MATURATION OF A LOW FLAME TEMPERATURE PEROXIDE-ALCOHOL MONOPROPELLANT PROPULSION UNIT FOR CUBESATS (MPUC)

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ABSTRACT

CU Aerospace (CUA) demonstrated the scaling of an alternative very low-toxicity monopropellant thruster to an 0.25 N thruster head at TRL-6. CUA's Monopropellant Propulsion Unit for Cubesats (MPUC) with CMP-X (CUA Monopropellant 10) is a system using a non-detonable yet energetic COTS fuel formulation that possesses many system-level advantages including lower cost, lower thermal load, water-like viscosity, and compatibility with common materials. CMP-X thrusters have demonstrated >180 s specific impulse at thrust levels between 100-500 mN during thrust stand testing and continuous firing times > 50 min. Multiple CMP formulations have demonstrated shelf life exceeding 1000 days. The flight-like, additively-manufactured thruster head passed environmental (vibration and thermal vacuum) acceptance testing. This thruster head is now TRL 6 and has been coupled with a TRL 5 propellant feed system to demonstrated 178-180 s specific impulse at 230-270 mN thrust using CMP-X with continuous firing times exceeding 55 minutes that were limited only by feed tank volume. The estimated total impulse of a 2U-sized flight MPUC is ~2100 N-sec with an operating power draw of ~1.5 W and ~180 s specific impulse.

INTRODUCTION

Commercial interest in nano- and microsatellites expanded by a factor of 10x between 2010 and 2019. In the 1-50 kg satellite sector, launches have been dominated by commercial interests, with Earth observation, remote sensing, and communications markets making up the majority of the missions [DelPozzo, 2020]. Various market analyses place the compound annual growth of this microsatellite market ~20% over the coming decade. Moving forward, it is more important than ever that these satellites have access to propulsion systems to extend their asset time on orbit. Such a system must occupy minimal bus volume and carry a high product of propellant density times Isp. Avoiding the use of toxic propellants such as hydrazine that significantly complicate the storage and handling of the propulsion system is also desirable [Hargus, 2010; Singleton, 2013]. Many green monopropellants reduce hydrazine's toxic complications, but have very high flame temperatures in excess of 1800 ºC and require thrust chambers made from refractory materials, which adds significant manufacturing time and expense. A survey of available propellants motivated the search for a less exotic and more easily

Figure 1. CMT-250 flight-like thruster glowing red hot during hot fire testing with CMP-X.

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sourced chemistry to be utilized in these small satellite propulsion systems. The authors leveraged this propellant to develop a new monopropellant thruster and integrated thruster system benefitting from the use of a fuel mixed from non-toxic, readily available reagents burning at lower temperatures. The candidate fuel needed a flame temperature low enough to avoid refractory construction of the thrust chamber, but high enough so the specific impulse and density were competitive with legacy propellants. The chosen propellant had heritage in bipropellant thrusters [Woschnak, 2013, Wieling, 2012] and in US Navy Mark 16 torpedoes [Clark, 1972], but had not been developed for a thruster in the 100 mN class.

Risk reduction (both in the laboratory and for the end product on the launch range) was paramount in the propellant development program. Desirable monopropellant safety properties were described by Hawkins, *et al.* [Hawkins, 2010] and include high thermal stability, low unconfined ignition explosive response, low impact sensitivity, low friction sensitivity, low detonability, insensitive adiabatic compression, low electrostatic discharge sensitivity, and low vapor toxicity. CMP-X has a flame temperature of ~900°C and has been tested to be non-detonable in UN test series. Long-term storage testing indicates undetectable fuel degradation in excess of one year in sealed containers.

RESULTS AND DISCUSSION

MONOPROPELLANT DEVELOPMENT

CUA monopropellant, “CMP”, is a stoichiometric mixture of hydrogen peroxide and pure ethyl alcohol. Various formulations included different alcohols and dilutions, with the tenth formulation, “CMP-X”, chosen as the most suitable for this thruster system. An earlier variant, CMP-8, was formulated to use the highest common concentration of commercially available stabilized hydrogen peroxide (50 % w/w) in order to be as performance-competitive as possible with the SOA green monopropellants for small satellite applications. The stoichiometric reaction of CMP-8 with ethanol is shown in Eq. 1, wherein the propellant is combusted over a catalyst. However, CMP-8 contains more than 40% total hydrogen peroxide (~45% by mass), which is not permitted for air transport by commercial carriers like UPS or FedEx because its H_2O_2 concentration exceeds 40%. Accordingly, CMP-X represents a dilute stoichiometric mixture of H_2O_2 and ethanol with a mixed H_2O_2 concentration of just under 40%, Eq. 2.

$$6\ H_2O_2\ (aq) + 1\ C_2H_6O\ (liq) + 11.28\ H_2O\ (liq) = 2\ CO_2(g) + (20.28)\ H_2O\ (g) + 2.34\ kJ/g \qquad (1)$$

$$6\ H_2O_2\ (aq) + 1\ C_2H_6O\ (liq) + 14.48\ H_2O\ (liq) = 2\ CO_2(g) + (23.45)\ H_2O\ (g) + 1.81\ kJ/g \qquad (2)$$

By its very nature, CMPs dilution results in an intrinsically safe material – like “burning water”. Until it is mixed with catalyst, CMP-X combusts no more vigorously than water-diluted ethanol. **Figure 2** shows a test series of ignition attempts demonstrating either no ignition or merely gentle alcohol burn off unless the CMP-X monopropellant is mixed with catalyst and a heat source. The ternary detonability of ethanol with hydrogen peroxide are shown in **Figure 3** [Shanley, 1958]. Detonation testing confirmed the detonability of the candidate propellants. CMP formulations were subjected to UN Test Series 1, 2, 3, and 6. The testing facility, Safety Consulting Engineers / Dekra Process Safety of Schaumburg, IL, recommended that “CMP-X liquid propellant be excluded from the explosives Class”. This rating permits far simpler logistics than those carried by ASCENT (UN Class 1.4C) or LMP-103s (UN Class 1.4S).

Catalysis only of CMP with no ignition source

Failed flame ignition of CMP and no catalyst

Ignition of CMP with flame and granular catalyst

Figure 2. Photographs showing inherent safety of CMP-X / catalyst system

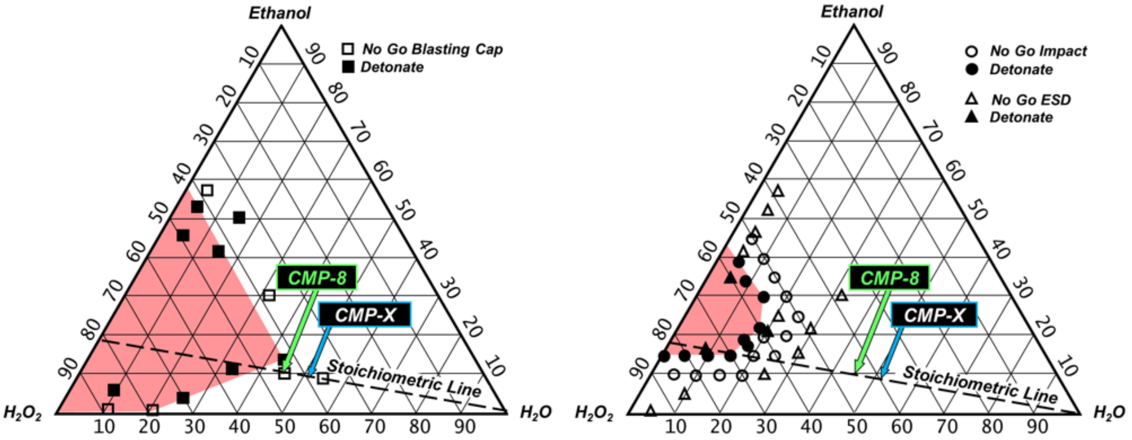

Figure 3. Ternary detonability plots of H_2O_2 / ethanol (blasting cap left, ESD and impact right).

