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Development of a Warm-Gas Butane System for Microsatellite Propulsion

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ABSTRACT

Satellite propulsion systems are typically complex and costly, and require extensive ground support equipment and personnel to implement. While these systems may be feasible for large commercial or government programs to execute, they are impractical for micro- and nano-satellite systems. In particular, these small spacecraft cannot utilize classical hazardous or high-pressure components, as small satellites are often hosted as secondary payloads with stringent restrictions imposed. In contrast to this paradigm, an effort to develop, manufacture, and validate a low-cost propulsion system suitable for small, secondary payloads was pursued. Intended for orbit insertion, station-keeping, and de-orbit operations, this system was developed to operate within the typical constraints of a small satellite. In addition to the aforementioned cost and safety concerns, small satellites possess limited mass, volume, and electrical power. Given these considerations, a straightforward warm gas system was chosen, with liquid butane as propellant. This configuration is not only inherently simple, but can also be manufactured using primarily off-the-shelf components, which can be quickly procured in small quantities. With a fully integrated sensor manifold and heating assembly, this design provides performance well suited to a small satellite mission of several years in low-Earth orbit.

INTRODUCTION

This paper discusses the design of a low-cost warm-gas propulsion system for microsatellites with less than approximately 50 kg on-orbit mass. For satellites of this class, the system provides modest (tens of meters per second) delta-V capability appropriate for orbit insertion, station-keeping, and de-orbit maneuvers.

Several satellite-level requirements drove the propulsion system design:

- Sufficient delta-V to maintain a 450 km orbit for at least 2 years, assuming worst-case atmospheric drag
- Low-cost: $10,000 recurring in materials and touch labor per flight unit
- Suitable for launch as a secondary payload, restricting high-pressure, hazardous, or overly complex components
- Size, weight, and power constraints of a microsatellite bus: approximately 30 liters, 10 kg, and 100 W, respectively

Commercially operated satellites with decade-long lifetimes drive high delta-V requirements; coupled with the inherent volumetric constraints of space systems, designers are forced into high-specific-impulse solutions to keep the overall size and mass of the system within budgets. Classical hydrazine-based mono- and bi-propellant systems are expensive to develop and produce, and rely on hazardous and difficult to procure propellants that drive up operations and launch vehicle integration costs. Hydrazine is effectively a non-starter for secondary payloads in all but the most extreme circumstances.

Furthermore, the exhaust temperatures associated with high specific impulse chemical propulsion can lead to the selection of exotic and expensive materials, such as refractory metals or high-temperature composites that are intricately machined with extreme precision. These propellant and material selections drive-up the cost—to design, manufacture, and integrate—and result in a system that is prohibitive for small programs whose entire budgets cannot be dedicated to legacy propulsion system approaches.

However, several factors unique to microsatellites can dramatically alter this calculus and enable a paradigm shift away from complex, high-performance systems towards low-cost simplicity.

The first important factor is the more modest on-orbit life requirements of a typical microsatellite: several years compared to a decade or more. Shorter life drastically reduces the delta-V required for orbit maintenance.

Perhaps even more important is the potential rise of small launch vehicles such as SWORDS and ALASA. In the near term, several entities are building proof-of-concept and technology demonstration satellites that can achieve their goals in a wide range of orbits and are thus served well by the currently existing secondary payload market. In the longer-term, when these entities switch their focus to deployment of satellite constellations or other spacecraft missions with precise orbital requirements, the availability of dedicated launch vehicles will enable microsatellites to be placed in custom orbits, eliminating the delta-V required for phasing and altitude change maneuvers.

Some combination of shorter life and dedicated launch relaxes the delta-V requirement enough to significantly expand the trade space of the microsatellite propulsion system designer and enable a low-cost, lightweight, and low-volume solution. The goal of this effort was to prove this assertion by developing, manufacturing, and testing a low-cost microsatellite propulsion solution.

