Executive Summary

There were five test in September (four succeeded & one failed). I varied the length of the PLA/$KMnO_4$ fuel core as follows: 16.5 cm, 15.0 cm, 13.5 cm, and 12.0 cm. All other parameters were the same. I used a blend of 55 ml of ~ 87% HTP and 2.1 ml of denatured ethanol (O/F = 26.2) as the oxidizer. The propellant tank was pressurized to 130 psig using CO_2 gas as the pressurant. I used a 1/4" stainless steel mist nozzle with a 1.0 mm orifice as the injector and a graphite phenolic nozzle with an initial throat diameter of 5 mm. The objective was to determine what effect the length of the fuel core had on the operation of the engine and to select the best length to continue. The ignition oxidizer flux of ~14 gm/cm^2-sec, the run-time oxidizer flux of ~6 gm/cm^2-sec, the fuel core regression rate of ~0.4 mm/sec, and the characteristic velocity of ~1390 m/sec were consistent on three out of the four successful test. The deciding factor was the oxidizer to fuel ratio, the thrust, and the burn time. For the 15 cm fuel core the O/F ratio was 2.3, close to theoretical. Ignition occurred in one second and lasted for ~7 sec. There was a net positive thrust of greater than 16.2 N at ignition and lasted throughout the burn. Based on these results, I've selected the 15 cm fuel core for the class I flight system.

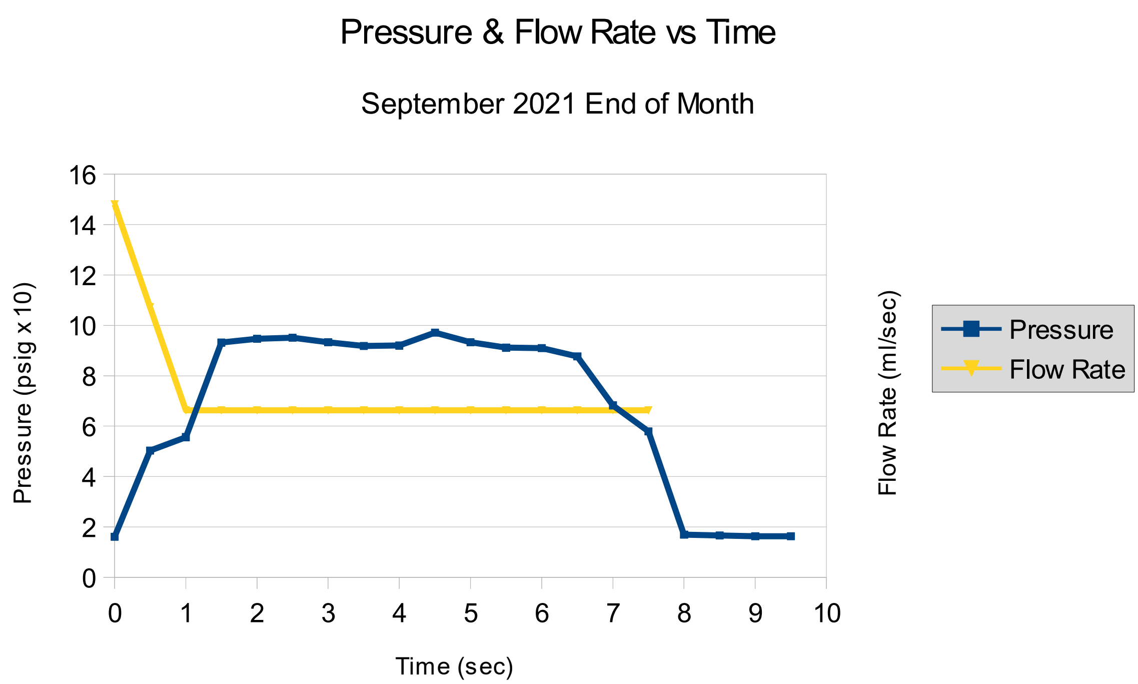

Figure. Test run for 15 cm PLA/$KMnO_4$ fuel core

Technical Stuff

By experiment, I determined that the flow rate of water through the 1/4" stainless steel mist nozzle with a 1.0 mm orifice at 130 psig was 14.8 ml/sec. As such, I used 14.8 ml/sec as the flow rate of HTPE at t = 0. I surmised that the flow rate decreased linearly as the pressure increased. The total area under the flow rate curve is 57.1 ml, (55 ml of 87% HTP plus 2.1 ml of Ethanol) for three of the fuel cores and 52 ml, (50 ml 87% HTP plus 2 ml Ethanol) for the 12 cm fuel core. Therefore,

$$r_1 + r_2 = 14.8 \text{ ml/sec}$$

$$1/2\ (t_1)(r_1) + (t_2)(r_2) = 57.1 \text{ ml}$$

where t_1 is time to ignition and t_2 is total run time. Solving for r_2 gives the flow rate for each of the four test during the ignition burn.

With the propellant tank pressurized to 130 psig and using an O/F ratio of 26.2, the initial HTPE flow rate of 14.8 ml/sec across the star chamber was converted to a mass flow rate of 19.83 gm/sec for the HTP and 0.43 gm/sec for ethanol. The cross sectional area for the "star chamber" was determined to be ~1.43 cm^2. As such, the ignition "oxidizer" flux for all four test was ~14.2 $gm/cm^2/sec$ (~0.2 $lb/in^2/sec$). Having calculated the burn flow rate (r_2) for each test, I used the same procedure to determine the oxidizer flux (G_{ox}) during the burn. The results were consistent for all four lengths and comparable to the available literature.