MONOPROPELLANT STORAGE

A sodium bisulfite potentiometric titration [Gimeno, 2013] was adopted for determination of the peroxide content in CMP mixtures, and proved to be a robust technique for all of the early CMP mixtures. Long-term storage samples (formulations described in **Table 1**) were prepared into polyethylene storage vials and kept at ~280 K. The samples were titrated at regular intervals throughout a ~3-year study period, maintaining their initial peroxide concentration to within experimental error, **Figure 4**.

Table 1. Stoichiometric CMP formulations for long-duration storage and titration (note that “u” indicates unstabilized/stabilizer-free peroxide and “s” denotes stabilized).

#	Sample Name	Fuel	mixed H_2O_2	mixed Fuel
1	201002 HTP.u	n/a	50.9%	0.0%
2	201002 Tech.s	n/a	50.2%	0.0%
3	201109 CMP-Xu	ethanol	39.9%	9.0%
4	201109 CMP-Xs	ethanol	39.9%	9.0%
5	211021 HTP.u	n/a	50.8%	0.0%
6	221013 HTP.s	n/a	49.8%	0.0%

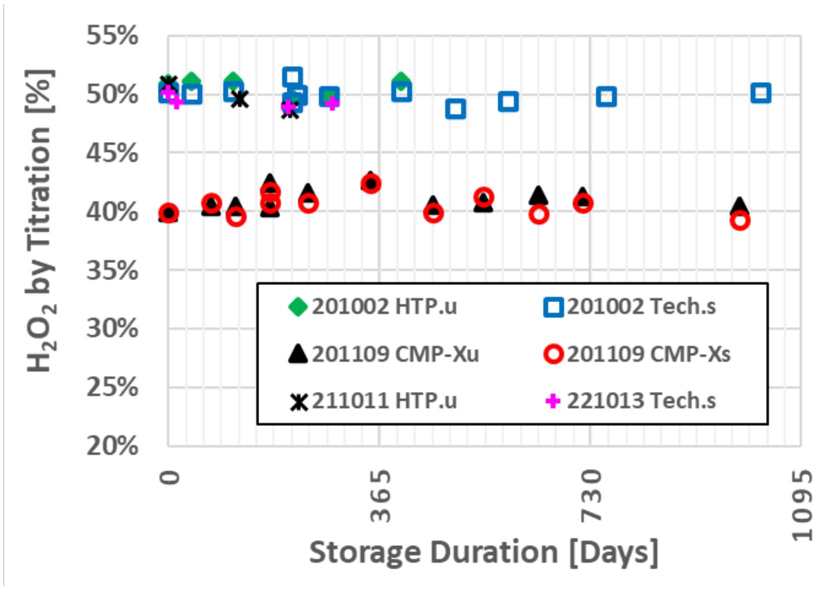

Figure 4: Long term storage titration results for individually sealed storage samples from October 2020 and bulk chemistry. Data indicate that the peroxide concentrations were maintained to within experimental error.

The properties and merits of the final CMP-X formulation are listed in **Table 2**.

Table 2. Characteristics of CMP-X.

Transport Hazzard Classification	UN/DOT 1.4S or better is anticipated
Total Vapor Pressure [psia, 20°C]	0.34
Partial Pressure of Hazardous Vapor [psia, 20°C]	0.027 (H_2O_2)
Vapor Toxicity - TLV / TWA [ppm] (Threshold limit value / time weighted average)	1 (H_2O_2)
Oral Toxicity - LD_{50} [mg/kg] (median lethal dose)	1000
PPE Required	Spill protection - gloves / goggles
Fuel Availability	>2M Metric tons COTS reagents produced annually
Price per kg [USD]	~$130
Thruster head materials	Non-refractory alloys (stainless steel)
Catalyst	$Ir–Al_2O_3$
Kinematic Viscosity [cSt, 20°C]	1.4
Minimum Operating Temperature [°C]	< –33
Typical Operational Mode	Continuous
Operational Pressure [psia]	60 – 280
Max Run Time [s]	$m_{propellant} / \dot{m}$
Max Theoretical Flame Temperature [°C]	950
Pre-Heat Temperature [°C]	220 (demonstrated as low as 150)
Cold Start Temperature [°C]	Incapable
Vacuum I_{sp}, measured (CMP-10) [s]	178
Propellant Density [g/cc]	1.15
*Density Impulse ($I_s d$), [g*s/cc, $\rho \times I_{sp}$]*	205
Volumetric Impulse, [N-s/liter]	~ 1060 †

INTEGRATED SYSTEM DEVELOPMENT

The CUA MPUC system features a pressurant-fed propellant, valved through a decoupling orifice and injected into a screen-retained granular catalyst bed. Bed compression is maintained by a pair of showerhead injector plates, a hard shoulder, and either a torqued hex jam nut or prescribed compression from engineered clearances taken up during assembly. After exiting the catalyst bed, the combustion gases enter the nozzle, where they are accelerated and exhausted, **Figure 5**. Various resistive heating solutions have been implemented to date, including nichrome wire, cartridge heaters, and most recently a band heater. The flight-like variant of this technology uses stainless steel cartridge heaters. Initial combustion testing of this propellant has been performed in Combustion Test Fixture (CTF) version “O3” (**Figure 6**). This CTF featured rapid reconfigurability with threaded inlet and exit fittings. Earlier “O” variants used a glow plug(depicted) for ignition assist, but this plug has since been removed and its port location left un-penetrated to help minimize leaks.

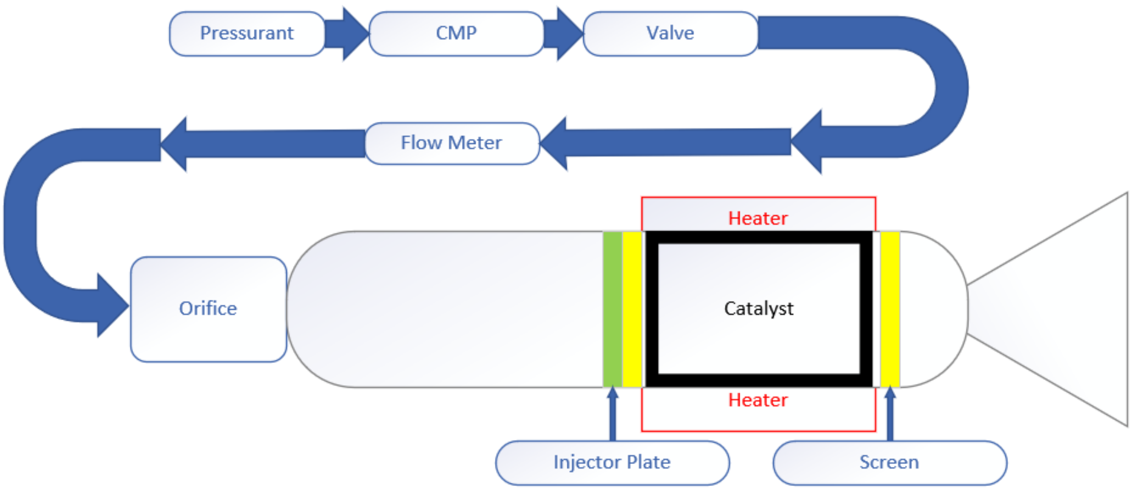

Figure 5. MPUC system diagram.

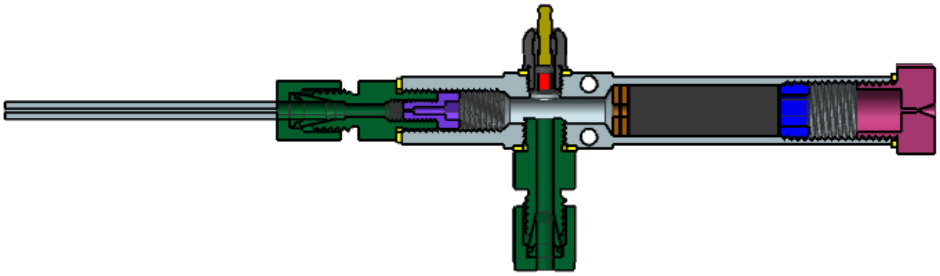

Figure 6. CTF O-3B (flow is left to right).

