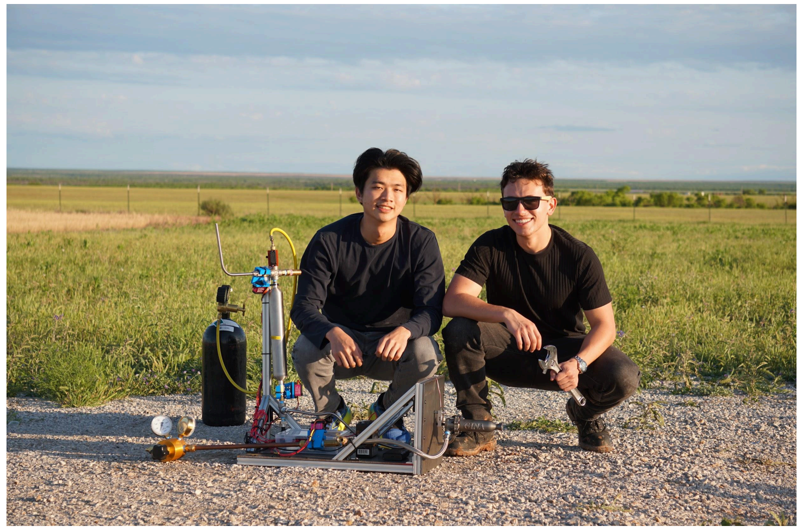
How to Build a Small Liquid Rocket Engine

[Kevin Ge], [Daniel Vayman]

5/11/2024



Contents

Introduction

This paper details the design, fabrication, and operation of UTD's first liquid rocket engine. The engine was built in under 4 months with less than $2000 as a prototype solely for static fires and was not meant for flight. Successes, failures, and future improvements will be discussed.

This paper is meant to provide serious amateur rocketeers with a detailed build process of a fully operational liquid rocket engine. Reference equations and physical concepts may be covered briefly but full derivations and in-depth explanations will be linked in the footnotes.

Propellant

At the time of this writing, liquid bipropellant rocket engines are the primary way to get to space. These engines utilize clever nozzle designs and the combustion byproducts of two liquid propellants to create thrust. Engines designed for flight typically use a cryogenic oxidizer (e.g. LOX) and sometimes a cryogenic fuel. For example, the RS-25 Space Shuttle engines used liquid hydrogen and liquid oxygen (LOX), two cryogenics, while SpaceX's Raptor engines use LOX/LCH4 (liquid oxygen and liquid methane).

Due to the prototypical nature of this engine, our propellant choice was motivated by 3 factors: obtainability, price, and ease of design/use. GOX/Ethanol fulfills each of these requirements (although IPA could have easily been chosen as well). We considered N2O as well, but decided against it due to its price point (it's significantly more expensive than GOX) and to avoid designing around a two-phase flow.

During the design process, we chose to run fuel-rich with an O/F (mass ratio) of 1.2 (rather than the ~2 stoichiometric mass ratio). Based on empirical evidence[1], we knew that a stainless steel chamber could probably survive a burn of at least 2.5 seconds.

[1] 2.5s burn with stainless steel engine (r/rocketry discord): https://discord.com/channels/723644976638066845/1316286060942196736/1316286060942196736

Sizing Equations

When it comes to sizing an engine, there are a few parameters one must choose: chamber pressure[2] and desired thrust. The size of your engine is directly proportional to thrust/mass flow rate and inversely proportional to chamber pressure.

We wanted our engine to be rather small, so we chose a small desired thrust of 300N and an amateur-feasible pressure of 20 bar (~290 PSI).

