EGR-330

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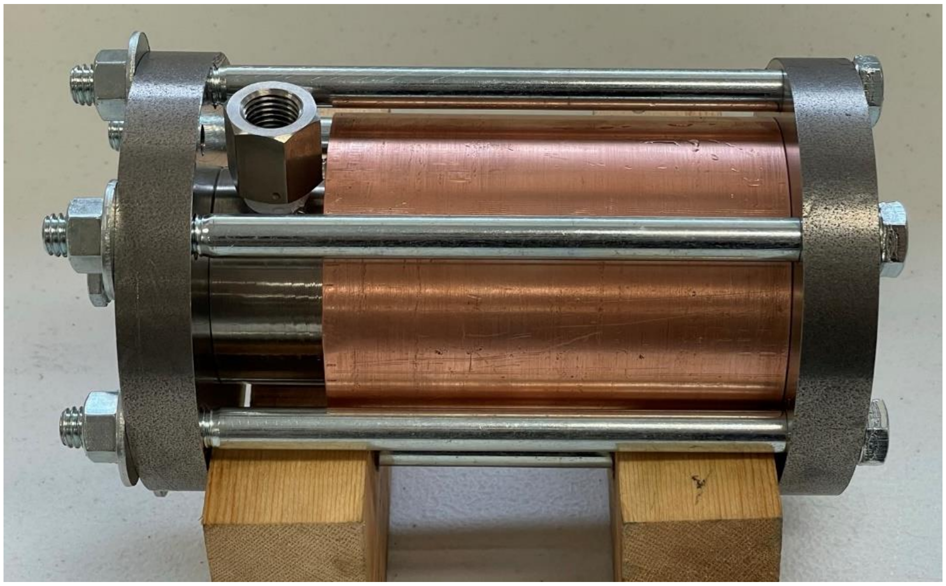
LIQUID FUELED ROCKET ENGINE REPORT

Space has long been known as “the final frontier,” and currently the most common technology used to traverse this frontier is the chemical rocket engine, or more simply: a rocket engine. Chemical rocket engines are complex machines, that work by burning fuel at tremendous pressures and temperatures in order to produce thrust. While the hunt to develop ever more efficient engine systems is an essential endeavor for the advancement of spaceflight, it is equally important to have a firm and intimate understanding of basic rocket propulsion technology, in order to be able to improve on it.

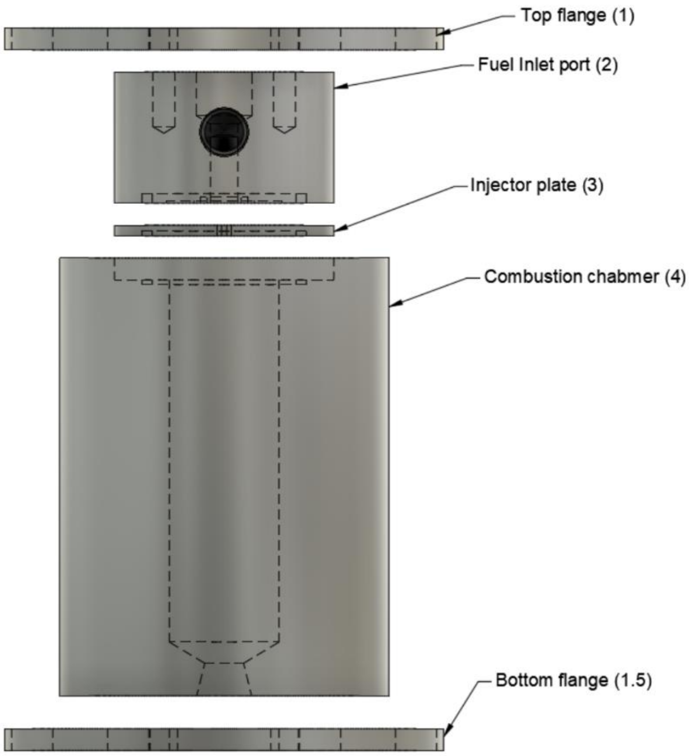
Therefore, it seems sensible for engineers interested in exploring the field of rocket propulsion to have experience with relatively simple rocket motors. Construction and testing of small-scale rocket motors allows an engineer to gain hands on experience in designing, fabricating, and testing rocket engines with a relatively high degree of safety. Additionally, a small and simple design has the added benefit of being much more affordable to fabricate, when compared to a state-of-the-art ion or magnetoplasma rocket engine.

The engine was designed to use readily available fuels: the oxidizer was gaseous oxygen, which was available in the College’s machine shop for use in welding. The fuel was 99% ethanol, which could be purchased relatively cheaply from McMasterCarr.com.



Components

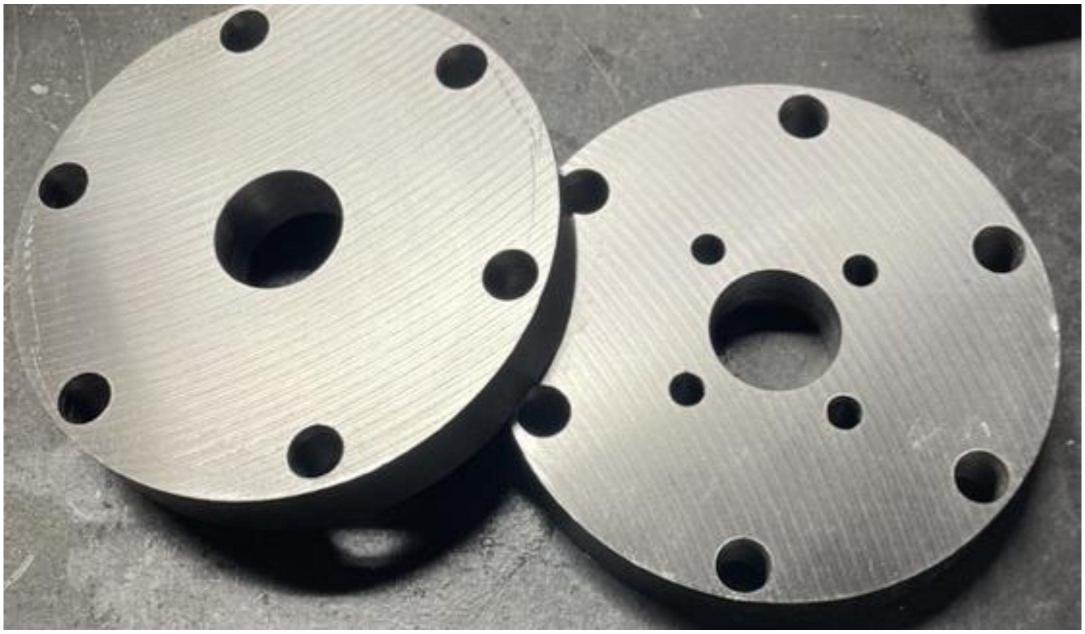
The engine consists of five primary parts: The fuel inlet port, the injector plate, the combustion chamber, and two flanges. In addition to the engine, a feed system was developed, as well as a method of monitoring the thrust of the engine using a force gauge.