Early development efforts used granular manganese dioxide as a catalyst. Although plentiful, inexpensive, and robust during low-performance tests, this mineral predictably demonstrated poor stability and life once internal temperatures of the thruster approached their stoichiometric limits. Accordingly, an in-house version of Shell's widely implemented 405 catalyst was created and used for further testing [King, 1969]. Iridium was loaded onto a granular white alumina substrate and the resulting catalyst grains were sifted for size uniformity before loading into CTFs for further testing.

Sealing the inlet and exit fittings is accomplished by copper crush washers between the Series 316 stainless steel body and fittings. Over time, wear on these threads and surfaces became pronounced and a move to a copper-free seal solution was made. The nozzle feature was integrated into the main body in CTF-S [King, 2021], and eventually the inlet fitting was removed in favor of a simple and robust compression fitting connection, **Figure 7**. During bench testing, a five-element thermocouple rake is placed onto the CTF to obtain axial temperature profiles while firing. Note that the CTF-S fixture uses an unoptimized easy-to-manufacture nozzle for testing purposes.

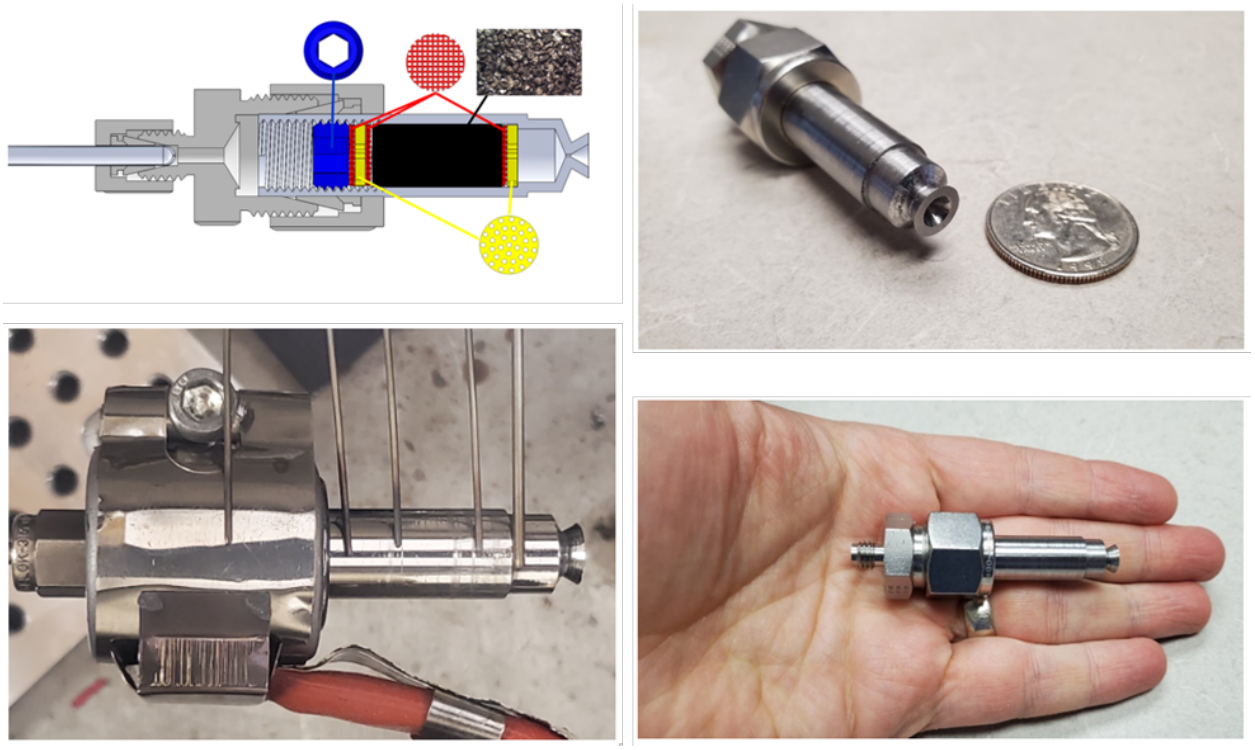

Figure 7. CTF-S (from upper left, clockwise - cross section of the thruster with inlet fitting, blue jam nut, red retention screens, and yellow injector plates; photos of the assembled thruster with items for scale; and the thruster ready for testing with the five-element thermocouple rake in place).

Run conditions were optimized in the CTF-S fixture until >10-minute, high-performance runs with steady fuel consumption were achieved. Decoupling pressure feedback from the thruster head into the feed system was the final step before proceeding to the “T” series of the CTF. This more flight-like thruster head leveraged advancement in additive manufacturing and was completely 3d-printed in 316-series stainless steel, **Figure 8** and **Figure 9**.

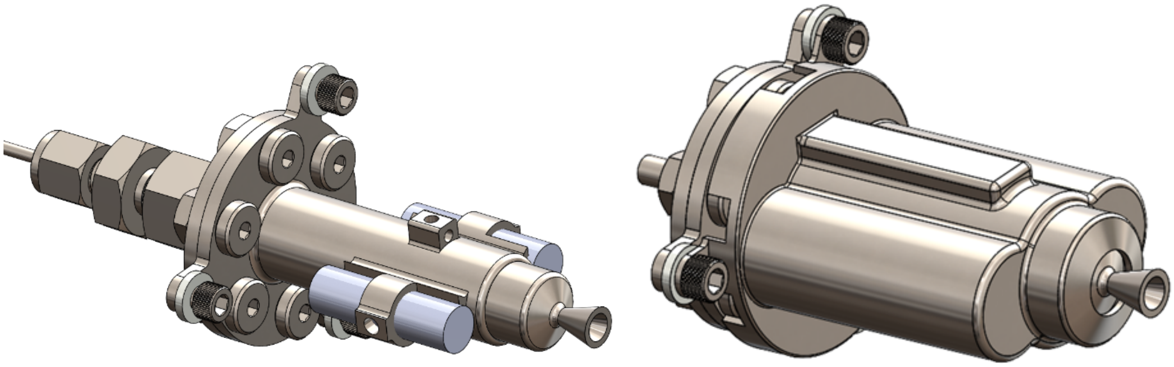

Figure 8. CTF-T without and with close-fitting heat shield.

Figure 9. CTF-T with integrated nozzle and flanged compression-fitting inlet

Further iteration with CTF-T located the optimum run conditions (**Table 3**) for its geometry, which approached theoretical maximum stoichiometric flame temperatures of ~910 °C. Performance measurements were carried out on CUA's Compact Thrust Stand [described in Wilson, 1997 and shown in **Figure 10**]. Thrust was responsive to fuel throttling (**Figure 11**).