Scope

This paper presents the driving system requirements and the systems engineering approach used to make fundamental configuration decisions. It discusses the modeling and analysis that initially sized the key subcomponents, the iterative approach used during detailed design and prototype manufacturing, and the test approach and objectives.

DESIGN APPROACH

Figure 1 shows the propulsion system design approach.

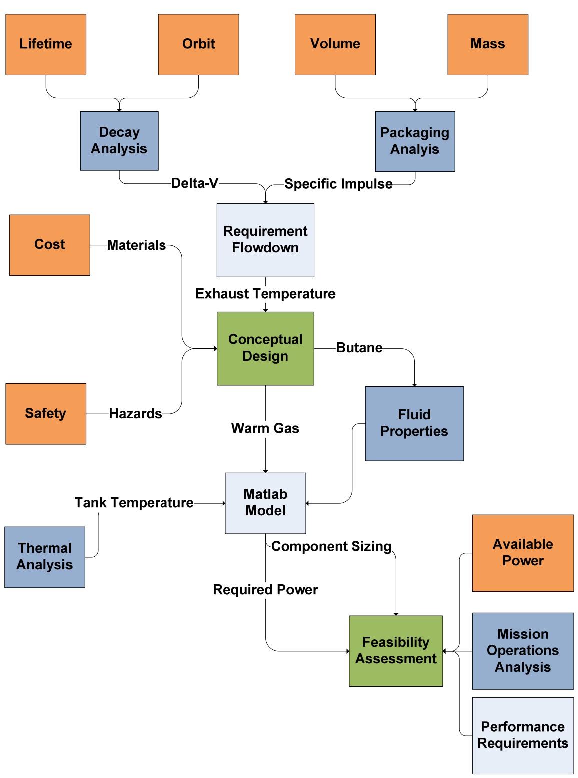


Figure 1: Overall design flow

Beginning with satellite-level requirements, the delta-V, specific impulse, and exhaust temperature were derived. These parameters, along with the cost and safety restrictions led to the initial warm-gas concept and fuel selection. An analytical model was then developed and used to determine necessary component size and electrical power. Finally, these results could be used to assess the system's feasibility, given the satellite's capabilities.

These requirements, modeling, and analyses are discussed in further detail in the following sections.

DRIVING REQUIREMENTS

Delta-Velocity

The principal performance requirement for this system is a total delta-V capability of 60 meters per second. This requirement was based (predominately) on an estimate of the aerodynamic drag of a characteristic microsatellite mass and cross-section (50 kg and 40 x 40 x 100 cm) during worst-case solar cycle at a spacecraft altitude of 520 km. Table 1 provides a summary of the delta-V required each year to maintain this orbit given different spacecraft and solar conditions.

Table 1: Delta-V requirement summary

Solar Panel Configuration	Ballistic Coeff	ΔV required per year (m/s)	
		Solar Min	Solar Max
Body-fixed	Min	1.5	32.6
	Max	0.9	19.5

This requirement drives many of the major design parameters of the system, including overall size and component selection.

Cost

Both recurring and development cost were design constraints on the system. The recurring cost target of $10,000 requires selection of mostly commercial-off-the-shelf (COTS) parts. There was a particular focus on non-aerospace components because the materials and process requirements driven into aerospace parts development by the large government and commercial contracts that funded their development typically result in sale prices far above the marginal cost to produce. The cost constraints also ruled out exotic materials that require special fabrication techniques or cannot be quickly and affordably sourced in small quantities.

The design also considered how to reduce the touch labor during fabrication and assembly. Commercially available (rather than custom) parts were selected when possible, and mechanical design focused on manufacturing simplicity and feasibility. The overall parts count was kept low, which included elimination of redundant components and of sensors not directly used by the propulsion system controller. The non-COTS items are machined from small amounts of inexpensive raw materials and designed for fabrication even in modest machine shops.

