. One valve was cycled with six, 1 Hz pulses at a time until the pressure was bled down to 100 kPa. At

825 kPa the valve did not open for the first pulse, but did function properly after the initial pulse. This is evidence that 825 kPa may be the maximum operating pressure even though the valve is rated at 760 kPa operating. The test setup is shown in the figure on the next page. Alex Diaz of Stellar Exploration built the circuit to control the valves for testing.

Figure 24. Valve Function test layout and equipment.

The next test performed was the system burst test. The goal of this test was to determine the failure mechanism of either the structure or the seals. It was decided that the valves would be left off of the system for this test. The stainless steel studs with the laser bored hole were replaced with a similar stud without the bored hole as shown in Figure 25 on the next page.

Figure 25. Assembled Micropropulsion system with non-bored stainless studs.

In addition, identical o-rings were used on the stud without the bored hole. The test was carried out at Stellar Exploration. The test area had a small cinderblock bunker which would be used to house the propulsion system during testing shown in Figure 26.

Figure 26. Cinderblock bunker used during system burst testing.

A wooden structure with a polycarbonate viewing section constructed as a blast shield was another safety feature used during the test and is shown in Figure 27.

Figure 27. Blast shield used during system burst testing.

The Nitrogen source bottle was kept away from the test bunker with the use of 6 m sections of stainless steel braided hoses. One hose was used for the pressure source while the other connected to the remote digital gauge mounted to the blast shield shown in Figure 28 on the next page.

Figure 28. Digital Pressure gauge connected to propulsion system.

In order to connect the Nitrogen source to the propulsion system as well as the digital gauge feed line, two adapters were made. Since both lines were already configured for use with high pressure quick connect fittings which use $\frac{1}{4}$ " NPT threads the adapters were machined to couple these fittings to the Schrader drain and fill valves on the propulsion system. The adapters along with the quick connect couplers are shown in Figure 29 on the next page mounted to the propulsion system.

Figure 29. Propulsion system with adapters and quick connect fittings.

After the adapters were installed, the Micropropulsion system was connected to the Nitrogen source bottle and digital pressure gauge and pressurization began.

Figure 30. Micropropulsion system connected and ready for pressurization.

The pressure inside the propulsion system was increased slowly until a drop in pressure was observed. This happened at 2310 kPa with the main o-ring failure. The cap was lifted off of the tank surface just enough for the internal pressure to push the o-ring out of the groove and just outside the mounting flange as shown in Figure 31 below.

Figure 31. O-ring shown protruding out of the tank and cap.

After the o-ring was pushed out, the drop in pressure during this time caused the tank cap to re-seat against the tank mounting flange and only a slow leak was observed. The pressure dropped from 2310 kPa to 800 kPa in 30 minutes after the o-ring failure. As shown in the figure below, the o-ring did not tear, however it did yield at the location that the tank and cap re-seated against the o-ring. Also shown in the figure below, is the damage to the bolts.

Figure 32. Micropropulsion bolts, o-ring, tank and cap after first burst test.

Although difficult to see in the figure, every bolt in the system was bent with the longest bolts located in the center of the cap on each side showing the most damage. In addition, small amounts of aluminum were left on the bolt threads from the tank. A second and third test was conducted with the same failure at pressures of 2206 kPa and 2027 kPa respectively. The location of the o-ring failure from the second test was marked on the cap in order to determine if local yielding in this location could have caused the failure and the test was run again. The third test had the same failure in the same location with respect to the cap. This may imply some kind of damage to the cap from the first test may be contributing to the failures of the second and third test.

The tank burst test was a success because it verified the design factor of safety as well as determined the failure mode. The measured factors of safety for these tests are 1.90, 1.82 and 1.68 respectively. This factor of safety is derived from the ratio of measured maximum pressure over the design maximum pressure of 1200kPa. The failure mode as determined by this test is a desirable one since it is not an actual “burst” but a simple o-ring failure which allows the pressurant gas to escape slowly. This is known as a

leak-before-burst failure mode which is a big advantage over a burst failure mode from a range safety standpoint. Although the Micropropulsion system performed successfully during this test, it exposed an error in the analysis used for the tension joint. From inspection of Figure 21 on page 59 the expected bolt yielding factor of safety at 2300 kPa is approximately 1.4 and the factor of safety for joint separation is approximately 2.75. The bolts yielded at this pressure and the joint separated allowing the o-ring to be pushed out. Possible sources of error may be the assumed spring rate of the mating surfaces, an incorrectly assumed bolt strength, or incorrect bolt preload during assembly.

The cold gas thrust test was conducted in order to verify proper nozzle function. The thrust and therefore specific impulse of the system will be measured and will indicate whether or not the nozzle is functioning properly. The specific impulse of this system would approach zero if the flow through the nozzle was not choked and therefore the gas was not supersonic at the exit plane. With a reasonable calculated specific impulse from this test it will be assumed that the flow is choked and the nozzle is functioning properly. The test setup is shown on the next page in Figure 33.

Figure 33. Cold gas thrust measurement test setup.

The test setup consists of the Micropropulsion system mounted to a bent steel rod acting as a pendulum. The end of the steel rod is coupled to an encoder to accurately measure the angular displacement. The encoder samples 10,000 points per rotation, allowing for small angular displacements to be measured accurately.

After a few trial runs a discrepancy was noticed between the portable pressure gauge used to measure the tank storage pressure and the regulator on the gas source. It was decided to mount a pressure gauge to the Micropropulsion system's Schrader valve so storage pressure could be read directly after the tank had been filled. Figure 34 shows the test setup during the first test.

Figure 34. Test setup shown after first test.

All four thrusters were fired at the same time with 2 Hz pulses during this test. Angular displacement was measured during the thruster firing. Table 9 on the next page contains the results from the three tests including the starting tank storage pressure, the encoder output for the beginning and end of the thruster firing as well as the angular displacement for the thruster firing. The encoder has 65536 points per revolution. Once the encoder passes point 65536 it resets to zero. As shown in Table 9, the encoder start position varied slightly for each test due to the sensitivity of the encoder.

Table 9. Cold gas test results for three tests with three 2 Hz pulses per test.