Table 3. CTF-T nominal operating conditions

Thruster	Fuel	Mdot [mg/s]	Thrust [mN]	Isp [s]
CTF-T	CMP-Xu	118	202	175

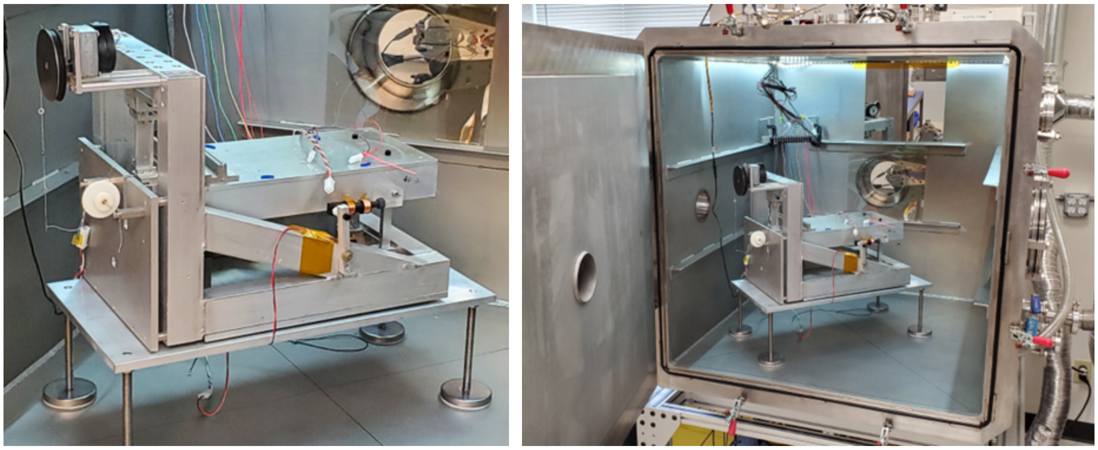

Figure 10. CUA compact thrust stand (left), and in CUA's ~0.8 m^3 vacuum chamber (right).

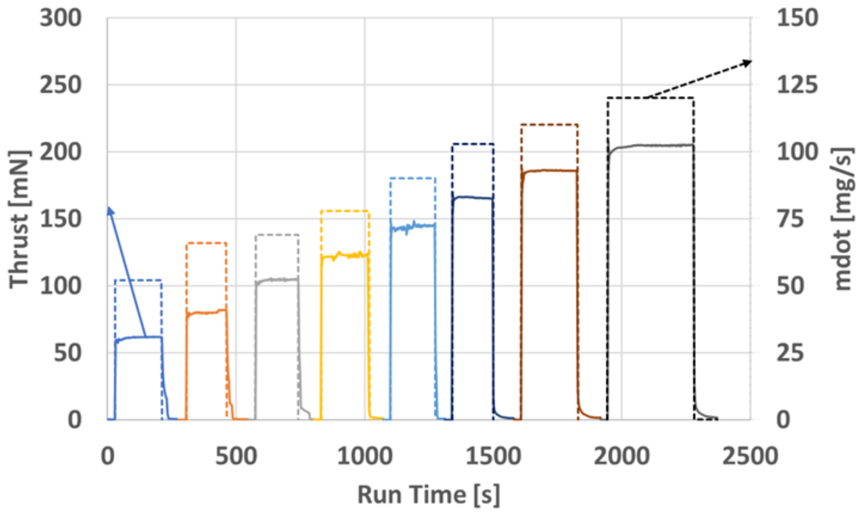

Figure 11. CTF-T14b thrust and mdot vs. time for throttle sweep test.

A long-life fuel load of CMP-Xu totaling over 900 mL (~1000 g) was burned in multiple back-to-back test fires with fuel cylinder refills (capacity ~225 mL) executed as quickly as possible between thruster operations. Nominal performance measurements for this higher-pressure, higher-flow rate condition are presented in **Table 4**. Performance remained steady for the first half of this life test, but began waning 5-6% over the second half, shown in **Figure 12**.

Table 4. CTF-T14 life test nominal operating conditions

Time	Thruster	Fuel	Mdot [mg/s]	Thrust [mN]	Isp [s]
beginning	CTF-T	CMP-Xu	154	270	179
end	CTF-T	CMP-Xu	140	232	169

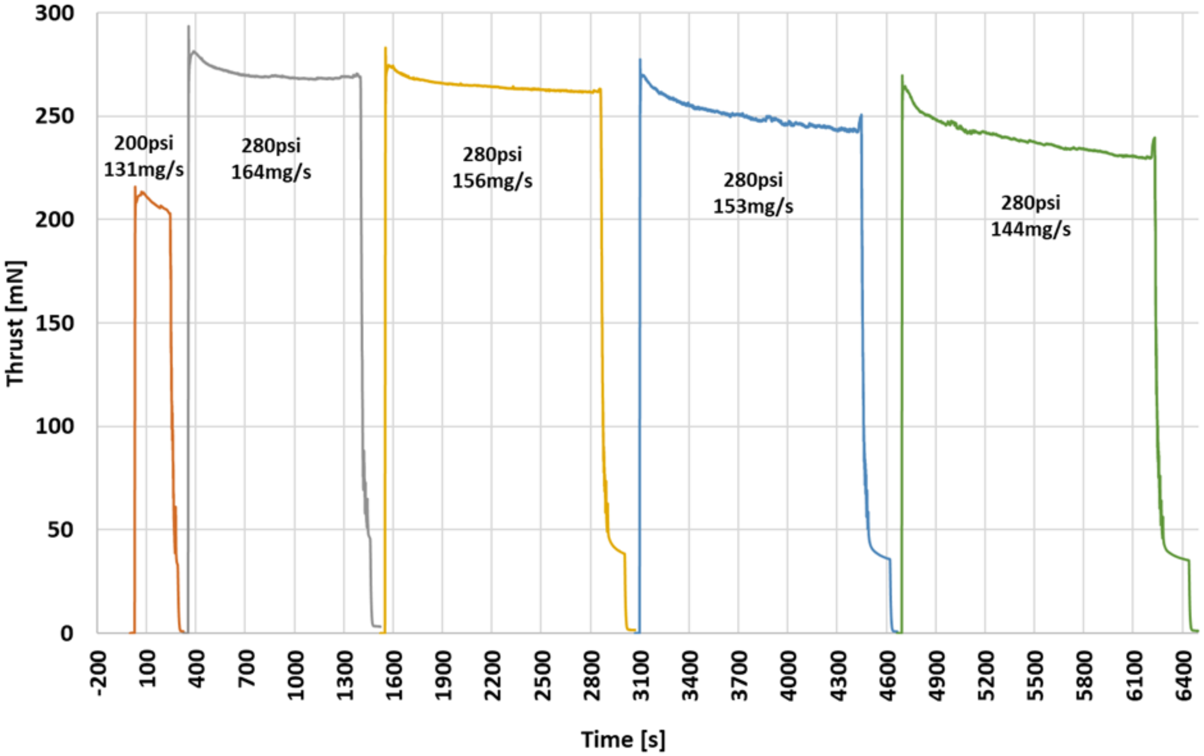

Figure 12. CTF-T long life testing excerpt with CMP-Xu.

Multiple causes are suspected for this late-test small percentage performance drop. In this breadboard configuration, the thruster rear seal (which will be welded on more flight-like hardware) is made from PTFE. This region should not experience the extreme temperatures present in the catalyst / nozzle regions, but are inherently coupled and may be failing to maintain a seal as test times lengthen. Additionally, post-test inspection reveals a small amount of “mortaring” between the spheres in the catalyst bed that may be presenting a flow restriction to the combustion gases.

INTEGRATED SYSTEM MANUFACTURE

Models of a 1.5U integrated CubeSat-sized flight MPUC feed system comprising a diaphragm-equipped stainless steel fuel tank of ~400 ml displacement, a small, inert pressurant tank, and relevant feed system components such as valves and internal control electronics were made, **Figure 13**. Valves from Lee Co. were chosen for the envisioned end-product system.

An optimized flight unit design will include all of the necessary components including dual-fault tolerance valving, pressurant/propellant tubing, bus mounting bolt hole pattern, simplified PCBs/connectors, and pressurant/propellant fill ports.