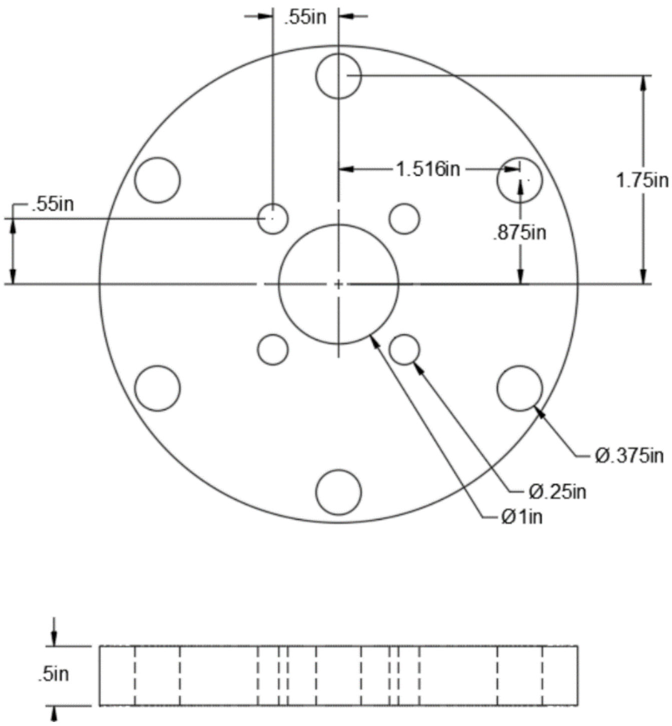


This design was selected as it could be easily machined and was relatively simple in its construction. The simplicity of the design traded efficiency and performance for safety and reliability, as even in the event of a failure of the engine, it was unlikely to explode catastrophically. The engine employed two additional systems: the propellant feed system, and an FSR force transducer.

Flanges

The flanges were constructed of ½" 1045 steel discs and were 4 inches in diameter. Their purpose was to hold the other parts together, in a similar manner to a vice using long bolts connecting the two flanges. The top flange has four ¼" holes to interface with the fuel inlet port, the bottom flange lacks these holes.

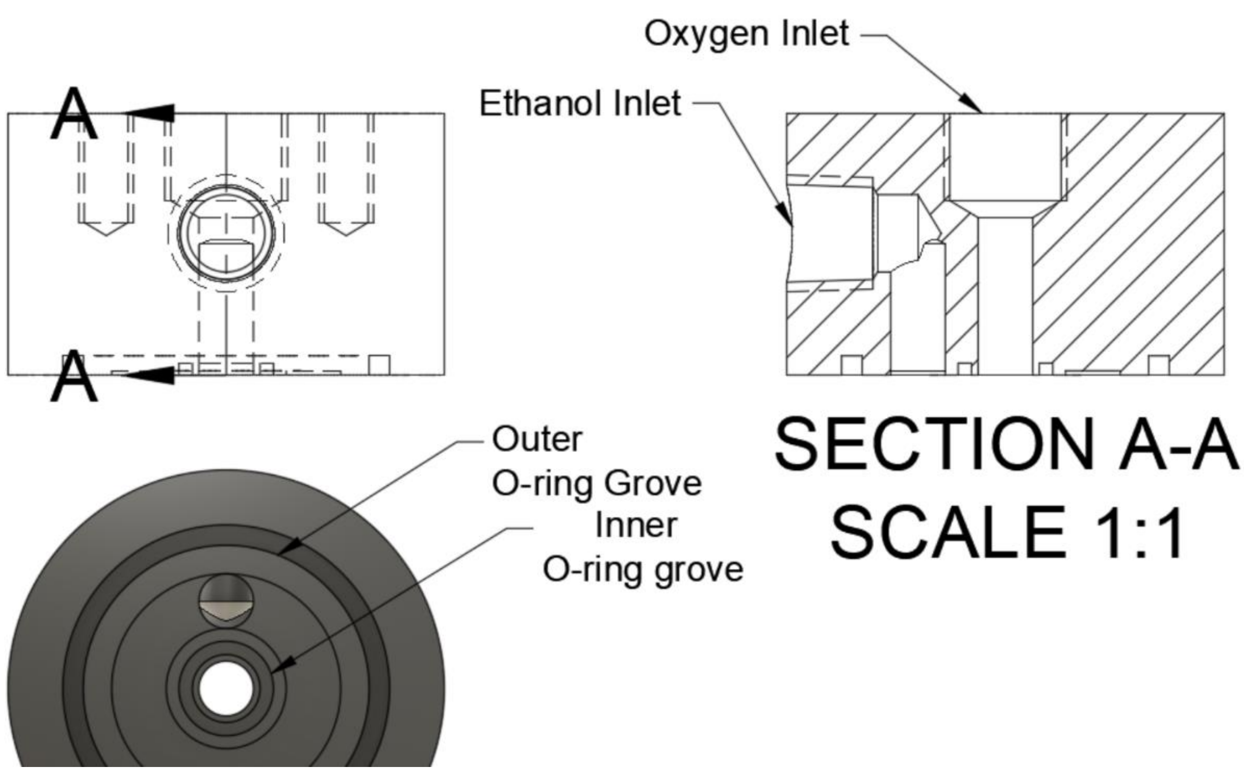




Fuel inlet port

The fuel inlet port is where the fuel (ethanol) and oxidizer (gaseous oxygen) enter the engine. The inlet port directs them to the injector plate, while ensuring that the fuel and the oxidizer do not come into contact with each other. To prevent premature mixing, the inlet port is fitted with a pair of O-rings, which ensure that the fuel cannot come into contact with the oxidizer or leave the engine by any means other than the injector.