	Tank Pressure (psig)	Encoder Output		Difference	Angular Displacement (rad)
Test 1		Start Position	End Position		
Pulse 1	95	65535	43	44	0.028
Pulse 2	73	65490	65524	34	0.021
Pulse 3	59	65500	65525	25	0.016
Test 2					
Pulse 1	91	2	49	47	0.030
Pulse 2	79	65489	65522	33	0.021
Pulse 3	60	65501	65525	24	0.015
Test 3					
Pulse 1	91	65511	22	46	0.029
Pulse 2	79	65488	65521	33	0.021
Pulse 3	60	65501	65527	26	0.016

As shown in the table, the angular displacements are small due to the low thrust to weight ratio of the cold gas system.

The nozzles used in this test have large expansion ratios defined as the throat area over the exit plane area, which is common in spacecraft. For atmospheric conditions a nozzle with an optimum expansion ratio is one that expands the gas to atmospheric pressure at the exit plane of the nozzle. For this case, that corresponds to an expansion

ratio on the order of 2 or 3 (Brown, 1996). Since the absolute pressure in space is zero, spacecraft generally have nozzles with large expansion ratios unless mass or volume constraints do not permit it. The reason for this is that the specific impulse increases as the exit pressure decreases for space applications. The nozzles used in this test have a throat diameter of 1mm and an expansion ratio of 43. It was decided to test the system with these nozzle expansion ratios because it is representative of what would actually be used in space. The only difference between these nozzles and those used with Hydrazine is a smaller throat diameter and the absence of the catalyst. Because of the over expansion of the gas by the nozzle, flow separation at the walls of the nozzle will occur decreasing the performance. However, the measured specific impulse will provide insight into the proper function of the nozzle. Choked flow conditions can be verified by a reasonable specific impulse of 10-50 s.

Figure 35. Nozzle used for cold gas test.

In order to calculate the specific impulse attained during testing, both the thrust and mass flow rate must be measured. The mass flow rate was calculated by measuring the pressure drop in the tank and calculating the change in mass of the contained gas using the ideal gas equation. Next, the change in mass was divided by the amount of time the valve was open (0.25 s), giving the mass flow rate. The thrust was calculated by equating the sum of the moments caused by impulse forces around a fixed point to the change in angular momentum of the system about the same point. The moment of inertia of the propulsion system with pressure gauge and adapter about the center of gravity was calculated using SolidWorks. The parallel axis theorem was then applied to get the moment of inertia about the point of rotation, which was the chosen fixed point for the analysis. Next the moment of inertia for the bent steel rod was calculated about the same point. A friction force was calculated using a coefficient for the friction between steel and wood surfaces (Roy, 2009). These values along with the encoder outputs were used to calculate the total thrust produced by the system. The results of the cold gas tests are shown in Table 10.

Table 10. Cold gas test calculated results.

	Tank Pressure (psig)	Thrust (N)	Mass Flow Rate (kg/s)	Isp (s)	Delta V (m/s)
Test 1					
Pulse 1	95	0.41	0.0034	12	0.2
Pulse 2	73	0.32	0.0022	15	
Pulse 3	59	0.24	0.0020	12	
Test 2					
Pulse 1	91	0.43	0.0019	24	0.2
Pulse 2	79	0.31	0.0030	11	
Pulse 3	60	0.24	0.0020	12	
Test 3					
Pulse 1	91	0.43	0.0019	23	0.2
Pulse 2	79	0.31	0.0030	11	
Pulse 3	60	0.25	0.0020	13	

As shown in Table 10 the thrust produced for each pulse during the three tests are all within 4-5% of each other and . However, the pressure drop from pulse 1 to pulse 2 during the first test is much larger than the pressure drop observed in tests 2 and 3. This results in a much larger mass flow rate for the pulse and thus a much smaller specific impulse when compared to tests 2 and 3. A possible explanation for this is simply human error in reading the pressure at pulse 2. A maximum specific impulse of 24 s was measured at the first pulse of test 2. This specific impulse is approximately half of common specific impulse for cold gas systems using air. This is probably due to the overexpansion of the gas through the nozzle. A specific impulse of 24 seconds is a reasonable result with a straight cut over expanded nozzle. This result gives high confidence of choked flow through the nozzle.

The ΔV of the system calculated for all three tests was 0.2 m/s. The ΔV was calculated using an average specific impulse of the three pulses and assumed a 3-unit CubeSat.

Future Work

For the current system, concern was raised over the cantilevered mounting of the solenoid valves. Under launch loads, launch vehicle vibrations will cause loading on the valve at the mounting interface. The concern was that these loads could cause valve leakage around the mounting stud from deflections or even fracture at the thread root of the valve. Another concern raised was that while in the P-POD under launch loads, the thrusters would be loaded since they would be in contact with the P-POD door. Because the thrusters are designed for minimum mass, this loading may produce significant damage. A solution to both of these problems could be a standoff bracket that simultaneously constrains the valve and takes the load off of the nozzles at the P-POD door interface. An initial design was created and is shown in Figure 36. This design consists of a CNC machined aluminum 7075-T6 bracket that interfaces with two bolts on the cap as well as the valve. The bracket also has a machined pad that will contact the P-POD door taking the load off of the thrusters. In the current form each bracket has a mass of 6 g. However the design will not be finalized until the combustion chamber dimensions are verified by a hot fire test.

Figure 36. Thruster standoff bracket design.

In order to increase the delta V of this propulsion system from a theoretical maximum of 420 m/s, the volume of propellant held by the tank must be increased. A large amount of the allowable volume for the propulsion system is used up by the thruster valves and long combustion chambers and nozzles. Bud Frazee proposed machining the thruster nozzle into the tank cap itself and also integrating the thruster valves into the tank cap in order to use up more of the allowable volume for propellant (Frazee, 2008-2009). This has the potential to greatly increase the propellant volume and therefore increase overall performance. However, care must be taken to keep the overall mass of the system below 1 kg.

A thrust control system will need to be developed in order to use the micro propulsion system. This system must be low mass and incorporate a navigation device to provide specific orientation of the spacecraft.

Much more testing needs to be conducted before a flight is planned. The system must be flight qualified which means passing vibration and leak testing as well as long term material compatibility. A high frequency vibration environment will be encountered by the propulsion system caused by the solid rocket motors during launch. No significant oscillations are expected during operation. In addition many hot fire tests will need to be conducted to test the thermal and structural integrity of the catalyst and nozzle. Additional thermal tests will include combustion chamber, valve head, and nozzle wall temperatures as a function of time. This will help determine a recommended pulse frequency to limit part damage due to thermal stresses.

