A 6U HYBRID PROPULSION MODULE FOR CUBESATS

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ABSTRACT

A 6U hybrid propulsion module is presented. It is designed to be used as a stage for a 6U CubeSat with a combined wet mass up to 25 kg. Hybrid rocket propulsion is utilized because it is responsive, high performance, capable of multiple burns and safe/non-toxic. Hybrid rockets employ a solid fuel and liquid or gaseous oxidizer. In this case, the fuel is PolyMethyl MethAcrylate and the oxidizer is gaseous oxygen. This propellant combination produces an Isp of greater than 300 seconds. The gaseous oxidizer simplifies the design and is used in dual mode with cold gas thrusters for attitude control. In general, hybrid propulsion is higher performance and higher thrust than conventional options considered for CubeSats (e.g. monopropellants), making it ideal for larger manuevers such as plane changes. The Attitude Control System has been sized to ensure that the CubeSat can be controlled under the thrust loads of the propulsion system. A hybrid rocket motor of this size is currently being tested at the Jet Propulsion Laboratory. Thirty-four successful tests have been completed at JPL with this propellant combination during the last 2.5 years. Current tests include long duration tests (> 95 seconds) and multiple starts (five). These tests and applicability to potential mission requirements will be discussed. Future test plans to move the motor into a vacuum facility to achieve a Technology Readiness Level of 5 by the end of September 2018 will be discussed, and an overview of the proposed TRL 5 to 6 maturity plan will also be presented.

INTRODUCTION

As CubeSats are considered for more data and science driven applications, the system requirements have become more challenging. Propulsion systems are being investigated for many of these CubeSats in order to provide attitude control and larger ΔV maneuvers. Current State of the Art (SOA) CubeSat propulsion systems are gas systems (MARCO), but chemical (e.g. monopropellant) and electric propulsion options are also being considered.

A hybrid rocket typically consists of a solid fuel and liquid oxidizer. Oxidizer is introduced into the combustion chamber, where the fuel is stored. A small amount of fuel is heated and vaporized. Combustion is self-sustaining until the oxidizer flow is shut off. Hybrid rockets can be ignited multiple times, an advantage that will be leveraged in this design. Hybrids find a special median between conventional liquid and solid propulsion systems. They enjoy performance similar to liquid bipropellant systems, with only half the complexity. In the design presented in this paper, using gaseous oxidizer further reduces the complexity of the feed system.

Many non-toxic or green options are available for the hybrid propellants. Typical fuels include: waxes, rubbers, and plastics. For this application, PolyMethyl MethAcrylate (PMMA) is used. PMMA, also known as acrylic, is used for many common applications including paint, eyeglass lenses, aquariums, airplane windows, hockey rink spectator protection, and artifical nails. Common non toxic oxidizers include oxygen and nitrous oxide. Gaseous oxygen was selected for ease of use and extensive flight heritage. Alternatives used in other applications include: the toxic Mixed Oxides of Nitrogen (N_2O_4 + NO) and (the not fully stable) hydrogen peroxide (85-90+%).

There are many benefits of using a hybrid propulsion system for small satellites. Small satellites are typically considered for rideshare opportunities on launch vehicles. Therefore, safety is paramount. The fuel and oxidizer are stored in different phases and are physically separate from each other. This makes it difficult to obtain a reactive mixture before it is desired. There is also flexibility in the oxidizer tank size and shape, giving packaging alternatives to the designer. For example, a single long, slender tank is possible if that aspect ratio is desired, or, as is the case with the current design, the oxidizer can be split into multiple tanks and placed around the motor for a more compact packing arrangement.

The propellant combination presented here is both space and Earth storable. The oxygen will remain in the gas phase above ~ 90.2K. Therefore, the tanks only need to be sized for the maximum expected temperature, and no heating is required. The PMMA is below its glass transition temperature at Earth ambient condition, where all the testing to date has been carried out. No adverse effects of combustion below the glass transition temperature have been observed.

