Executive Summary

There were two test in October. I eliminated the glow wire for ignition and decreased the cross sectional area of the 15 cm PLA/$KMnO_4$ fuel cell to increase the oxidizer flux. All other parameters were the same. I used a blend of 55 ml of ~ 85% HTP and 1.7 ml of denatured ethanol (O/F = 37.4) as the oxidizer. 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 increased flux had on the operation of the engine and if auto ignition would occur without the glow wire.

The ignition oxidizer flux was ~14 gm/cm^2-sec and was the same for both test. Ignition occurred in ~1.9 sec for the low flux fuel core and ~1.5 sec for the high flux fuel core. The ignition times are about the same as with a glow wire ignitor. Eliminating the ignitor simplifies the system.

The run-time oxidizer flux was ~5.4 gm/cm^2-sec for the low flux fuel core and 9.1 gm/cm^2-sec for the high flux fuel core. The fuel core regression rate and O/F ratio was approximately the same in both test. The deciding factor was the chamber pressure and the characteristic velocity. The propellant tank was pressurized to 130 psig using CO_2 gas as the pressurant in both test. The low flux test had a higher chamber pressure with corresponding higher characteristic velocity with a c* efficiency of ~91%. Based on these results and despite the longer ignition time, I've selected the low flux 15 cm fuel core for the class I flight system.

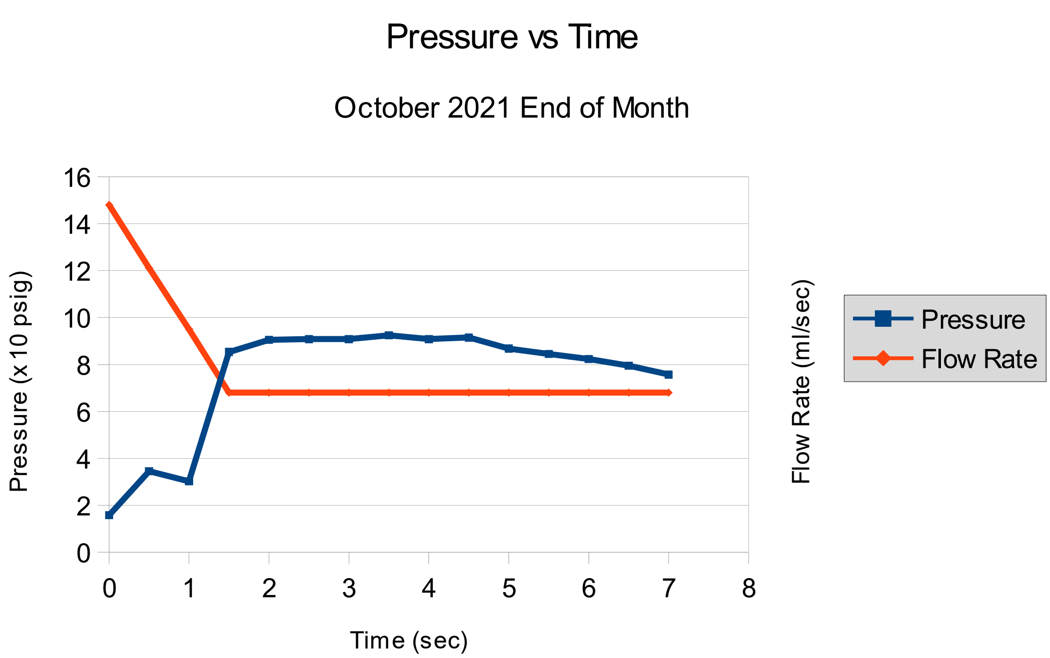

Figure. Test run for 15 cm PLA/$KMnO_4$ high flux fuel core

Technical Stuff

By experiment, I had 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 56.7 ml, (55 ml of 85% HTP plus 1.7 ml of Ethanol) for the high flux fuel core test. Therefore,

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

$$1/2\,(t_1)(r_1) + (t_2)(r_2) = 56.7 \text{ ml}$$

where t_1 is time to ignition (~1.5 sec) and t_2 is total run time (~7.6 sec). Solving for r_2 gives the oxidizer flow rate 6.8 ml/sec for the test during the ignition burn.