Conclusions

The development of the Micropropulsion system for CubeSats is a difficult task requiring balancing multiple variables at a time while meeting the constraints imposed by the CubeSat standard. Each variable in the design of the micropropulsion system is coupled with at least one more variable. This makes changes to the system very difficult due to the undesirable effects a variable may have on another. It also makes the development of the micropropulsion system an iterative type of project because an “optimum” solution for one variable may drive another to an undesirable value. In order to look at the big picture concerning the design of this system, a flow chart was created to show the design process and how each variable is coupled to the assembly. This chart is shown in Figure 36. With the CubeSat Standard, the general shape, mass and dimensional constraints helped to decrease the number of open ended variables. Also, since a custom valve design was not part of this project, the valves became the uncontrolled fixed variable. This meant that the rest of the propulsion system was designed around the valve.

Figure 37. Design Flow chart for Micro-propulsion system.

This flow chart may be used during future iterations of the Micropropulsion system. This flow chart also illustrates how small changes to certain variable can affect the entire system.

The use of as many commercial-off-the-shelf components as possible is very important for developing a system like this rapidly. Also commercial-off-the-shelf

components can provide a starting point for a design and allow for a big picture of the system to be developed early on. Significant cost savings are another result of using commercial-off-the-shelf components when available as opposed to a custom part. These components along with keeping the system as simple as possible at the beginning design stages helps insure project success. The design philosophy used throughout this project was simplification, the use of commercial off the shelf components, and rapid prototyping to support component and system testing rapidly. Many empirical relationships were used to size components with the intention of rigorous testing programs for component and system verification.

As discussed in the test results section of this paper, the leak-before-burst failure mechanism is very desirable because it is not a catastrophic failure. This failure mechanism poses much less threat to the range safety personnel, launch vehicle, and primary payload than a tank burst failure.

While much work still needs to be completed, the Micropropulsion system development up to this point is a great step towards a viable high performance propulsion system for CubeSats. A theoretical ΔV of 419 m/s is approximately 84% of the performance goal. This is encouraging because all required constraints were met as per the CubeSat Standard with the overall performance possibly being within 20% of the performance goal. The design goal of $\Delta V = 500$ m/s may be ambitious currently, but with new innovations with respect to the thruster valves it may be an attainable goal in the near future.

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Appendix A. Mass Fraction vs. I_{sp} Required Hand Calculation

Constants

$$W_f = 3 \text{ kg}$$

$$\Delta V = 500 \text{ m/s}$$

Eqs

$$\Delta V = g I_{sp} \left(\ln \left(\frac{W_i}{W_f} \right) \right) \quad (5)$$

$$W_p = W_f \left[\exp \left(\frac{\Delta V}{g I_{sp}} \right) - 1 \right] \quad (6)$$

For $I_{sp} = 175$

$$W_p = 3 \text{ kg} \left[e^{\frac{(500 \text{ m/s})}{9.81 \text{ m/s}^2 (175 \text{ s})}} - 1 \right]$$

$$\underline{W_p = 1.01 \text{ kg}} \quad (\text{not possible}) \quad W_p \text{ must be } < 1 \text{ kg}$$

I_{sp}	W_p
200	0.87
225	0.76
250	0.66
275	0.61
300	0.56
325	0.51
350	0.47
375	0.44

Mass fraction:

$$\frac{W_p}{W_f}$$

Appendix B. Performance vs. Dry Mass Fraction data for Figure 3.

Assumes 3 kg (fueled) spacecraft							
Isp = 150		Isp = 180		Isp = 210		Isp = 240	
Propulsion Dry Mass fraction	Delta V (m/s)	Propulsion Dry Mass fraction	Delta V (m/s)	Propulsion Dry Mass fraction	Delta V (m/s)	Propulsion Dry Mass fraction	Delta V (m/s)
0.67	589	0.67	707	0.67	825	0.67	943
0.68	568	0.68	681	0.68	795	0.68	908
0.69	546	0.69	655	0.69	764	0.69	874
0.70	525	0.70	630	0.70	735	0.70	840
0.71	504	0.71	605	0.71	706	0.71	806
0.72	483	0.72	580	0.72	677	0.72	773
0.73	463	0.73	556	0.73	648	0.73	741
0.74	443	0.74	532	0.74	620	0.74	709
0.75	423	0.75	508	0.75	593	0.75	677
0.76	404	0.76	485	0.76	565	0.76	646
0.77	385	0.77	462	0.77	538	0.77	615
0.78	366	0.78	439	0.78	512	0.78	585
0.79	347	0.79	416	0.79	486	0.79	555
0.80	328	0.80	394	0.80	460	0.80	525
0.81	310	0.81	372	0.81	434	0.81	496
0.82	292	0.82	350	0.82	409	0.82	467
0.83	274	0.83	329	0.83	384	0.83	439
0.84	257	0.84	308	0.84	359	0.84	410
0.85	239	0.85	287	0.85	335	0.85	383
0.86	222	0.86	266	0.86	311	0.86	355
0.87	205	0.87	246	0.87	287	0.87	328
0.88	188	0.88	226	0.88	263	0.88	301
0.89	171	0.89	206	0.89	240	0.89	274
0.90	155	0.90	186	0.90	217	0.90	248
0.91	139	0.91	167	0.91	194	0.91	222
0.92	123	0.92	147	0.92	172	0.92	196
0.93	107	0.93	128	0.93	150	0.93	171
0.94	91	0.94	109	0.94	127	0.94	146
0.95	75	0.95	91	0.95	106	0.95	121
0.96	60	0.96	72	0.96	84	0.96	96
0.97	45	0.97	54	0.97	63	0.97	72
0.98	30	0.98	36	0.98	42	0.98	48
0.99	15	0.99	18	0.99	21	0.99	24
1.00	0	1.00	0	1.00	0	1.00	0

Appendix C. Graham Edney catalyst design sketches.

Appendix D. Nozzle Area Ratio Calculation and Derivation.