Keeping development costs low drove fast-paced development cycles; acceptable solutions were selected and advanced quickly, rather than iterating exhaustively for an ideal result. Sound systems engineering analysis enabled closing off significant portions of the potential trade tree early in the design process to enable greater focus on detailed design and prototyping and less on building analytical models and PowerPoint™ charts.[1]

This detailed design and prototype phase focused on system simplicity. Architectures or components that required additional support equipment or increased the complexity of the interface between the propulsion system and the spacecraft avionics were avoided. Components or propellants that would impact launch availability or compatibility with other payloads were removed. Every component of the system had to earn its way into the final design.

Specific Impulse

For a notional 50 kg and 40 x 40 x 100 cm microsatellite, 10 kg and a 40 x 40 x 20 cm volume was allocated to the propulsion system. Coupled with the 60 m/s delta-V requirement described above, and an estimate of the feasible propellant mass fraction, the rocket equation was used to estimate the required specific impulse of the system. The equations below show this calculation:

$$\Delta V = I_{SP} * g_0 * ln\left(\frac{m_0}{m_f}\right) \quad (1)$$

$$60 = I_{SP} * 9.8 * ln\left(\frac{50}{46}\right) \quad (1)$$

$$I_{SP} \approx 70\ seconds \quad (3)$$

Thus, a specific impulse target of 70 seconds was established. The mass fraction of approximately 8% was based on available volume for propellant storage, and influenced propellant selection.

[1] Ultimately both an analytical model and several PowerPoint charts were created. The analytical model proved to be very useful.

Power Consumption and Electrical Compatibility

Microsatellites are typically power limited. Even given an assumption that the propulsion system can consume a large fraction of the available electrical energy when it is operating, available power (for sensors, valves, and heaters) is practically limited by the size of the solar arrays (often not deployable) and the power system avionics.

Additional constraints are compatibility of electrical components. Microsatellite propulsion systems are far from ubiquitous, meaning there may be an implicit requirement to interface with an existing avionics system, a system whose design was established without consideration for a propulsion system interface, or at least without the detailed interfaces being designed. This means the number and type of interfaces available in the existing avionics architecture constrains electrical component selection. It also means there is value in a propulsion system design where specific components (e.g. pressure sensors) can be replaced with alternative part numbers that offer the same performance and physical interface, but with alternative electrical interfaces. In this way, the cost of tailoring the propulsion system for electrical compatibility with myriad bus configurations is straightforward and low-cost and can be done without invalidating the environmental qualification of the system.

PRELIMINARY DESIGN

Given the above requirements, several initial trades determined the most suitable overall concept. Based on these early decisions an analytical model was developed to estimate major performance parameters and size the key components in support of component selection and detailed design.

The driving requirements discussed above easily eliminated several configurations: mono-propellant hydrazine and high pressure systems are incompatible with most secondary launch providers, and bi-propellants and designs requiring secondary pressurization are overly complex, and may not provide sufficient performance given mass and volume constraints.

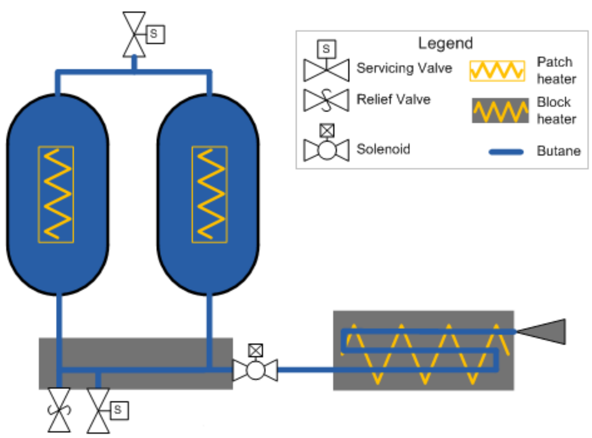
Thus a monopropellant warm-gas system was selected and the design effort switched to identifying an inexpensive, safe, and readily available propellant. A warm gas system has several characteristics that are highly favorable for this application:

First, as one of the most basic propulsion schemes available, the system is inherently simple and low-cost. Both of these are essential for providing a realistic propulsion solution for small spacecraft applications.