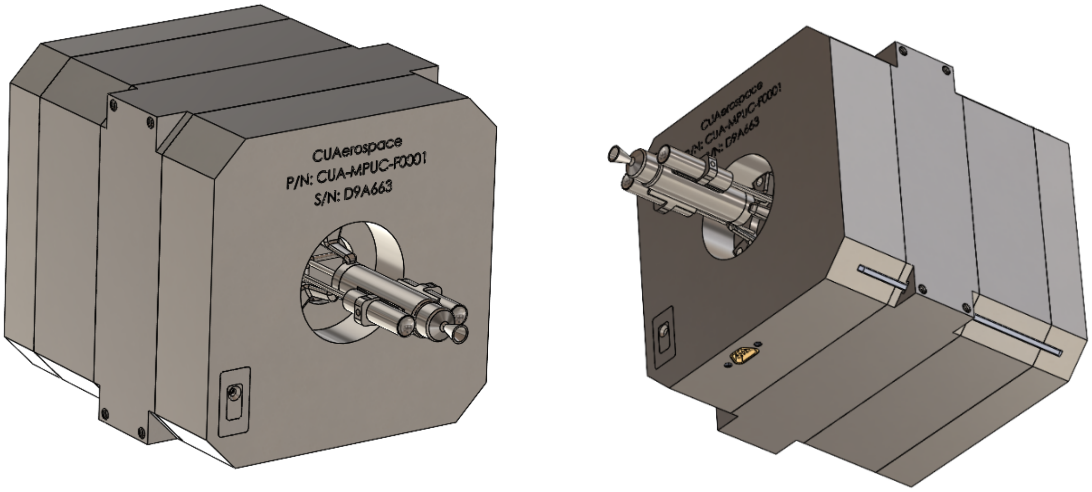

Figure 13. Trimetric view of CTF-T flight-like thruster head with flight feed system.

A custom control PCB was designed and manufactured to operate the necessary heaters and valves in the integrated system, **Figure 14**. A brassboard version of the MPUC system including a flight-like thruster head and brassboard feed system was manufactured and assembled prior to the end of the development program, **Figure 15**. Along with its custom electronics control box, the integrated unit was assembled on a small optical breadboard, **Figure 16**. A silicone diaphragm is used inside the pressurized feed tank, which was cast in-house, **Figure 17**.

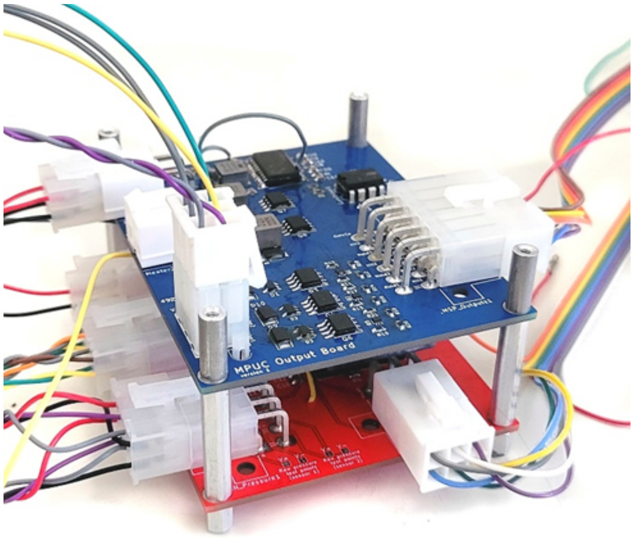

Figure 14. Custom MPUC control electronics.

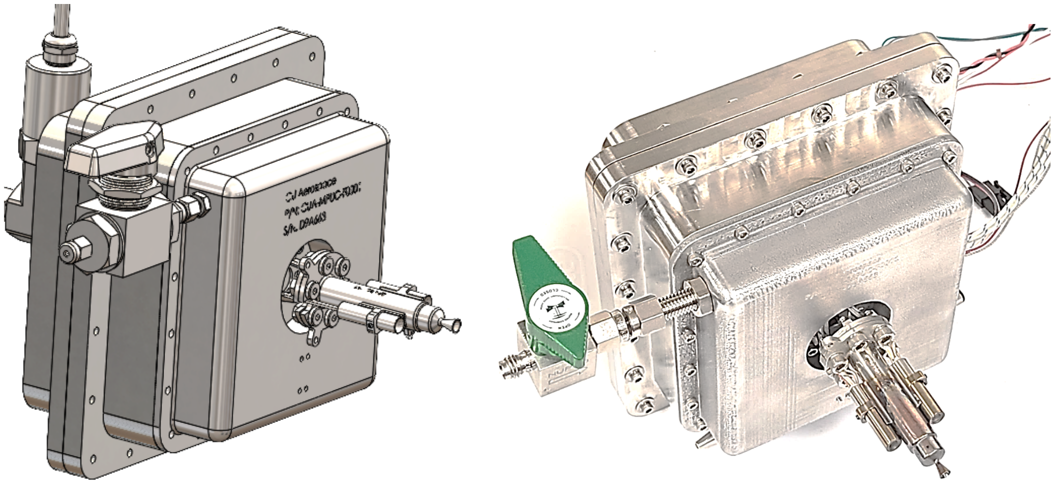

Figure 15. Integrated MPUC brassboard feed system with flight-like TRL 6 thruster head design (left) and as manufactured (right).

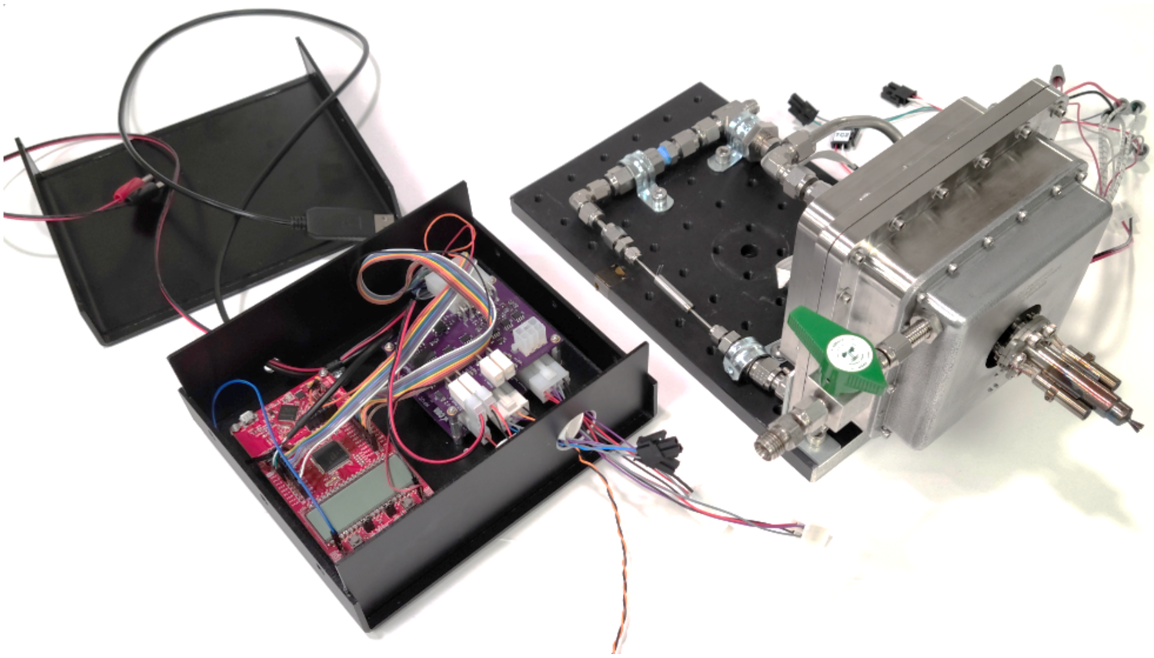

Figure 16. MPUC brassboard system and flow control electronics delivered to NASA.