Nozzle Area ratio calculation / Derivation

1) Continuity equation

$$\dot{m}_1 = \dot{m}_2$$

$$\rho_1 V_1 A_1 = \rho_2 V_2 A_2$$

2) Acoustic velocity

$$a = \sqrt{\gamma R T}$$

Mach number:

$$M = \frac{V}{a} = \frac{V}{\sqrt{\gamma R T}}$$

arranging for V :

$$V = M \sqrt{\gamma R T}$$

Assumptions

(steady flow)
one-dimensional flow
Ideal gas
Isentropic process

Solve for area ratio using continuity equation

$$\frac{A_2}{A_1} = \frac{\rho_1 V_1}{\rho_2 V_2}$$

substitute V

$$\frac{A_2}{A_1} = \frac{\rho_1 M_1 \sqrt{\gamma R T_1}}{\rho_2 M_2 \sqrt{\gamma R T_2}}$$

$$\frac{A_2}{A_1} = \frac{M_1 \sqrt{\rho_1^2 T_1}}{M_2 \sqrt{\rho_2^2 T_2}}$$

For Isentropic flow the following relationships apply

$$\frac{T_2}{T_1} = \left(\frac{p_2}{p_1} \right)^{\frac{\gamma-1}{\gamma}}$$

$$\frac{p_2}{p_1} = \left(\frac{T_2}{T_1} \right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{p_2}{p_1} = \left(\frac{\rho_2}{\rho_1} \right)^{\gamma}$$

Assumptions

Isentropic flow
one dimensional flow
steady flow
Ideal gas

Stagnation conditions apply at combustion chamber

$$\frac{T_0}{T} = 1 + \frac{\gamma-1}{2} M^2$$

Combining stagnation conditions with Isentropic flow equations:

$$\frac{p_0}{p} = \left(1 + \frac{\gamma-1}{2} M^2 \right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{\rho_0}{\rho} = \left(1 + \frac{\gamma-1}{2} M^2 \right)^{\frac{1}{\gamma-1}}$$

substituting into area ratio equation:

$$\frac{A_2}{A_1} = \frac{M_1}{M_2} \sqrt{\frac{\left(1 + \frac{\gamma-1}{2} M_2^2 \right) \left(\frac{\gamma+1}{\gamma-1} \right)}{\left(1 + \frac{\gamma-1}{2} M_1^2 \right)}}$$

If location 1 is assumed to be the throat the equation becomes:

$$\frac{A_2}{A_t} = \frac{1}{M_2} \sqrt{\left[\left(\frac{2}{\gamma+1} \right) \left(1 + \frac{\gamma-1}{2} M_2^2 \right) \right]^{\frac{\gamma+1}{\gamma-1}}}$$

This equation can be used to relate the throat conditions to the combustion chamber inlet (contraction ratio) or nozzle exit (Expansion Ratio)

Appendix E. I_{sp} vs. Expansion Ratio data for Figure 6.

Chamber Pressure Pa	550000							Assumes 30degree straight cone	
gamma	1.238	Exit Pressure Pa	Exit Velocity (Ma)	Nozzle Expansion (Area)Ratio	Isp (seconds)	Slope (disp)/(dExp. Ratio)	Nozzle Diameter (mm)	Length of Nozzle (mm)	Isp Increase (%)
Tc=2930K	2930	55000.00	2.16	2	227.59		2.69	5.02	
M=20 kg/kmol	20	33875.00	2.44	3	237.36	11.98	3.15	5.88	4.292
c*=1672.3m/s	1672.3	22250.00	2.68	4	244.98	7.63	3.64	6.79	3.212
lambda (nozzle efficiency)	0.9	16200.00	2.85	5	250.26	5.28	4.07	7.59	2.152
Pa=0	0	12545.00	3.00	6	254.23	3.97	4.46	8.32	1.586
Throat Diameter (mm)	1.82	10125.00	3.12	7	257.37	3.14	4.81	8.98	1.238
		8475.00	3.22	8	259.87	2.58	5.14	9.58	0.970
		7200.00	3.31	9	262.07	2.18	5.45	10.17	0.846
		6200.00	3.39	10	264.01	1.86	5.76	10.75	0.742
		5400.00	3.47	11	265.75	1.61	6.06	11.31	0.658
		4850.00	3.53	12	267.06	1.43	6.31	11.77	0.494
		4350.00	3.59	13	268.36	1.28	6.57	12.26	0.486
		3925.00	3.65	14	269.56	1.15	6.83	12.74	0.446
		3600.00	3.70	15	270.54	1.05	7.05	13.16	0.365
		3300.00	3.75	16	271.51	0.96	7.28	13.59	0.359
		3000.00	3.81	17	272.56	0.88	7.55	14.09	0.384
		2825.00	3.84	18	273.20	0.81	7.72	14.41	0.237
		2650.00	3.88	19	273.88	0.76	7.91	14.76	0.248
		2450.00	3.92	20	274.70	0.71	8.15	15.20	0.299
		2300.00	3.96	21	275.35	0.66	8.35	15.57	0.236
		2150.00	4.00	22	276.03	0.62	8.56	15.97	0.248
		2000.00	4.04	23	276.75	0.58	8.80	16.42	0.261
		1875.00	4.08	25	277.39	0.54	9.02	16.83	0.229
		1750.00	4.12	26	278.05	0.51	9.26	17.27	0.241
		1625.00	4.16	27	278.76	0.47	9.52	17.77	0.254
		1500.00	4.21	29	279.51	0.44	9.82	18.32	0.269
		1375.00	4.26	31	280.31	0.40	10.15	18.93	0.286
		1275.00	4.31	33	280.99	0.37	10.44	19.49	0.243
		1175.00	4.36	35	281.71	0.34	10.77	20.11	0.258

Appendix F. Micropropulsion system data sheets and drawings.

ASCO miniature solenoid valve

2/2
3/2
Series
AL

General Service

2 and 3-Way Miniature Solenoid Valves

10-32 UNF Porting, In-line Porting

ASCO SCIENTIFIC®

The Series AL Valves are suitable for a wide range of OEM applications where small size, low power and long life are a must.

- Cycle life in the hundreds of millions
- Corrosion resistant materials of construction
- In-line porting for use with standard metal or plastic #10-32 UNF fittings
- Magnetic latching construction available to maintain valve position with loss of power, eliminate coil heat rise, and extend battery life

Construction

Valve Parts In Contact with Fluids	
Body	POM or 300 Series Stainless Steel
Disc	FKM
Gaskets	FKM
Bobble/CoreTube	PBT
Core and Plugnut	400 Series Stainless Steel
Springs	300 Series Stainless Steel

Electrical

Standard Voltages	6, 12, 24 VDC ± 10%, -5% 115 VAC (with rectifier in lead wires)
Power Consumption	0.65-2.0 Watts (10 watts for latching version)
Duty Cycle Rating	Continuous (intermittent for latching version)
Coil Insulation	266°F (130°C)
Electrical Connection	26 gage lead wire

Valve

Response Time	~5 ms at rated voltage (2 watt coil) ~12 ms at rated voltage (0.65 watt coil)
Internal Volume	2-way NC = 330 µL, 2-way NO = 310 µL, 3-way = 370 µL
Mounting Bracket	Optional mounting clip, pin 5188-02 (see drawing)
Options	• Oxygen service construction available • Lubricant free construction available
Vacuum Rating	29" Hg

Alternative Constructions

Many alternative constructions are available and include a variety of voltages, electrical connectors, and materials of construction. ASCO Scientific can also custom design a valve for your specific application.