Potential applications of this small-scale hybrid system include orbit insertion, change of orbit, breaking burns, flybys, and deorbit burns. [1] The hybrid is especially advantageous for these larger/longer burns because of its increased thrust compared to typical monopropellant systems. Larger hybrids are being considered for planetary ascent.[2]

REGRESSION RATE

The regression rate, or rate at which the fuel burns/regresses with time during combustion, is one of the most important design parameters for a hybrid motor. The regression rate is given by Equation 1,

$$\dot{r} = a_o G_{ox}^n$$

where $\dot{r}$ is the regression rate, G_{ox} is the oxidizer mass flux (oxidizer mass flow rate divided by the instantaneous cross-sectional area of the fuel grain port), and a_o and n are empirically derived constants. The constants a_o and n are unique to each propellant combination. Regression rate parameters (a_o and n in Equation 1) were determined through testing at JPL.

While data exists in literature for the propellant combination selected here[3], it was not found to be consistent with the testing carried out at JPL. PMMA can vary in chemical composition between sources, so this is seen as the potential cause for this difference. The PMMA used in these tests is from McMaster Carr (clear) and Professional Plastics (blackened) and purchased as 2-inch diameter extruded rods. This will be discussed further in the results section of this paper.

6U PROPULSION MODULE DESIGN

The design presented here is intended to be a 6U module to be used with a larger spacecraft. The payload capability is discussed in Table 1. In this case, the payload is taken to be all non-propulsion system dry mass. This could be avionics, telecom, science instruments, etc. as well as all the structure needed to support those items. The propulsion system is self-contained except for the avionics and battery, which is shares with the main system to reduce mass. The 6U module uses oxygen cold gas systems for Thrust Vector Control (TVC) and Reaction Control System (RCS). The TVC system uses four 1.1 N thrusters and RCS is made up of eight 0.05 N thrusters. These operate in dual mode meaning that the gaseous oxidizer is used as the propellant in these cold gas thrusters. Two percent of the useful propellant goes towards TVC and 158 grams of oxygen is available for the RCS.

Table 1: Payload and ΔV capability dependence on total Wet Spacecraft Mass

Total Spacecraft Mass (Wet)	15 kg	20 kg	25 kg
Payload (Available Non-prop Mass)	3.4 kg	8.35 kg	13.36 kg
ΔV	540 m/s	400 m/s	315 m/s

As expected, the geometric constraints are the most challenging for the CubeSat hybrid propulsion system., Typically, 1U is taken to be a 10 cm by 10 cm by 10 cm cube. So, 6U would be 30 cm by 20 cm by 10 cm. However, as the platforms for launching these payloads are becoming available, the form factor has increased slightly. The 6U system is taken to be the maximum volume that can be provided by the launch system or 36.6 cm by 23.9 cm by 11.6 cm.[4] Most of the volume is taken up by two oxidizer tanks and the motor. There is ~ 2 cm on top of the tanks for the components, ~ 1 cm on the bottom of the system for the RCS and ~ 3 cm available above the motor for the oxidizer line and main valve. Miniature components are used where ever possible to minimize both mass and volume.

The design incorporates one motor down the center of the volume and two main oxidizer tanks, holding gaseous oxygen at high pressure (10,000 psi). The total dry mass of the 6U propulsion module is 8.33 kg. This includes 5.67 kg of components (including margin). Primary structure accounts for 15% of the spacecraft wet mass. The propulsion system dry mass has been sized at a component level with an additional 25% added to the total dry mass for secondary structure, thermal and cabling. AIAA S-120 margins are applied to all estimated masses.[5]

The hybrid motor is located in the center of the 6U module. It is 7.6 cm in diameter by 28.6 cm long plus a 40:1 area ratio nozzle. The usable fuel is about 2.4 kg at an oxidizer to fuel mixture ratio of 1.6. The mean chamber pressure is 200 psi, with the predicted chamber pressure shown in Figure 1. The propulsion system relies on the main avionics and battery for power and control, but it otherwise self-sufficient. This is to minimize the system mass.