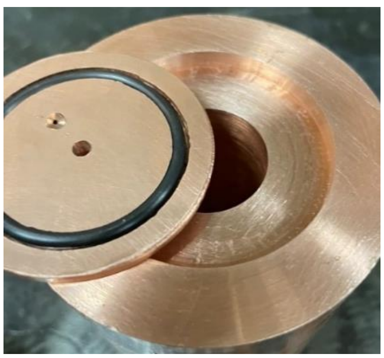
It was discovered that the fuel inlet could not be tapped with a ¼” NPT thread, as was initially intended, (due to the fact that no NPT taps were available) so it was tapped using a 9/16”-18 tap instead. The tap could not cut very far into the hole, so PTFE tape was used abundantly to ensure proper sealing. Additionally, the holes on the top, which were used to align the inlet port with the top flange, were not drilled properly and only two could be used at a time.

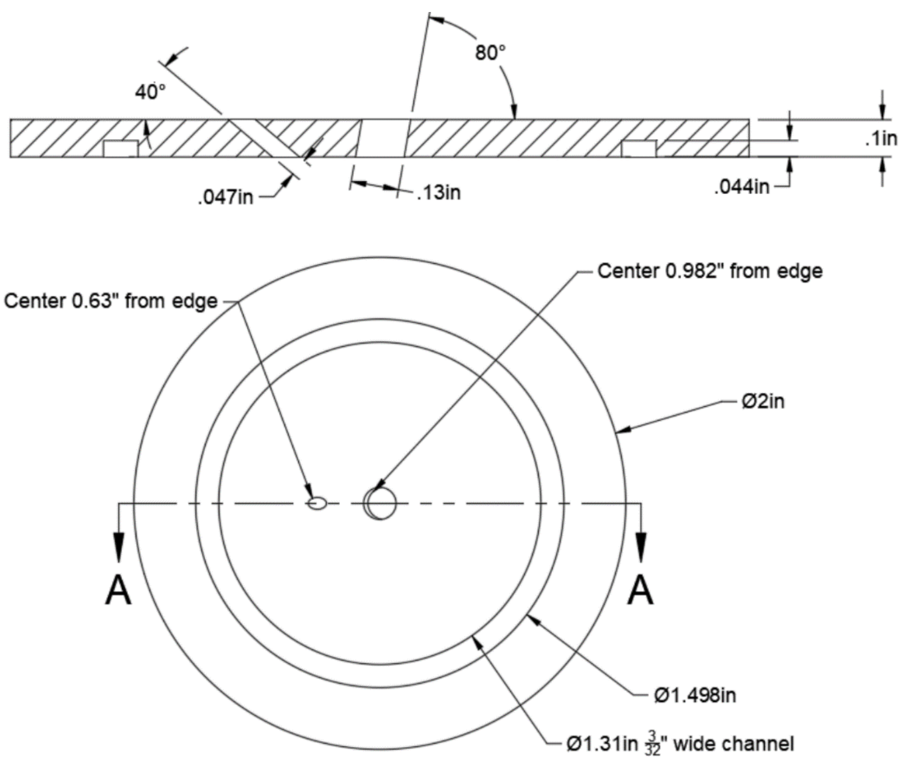


Injector plate

The injector plate was manufactured from a 1/8" copper disc which was 2" in diameter. It had two holes angled in towards each other, which was designed to cause the oxidizer and fuel to collide and atomize as they entered the combustion chamber, which is essential in ensuring proper combustion, as is the ratio of fuel to oxidizer (1:1.45 O/F), which is ensured by the sizing of the injector orifices. The injector plate fits into a slot cut into the combustion chamber and is sealed with an O-ring.

The center hole supplied gaseous oxygen, with a pressure drop of ~43psi, while the smaller radial hole supplied ethanol with a pressure drop of ~30psi. The oxygen hole size was selected partly due to the fact that a 3.3mm drill was readily available. The small size of the ethanol hole required a larger pilot drill to drill part of the way through, to avoid the small drill deflecting and breaking.