Contact your local ASCO sales office for more information.

Temperature Range:

Ambient & Media:

32°F to 77°F (0°C to 25°C) continuous duty
up to 104°F (40°C) intermittent duty

Approvals:

Meets applicable CE directives

Specifications

Ports	Orifice Size (in.)	Cv Flow Factor	Maximum Pressure (psig)	Catalog Number	Watt Rating @ 24°C	Weight (oz.)
2-WAY NORMALLY CLOSED (Closed when de-energized)						
#10-32 UNF	0.025	0.015	110	AL11xx	0.65	1.3
#10-32 UNF	0.065	0.038	±0	AL21xx	0.65	1.3
#10-32 UNF	0.065	0.038	100	AL31xx	2.0	1.3
#10-32 UNF	0.090	0.07	30	AL41xx	2.0	1.3
2-WAY NORMALLY OPEN (Open when de-energized)						
#10-32 UNF	0.025	0.013	110	AL12xx	0.65	1.3
#10-32 UNF	0.048	0.033	±0	AL22xx	0.65	1.3
#10-32 UNF	0.048	0.033	100	AL32xx	2.0	1.3
#10-32 UNF	0.078	0.06	30	AL42xx	2.0	1.3
3-WAY UNIVERSAL OPERATION (Pressure at any port)						
#10-32 UNF	0.025/0.025	0.015/0.013	110	AL13xx	0.65	1.3
#10-32 UNF	0.065/0.048	0.038/0.033	±0	AL23xx	0.65	1.3
#10-32 UNF	0.065/0.048	0.038/0.033	100	AL33xx	2.0	1.3
#10-32 UNF	0.090/0.078	0.07/0.06	30	AL43xx	2.0	1.3
2-WAY LATCHING						
#10-32 UNF	0.025	0.015	110	AL11xxL	10*	1.8
#10-32 UNF	0.065	0.038	100	AL31xxL	10*	1.8
#10-32 UNF	0.090	0.07	30	AL41xxL	10*	1.8
3-WAY LATCHING						
#10-32 UNF	0.025/0.025	0.015/0.013	110	AL13xxL	10*	1.8
#10-32 UNF	0.065/0.048	0.038/0.033	100	AL33xxL	10*	1.8
#10-32 UNF	0.090/0.078	0.07/0.06	30	AL43xxL	10*	1.8

Notes

*xx Denotes place in catalog number for voltage, three characters may be used when required.
Latching valves are designed for intermittent duty only. Wattage rating applies to 20 – 30 ms duration required to actuate valve. Once switched no additional power is required to hold the valve in its position.

Catalog Number Description and Options

AL33	xx	L	O	S
Base Catalog Number	Voltage - 06 (DC) - 12 (DC) - 24 (DC) - 115 (AC)	Latching Suffix	Options Suffix A = AC service (rectifier in lead wire) O = No Lubricant K = Oxygen Service Construction PBT Valve Body, FKM Seals, PFPE Lubricant	Optional Body Material S = 300 Series Stainless Steel

To Construct Catalog Number

- Select base catalog number
- Insert voltage into the 5th, 6th, (and 7th when required), digits denoted by "xx"
- Add suffix for optional features to end of base catalog number

Examples

AL1124 = 2-way normally closed valve with 0.025" orifice, 110 psig max pressure rating, and 24 VDC coil rated at 0.65 watts.
AL32115A = 2-way normally open valve with 0.048" orifice, 100 psig max pressure rating, and 115 VAC coil with rectifier.
AL4306LK = 3-way latching valve with 0.090" and 0.078" orifices, 30 psig max pressure rating, 6 VDC coil and suitable for oxygen service.

2/2
3/2
Series
AL

ASCO SCIENTIFIC®

Dimensions 2 and 3-Way Standard Solenoid: Inches [mm]

Dimensions 2 and 3-Way Latching Solenoid: Inches [mm]

Vented Cap Screw Drawing

Parker EDPM O-Ring Data Sheets

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Parker O-Ring Handbook

COMPOUND COMPATIBILITY RATING
 1 - Satisfactory
 2 - Fair (usually OK for static seal)
 3 - Doubtful (sometimes OK for static seal)
 4 - Unsatisfactory
 x - Insufficient Data