With your O/F ratio and/or propellants, you should be able to find a known Isp of your engine. With online resources[3], we were able to approximate our Isp to 244, allowing us to calculate our total mass flow rate.

$$I_{sp} = \frac{F}{\dot{m} g_0} \rightarrow \dot{m} = \frac{F}{I_{sp} g_0}$$

Using NASA's CEA tool[4], we can get the specific heats and temperature at certain points along our de Laval nozzle. Using equations following the isentropic flow model, we are able to derive the following engine sizing equations.

$$A_t = \frac{\dot{m}}{P_t}\sqrt{\frac{R_s T}{\gamma}}$$

$$M_e = \sqrt{\frac{2}{\gamma - 1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}$$

$$\frac{A_e}{A_t} = \frac{1}{M_e}\left[\frac{1 + \frac{\gamma-1}{2} M_e^2}{(\gamma+1)/2}\right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

Note: Rs is the specific gas constant. Rs = R/M.
The isentropic flow model should be within ~2-5% of approximating engine performance during nominal conditions. Full derivations can be found in Rocket Propulsion Elements by George Sutton or various online resources[5].

[2] Chamber pressure may seem like an odd design parameter. After all, doesn't this pressure occur naturally from combustion? This is a misconception, as the flow can be accurately modeled as quasi-steady and quasi-one dimensional (i.e. pressure is self-regulated/constant with respect to the inlet pressure and flow is axial)

[3] GOX/Ethanol Isp can be found here: https://spacha.github.io/How-to-Rocket/#propellant-choice-and-properties

[4] NASA CEA: https://cearun.grc.nasa.gov. Using your O/F ratio and chamber pressure, CEA provides the theoretical chamber, throat, and exit temperatures, pressures, and specific heats.

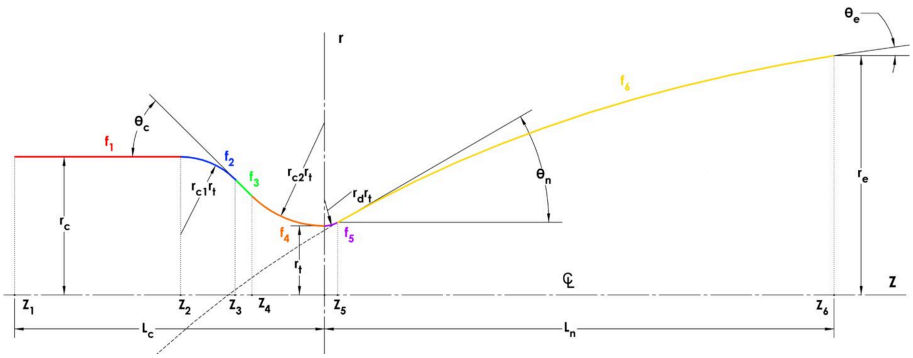
[5] Derivations: https://ftp.idu.ac.id/wp-content/uploads/ebook/tdg/DESIGN%20SISTEM%20DAYA%20GERAK/Rocket%20Propulsion%20Elements.pdf (Ch 3.1) or https://www.youtube.com/watch?v=sTZhUrflZF4

To calculate the length and cross-sectional area of the chamber, we can use $L^* = \frac{V_c}{A_t}$, a ratio that describes the chamber volume ideal for optimal combustion. It is important to note that L* is some function of operating conditions that are unique to each engine design[6].

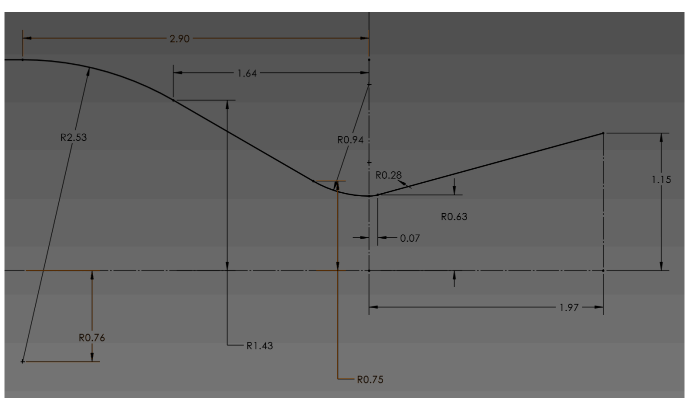
Nozzle Design

The sizing equations may provide us with the dimensions of our chamber, throat, and exit plane, but we are still left with *how* to design the parabolic curve in which the nozzle converges and diverges. Seemingly simplistic, it is difficult to CAD without a systematic approach.