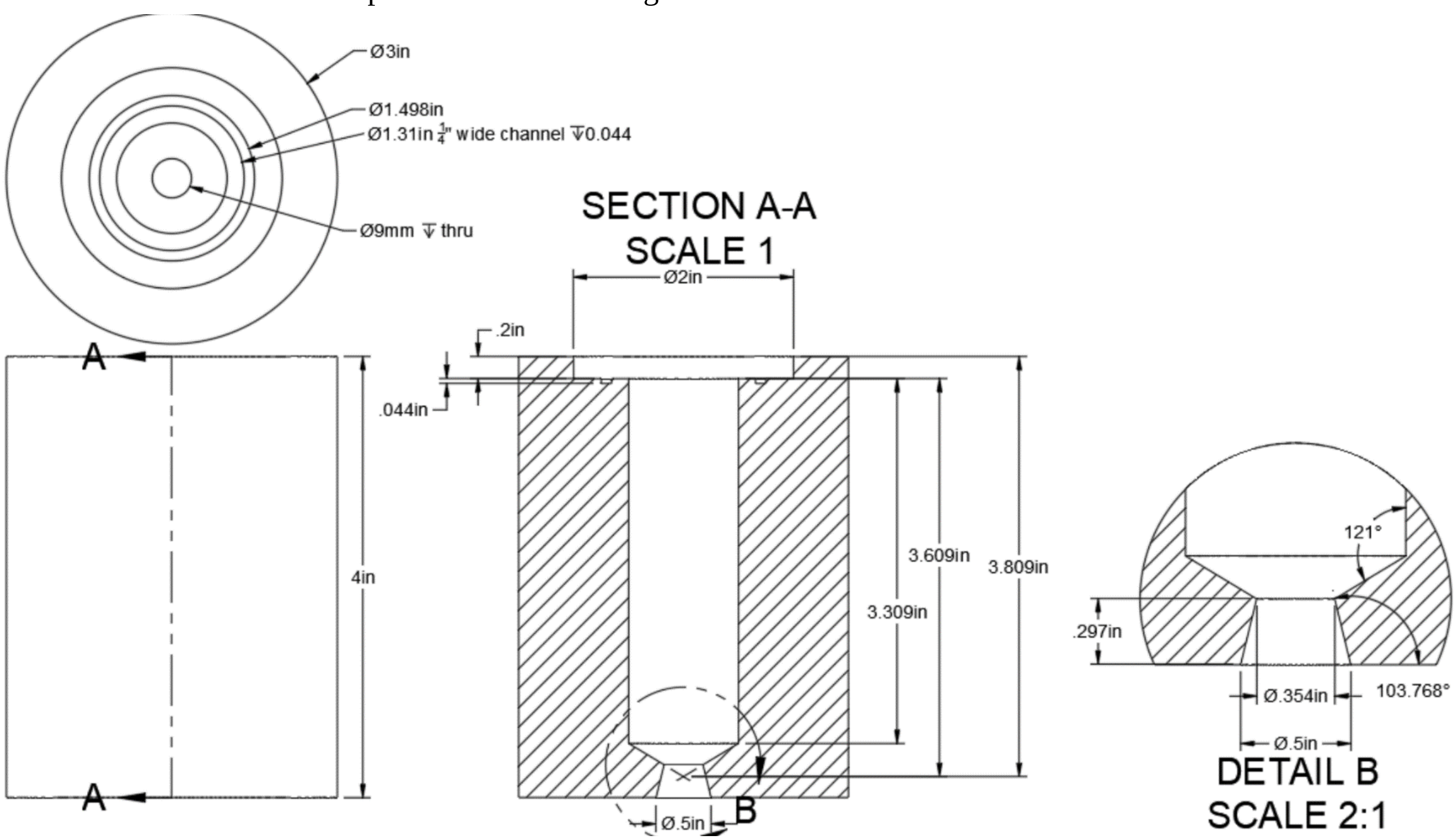
Combustion chamber

The combustion chamber is where the fuel is burned at a designed pressure of 150psi. The hot combustion gases are then directed through the nozzle section, which converts the high pressure inside the engine into flow velocity and ejects the hot gases at high velocity and low pressure.

The throat was 9mm in diameter, and was made using a parabolic drill bit, meaning that it could be drilled through the entire stock and used as a pilot for larger drill bits. The combustion chamber was drilled using a 1" drill and the convergent section of the nozzle (before the throat) was made using the end of the drill. Meaning that, while not the most efficient convergent angle, it did not require any further machining after the initial hole was drilled.

For the sake of simplicity, the nozzle section was drilled at an angle that ensured the exit diameter would be ½" and thus easier to mark out on a lathe. This resulted in the angle of the nozzle's diverging section being just under 14°.

The gases in the combustion chamber are incredibly hot (~3050°K, or ~2775°C), and so the mass of the copper used in the chamber's construction comes into play: if fired for only a short time, the thick copper walls of the combustion chamber act as a heat-sink and employs the high thermal conductivity of the copper to draw the thermal energy away from the combustion-side surface in order to prevent it from melting.

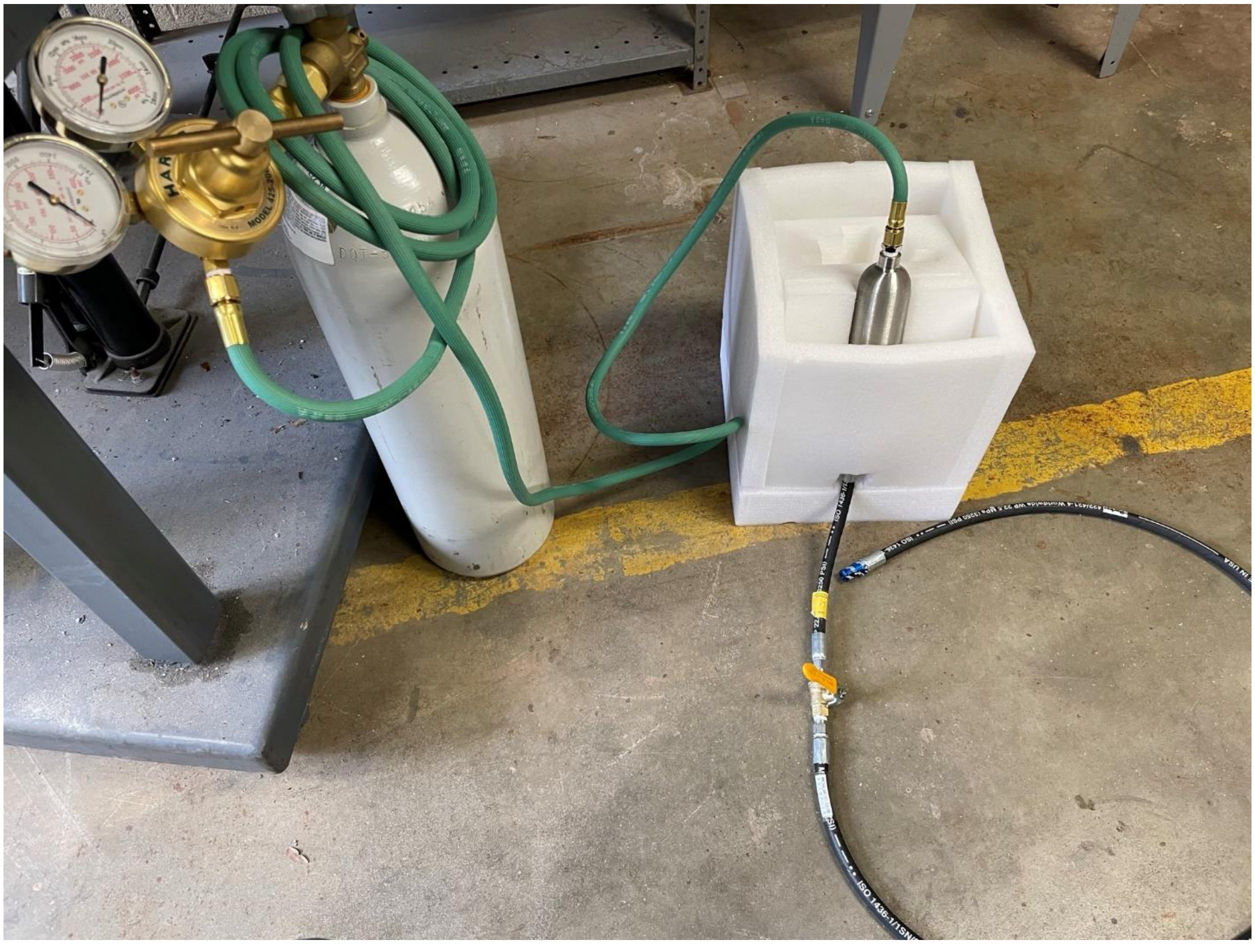


Propellant feed system

The propellant feed system consisted of two separate feed lines: one for the oxygen, and another for the ethanol. Both lines employed a pressure regulator set to 200psi, in order to ensure that the chamber pressure remained below 200psi.