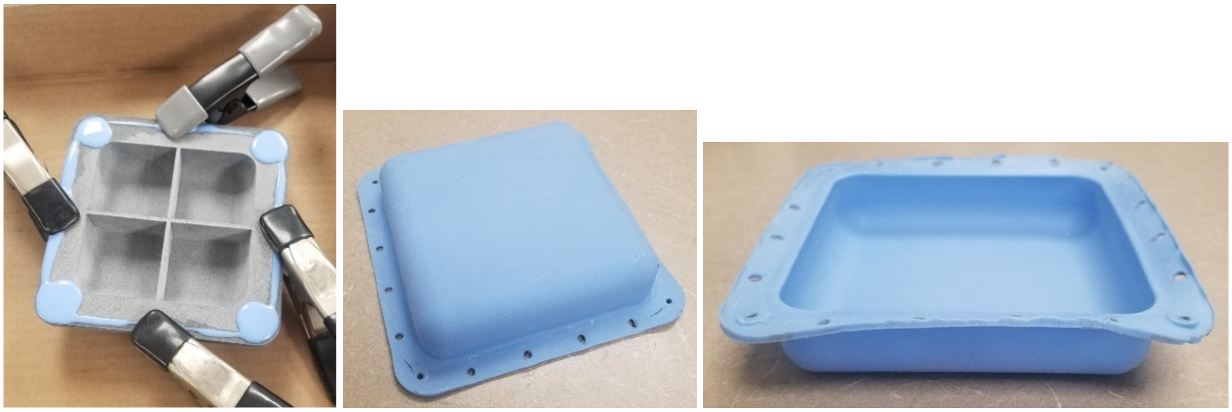

Figure 17. MPUC brassboard feed system silicone diaphragm mold (left) and result (middle and right).

The flight-like thruster head was subjected to an acceptance test procedure including inspections, functional tests, cold and hot fire tests, a 1.5-U fuel life-load consumption test, and thermal vacuum / GEVS-7000A vibrational (14.1 Grms) environmental test series. The testbed system diagram is depicted in **Figure 18**. In preparation for environmental testing of the flight-like T-14 thruster head, the head was loaded with catalyst, assembled, and pre-tested to establish baseline performance. It was then fixed to an adapter plate with mounting holes compatible with our environmental test service provider's equipment, **Figure 19**, and brought to the environmental testing facility. The plate was mounted in two planes and rotated once for 3-axis vibration, **Figure 20**, and simply placed inside a TVAC oven for thermal testing, **Figure 21**.

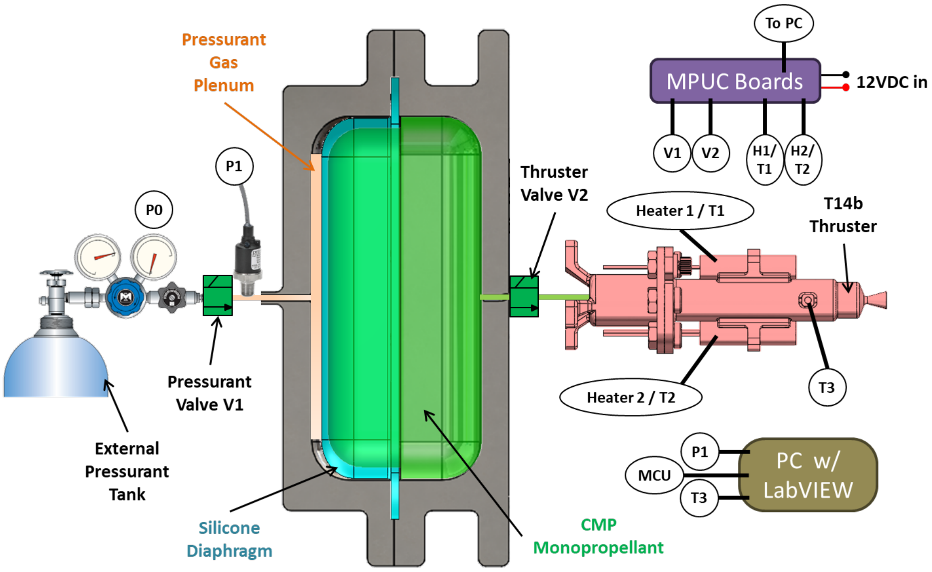

Figure 18. Brassboard MPUC system diagram.

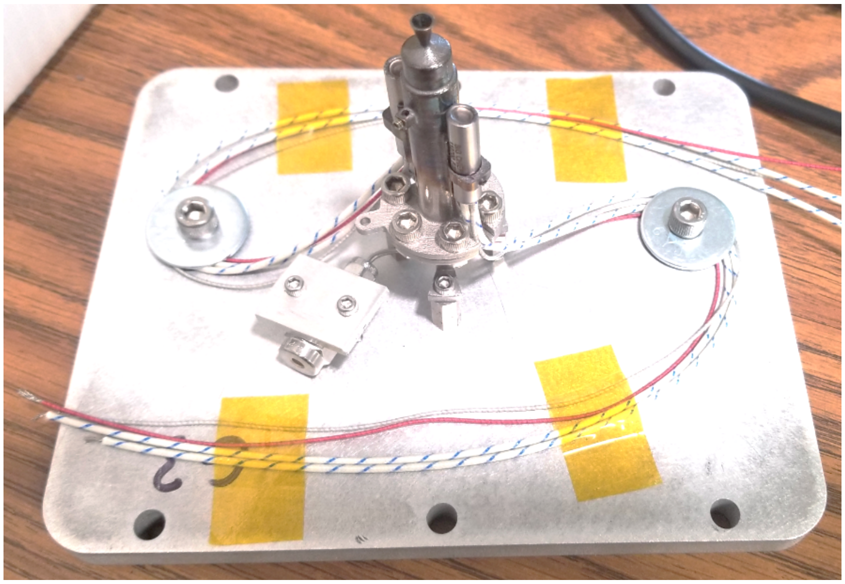

Figure 19. T-14 mounted to base plate ready for vibe and TVAC.

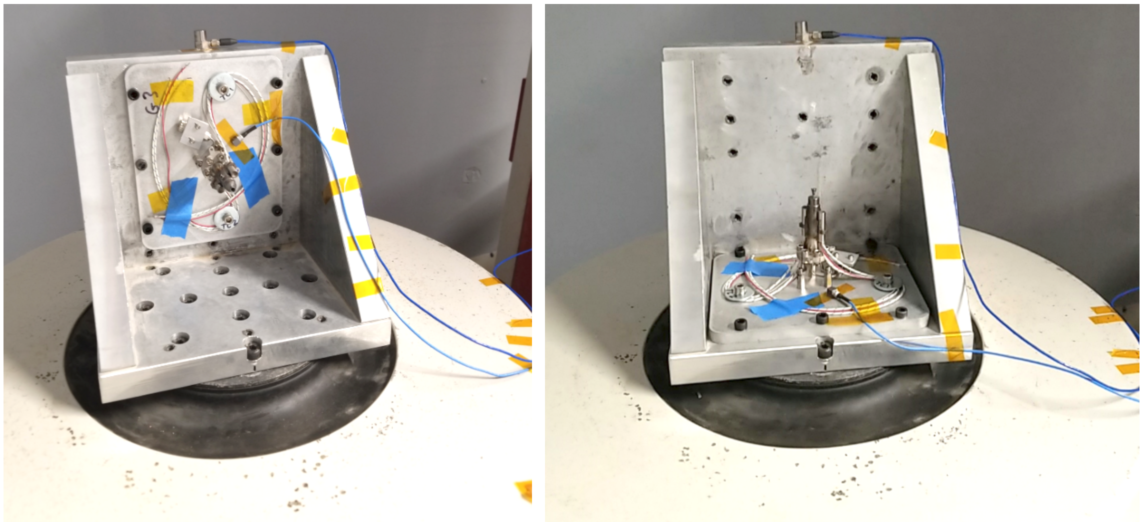

Figure 20. T-14 mounted to base plate ready and fixed to vibe system.

