Executive Summary

This month, I finally collected all of my data and entered it into a spreadsheet. After two years of scattered data, I saw a trend in the rocket engine testing. Using an oxidizer injector with a 1.5 mm orifice produces an initial flow rate of ~ 21 ml/sec at 140 psig. As such, I get an initial surface flux of ~ 0.2 $gm/cm^2/sec$ using a 13.5 cm long PLA/$KMnO_4$ fuel core. The nozzle throat diameter is 6.0 mm and the rocket engine thrust is ~ 22 N. Assuming the maximum weight of ~ 15 N for my rocket glider, gives a thrust to weight ratio of 1.5. Over the next several months, I plan to test this design for consistency of time to ignition and thrust.

Technical Stuff

This month, I began to formalize my testing results. Previously, I've been going back and forth with scaling parameters on a whim, going in directions according to what looked like a promising direction (i.e., I was learning a lot and having fun). Toward the end of last month, I began developing a format for the test results. As such, I've seen a trend in the data.

Table I below shows the scaling parameters I've settle on and table II shows the performance results. The best characteristic velocity and efficiency is with using a nozzle with a 6.0 mm throat diameter. The video of the test on the 16^{th} shows no ignition and PLA/$KMnO_4$ streaming out of the nozzle. The physical examination revealed that the ½" stainless steel injector was to far away from the fuel core. This was the fault of the process I was using to assemble the fuel core in its housing. I have since altered the assembly process and the results are more consistent. I suspect some of my previous inconsistent test as having the same problem.

Table I
Scaling Parameters

Noz Orifice (mm)	V_{dot} (ml/sec) @140 psig	L (cm)	ISF ($gm/cm^2/sec$)	d_{eq} (cm)	L/D
1.5	20.84	13.5	0.21	1.52	8.9

Note: V_{dot} is the initial flow rate at valve opening, L is the length of the fuel core, ISF is the initial surface flux of the oxidizer at valve opening, d_{eq} is the equivalent inner port diameter of the fuel core (found by dividing the open surface area at the inlet by pi, taking the square root and multiplying by two), and L/D is the length to diameter ratio of the fuel core.

Table II
Performance Results

d_t (cm)	t_{ig} (sec)	L* (cm)	c* (m/s)	c^*_{eff}	O/F	Ref
0.55	1.1	79	1223	0.77	6.2	01/26/23
0.6	NI	66	N/A	N/A	N/A	01/16/23
0.6	0.97	66	1716	1.08	4.5	01/20/23
0.6	0.97	66	1607	1.01	4.5	01/30/23
0.65	0.93	54	1195	0.75	4.2	01/05/23

Note: d_t is the nozzle throat diameter, t_{ig} is the time from opening the valve to ignition, L* is the characteristic length, c* is the characteristic velocity, c^*_{eff} is the experimental c* divided by the theoretical c*, O/F is the oxidizer to fuel ratio, and Ref is the date of the test.

Figure 1 below shows the average thrust at ~ 22 N. The test on 20th had a thrust of ~ 23 N. However, the test on the 20th also had a longer ramp to thrust (1.5 sec vs 1.0 sec). I believe this is an effect of the edzieba* flow restrictor (4.0 mm orifice vs 5.0 mm orifice). So it appears as though a trade off between thrust and ramp to thrust may be necessary. Next month, I'll test a 6.0 mm orifice and we'll see.

Also, my data sampling rate is every 500 ms. I'm using Arduino boards for my diagnostics. If I try a faster sampling rate, I miss data points altogether. But, that's okay because I'm learning a lot about diagnostics as well. The Arduino boards are inexpensive, readily available, and the programming sketches are free.

Figure 2 is the characteristic velocity versus the characteristic length. It shows a peak of ~ 1600 m/sec at a characteristic length of ~ 66 cm. I would expect the curve to be more rounded with more data points. But, this is good enough to freeze the rocket engine design.

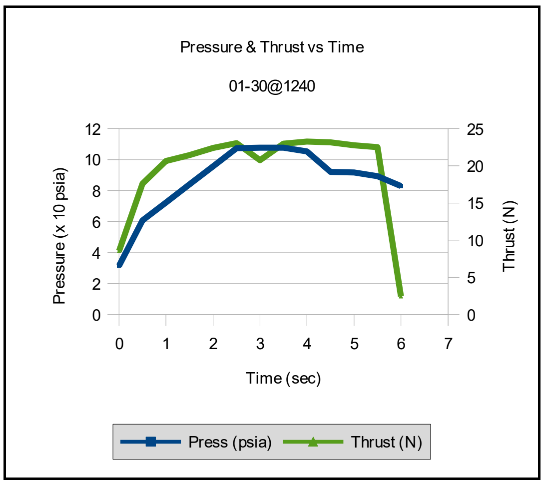

Figure 1

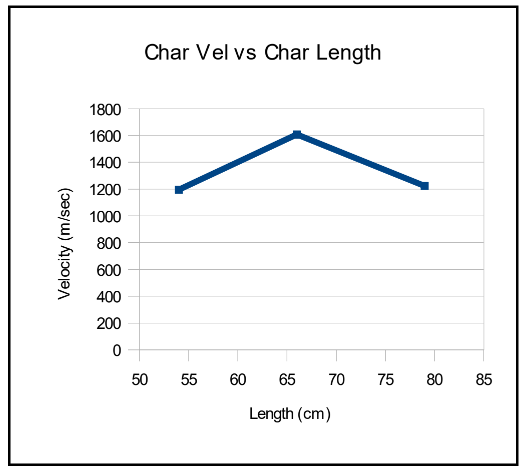

Figure 2

Once I decrease the ramp to thrust as much as possible, I will do a few more experiments to show consistency and then, freeze the design for a Mk I Viper Launch.

* edzieba, Reply #28, NASASpaceflight.com Forums>>General Discussion>>Advanced Concepts>> HTP/PLA/KMnO4 Hybrid Rocket Engine, 11/05/2021