Second, with few overall components and a single moving part, system integration is streamlined. Moreover, the minimal number of distinct components reduces the potential for leaks, contamination, and other sources of inefficiency.

Finally, as will be presented below, this configuration satisfies the fundamental performance requirements while allowing for operation within spacecraft's capabilities.

A basic schematic diagram of this warm-gas system is depicted in F*igure 2*. Fuel is stored as a liquid, and flows through a block heater before being exhausted. Heat is provided through resistive heating elements, and the flow is controlled by a single solenoid valve.



F**igure 2: Propulsion system schematic diagram**

Propellant

Several warm-gas monopropellants were considered. As discussed below, several were easily ruled out. The remaining were easily analyzed using simple propulsion and thermodynamic equations (discussed in more detail later) and the NIST Refprop fluids property database (automating an otherwise painfully tedious task of looking up fluid properties manually).

Some of the same considerations that ruled out bi-propellants also applied to a selection of a specific monopropellant. Fuels that are difficult to store or hazardous to work with were eliminated. A spacecraft thermal analysis showed the propellant tanks are relatively cold due to their location and arrangement within the satellite bus. This result encouraged options whose density and vapor pressure were high at the anticipated operational temperatures of 250 K. Table 2 shows the target propellant density, given available tank volume and required propellant mass fraction.

Table 2: Summary of necessary fuel storage density

Parameter	Value
Tank volume	8 L
Propellant mass	4 kg
Target density	500 kg/m^3

A low freezing point eliminated the need to run tank heaters, liquid storage allows a greater mass of fuel to be carried in the limited spacecraft volume (an important consideration for meeting the total delta-velocity requirement), and low-pressure storage is important for range safety, cost, and leak reduction factors.

A fuel that satisfies these requirements is butane. Readily available, inexpensive, and easy to work with, butane was chosen as a suitable propellant for this system. Table 3 provides a qualitative summary of several fuels that were considered. Some of these were rendered unusable due to their limited availability, hazardous nature, or high cost. Others were handicapped by poor performance or unrealistic demands on the satellite. The table is not exhaustive in listing all the options that were considered, but reflects the general approach and several obvious candidates.

Table 3: Relevant qualities of potential fuels

Fuel	Hazard	Cost	Comments
Butane		<$200	~8 psi operating ~75 psi storage
Benzene		--	~0.2 psi operating ~6 psi storage
Propane		<$10	Unrealistic electrical power requirements
Nitrogen	--	--	Cryogenic at required density
Helium	--	--	Cryogenic at required density
Water	--	--	Freezes at operating temp
Ethanol		<$200	~0.05 psi operating ~3 psi storage
IPA		<$1,000	--
Methanol		<$200	~0.2 psi operating ~6 psi storage

Mechanical Configuration

Using butane as a baseline fuel, the initial requirements were then analyzed to determine driving performance constraints. As discussed above, the delta-velocity and volume constraints yielded a minimum specific impulse derived requirement of 70 seconds. Using butane fuel, and assuming a simple mechanical nozzle, this specific impulse results in a required exhaust temperature of 450 K.

The nozzle is a simple cone. While this geometry is not ideal for maximum performance, the nozzle could be procured as a single, readily available, fully integrated, off-the-shelf unit. Future work may include improving on this baseline.

This high exhaust temperature further restricted the design space. Normal metals like aluminum and steel can easily reach these temperatures, but the seals on most valves, typical pressure sensors, and many types of heaters cannot. This meant whatever the design of the upstream portions of the system, the final components would be a heat exchanger to raise the propellant temperature to at least 450 K followed immediately by the exhaust nozzle. This observation, combined with a desire to reduce the number of solenoid valves, and components in general, ruled out one of the early concepts, which was a system with multiple heated sections, such as those containing a plenum, to heat and pressurize the propellant before venting.