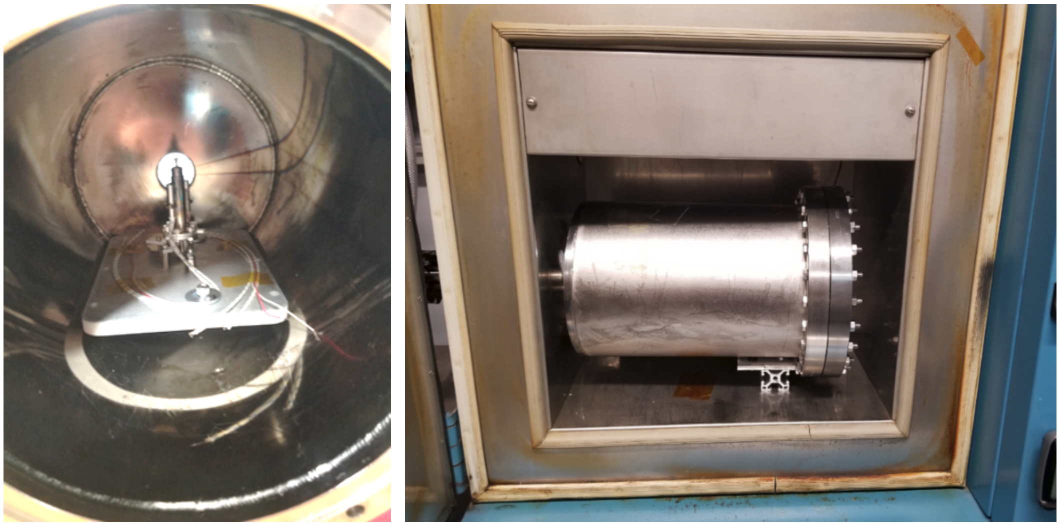

Figure 21. T-14 mounted to base plate ready and inserted into TVAC chamber.

The thruster head survived environmental testing and to within experimental error demonstrated repeatable before pre- and post- TVAC and vibe. A sample thrust trace of the system's pre- and post-environmental performance is shown as **Figure 22**. A long-duration thrust trace of the system post-environmental testing with a brassboard life-load of propellant is shown as **Figure 23** demonstrating very stable thrust over the life of the 410 g brassboard propellant load.

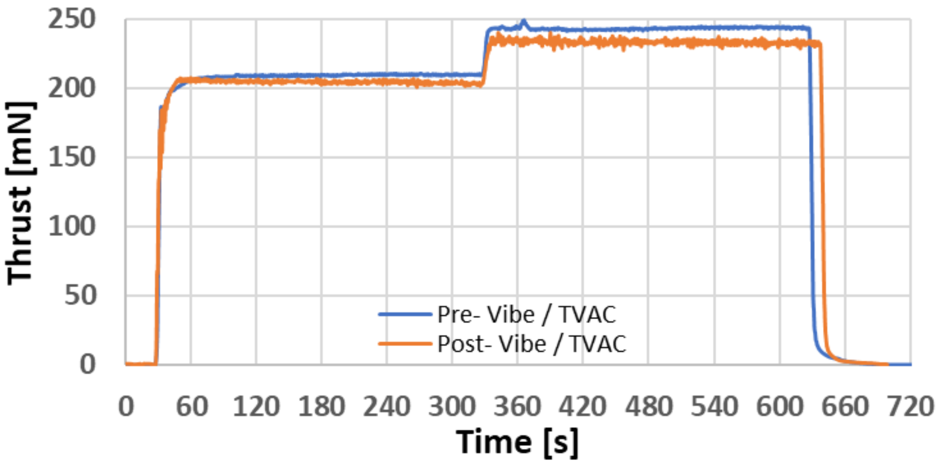

Figure 22. MPUC nominal thrust [mN] vs time [s] trace for T-14b thruster integrated with brassboard feed system (throttle up at t=330s from 200 to 240 psi feed pressure).

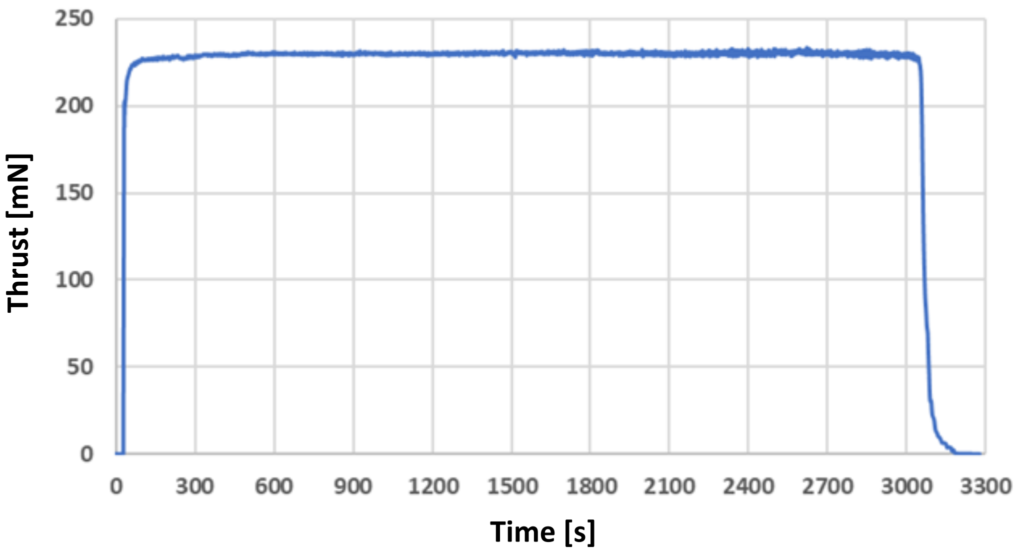

Figure 23. MPUC nominal thrust [mN] vs time [s] trace for T-14b thruster integrated with brassboard feed system (200 psi external feed pressurant).

PERFORMANCE ANALYSES

A critical factor in the future success of the thruster lies in identifying potential mission profiles and tailoring the preliminary thruster design to satisfy as many of the mission requirements as possible. We anticipate on-orbit life of the Phase III thruster at > 24 months. A sample 6U, 10kg satellite with a 2U MPUC containing a CMT would see a total ΔV of 226 m/s. This assumes a 10 kg total satellite mass, ~178 s specific impulse, and 1214 g fuel. The same satellite beginning at a circular orbit of 370 km could have its orbit raised to ~785 km, or could begin at 910 km and end at ~484 km, allowing the satellite to meet the 5-year FCC deorbit requirement.

Based upon potential customer/user feedback, CUA chose to develop a ΔV-only design (without ACS) in Phase II from the following rationale:

- Almost all CubeSats now use reaction wheel/torque rod packages for ACS already
- Largest market for CubeSats is LEO where torque rods are used to desaturate reaction wheels
- Additional valving for ACS adds complexity, reduces tank volume for ΔV maneuvers, and adds considerable cost per unit
- Greatest need for propulsion is therefore pure ΔV

There are significant missions such as Mars Cube One (A&B), BioSentinal, NEAScout, and CAPSTONE to name a few that could have benefited from an MPUC system. More generally, lunar exploration precursor missions, ΔV systems in LEO or deep space, or SLS ARTEMIS-deployed CubeSat Moon missions could utilize MPUC technology. However, until the MPUC system is increased in TRL (the result of this effort), it will be difficult to align with any specific mission. **Table 5** presents a list of performance capabilities for different MPUC system sizes. Note that the package volumes listed assume the length of the thruster head can be housed, for example, inside a tuna-can-sized deployer spring in the CubeSat deployer.

Table 5. Estimated MPUC performance capabilities.