The oxygen line was relatively straightforward: the pressure regulator was mounted to a high-pressure oxygen cylinder, and a welding gas hose connected it to a land-operated valve. The valve was then connected to the engine by another welding hose.

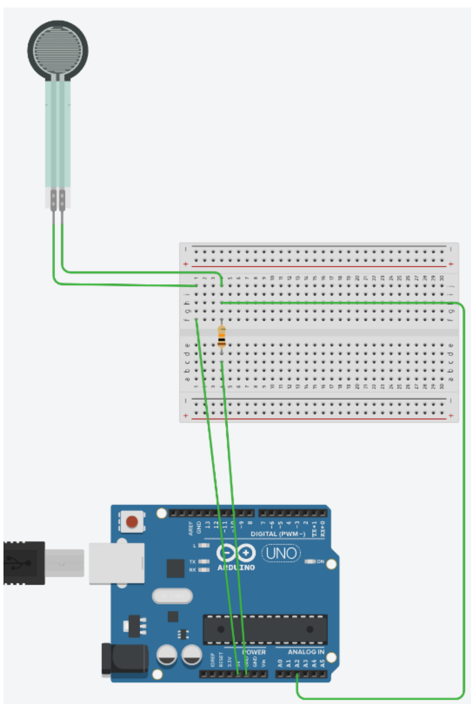
The fuel feed line was somewhat more complex. A regulator was mounted to a high-pressure cylinder containing an inert mixture of CO_2 and argon, which was connected to a small 300ml double-ended sample cylinder, which was held vertically using a Styrofoam cube (it's what was on hand at the time). The sample cylinder was filled with ethanol to about ¾ full, and the CO_2-argon mixture was used to pressurize it to just over 200psi. The bottom of the sample cylinder was connected to a 1ft hydraulic fluid hose, which was attached to a hand-operated valve. The valve was connected to the engine by a 5ft hydraulic hose.

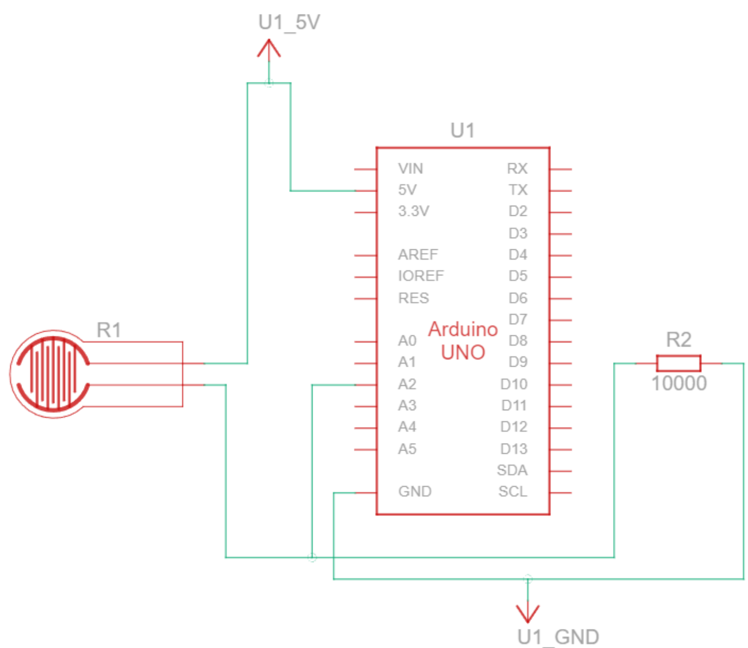


Force transducer

To measure the thrust generated by the engine, a force sensitive resistor (FSR) was used. The FSR generates variable resistance in proportion to the pressure applied to it. It is able to measure forces up to 100N.

Unfortunately, the FSR stopped working for unknown reasons prior to testing, and so thrust measurements could not be taken.