	Recommended	Nitrile NBR	Hydrogenated Nitrile HNBR	Ethylene Propylene EPDM	Fluorocarbon FKM	Hilufur FKM	Perfluoroelastomer FRKM	Aflas (TFE/Propylene) FEP/FP	Neoprene/Chloroprene CR	Styrene-Butadiene SBR	Polyacrylate ACM	Polyurethane AU, EU	Butyl IIR	Butadiene BR	Isoprene IR	Natural Rubber NR	Hypalon CSM	Fluorosilicone FVMQ	Silicone MQ, VMQ, PVMQ
Hexamethyldisilazane	V8545-75	X	X	X	X	1	1	X	X	X	X	X	X	X	X	X	X	X	X
Hexamethylene (Cyclohexane)	N0674-70	1	1	4	1	1	1	X	2	4	1	1	4	4	4	4	2	1	2
Hexamethylene Diammonium Adipate	V1164-75	2	2	4	1	1	1	X	4	4	4	3	4	4	4	4	4	2	X
Hexamethylenediamine	E0540-80	3	3	1	3	2	2	X	1	1	4	4	1	1	1	1	1	1	2
Hexamethylenetetramine	E0540-80	3	3	1	3	2	2	X	1	1	4	4	1	1	1	1	1	1	2
Hexane or n-Hexane	N0674-70	1	1	4	1	1	1	X	2	4	1	2	4	4	4	4	2	3	4
Hexene-1 or n-Hexene-1	V1164-75	2	2	4	1	1	1	X	2	4	1	2	4	4	4	4	2	4	4
Hexone (Methyl Isobutyl Ketone)	E0540-80	3	3	1	3	1	1	X	1	1	4	4	1	1	1	1	1	1	2
Hexyl Acetate	N0674-70	1	1	4	1	1	1	X	2	4	1	1	4	4	4	4	2	1	2
Hexyl Alcohol	N0674-70	1	1	3	1	1	1	X	2	1	4	4	3	1	1	1	1	2	2
Hexylene Glycol	E0540-80	3	3	1	3	1	1	X	1	1	4	4	1	1	1	1	1	1	2
Hexyresorcinol	V1164-75	2	2	4	1	1	1	X	4	4	4	3	4	4	4	4	4	2	X
HFC-245fa	C0873-70	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
High Viscosity Lubricant, H2	N0674-70	1	1	1	1	1	1	X	2	1	4	4	1	2	X	X	X	2	1
High Viscosity Lubricant, U4	N0674-70	1	1	1	1	1	1	X	2	1	4	4	1	2	X	X	X	2	1
HiLo MS #1	E0540-80	4	4	1	4	1	1	X	4	4	4	4	2	4	4	4	4	3	3
Houghto-Safe 1010 phosphate ester	E0540-80	4	4	1	1	1	1	X	4	4	4	X	1	4	4	4	4	2	3
Houghto-Safe 1055 phosphate ester	E0540-80	4	4	1	1	1	1	X	4	4	4	X	1	4	4	4	4	2	3
Houghto-Safe 1120 phosphate ester	V1164-75	4	4	2	1	1	1	X	4	4	4	1	4	4	4	4	4	2	3
Houghto-Safe 271 (Water & Glycol Base)	N0674-70	1	1	1	2	1	1	X	2	1	4	4	2	X	X	X	X	2	2
Houghto-Safe 416 & 500 Series	N0674-70	1	1	1	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Houghto-Safe 5040 (Water/Oil emulsion)	N0674-70	1	1	4	1	1	1	X	2	4	4	4	4	4	4	4	4	2	3
Houghto-Safe 620 Water/Glycol	N0674-70	1	1	1	2	1	1	X	2	1	4	4	2	X	X	X	X	2	2
Hydraulic Oil (Petroleum Base, Industrial)	N0674-70	1	1	4	1	1	1	X	2	4	1	1	4	4	4	4	2	1	2
Hydraulic Oils (Synthetic Base)	V1164-75	2	2	4	1	1	1	X	4	4	4	3	4	4	4	4	4	2	X
Hydrazine	E0540-80	2	2	1	4	1	1	X	2	2	X	4	1	X	X	1	2	4	2
Hydrazine (Anhydrous)	E0540-80	4	4	2	4	1	1	2	2	1	4	4	2	4	4	4	2	4	X
Hydrazine Dihydrochloride	E0540-80	3	3	1	3	1	1	X	1	1	4	4	1	1	1	1	1	1	2
Hydrazine Hydrate	E0540-80	3	3	1	3	1	1	X	1	1	4	4	1	1	1	1	1	1	2
Hydroiodic Acid	V1164-75	2	2	4	1	1	1	X	4	4	4	3	4	4	4	4	4	2	X
Hydroisobutyl Alcohol	V3819-75	X	X	X	X	1	1	X	X	X	X	X	X	X	X	X	X	X	X
Hydrobromic Acid	E0540-80	4	4	1	1	1	1	X	4	4	4	4	1	4	1	1	1	3	4
Hydrobromic Acid 40%	E0540-80	4	4	1	1	1	1	X	2	4	4	4	1	4	1	1	1	3	4
Hydrocarbons, Saturated	N0674-70	1	1	4	1	1	1	X	2	4	1	2	4	4	4	4	3	1	4
Hydrochloric Acid (cold) 37%	V1164-75	4	X	3	1	1	1	1	4	X	X	X	X	X	X	X	X	X	X
Hydrochloric Acid (hot) 37%	V1164-75	4	X	3	1	1	1	X	4	4	3	4	4	4	4	4	4	2	X
Hydrochloric Acid, 3 Molar to 158°F	V1164-75	2	2	1	1	1	1	X	2	3	3	4	1	X	X	3	1	3	4
Hydrochloric Acid, Concentrated Room Temp.	V0834-70	2	2	2	1	1	1	X	X	X	X	X	X	X	X	X	X	X	X
Hydrochloric Acid, Concentrated to 158°F	V1164-75	4	4	4	1	1	1	X	4	4	4	4	4	X	X	4	X	4	4
Hydrocyanic Acid	E0540-80	2	2	1	1	1	1	X	2	2	4	X	1	2	1	1	1	2	3

Approximate Service Temperature Ranges for Commonly Used Basic Polymer Types*

Nitrile (General Service)	-34°C to 121°C (-30°F to 250°F)*	AFLAS	-9°C to 232°C (15°F to 450°F)*
Nitrile (Low Temperature)	-55°C to 107°C (-65°F to 225°F)*	Neoprene	-51°C to 107°C (-60°F to 225°F)*
Hydrogenated Nitrile	-32°C to 149°C (-23°F to 300°F)*	Polyacrylate	-21°C to 177°C (-5°F to 350°F)*
Ethylene Propylene	-57°C to 121°C (-70°F to 250°F)*	Polyurethane	-40°C to 82°C (-40°F to 180°F)*
Fluorocarbon	-26°C to 205°C (-15°F to 400°F)*	Butyl	-59°C to 120°C (-75°F to 250°F)*
Hilufur	-26°C to 205°C (-15°F to 400°F)*	Fluorosilicone	-73°C to 177°C (-100°F to 350°F)*
Perfluoroelastomer (Perfluor)	-26°C to 320°C (-15°F to 608°F)*	Silicone	-115°C to 232°C (-175°F to 450°F)*

NOTE: *These temperature ranges will apply to the majority of media for which the material is potentially recommended. With some media however, the service temperature range may be significantly different. ALWAYS TEST UNDER ACTUAL SERVICE CONDITIONS.

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 2380 Palumbo Drive, Lexington, KY 40509
 Phone: (606) 269-2351 • Fax: (606) 335-5128
www.parkerorings.com

Compatibility Tables for Gases, Fluids, Solids

7-27

Face Seal Glands

O-Ring Face Seal Glands These dimensions are intended primarily for face type O-ring seals and low temperature applications.