The de facto approach is Rao's nozzle, described in this [article](). We wrote a handy script that generates Rao nozzle points (as seen in the diagram below) and a STEP file, given certain nozzle dimensions.

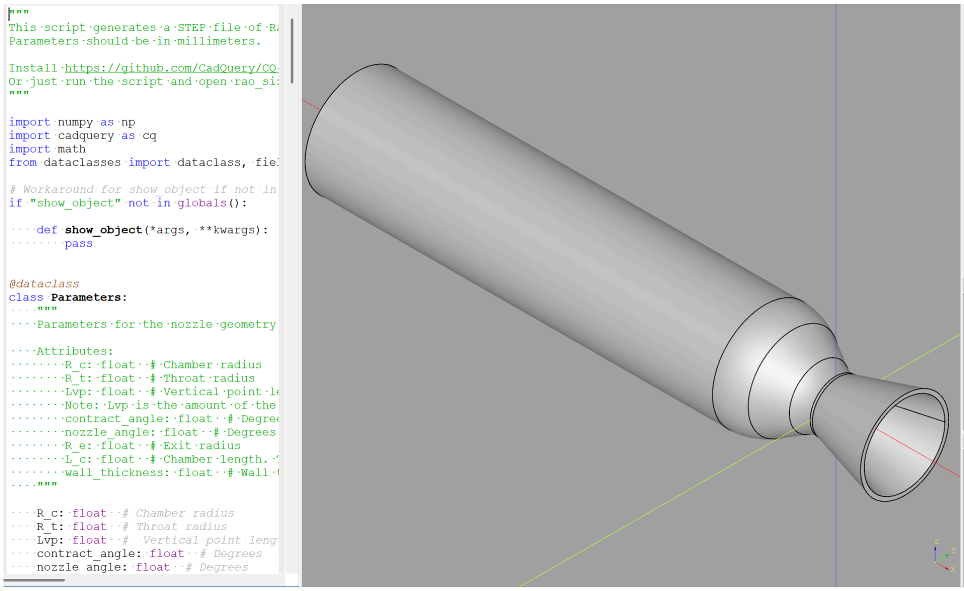


Rao's Nozzle Design Points



Nozzle Constraints in CAD Software manually sketched.

[6] "Under a given set of operating conditions, such as type of propellant, mixture ratio, chamber pressure, injector design, and chamber geometry, the value of the minimum required L* can only be evaluated by actual firings of experimental thrust chambers." http://www.braeunig.us/space/propuls.htm#engine



Code-generated CAD model.

Injector

We opted for a static pintle injector design (non-throttled), with a gaseous oxygen center (radial sheet exit) and six impinging fuel jets. Injector orifices were calculated using the following formulas, relating mass flow rate, pressure drop (25% in our case), and a continuous discharge factor. Pressure drop is considered both for the velocity of the exit fluid, which affects momentum, and protection against combustion backflow. We found that at least a 20% pressure drop from the injector inlet to outlet is recommended so any reactant or combustion does not travel backward.

$$A = \frac{\dot{m}_p}{C_d \cdot \sqrt{2 \cdot \rho_p \cdot \Delta P}}$$

p = propellant (fuel or oxidizer)

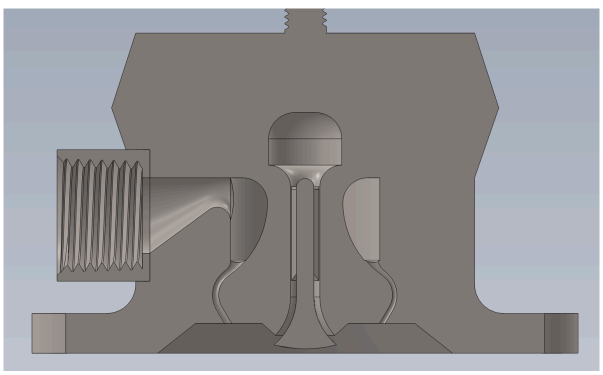
Next, we calculated a momentum flux ratio[7]. This is an accurate metric to judge how effective our sheet/jet impingement and thus atomization will be. For example, if the momentum flux per unit area ratio is 1, then the gaseous oxygen sheet and liquid ethanol jet will be colliding with the same force, which should result in the fluids breaking up and atomizing. A ratio less than or greater than 1 will result in either fuel or oxidizer