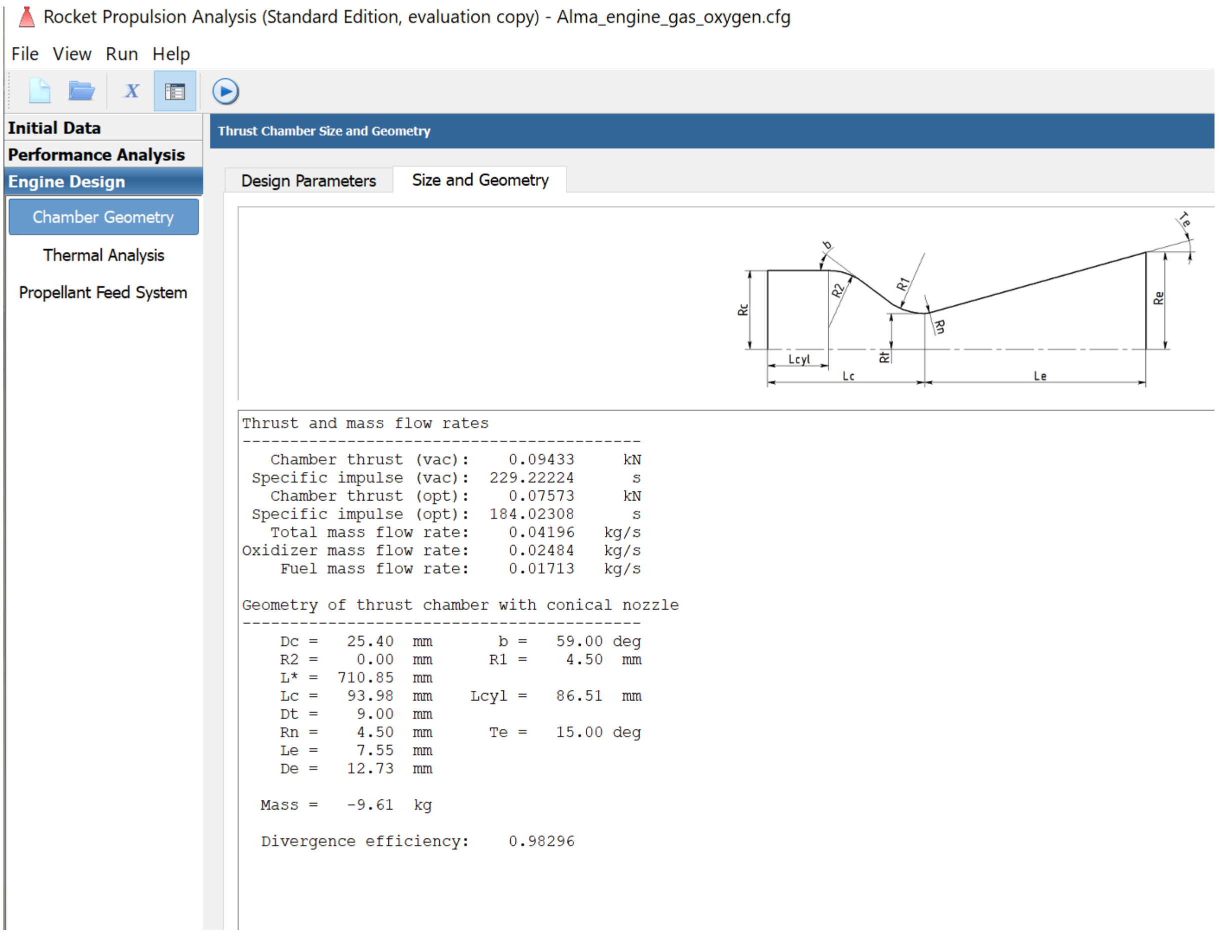
Calculated performance

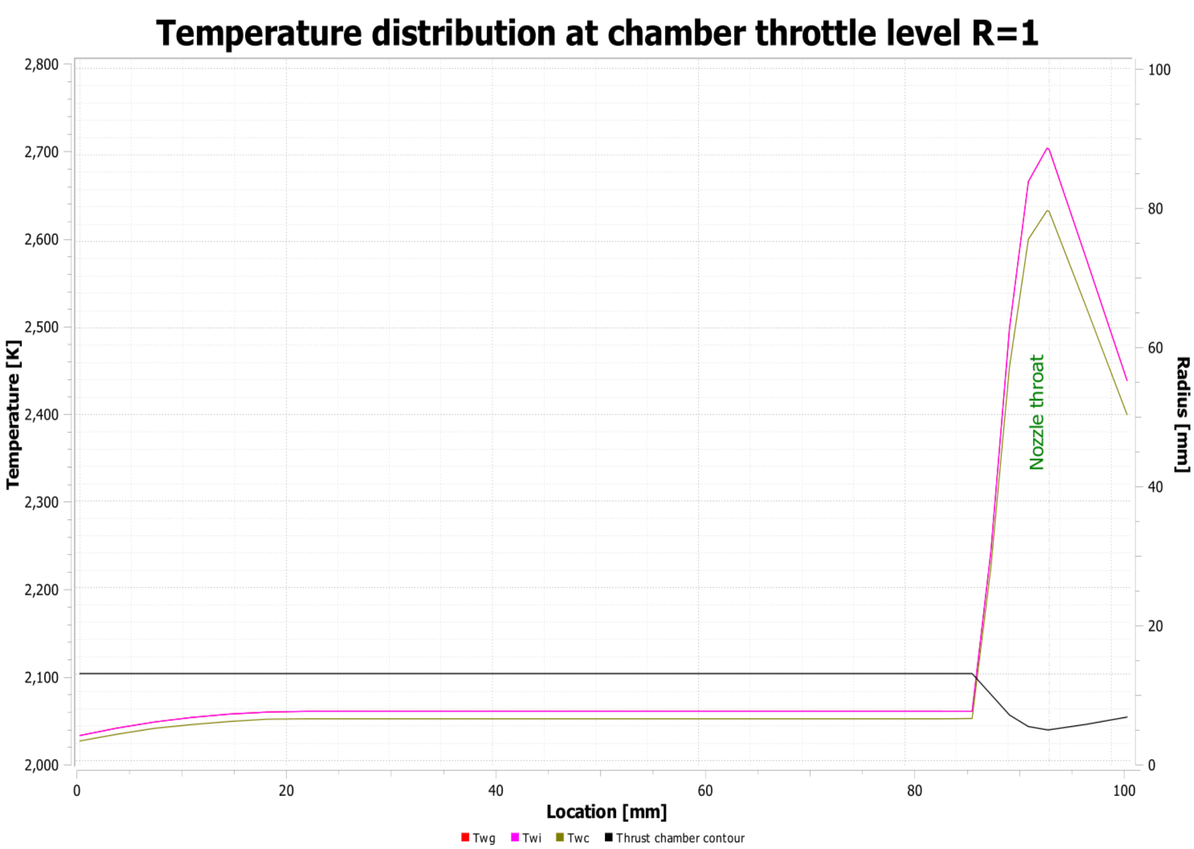
Using a program called "Rocket Propulsion Analysis" (RPA), it is possible to estimate the performance of the rocket engine. The program is very useful in quickly estimating the performance characteristics of a rocket engine design. This particular software was used to estimate performance for three different fuel/oxidizer combinations before the current mixture was selected. Under ideal conditions, it was calculated that the engine would produce about 81N of thrust at sea level, or around 18 lbf.

Below are some selected screenshots from the RPA program.

Altitude performance (theoretical ideal performance)

Altitude, km	Pressure, atm	Effective exhaust velocity, m/s	Specific impulse (by weight), s	Thrust coefficient	Thrust, kN
0.000	1.00000	2079.750	212.076	1.2754	0.084
0.273	0.96793	2090.034	213.124	1.2817	0.084
0.548	0.93670	2100.048	214.145	1.2878	0.084
0.822	0.90630	2109.798	215.140	1.2938	0.085
1.096	0.87670	2119.289	216.107	1.2996	0.085
1.370	0.84789	2128.527	217.049	1.3053	0.086
1.644	0.81985	2137.517	217.966	1.3108	0.086
1.918	0.79258	2146.264	218.858	1.3161	0.086
2.192	0.76604	2154.772	219.726	1.3214	0.087
2.467	0.74023	2163.049	220.570	1.3264	0.087
2.741	0.71513	2171.097	221.390	1.3314	0.087
3.015	0.69073	2178.922	222.188	1.3362	0.088
3.289	0.66701	2186.529	222.964	1.3408	0.088
3.563	0.64395	2193.922	223.718	1.3454	0.088
3.837	0.62155	2201.106	224.450	1.3498	0.089
4.111	0.59978	2208.086	225.162	1.3541	0.089
4.385	0.57863	2214.866	225.853	1.3582	0.089
4.659	0.55810	2221.451	226.525	1.3622	0.089
4.933	0.53816	2227.845	227.177	1.3662	0.090
5.207	0.51880	2234.051	227.810	1.3700	0.090
5.481	0.50002	2240.076	228.424	1.3737	0.090