Ultimately the design was a simple system. Two propellant tanks (tank shape, quantity, and layout driven by packaging constraints) are connected via a shared fill/drain line and manually operated valve. Ports on the opposite sides of the tank are fed directly to a manifold with a service valve, relief valve, and temperature and pressure sensors. This layout with the second tank depicted in wireframe is shown in Figure 3.

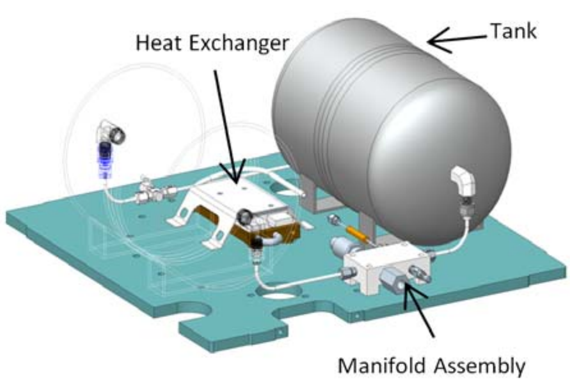


Figure 3: CAD model of propulsion system assembly

During fill the manual service valve is opened to vent the tanks while butane is gravity fed into the tanks through the fill valve.

During operation, flow out of the manifold is controlled with the solenoid valve, after which it enters a heat exchanger and is vented through the integrated nozzle.

ANALYSIS

The following section describes the analytical model that was developed to size the propulsion system components and predict performance, as well as the specific analyses that were performed.

The model, as with the analysis process, is broken into two sections. The first section was used during preliminary design to size the heat exchanger based on simple rocket performance and heat transfer equations. This analysis provided inputs to the mechanical designer and guided heat exchanger design and material and component selection.

The second section uses the preliminary design as an input, predominately the geometric configuration of the system, including tanks, manifold, heat exchanger, nozzle, and piping. The outputs of this section were used to estimate heat loss sources, pressure drops, and otherwise validate performance against the initial requirements and provide test predictions to be validated by the demonstration system.

This detailed analysis demonstrates that the warm gas configuration can provide the required performance within the set of constraints required to maintain compatibility with the rest of the spacecraft. This modeling and analysis also led to the development of more precise subsystem and component requirements for further refinement of the preliminary concept.

Heat Exchanger Sizing

The analytical model was constructed using Matlab software, supported by NIST's Reference Fluid Properties (REFPROP) for properties of specific fuels. The model simulates the fluid and thermodynamic properties of the system to calculate major performance parameters and design requirements. The combination of Matlab and REFPROP allows for rapid reevaluation of performance parameters given different input conditions.

During normal operations the spacecraft will be venting to vacuum and the flow will be choked at some point in the system. Performance is maximized by choking the flow at the nozzle throat and expanding it through a divergent nozzle (conical in our case for ease of manufacturing). The first step in the analysis is to calculate the mass flow rate as a function of nozzle geometry. Given the nozzle parameters described above, the required specific impulse, and the stagnation pressure (established by the vapor pressure of the propellant in the tanks and an assumed pressure drop) the throat and stagnation conditions as well as the mass flow rate can be calculated.

The mass flow rate is a critical parameter because it dictates feasible plumbing sizes (so that the pressure drop is consistent with the assumed value), and more importantly because it drives the rate at which heat must be added to the flow to achieve the required performance (thus setting the electrical power consumption).Next, a thermodynamic analysis is performed to size a heat exchanger that can start with liquid tank propellant, vaporize it, and raise its temperature to the assumed value. This model has two distinct stages. The first stage assumes fully liquid flow is supplied from the tanks to the heat exchanger and calculates the length of heat exchanger required to fully vaporize it. The second stage assumes fully gaseous flow and calculates the length of heat exchanger required to sufficiently heat the gas. In reality there is two-phase flow within a significant portion of the heat exchanger, but this simplified analysis is of sufficient fidelity to perform preliminary heat exchanger sizing. In micro-gravity operations, especially when a significant quantity of the propellant has been consumed and the tanks contain a sizable ullage volume, the quality of the fuel exiting the tank will be unknown; this two-phase analysis represents the worst-case scenario from a required power perspective, when sufficient energy and heat exchanger surface area is required to fully vaporize the nominal mass flow rate.