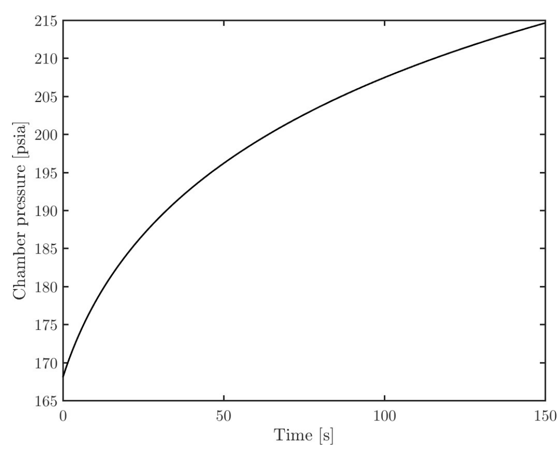

Figure 1: Predicted chamber pressure for the flight design.

Power is a scarce resource on CubeSats; the design has leveraged low power options wherever possible. The propellant was especially chosen to ensure no heating power would be required during the mission. The propulsion system masses have been estimated assuming a 28 V bus. The switch to a 12 V bus, which is often desirable for CubeSats, can be made for a small mass penalty. The most power intensive component is the ignition system, which requires a spark plug. However, operation of the spark plug is limited to ignition, and thus will only consume power for fractions of a second during each burn. It is anticipated that the CubeSat will not be doing science at the times of its burns, so this power requirement will not be an issue.

TEST SETUP

Testing for this propulsion system has been completed at JPL (see Figure 2). The feed system for gaseous oxygen, nitrogen purge, and methane/oxygen igniter systems are shown. This set up has been described previously.[6,7]

Approximately 30 tests with this propellant combination have been carried out from 1 to 95 s in duration. The goal of this hotfire testing is to demonstrate the performance of the motor and to mature the CubeSat design. This included several objectives: confirming (or in this case revising) the regression rate coefficients, demonstrating multiple restart capability, confirming the feasibility of burning the fuel to near depletion (small residual), and high performance. These objectives have been met and the results will be described in the following sections.