Rocket Propulsion Analysis (Standard Edition, evaluation copy) - Alma_engine_gas_oxygen.cfg
File View Run Help
Initial Data
Performance Analysis
Engine Design
Chamber Geometry
Thermal Analysis
Propellant Feed System
Thrust Chamber Size and Geometry
Design Parameters
Size and Geometry
Rc
R2
b
R1
Rt
Rn
Re
Te
Lcyl
Lc
Le
Thrust and mass flow rates
Chamber thrust (vac): 0.09433 kN
Specific impulse (vac): 229.22224 s
Chamber thrust (opt): 0.07573 kN
Specific impulse (opt): 184.02308 s
Total mass flow rate: 0.04196 kg/s
Oxidizer mass flow rate: 0.02484 kg/s
Fuel mass flow rate: 0.01713 kg/s
Geometry of thrust chamber with conical nozzle
Dc = 25.40 mm b = 59.00 deg
R2 = 0.00 mm R1 = 4.50 mm
L* = 710.85 mm
Lc = 93.98 mm Lcyl = 86.51 mm
Dt = 9.00 mm
Rn = 4.50 mm Te = 15.00 deg
Le = 7.55 mm
De = 12.73 mm
Mass = -9.61 kg
Divergence efficiency: 0.98296


Temperature distribution at chamber throttle level R=1
Temperature [K]
2,000
2,100
2,200
2,300
2,400
2,500
2,600
2,700
2,800
Radius [mm]
0
20
40
60
80
100
Location [mm]
0
20
40
60
80
100
Nozzle throat
Twg
Twi
Twc
Thrust chamber contour

Thermodynamic properties (O/F=1.450)

Parameter	Injector	Nozzle inlet	Nozzle throat	Nozzle exit	Unit
Pressure	1.0342	1.0277	0.6000	0.1400	MPa
Temperature	3048.5364	3047.4445	2918.6801	2601.1579	K
Enthalpy	-3548.4431	-3551.7890	-4106.5160	-5464.3313	kJ/kg
Entropy	11.7415	11.7426	11.7426	11.7426	kJ/(kg·K)
Internal energy	-4608.4447	-4611.3678	-5109.1557	-6331.3442	kJ/kg
Specific heat (p=const)	7.2472	7.2490	7.0674	6.2926	kJ/(kg·K)
Specific heat (V=const)	6.2915	6.2933	6.1697	5.5617	kJ/(kg·K)
Gamma	1.1519	1.1519	1.1455	1.1314	
Isentropic exponent	1.1156	1.1156	1.1132	1.1093	
Gas constant	0.3477	0.3477	0.3435	0.3333	kJ/(kg·K)
Molecular weight (M)	23.9122	23.9132	24.2034	24.9446	
Molecular weight (MW)	0.02391	0.02391	0.0242	0.02494	
Density	0.9757	0.9699	0.5984	0.1614	kg/m^3
Sonic velocity	1087.4666	1087.2359	1056.4795	980.7232	m/s
Velocity	0.0000	81.8033	1056.4795	1957.4924	m/s
Mach number	0.0000	0.0752	1.0000	1.9960	
Area ratio	7.9650	7.9650	1.0000	2.0000	
Mass flux	79.3439	79.3439	632.2170	315.9867	$kg/(m^2·s)$
Mass flux (relative)	0.767e-04	0.772e-04			kg/(N·s)
Viscosity	0.0001044	0.0001044	0.0001016	9.473e-05	kg/(m·s)
Conductivity, frozen	0.3254	0.3253	0.3132	0.2827	W/(m·K)
Specific heat (p=const), frozen	2.192	2.192	2.181	2.15	kJ/(kg·K)
Prandtl number, frozen	0.7031	0.7031	0.7077	0.7202	
Conductivity, effective	1.367	1.367	1.279	1.014	W/(m·K)
Specific heat (p=const), effective	7.247	7.249	7.067	6.292	kJ/(kg·K)
Prandtl number, effective	0.5535	0.5535	0.5617	0.5876	

Testing

First test

The first resulted in an ignition failure, and only succeeded in spewing a large mist of oxygen and ethanol, much to the confusion of the gathered crowd (as well as the two of us):

The ignition failure was due to the method of ignition: prior to opening the fuel/oxidizer valves, a match was soaked in kerosene, lit, and inserted into the nozzle. However, when the fuel and oxidizer valves were opened, the match was ejected from the engine before the fuel could be ignited.

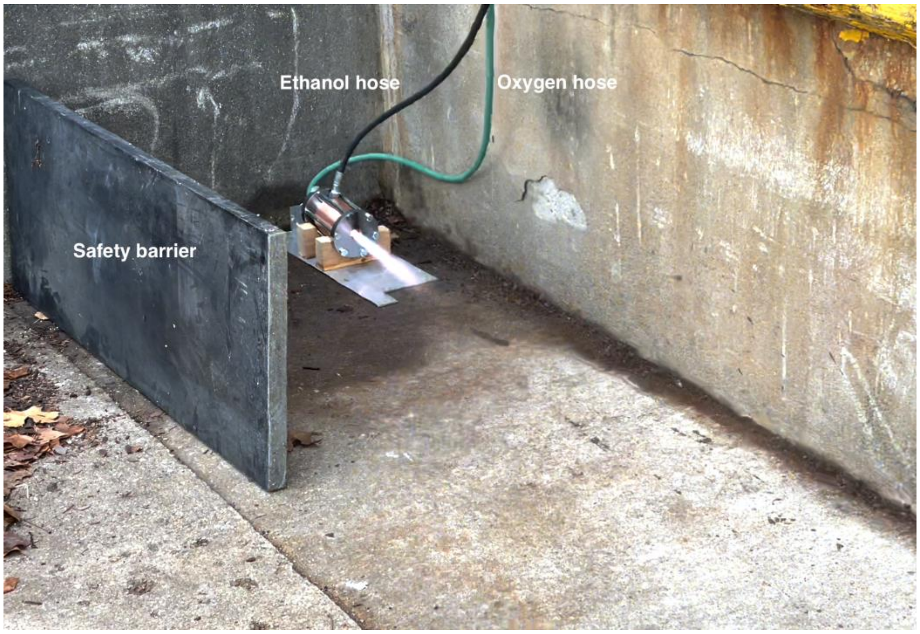
Second test

After footage of the first test was reviewed, and changes were implemented to the ignition procedure: for the next test, a small amount of kerosene was poured directly into the combustion chamber and following this, two pieces of cloth were soaked in kerosene and inserted into the combustion chamber through the nozzle. It was also decided that a different testing location should be selected: the new testing location would be in the loading bay of the Dow Science Center, which was both much closer to the workshop, and much less combustible than the field in which the first test was conducted **(A.1)**.