Standard convection heating relations for a circular cross-section tube at uniform temperature are used to determine the length of tubing required to fully vaporize the flow. This temperature was a user-defined input parameter iterated during analysis, but fundamentally driven by maximum heater temperatures and heat exchanger material properties. The stagnation pressure loss across this section is also determined using the calculated length and input geometric parameters. The hydrodynamic and kinematic entrance lengths are calculated to verify correctness of the laminar vs. turbulent kinematic and thermal boundary layers, which dictate the correct equations to use.

The second stage of analysis determines the length of heat exchanger tubing required to heat the gaseous fuel to the target exit temperature. At this point, the fuel has been fully vaporized by the first section of the heat exchanger. Pressure drops across this section are also calculated. The calculations are fundamentally the same as above except for the fluid property assumptions, most importantly that during vaporization the bulk fluid stayed close to the vaporization temperature.

Heat Exchanger Preliminary Design

A preliminary heat exchanger design was generated based on this required fluid passage length and the packaging constraints of the system.

The selection of the heat exchanger temperature was an iterative process. Unfortunately the maximum temperature of typical Kapton patch heaters was below the required exhaust temperature, so a higher-temperature ceramic heater was used instead. The maximum temperature of these heaters over 1,000°F, so the maximum heat exchanger temperature was instead set by the heat transfer problem: as hot as the block could sustain a temperature given the available input power and the heat consumed by heating the propellant and lost through conduction and radiation.

Iterations of this analysis were performed to compromise between heat exchanger temperature and size. As temperature increases, the length of tube in the exchanger can be decreased, but the conductive and radiative heat loss increase. Too low and the size and mass of the heat exchanger become infeasible. With full utilization of available power, a set temperature of 600°F was selected, which corresponded to approximately 20 centimeters of heating length.

Initial allocations were made to these heat loss sources and an initial heat exchanger was designed. Thermal conduction paths that permit heat loss from the heat exchanger are through the inlet pipe and mounting bracket. The initial analysis showed mounting the heat exchanger directly to the propulsion deck would have too much conductive heat loss, even if separated by a relatively thick insulator. The primary issue is the ceramic heaters are mechanically mounted to the heat exchanger under high pressure, and that no suitable insulation could be found that would provide enough isolation and carry this mounting load. Furthermore, the mounting screws acted as thermal shorts, largely undermining the effectiveness of the insulation. The solution was a re-design that used a high-strength (but less effective) insulator with large surface area to take the heater mounting pressure and to add additional thermal isolation by suspending the heat exchanger from a bracket mounted to the propulsion deck specifically designed with thermal choke points (akin to the design of the choke point on the handle of a good pot or pan) to reduce conductive loss to the deck.

As a bounding analysis, thermal radiation losses were assumed to occur from the heat exchanger to deep space given no insulation. This assumption is conservative, as the primary view factors after installation would be to the spacecraft itself. Nonetheless, the calculated losses were acceptable, so no additional radiation modeling was performed.

These two sources of heat loss (plus another calculation for the conduction back through the long, thin-walled supply tube) were added to the heat provided to the flow to determine the electrical power necessary to achieve the desired heat exchanger temperature. This parameter, along with the overall heat exchanger size, was compared to the available spacecraft power to assess overall system feasibility. After several iterations (mostly to the materials selected for the heat exchanger and other relatively minor but important mechanical details) a design was selected that closed. The model predicts the required delta-velocity is achievable and that the required power input to the system falls within the capabilities of the spacecraft.