For the regression rate analysis, I had to determine the surface area of the fuel core and calculate the PLA volume flow rate (density of PLA = 1.21 gm/cm^3) during the burn to get the regression rate in mm/sec. Using the 15 cm fuel core as an example, the initial surface area of the fuel core was 124.5 cm^2 and the volume flow rate was ~3.6 cm^3/sec. As such, the regression rate for the 15 cm fuel core was ~0.25 mm/sec. With the exception of the 13.5 cm fuel core, the results were consistent and comparable to the available literature.

The table below is a summary of the run-time test.

Length (cm)	t_1 (sec)	t_2 (sec)	Flow Rate (ml/sec)	G_{ox} ($gm/cm^2/sec$)	G_{ox} ($lb/in^2/sec$)	Regression (mm/sec)
12.0	0.9	7.3	6.62	6.3	0.09	0.22
13.5	0.9	8.4	6.34	6.1	0.09	0.36(?)
15.0	1.0	8.0	6.63	6.3	0.09	0.25
16.5	1.5	8.1	5.29	5.1	0.07	0.22

? - Some anomalies in the test lead me to question these results.

I used a pressure probe at the inlet to the nozzle to measure the nozzle stagnation pressure. I calibrated the pressure sensor using an air pressure gauge and added 14.7 psi to the gauge pressure to get the absolute pressure. At ignition, the pressure was at peak and declined slightly during the burn as the throat diameter eroded. The recorded pressure is the average pressure during the burn (t_2 - t_1). The average stagnation pressure on all four test was ~102 psia.

During each of the four burns, the diameter of the graphite nozzle eroded from it's initial 5.0 mm to between 5.6 and 6.0 mm. As such, the cross sectional area of the throat changed during the burn. I used the average cross section to calculate the characteristic velocity.

The total mass flow rate was determined by first calculating the HTP volume flow rate and the ethanol volume flow rate (O/F=26.2), converting to mass flow rate, adding those together, and then adding the PLA mass flow rate. Using the 15 cm fuel core, the HTP mass flow rate was 8.9 gm/sec, the ethanol mass flow rate was 0.2 gm/sec, and the PLA mass flow rate was 3.7 gm/sec. Adding together gives a total mass flow rate of 12.8 gm/sec to the 15 cm fuel core.

The oxidizer to fuel ratio was determined by adding the ethanol mass flow rate to the PLA mass flow rate and then dividing the oxidizer mass flow rate by the fuel mass flow rate during the burn. Again using the 15 cm fuel core, the HTP mass flow rate was 8.9 gm/sec, the ethanol mass flow rate was 0.2 gm/sec, and the PLA mass flow rate was 3.7 gm/sec. That gives 8.9/3.9 = 2.3 for the oxidizer to fuel ratio of the 15 cm fuel core.

The characteristic velocity was calculated by multiplying the chamber pressure (in N/m^2) with the average throat diameter (in m^2) and then dividing by the total mass flow rate (in kg/sec). Using the NASA CEA code, the c* is 1485 m/sec (frozen composition, 100 psia, 85% HTP, 94% PLA/$KMnO_4$, and 6% ethanol) and the O/F ratio is 2.75. In practice, it is difficult to determine the mass percentage of PLA vs $KMnO_4$. Without more sophisticated diagnostics, I can not determine the percentage of PLA vs the percentage of $KMnO_4$ in the infusion process. For now, I'm using 92% PLA and 2% $KMnO_4$.

The table below is a summary of the run-time test.

Length (cm)	$(P_c)_{ns}$ (psia)	Avg A_t (cm^2)	m_t flow rate (gm/sec)	O/F	c* (m/sec)	c^*_{eff} (%)
12.0	98.7	0.24	11.7	3.1	1395	94
13.5	103.9	0.24 (!)	12.8	2.0	1348 (?)	91
15.0	107.4	0.24	12.8	2.3	1391	94
16.5	99.4	0.27	10.9	1.9	1691 (?)	114

! - Assumed 0.24 cm^2. PLA melt in the nozzle prevented an accurate measurement.
? - Some anomalies in the test lead me to question these results.

As shown in the video of the 15 cm fuel core test, there was a net positive thrust, a steady burn, and a rapid shutdown. The mass of the system (rocket engine, propellant tank, plumbing, pressure sensor, test frame, propellant, and the additional 212 gm mass) was 1.65 kg giving a thrust of greater than 16.2 N (3.6 lb) for ~7.0 sec. There was a small pressure leak at the end of the test. I was testing a new adapter. It had little effect on the test. The remaining fuel core tests had similar results. After some review, the 15 cm fuel core was the best.

Please note that these are very preliminary results. There was only one test for each fuel core lengths. Multiple runs, especially for the 13.5 cm and the 15.0 cm fuel cores, will take place over the next year to show consistency and reliability in operation. As for now, I believe I have enough information to built the class I flight system. As such, over the next several months, I'll be integrating the HTP/PLA/$KMnO_4$ engine into a light weight air-frame. I may be overly optimistic, but I hope to have a first launch before the end of November.

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