The test went as follows: Once the cloth was lit, the oxygen valve was partly opened, allowing the kerosene to combust. **(A.2)** The kerosene quickly combusted, and shortly after this, the burning cloth was ejected from the nozzle with a loud "thud" **(A.3)**. The ethanol valve was then opened, and ignition was confirmed as a spray of partially combusted ethanol spewed out of the nozzle **(A.4)**, accompanied by a shrill "shriek," the cause of which is unclear. The oxygen valve was then opened fully, and proper combustion commenced with a deafening roar **(A.5)**. After regaining his senses (which had been momentarily blown away by the volume of sound produced by the engine), the valve operator closed the oxygen valve, halting combustion (**A.6)**. The entire test lasted approximately 10 seconds, with steady-state flow lasting for roughly 5 seconds. Unfortunately, due to the failure of the FSR unit, thrust data could not be gathered.

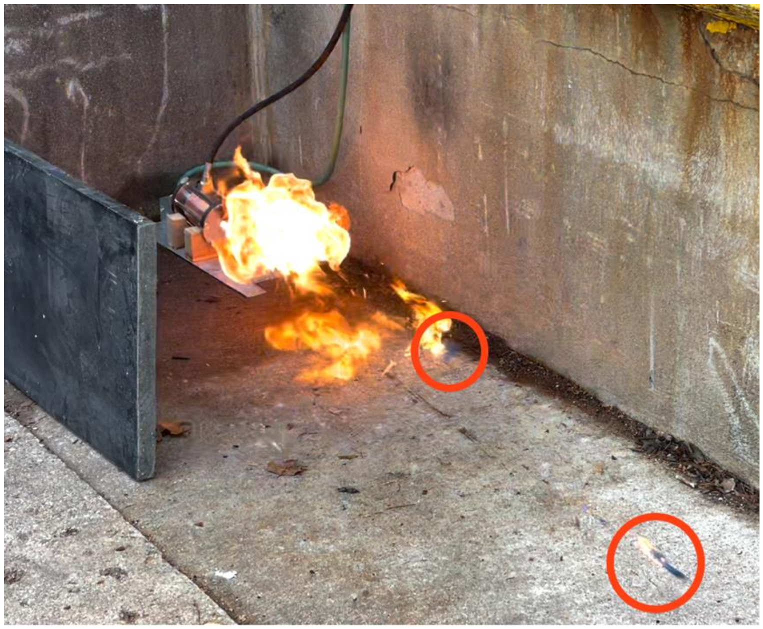
Test 2 images

A.1 – Test site



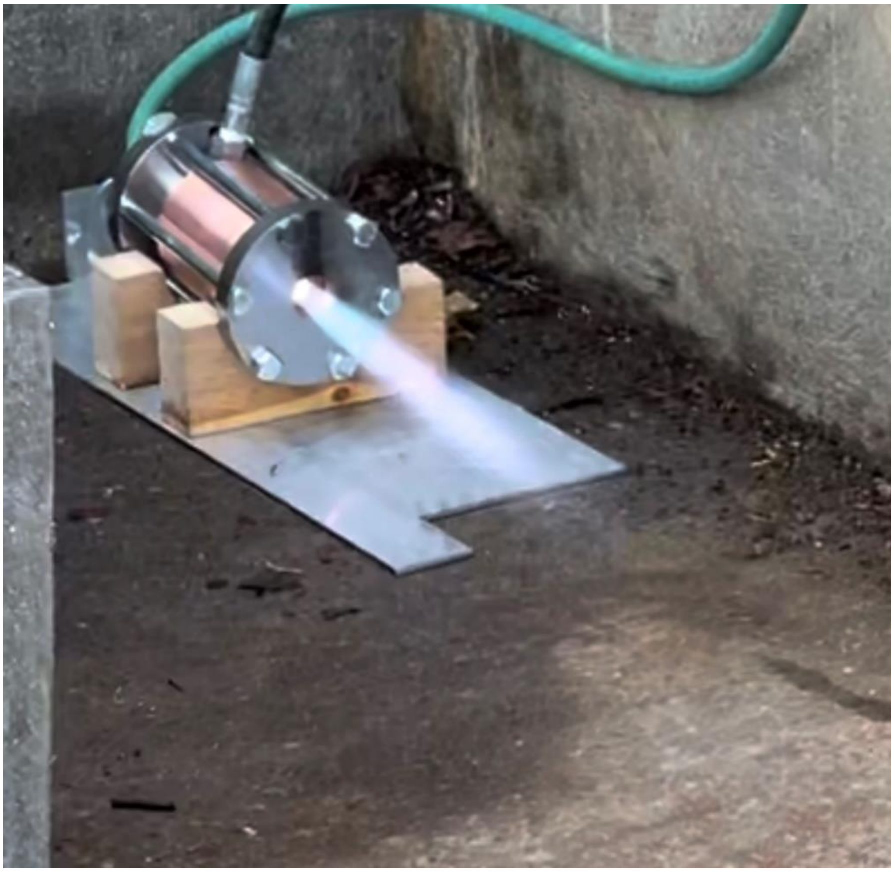
A.2 – pre-ignition; the burning cloth can be seen smoldering in the nozzle

A.3 – Oxygen valve partly opened & burning cloth is ejected (circled in red) as the kerosene ignites



A.4 – Ethanol valve opened; partial ignition of the fuel observed

A.5 – Oxygen valve opened fully; proper combustion achieved, note the white-blue color of the exhaust jet



A.6 – Oxygen and ethanol valves closed, smoke and unburnt ethanol spurts from the nozzle

Future work

While the engine was successfully test-fired, there is still much to be desired. No thrust readings could be taken during the tests, and the method of ignition is very crude. If further testing is to be done, some steps must be taken to ensure reliable ignition, as well as improving control over the fuel/oxidizer valves.

For future tests, we plan to make a stand on which the fuel/oxidizer valves can be mounted, so that they can be operated simultaneously rather than one at a time. Additionally, research will be done into more reliable ignition methods, including spark-gap ignition by either the use of a commercial spark plug, or by a pair of wires. If a spark plug is used, then a sizable hole must be bored into the side of the combustion chamber, and great care must be taken to ensure that its structural integrity is not compromised; if wires are used for the spark-gap igniter, then either a very small hole would need to be bored into the combustion chamber, or the wires could be inserted through the nozzle.