IMPLEMENTATION

Implementation of this design has been focused on constructing a ground-test system to demonstrate the concept. Deviations from the flight configuration have been kept to a minimum, and are restricted to mounting hardware and number of sensors (more sensors were used in the test system to provide additional performance data). In an effort to reduce development time and cost, readily available commercial off-the-shelf (COTS) have been used when possible. Custom mechanical components have been manufactured on-site.

COTS Components

COTS parts were selected for all electrical devices, and several mechanical components. Fuel tanks, piping, fittings, heating elements, sensors, valve, and the nozzle were all procured from commercial suppliers.

Tank selection was simplified by the presence of a local vendor. Tanks are simple, welded aluminum cylinders with significant margin to the maximum expected operating pressure of the system.

Piping and fittings are all standard stainless steel and aluminum components, available from a number of vendors.

Heater selection was driven primarily by the high temperatures required in the heat exchanger. This requirement eliminated popular Kapton heaters, and drove the selection toward ceramic devices. A pair of mica heaters that can be clamped to the heat exchanger were selected.

Low cost, high temperature, resistance temperature detectors (RTDs) were selected as temperature sensors. These devices can withstand the high temperatures, and are compatible with a wide variety of electronic measurement systems. A commercial pressure transmitter was chosen for the manifold measurements. The device has a tight measurement range, suitable for this relatively low-pressure system. Additionally, as a current source it will not be subjected to errors associated with lengths of wiring.

There were several requirements that made valve selection very difficult. The mass flow rate is low, the system must be leak tight for years on orbit, and the signal frequency to pulse the valves was high. The working fluid is butane, the operational environment is hard vacuum, with low operating temperatures (<273 K) during start of operations and potentially high (475 K) near the end (when heat soak back through the supply tube has increase the valve temperature). Lastly, the valve needed to be compact, low-power, highly reliable (as it is a single-string failure component) and easy to control.

The power and packaging requirements were fairly easy to meet. As were the temperature extremes, microgravity operation, and even the butane working fluid (mostly a factor in valve seat material selection). The most stressing requirement proved to be leak rate. Most commercial valves of this size class (especially manifold or surface-mounted valves) have leak rates that exceed the operational mass flow rate of the system. We suspect this particular issue has been observed by many developers of cubesat and small satellite propulsion systems. A fairly exhaustive survey was conducted and a LeeCo valve specifically designed for aerospace applications was ultimately selected for its operational capabilities and leak-proof design. A photo of this valve is shown in Figure 4.

Figure 4: LeeCo solenoid valve

The nozzle is a simple NPT fitting with a machined conical orifice. Because the flow is choked by this nozzle the system can be tuned to tailor the mass flow rate (very late in production) by machining an orifice and nozzle contour into a NPT blank and screwing (and likely then welding) the nozzle into the heat exchanger base. This enables the system to be tuned to the particular orbit and application (e.g. the mass flow rate is a strong function of the stagnation pressure in the tanks, which is a strong function of the tank temperature).

Similarly, because the heat exchanger is connected to the manifold by a relatively long run of aluminum tubing, the location of the thrust vector can be tailored somewhat to better align with the as-measured spacecraft center-of-mass.

In-house Manufacturing

Several mechanical components required custom manufacturing. These include the manifold block, heat exchanger, and mounting brackets. All custom components were designed to allow for manufacturing with available machine shop equipment, and do not require multi-axis control, tight tolerance, or other costly processes.

The manifold and heat exchanger are each machined from a single block of aluminum. Drilled and tapped holes are used to provide flow paths and access for the valves and sensors in the manifold. A diagram of the manifold assembly is shown in Figure 5.