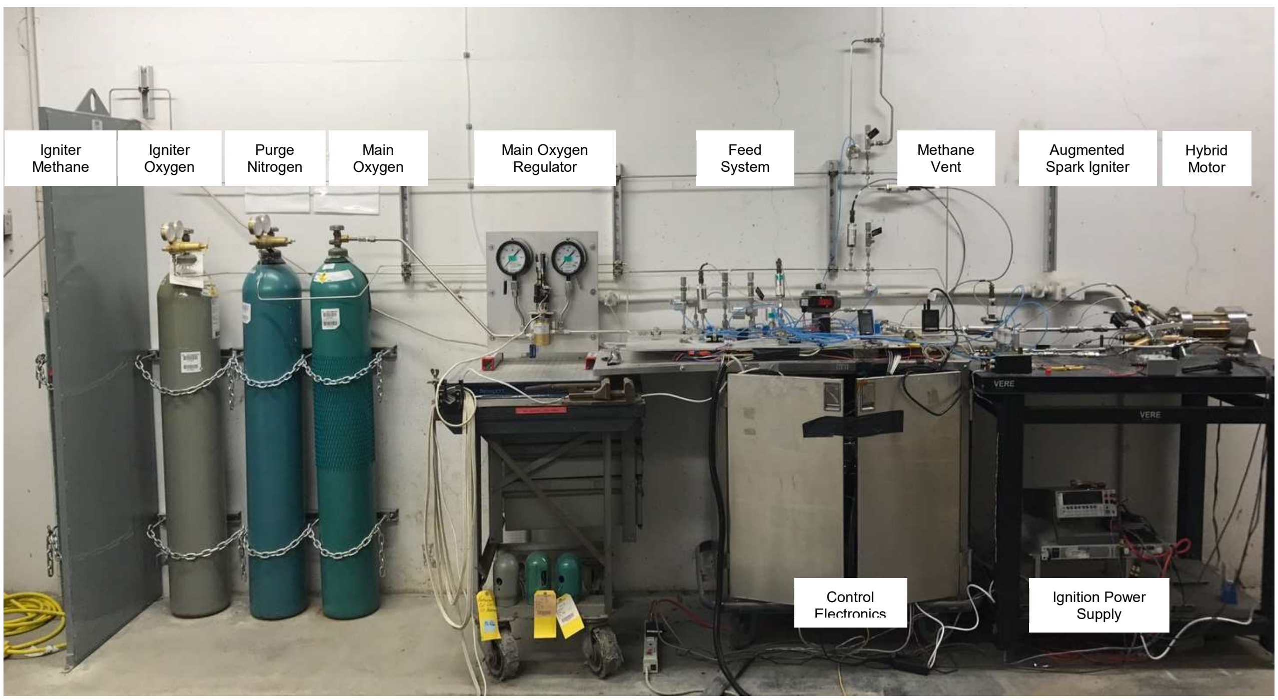

Figure 2: Hybrid CubeSat Propulsion System Test Set Up at JPL.

RESULTS DISCUSSION

Test results were obtained to confirm the performance necessary to meet flight objectives including regression rate, multiple starts, fuel utilization, and performance.

REGRESSION RATE

The empirical constants (Equation 1) for PMMA/GOx were found to be: a_o = 9.40 x 10^{-5} and n = 0.34 (using SI units for the regression rate and oxidizer mass flux).[8] This combination results in a progressive burn, meaning the thrust increases over time in a predictable manner. These regression rate parameters differ significantly from those presented previously in literature (a_o = 2.11 x 10^{-5} and n = 0.615 in Reference 3). The higher regression rate observed at JPL makes a single, wider motor possible for this application, where previously, multiple motors had been considered. This is a much more robust solution for a CubeSat propulsion system because it is substantially easier to control since all the thrust is aligned with the center of the vehicle.

MULTIPLE STARTS

Five ignitions have been demonstrated autonomously on a single motor. This has now been confirmed through two separate tests. Figure 3 shows the chamber pressure from the earlier of these tests. Each start is stable and repeatable. The number of starts is not a limit to the system, but just the initial goal set for the system. Depending on the final application, more or fewer starts would be possible. This testing was intended to give confidence to the requirement for multiple burns. A qualification program will be necessary for the final flight design.

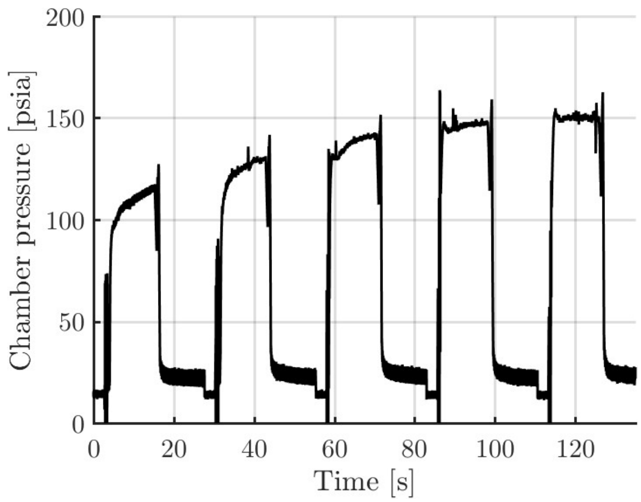

Figure 3: Multiple starts during the same test (Test 69). Plot originally published in Ref. Error! Bookmark not defined.**.**

FUEL ULITIZATION

Two near full duration tests have been completed to date. The first was 95 s (commanded) and represented 94.3% fuel utilization in the 2 in diameter motor. A second test of 95 seconds, spilt up between 5 burns, achieved 97.5% fuel utilization. The injector was modified between the two tests in order to allow the oxidizer to more completely mix with the fuel. As the diameter grows in the flight like configuration, longer burn times will be enabled. Figure 4 shows the combustion chamber pressure from the single burn, long duration test. The progressive nature of this propellant combination can clearly be identified in this plot as the pressure increases noticeably over time. The rapid decrease in pressure at about 80 seconds is caused by the fuel burning out at the aft end of the combustion chamber before being consumed at the front of the motor. Approximately 5.7% of the fuel remained after the test, all at the fore end of the motor. At that point, insulation begins to burn over an increasing area of the fuel grain as the fuel burns forward. Work at JPL has been conducted to improve the uniformity of the burn and thereby minimize the length of time during which the insulator is exposed.