O-Ring Size Parker No. 2	W Cross Section		L Gland Depth	G Groove Width		R Groove Radius
	Nominal	Actual		Liquids	Vacuum and Gases	
004 through 060	1/16	.070 ±.003 (1.78 mm)	.050 to .054	.013 to .023	.101 to .107	.006 to .015
102 through 178	3/32	.103 ±.003 (2.62 mm)	.074 to .090	.020 to .032	.136 to .142	.006 to .015
201 through 284	1/8	.139 ±.004 (3.53 mm)	.101 to .107	.028 to .042	.177 to .187	.010 to .025
309 through 395	3/16	.210 ±.005 (5.33 mm)	.152 to .162	.043 to .063	.270 to .290	.020 to .035
425 through 475	1/4	.275 ±.006 (6.99 mm)	.201 to .211	.058 to .080	.342 to .362	.020 to .035
Special	3/8	.375 ±.007 (9.52 mm)	.276 to .296	.082 to .106	.475 to .485	.030 to .045
Special	1/2	.500 ±.008 (12.7 mm)	.370 to .390	.112 to .138	.638 to .645	.030 to .045

Design Chart 4-3: Design Chart for O-Ring Face Seal Glands

Gland Dimensions for Industrial O-Ring Static Seals, 103.5 Bar (1500 psi) Max.† (Continued)

O-Ring Size Parker No. 2-	Dimensions				A	A-1		B	B-1		C	D	G [†]
	ID	±	W	Mean OD (Ref)	Bore Dia. (Male Gland) +0.002 -0.000	Groove Dia. (Female Gland) -0.000	+	Tube OD (Female Gland) +0.000 -0.002	Groove Dia. (Male Gland) +0.000	-	Plug Dia. (Male Gland) +0.000 -0.001	Throat Dia. (Female Gland) +0.001 -0.000	Groove Width +0.005 -0.000
044	3.739	.027	↑	3.879	3.875	3.850	↑	3.750	3.775	↑	3.873	3.752	↑
045	3.989	.027	.070	4.129	4.125	4.100	.002	4.000	4.025	.002	4.123	4.002	.093
046	4.239	.030	±.003	4.379	4.375	4.350		4.250	4.275		4.373	4.252	
047	4.489	.030		4.629	4.625	4.600		4.500	4.525		4.623	4.502	
048	4.739	.030		4.879	4.875	4.850		4.750	4.775		4.873	4.752	
049	4.989	.037		5.129	5.125	5.100		5.000	5.025		5.123	5.002	
050	5.239	.037	↓	5.379	5.375	5.350	↓	5.250	5.275	↓	5.373	5.252	↓
102	.049	.005	↑	.255	.247	.224	↑	.062	.085	↑	* .245	.064	↑
103	.061	.005		.287	.278	.256		.064	.116		* .276	.085	
104	.112	.005		.318	.310	.287		.125	.148		* .308	.127	
105	.143	.005		.349	.342	.318		.156	.180		* .340	.158	
106	.174	.005		.380	.374	.349		.187	.212		* .372	.189	
107	.206	.005		.412	.405	.381		.219	.243		* .403	.221	
108	.237	.005		.443	.437	.412		.250	.275		* .435	.252	
109	.299	.005		.505	.500	.474		.312	.338		* .498	.314	
110	.362	.005		.568	.562	.537		.375	.400		* .560	.377	
111	.424	.005		.630	.625	.599		.437	.463		* .623	.439	
112	.487	.005		.693	.687	.662		.500	.525		* .685	.502	
113	.549	.007		.755	.750	.724		.562	.588		* .748	.564	
114	.612	.009		.818	.812	.787		.625	.650		.810	.627	
115	.674	.009		.880	.875	.849		.687	.713		.873	.689	
116	.737	.009		.943	.937	.912		.750	.775		.935	.752	
117	.799	.010		1.005	1.000	.974		.812	.838		.998	.814	
118	.862	.010		1.068	1.062	1.037		.875	.900		1.060	.877	
119	.924	.010	.103	1.130	1.125	1.099	.002	.937	.963	.002	1.123	.939	.140
120	.987	.010	±.003	1.193	1.187	1.162		1.000	1.025		1.185	1.002	
121	1.049	.010		1.255	1.250	1.224		1.062	1.088		1.248	1.064	
122	1.112	.010		1.318	1.312	1.287		1.125	1.150		1.310	1.127	
123	1.174	.012		1.380	1.375	1.349		1.187	1.213		1.373	1.189	
124	1.237	.012		1.443	1.437	1.412		1.250	1.275		1.435	1.252	
125	1.299	.012		1.505	1.500	1.474		1.312	1.338		1.498	1.314	
126	1.362	.012		1.568	1.562	1.537		1.375	1.400		1.560	1.377	
127	1.424	.012		1.630	1.625	1.599		1.437	1.463		1.623	1.439	
128	1.487	.012		1.693	1.687	1.662		1.500	1.525		1.685	1.502	
129	1.549	.015		1.755	1.750	1.724		1.562	1.588		1.748	1.564	
130	1.612	.015		1.818	1.812	1.787		1.625	1.650		1.810	1.627	
131	1.674	.015		1.880	1.875	1.849		1.687	1.713		1.873	1.689	
132	1.737	.015		1.943	1.937	1.912		1.750	1.775		1.935	1.752	
133	1.799	.015		2.005	2.000	1.974		1.812	1.838		1.998	1.814	
134	1.862	.015		2.068	2.062	2.037		1.875	1.900		2.060	1.877	
135	1.925	.017		2.131	2.125	2.099		1.937	1.963		2.123	1.939	
136	1.987	.017		2.193	2.187	2.162		2.000	2.025		2.185	2.002	
137	2.050	.017	↓	2.256	2.250	2.224	↓	2.062	2.088	↓	2.248	2.064	↓

† This groove width does not permit the use of Parbak rings. For pressures above 103.5 Bar (1500 psi), consult Design Chart 4-2 for groove widths where back-up rings must be used.

* These designs require considerable installation stretch. If assembly breakage is incurred, use a compound having higher elongation or use a two-piece piston.

Design Table 4-2: Gland Dimensions for Industrial O-Ring Static Seals, 103.5 Bar (1500 psi) Max.

Tank Fasteners

Micropropulsion tank and cap detailed drawings

Appendix G. COSMOS simulation results with table of mechanical properties used.

Tank Simulations

Stress simulation with 1200kPa pressure and all cap mounting holes and CubeSat interface holes constrained.

Stress simulation with lowest yield factor of safety = 1.42

Deflection simulation with 1200 kPa internal pressure and all cap mounting holes and CubeSat interface holes constrained.

Maximum deflection of 1.9 mm at bottom base of cap.

Cap simulations

Stress simulation on cap with 1200 kPa internal pressure with all cap mounting holes constrained.

Lowest yield factor of safety for conditions listed above = 1.20.

Deflection simulation for 1200 kPa internal pressure and all cap mounting holes constrained.