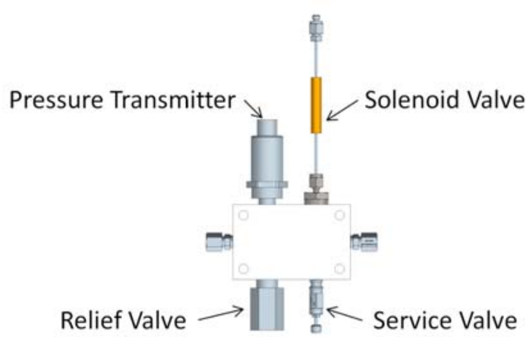


Figure 5: Diagram of manifold assembly

The heat exchanger block is machined with several parallel flow paths to allow for the most efficient use of available volume. These parallel tubes are connected with two small U-sections of piping to create a single serpentine path. This block forms the base of the heat exchanger assembly, depicted in Figure 6.

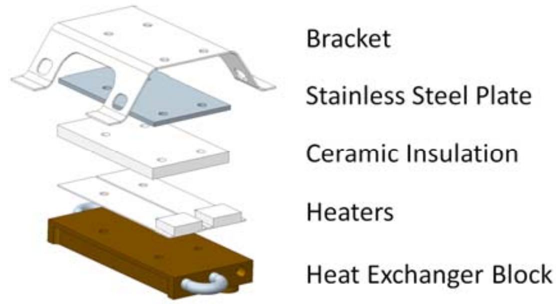


Figure 6: Exploded view of heat exchanger assembly

The next four layers are directly bolted to the main heat exchanger body. The next layer is the heaters themselves, which are in direct contact with the heat exchanger. Above the heaters is a block of ceramic insulation. A stainless steel plate is used to distribute mechanical loading and provides an additional thermal interface to overcome. Finally, a custom designed mounting bracket is used to support the heat exchanger and provide thermal isolation between this assembly and the satellite bus. The two conduction paths from the heaters to the bracket are through the insulation and steel plate and through the bolts. Heat in the bracket conducts down the bracket walls, past the thermal cut-outs and into the propulsion deck via direct contact and the bolts.

TESTING

At the time of this paper's writing, efforts are focused on the construction of the ground test system. The electronic components have been tested (most importantly the valve) and are being electrically integrated into EDU avionics.

Manifold construction and population is complete, as is the plumbing to the tanks. The heat exchanger has been machined, assembly is underway and fill/drain operations have been demonstrated. Figure 7 shows a photograph of some of these components.

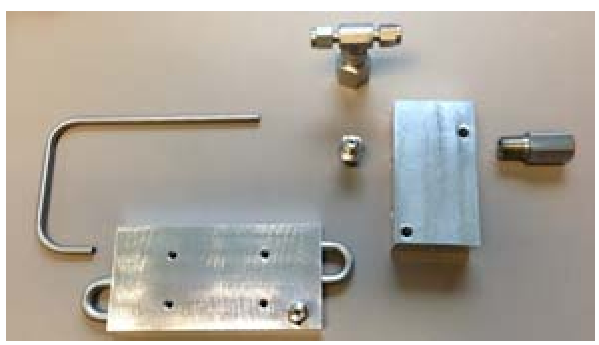


Figure 7: Selection of propulsion system components

Initial testing will validate the electrical setup: the ability to collect sensor data and command the valve and heaters. Temperature data will be collected at ambient conditions and in our in-house thermal vacuum chamber to validate the thermal model. The system will be operated at ambient and vacuum conditions to directly measure the pressure drops across the system and the exhaust temperature. The system will be left pressurized for a long period of time to test for slow leaks and Helium leak checked.

Testing is expected to begin in June 2013 and be completed by the end of the summer.

CONCLUSIONS

A low-cost propulsion system for use in typical small satellite applications has been developed and is currently undergoing integration and testing. In recognition of the constraints placed on micro- and nano-satellites, this system has been designed to operate given size and power restrictions of small satellites. Additionally, this design provides performance suitable to a mission of several years in low-Earth orbit.

Further efforts will focus on verifying the design and modeling and determining performance parameters over a range of operating conditions. This testing and future work will enable this system to be adapted to meet a wide variety of micro- and nano-satellite missions.