The injector in this first long duration test only had a single hole and no effort was made to diffuse the oxidizer flow. Therefore, it appears that the oxidizer flow was not reacting with the upstream section of the fuel grain. A simple change in injector enabled a fuel residual of only 2.5%. The current design assumes 3% fuel residual. Therefore, it has been shown that this level can be met for flight. Optimization of the injector could continue to improve the fuel utilization.

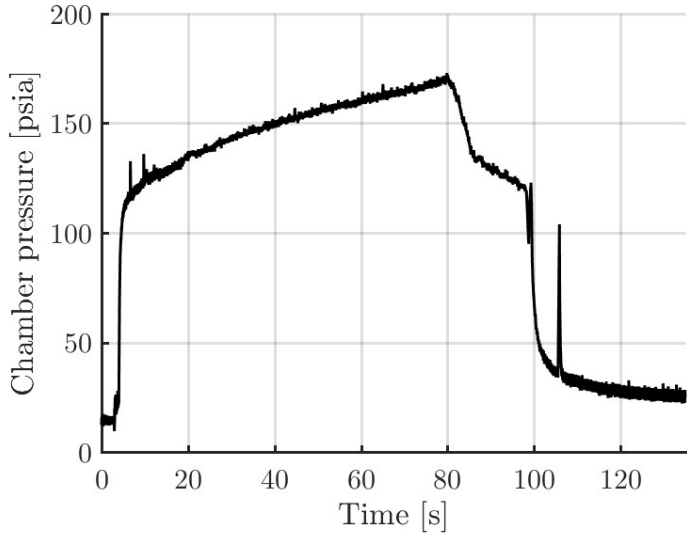

Figure 4: Long Duration Test (Test 73). Decrease in chamber pressure at 80 s is due to the fuel burning out at the aft end of the combustion chamber before the fore end.

SPECIFIC IMPULSE

The design Isp for the 6U propulsion module is 311 s, which requires an efficiency of 93%, assuming equilibrium combustion. The ideal vacuum Isp for this propellant combination is achieved at oxidizer to fuel ratio, O/F, of 1.6 and is 334.5 s. There will be a slight change in Isp during the burn as shown in Figure 5. The mean ideal Isp during the burn is 332.1 s, and thus the O/F shift during the burn will reduce the ideal specific impulse by less than 1%. The assumed 93% efficiency accounts for this O/F shift as well as combustion efficiency. Testing at JPL has demonstrated combustion efficiencies in the range of 80-95% and has therefore validated the design assumption of 93%.

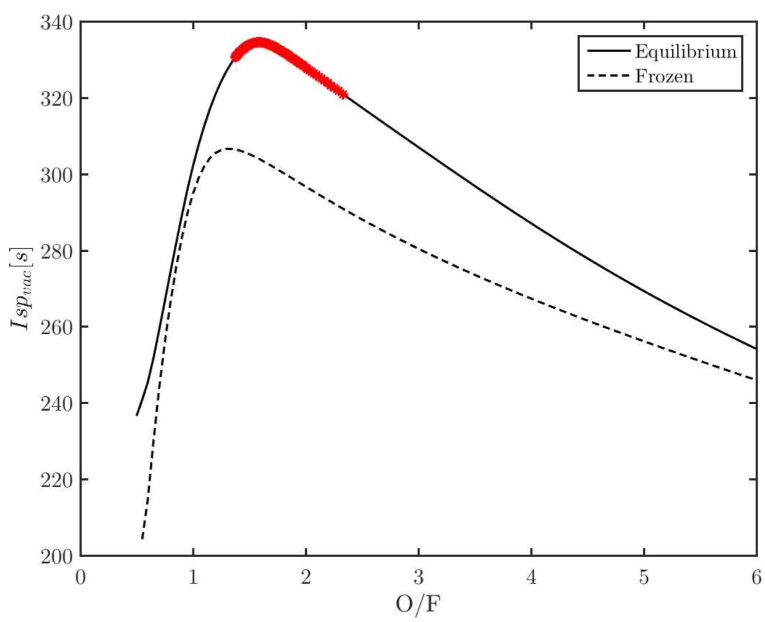

Figure 5: Expected ideal vacuum specific impulse versus oxidizer to fuel ratio for this propellant combination with an area ratio of 40 and a chamber pressure of 200 psi. The expected conditions for this design are shown in red. This plot is produced with Chemical Equilibrium with Applications[9]