Maximum deflection at center of cap = 0.88mm.

Cap mounting hole stress concentrations shown produce lowest factor of safety.

O-ring stress concentrations also limiting factor in cap strength.

Aluminum 7075-T6 mechanical
properties used in simulations

Density (kg/m3)	2800
Elastic Modulus (GPa)	71.7
Tensile Strength (MPa)	593
Shear Modulus (GPa)	542
Poissons Ratio	0.33

Appendix H. Tension Joint Analysis data for Figure 20.

C	0.2						
At	5.03E+00						
Sy	1100		Recommended				
Sp	970		75% of proof load				
	Tank Pressure (kPa)	External Force per Bolt (N)	Preload Force (N)	Bolt Torque Required (N*m)	Bolt Stress (Mpa)	Bolt Yield Factor of Safety	Joint Separation Factor of safety
	1100	775.87	3659.33	2.20	758.35	1.45	5.90
	1200	846.40	3659.33	2.20	761.15	1.45	5.40
	1300	916.93	3659.33	2.20	763.96	1.44	4.99
	1400	987.47	3659.33	2.20	766.76	1.43	4.63
	1500	1058.00	3659.33	2.20	769.57	1.43	4.32
	1600	1128.53	3659.33	2.20	772.37	1.42	4.05
	1700	1199.07	3659.33	2.20	775.18	1.42	3.81
	1800	1269.60	3659.33	2.20	777.98	1.41	3.60
	1900	1340.13	3659.33	2.20	780.79	1.41	3.41
	2000	1410.67	3659.33	2.20	783.59	1.40	3.24
	2100	1481.20	3659.33	2.20	786.39	1.40	3.09
	2200	1551.73	3659.33	2.20	789.20	1.39	2.95
	2300	1622.27	3659.33	2.20	792.00	1.39	2.82
	2400	1692.80	3659.33	2.20	794.81	1.38	2.70
	2500	1763.33	3659.33	2.20	797.61	1.38	2.59
	2600	1833.87	3659.33	2.20	800.42	1.37	2.49
	2700	1904.40	3659.33	2.20	803.22	1.37	2.40
	2800	1974.93	3659.33	2.20	806.03	1.36	2.32
	2900	2045.47	3659.33	2.20	808.83	1.36	2.24
	3000	2116.00	3659.33	2.20	811.64	1.36	2.16
	3100	2186.53	3659.33	2.20	814.44	1.35	2.09
	3200	2257.07	3659.33	2.20	817.24	1.35	2.03
	3300	2327.60	3659.33	2.20	820.05	1.34	1.97
	3400	2398.13	3659.33	2.20	822.85	1.34	1.91
	3500	2468.67	3659.33	2.20	825.66	1.33	1.85
	3600	2539.20	3659.33	2.20	828.46	1.33	1.80
	3700	2609.73	3659.33	2.20	831.27	1.32	1.75
	3800	2680.27	3659.33	2.20	834.07	1.32	1.71
	3900	2750.80	3659.33	2.20	836.88	1.31	1.66
	4000	2821.33	3659.33	2.20	839.68	1.31	1.62
	4100	2891.87	3659.33	2.20	842.48	1.31	1.58
	4200	2962.40	3659.33	2.20	845.29	1.30	1.54
	4300	3032.93	3659.33	2.20	848.09	1.30	1.51
	4400	3103.47	3659.33	2.20	850.90	1.29	1.47
	4500	3174.00	3659.33	2.20	853.70	1.29	1.44
	4600	3244.53	3659.33	2.20	856.51	1.28	1.41
	4700	3315.07	3659.33	2.20	859.31	1.28	1.38
	4800	3385.60	3659.33	2.20	862.12	1.28	1.35
	4900	3456.13	3659.33	2.20	864.92	1.27	1.32
	5000	3526.67	3659.33	2.20	867.73	1.27	1.30

Appendix I. Thrust Calculation for Cold Gas Test Hand Calculations.

Cold Gas Thrust test calculations

FBD

sum of angular impulses = change in angular momentum

$$\int_{t_1}^{t_2} \sum M_o dt = H_{o_2} - H_{o_1} \quad \omega_1 = 0$$

$$F_T R (t_2 - t_1) - M_{friction} (t_2 - t_1) - I_o \alpha (t_2 - t_1) = H_{o_2}$$

$$F_T R (t_2 - t_1) - M_{fr} (t_2 - t_1) - I_o \alpha (t_2 - t_1) = m \omega_2 R^2$$

$$F_T = \frac{m \omega_2 R^2 + I_o \alpha (t_2 - t_1) + M_{fr} (t_2 - t_1)}{R (t_2 - t_1)}$$

$$F_T = \frac{m \left(\frac{\theta_2 - \theta_1}{t_2 - t_1} \right) R^2 + I_o \left(\frac{\omega_2 - \omega_1}{t_2 - t_1} \right) (t_2 - t_1) + M_{fr} (t_2 - t_1)}{R (t_2 - t_1)}$$

$$F_T = \frac{m (\theta_2 - \theta_1) R}{(t_2 - t_1)^2} + I_o \frac{(\theta_2 - \theta_1)}{R (t_2 - t_1)^2} + \frac{m_{fr}}{R}$$

$$F_T = \frac{m R \theta_2}{t_2^2} + \frac{I_o \theta_2}{R t_2^2} + \frac{m_{fr}}{R}$$

Now Moment due to friction calculation

$$M_{fr} = F_{fr} R$$

$$F_{fr} = f \left(\frac{W}{R} \right) \quad f = 0.0012 \text{ m (published value)}$$

$$= 0.0012 \text{ m} \left(\frac{9.81 \text{ N}}{0.00472 \text{ m}} \right)$$

$$= 2.59 \text{ N}$$

$$M_{fr} = 2.59 \text{ N} (0.00472 \text{ m}) = \underline{0.0117 \text{ N}\cdot\text{m}}$$

$$I_{o \text{ propulsion module}} = 0.2383 \text{ kg}\cdot\text{m}^2 \quad \left(\text{from Solidworks and parallel axis theorem} \right)$$

$$I_{o \text{ rod}} = 0.0693 \text{ kg}\cdot\text{m}^2$$

$$I_o = I_{o \text{ propulsion module}} + I_{o \text{ rod}} = \underline{0.2394 \text{ kg}\cdot\text{m}^2}$$

$$t_1 = 0$$

$$t_2 = 0.25 \text{ sec} \quad (\text{valve open time})$$

$$\Theta_2 = \text{read from encoder output}$$