With the propellant tank pressurized to 130 psig and using an O/F ratio of 37.4, the initial HTPE flow rate of 14.8 ml/sec across the star chamber was converted to a mass flow rate of 19.9 gm/sec for the ~85% HTP and 0.3 gm/sec for ethanol for a total of 20.2 gm/sec. The ignition flow rate was the same for both the low flux and high flux fuel core segments. The cross sectional area for the low flux and high flux fuel cores was determined to be ~1.4 cm^2 and ~1.0 cm^2 respectively. As such, the ignition "oxidizer" flux for the low flux fuel core was ~ 14.1 $gm/cm^2/sec$ and for the high flux fuel core was ~19.9 $gm/cm^2/sec$.

Having calculated the burn flow rate (r_2) for each test, I used the same procedure to determine the oxidizer flux (G_{ox}) for the early part of the burn when the flow rate drops from 14.8 ml/sec to 5.7 and 6.8 ml/sec respectively. For the low flux core the flux was ~ 5.4 $gm/cm^2/sec$ and for the high flux core was ~9.1 $gm/cm^2/sec$. The oxidizer flux decreases as the cross sectional area increases for a solid core hybrid rocket engine. One can compensate for this by increasing the mass flow rate (i.e. throttle up).

For the regression rate analysis, I determined the surface area of the fuel cores and calculate the PLA volume flow rate (density of PLA = 1.21 gm/cm^3) during the burn. With the low flux 15 cm fuel core, the initial surface area was ~124.5 cm^2 and the volume flow rate was ~4.0 cm^3/sec. As such, the regression rate was ~0.31 mm/sec. With the high flux 15 cm fuel core, the initial surface area of the fuel core was 91.8 cm^2 and the volume flow rate was ~3.1 cm^3/sec. As such, the regression rate for the 15 cm high flux fuel core was ~0.34 mm/sec.

The table below is a summary of the run-time test. The first row is low flux 15 cm fuel core and the second row is the higher flux 15 cm fuel core. Both test used the same batch of distilled HTP.

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)
15.0	1.9	8.4	5.7	5.4	0.07	0.31
15.0*	1.5	7.6	6.8	9.1	0.13	0.34

* High flux fuel core

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 for each test was ~106.2 psia and ~101.4 psia respectively.

During the burn, the diameter of the graphite nozzle erodes. As such, the cross sectional area of the throat changes 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=37.4), converting to mass flow rate, adding those together, and then adding the PLA mass flow rate. For the low flux fuel core, the HTP mass flow rate was 7.6 gm/sec, the ethanol mass flow rate was 0.12 gm/sec, and the PLA mass flow rate was 3.1 gm/sec. Adding together gives a total mass flow rate of 10.9 gm/sec. For the high flux fuel core, the HTP mass flow rate was 9.1 gm/sec, the ethanol mass flow rate was 0.14 gm/sec, and the PLA mass flow rate was 3.8 gm/sec. Adding together gives a total mass flow rate of 13.0 gm/sec.

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. That gives 7.6/3.3 = 2.4 for the low flux fuel core oxidizer to fuel ratio and 9.1/3.9 = 2.3 for the high flux oxidizer to fuel ratio.

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} (%)
15.0	106.2	0.20	10.9	2.4	1346	91
15.0*	101.4	0.24	13.0	2.3	1290	87

* High flux fuel core

As shown in the video of the high flux 15 cm fuel core test, there was an initial jump at ignition and then the engine settled down to it's rest position. Four and half seconds into the burn, there was a net positive thrust of greater than 17 N. It was a steady burn with little oscillations and a rapid shutdown. I left the CO_2 gas valve open which allowed the engine to cool with little warping.

Please note that these are very preliminary results. There was only one test for high flux fuel core. I've down selected to the low flux 15 cm fuel core for the class I flight system. The low flux fuel core is lighter and the c* efficiency is higher. This is probably due to the higher chamber pressure. The propellant tank in both test were pressurized to 130 psig and the initial nozzle throat diameter was 5 mm. So, I am uncertain as to why the chamber pressure was lower in the high flux test. 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.