COMPARISON TO STATE OF THE ART

A comparison of the hybrid system to the current monopropellant state of the art options for CubeSats is presented in Table 2. The MARCO CubeSat propulsion system was mentioned earlier because it is the current SOA CubeSat propulsion system. MARCO is scheduled to launch with the NASA InSight mission, whose launch window opens on May 15, 2018. However, this simple system is very low performance compared to the alternatives, MARCO has a total impulse of 755 Ns and is much smaller than the system presented here (approximately 20.5 cm by 15 cm by 4 cm).[10] For a more competitive comparison, the table only includes monopropellant options. These simple systems are ideal at small scale. What they lack in performance, they make up in simplicity and therefore dry mass. The table is color coded, where green is desirable, yellow is moderate and red is not desirable.

Table 2: Hybrid CubeSat Propulsion Compared to the State of the Art

	Hybrid (PMMA/GOx)	**State of the Art**		
		Monopropellant		
		Hydrazine	HAN 10% water (AF-M315)	ADN (LMP-103S)
Available Non-Propulsion Mass/ Payload	16.67			
Main Advantage	High Isp/Thrust	TRL	Higher Density than hydrazine	
Isp (Performance)	311 s	227 s	213 s	233 s
Thrust [N]	50	21	20	20
Delta V for 12U, 25 kg	315	148	61	89
Prop System % Volume	61.4% of 6U	53.7% of 6U	50.5% of 6U	47.1% of 6U
Survival Temperature [C] (5 C margin)	~-150	8	-75	-85*
Toxicity/Issues	Nontoxic	Toxic	Acidic, titanium components only	Similar to Ammonia, Precipitate is 1.4 explosive
Single Burn Impulse [Ns](e.g. 1 hr OI burn)	5,050	2,520	4,950	2,640

FUTURE WORK

As of the time of this paper, this technology is at TRL 4: component validation in laboratory environment. The immediate goal of this technology development program is to increase the Technology Readiness Level (TRL) of the CubeSat hybrid propulsion system to TRL 5: component validation in a relevant environment. This will be completed during the summer of 2018 through short duration vacuum tests at about 10 torr.

A light-weight Moog oxidizer valve capable of being used for flight will be tested with the flight-like hybrid motor as well. Therefore, elevating the test to be a partial system level demonstration (pushing towards TRL 6). The main oxidizer valve was selected for testing because it will have the largest impact on the performance of the motor. Additionally, flight-capable pressure transducers (Paine) will demonstrate the telemetry that can be expected from space. Other system level components will be tested in the future, but do not have such a direct impact on the performance of the system. Due to funding constraints, we were not able to test the complete system as part of this task. The development of high pressure, miniature, oxygen compatible components will be required for the flight system. This is considered to be low risk since oxygen containment and handling at 10ksi has flight heritage. An example is the spare oxygen tanks Apollo astronauts carried for emergencies.[11]

At the end of this planned testing, the motor will be nearly ready for integration. However, the rest of the propulsion system will still need to be built including the feed system with TVC and RCS. The authors anticipate this will take approximately 2.5 years with adequate funding. The high-pressure components will drive the cost and schedule of this system. However, they also enable the high performance in the small propulsion module.

SUMMARY AND CONCLUSIONS

A 6U hybrid propulsion module for CubeSats has been presented. Depending on the mass of the rest of the system, the propulsion module, taking up 6U and 1/3 of the spacecraft mass, can provide 315 m/s of ΔV. This is a substantial improvement over the state of the art. Test data confirm the performance of the PMMA/GOx motor as well as the high-level requirements for the system: multiple starts and reasonable fuel utilization. The TRL of the propulsion system is currently 4 and is expected to be increased to 5+ by the end of this fiscal year.

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