PARAMETER / PROPULSION SYSTEM	MPUC (1.5U)	MPUC (2.0U)	UNITS
Thruster System Package Volume	1,500	2,000	cm^3
Propulsion Technology	Monopropellant	Monopropellant	–
Propellant	Low Toxicity	Low Toxicity	–
Warm-Up Power Draw	12	12	W
Nominal Power Draw	1.5	1.5	W
Specific impulse	> 175	> 175	sec
Thrust	> 200	> 200	mN
Total impulse	1407	2122	N-s
Vol. Impulse (total impulse / system vol)	938	1061	N-s/liter
Propellant Mass	805	1214	g
Propulsion System Wet Mass	2446	3250	g
Delta-V capability (**S/C Mass = 5 kg**)	307	486	m/s
Delta-V capability (**S/C Mass = 10 kg**)	147	226	m/s
Delta-V capability (**S/C Mass = 12 kg**)	121	186	m/s

COLD START INVESTIGATION

CUA demonstrated a nominal-preheat run to baseline lower-temperature start conditions. The flight-like thruster candidate, “T-14b”, was used for this test series. Operation was started normally, with a nominal pre-heat target of 200 °C. After demonstrating successful ignition and operation, fuel flow is stopped, and the thruster is allowed to begin cooling.

In this state, both in-heater thermocouples are monitored and the system temperature is allowed to fall to increasingly lower pre-heat temperatures. A summary table of this investigation is shown in **Table 6** for both CMP-8 and CMP-X propellants. Early experiments indicated that CMP-X would not be favorable for cold start operation, so this study focused on the more energetic CMP-8 monopropellant mixture.

We were unable to achieve true cold start with either CMP-X or CMP-8. CMP-8 previously achieved cold start with the addition of an ignition plug into the catalyst bed, but required pulses of fuel injection to gradually raise the temperature of the catalyst bed to a higher temperature where fuel could be run continuously and ignite. Without the ignition plug, CMP-8 also could not achieve a true cold start.

Table 6. Summary of reduced-preheat test series.

	Ignition Achieved?	
$T_{preheat}$ [°C]	**CMP-8**	**CMP-X**
250	Yes	Yes
200	Yes	Yes
180	Yes	Yes
150	Yes	Not Tested
130	Yes	Not Tested
110	Yes	Not Tested
90	Yes	No
85	No	No
70	No	No

MODELING RESULTS TO EVALUATE FLAME TEMP

To evaluate the flame temperature of the flight-like MPUC thruster head, CUA utilized its internally developed BLAZE Multiphysics™ Simulation Suite [Palla, 2011] <http://www.blazemultiphysics.com> in order to construct additional high-fidelity simulations of the MPUC micro-nozzle. BLAZE is comprised of a number of inter-operable and highly scalable parallel finite-volume models for the analysis of complex physical systems dependent upon laminar and turbulent fluid-dynamic (incompressible and compressible subsonic through hypersonic regimes), non-equilibrium gas- and plasma-dynamic, electrodynamic, thermal, and optical physics (radiation transport and wave optics) using any modern computational platform (Windows, Mac, Unix/Linux). BLAZE is compatible with a number of free, open-source, yet commercial quality grid generation and post-processing software packages which greatly reduces training and operating costs. BLAZE is also compatible with state-of-the-art commercial grid generation and post-processing solutions. BLAZE was previously utilized to provide a more detailed understanding of the performance of the MPUC nozzle and aid in design to minimize the impact of the boundary layer and maximize nozzle efficiency [King, 2021].

The flight-like nozzle selected was conical with an area ratio of 100 with a 15° divergence angle, **Figure 15**. The throat diameter was 0.03556 mm (0.014”) and the acceptance test flow rate was 144 mg/s of combusted CMP-X. A series of BLAZE simulations were performed with different flame temperatures (**Figure 23**) in an effort to deduce the actual flame temperature being produced in the combustor. Based upon experimental data, a thruster wall temperature of 600 K was used. For simplicity, thermal expansion of the nozzle was ignored in these simulations. This series of simulations suggests that the flame temperature in the flight-like combustor is approximately 1175 K, which is 94.5% of the theoretical maximum (a reasonably efficient reaction fraction for a micro-combustor).

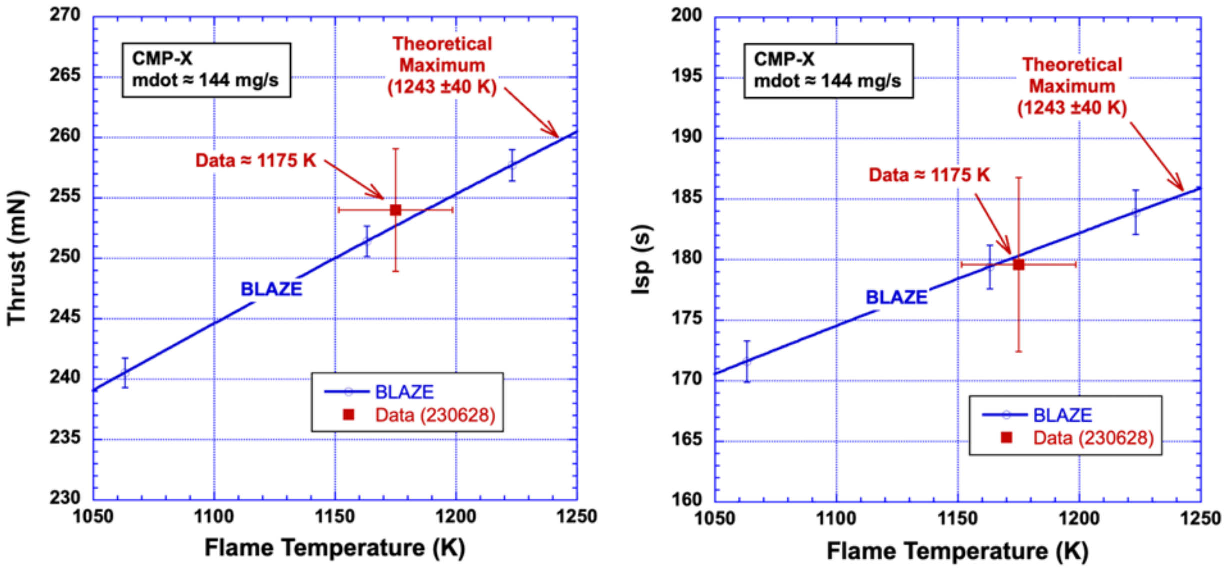

Figure 23. BLAZE predictions of MPUC thruster performance as a function of flame temperature: (left) thrust, and (right) exit Isp. The conical nozzle had a 0.014" throat, an area ratio of 100, and a 15° divergence angle. Simulations were run for CMP-X with a flow rate of 144 mg/s and a wall temperature of 600 K. (Note: the uncertainty in the theoretical maximum flame temperature arises from uncertainties in the gas mixture Cp at elevated temperatures).

SUMMARY AND CONCLUSIONS

CU Aerospace (CUA) demonstrated the scaling of an alternative very low-toxicity monopropellant thruster to a 0.5 N class engine. CMP-X is a non-detonable yet energetic COTS formulation that possesses many system-level advantages including lower cost (COTS propellant and non-refractory thruster construction), lower thermal load (~900°C flame temperature), water-like viscosity, and common materials compatibility (aluminum, stainless steels, and most elastomers). The flight-like CMP-X thruster demonstrated 178-200 s specific impulse at 230-270 mN thrust during thrust stand testing and continuous firing times > 55 min. Recent work demonstrated an improved catalyst for CMP-X with higher reactivity and longer life with minimal warmup time in a prototype unit that can be scaled to >500 mN. Both stabilized and unstabilized feedstock formulations of CMP-X have demonstrated shelf life exceeding 1200 days. Studies included UN/DOT Series 6 testing to establish a formal hazard classification of CMP-X (permitted on common commercial air transport such as UPS), further catalyst risk reduction studies / characterizations, and brassboard feed system development. CMP-X is designed not for highest performance Isp, but as a monopropellant option for customers who can accept a modest 20% performance penalty (relative to AF-315E and LMP-103S) for the advantages of lower cost, air transportability, considerably fewer range safety concerns, lower thermal soakback into the spacecraft, and longer continuous thrust burns. The estimated total impulse of a 2U-sized flight MPUC is ~2120 N-sec with an operating power draw of ~1.5 W and ~180 s specific impulse.

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