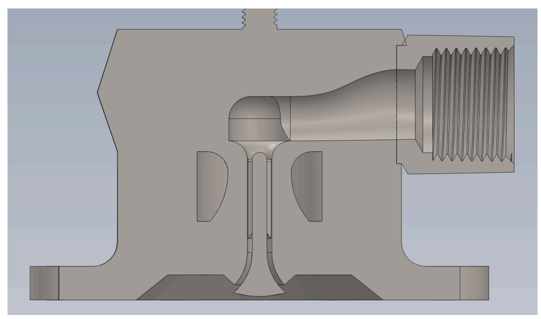
[7] Note, this is not a momentum ratio. Continuous masses of propellants are moving and crashing into each other at different speeds, so the momentum per unit time would be more useful. Then, if we cancel out area, then we can calculate a ratio of momentum flux *per unit area.*

overpowering the other, resulting in poor atomization. Reminder, atomization is important for efficient combustion because the surface area of our propellants greatly increases.[8]

$$MR = \frac{\rho_{ox} \cdot v_{ox}^{2}}{\rho_{f} \cdot v_{f}^{2}}$$

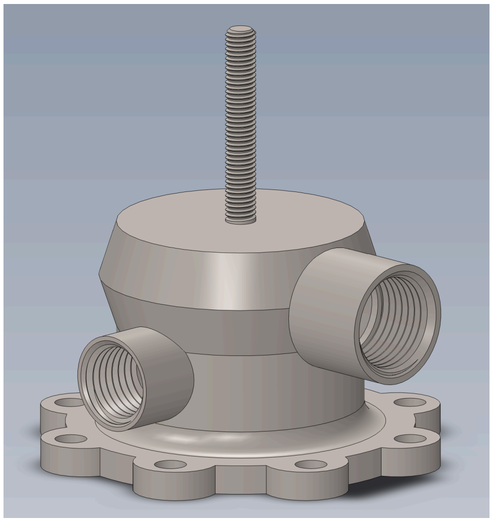
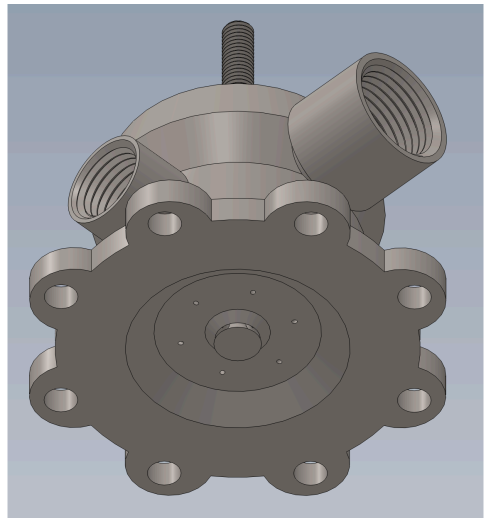
The rest of the injector geometry was designed around a few key ideas we found in literature[9]. First, our chamber diameter should be 3-5x the diameter of the pintle. Second, the pintle exit angle should be at least 20°. These design parameters were implemented to (hopefully) induce combustion vortices/recirculation zones in the space downstream of the pintle. These are important for increasing the stability and completeness of combustion. The injector, including the chamber, was 3D printed with a 17-4 Stainless Steel SLM printer. SLM printing results in rough surfaces, which was not ideal for the mating face of our flange. So we found a local machine shop and lathed <1mm off each face for a clean connection. We also chased the printed threads with a tap. Pictures of the injector model and printed face are included below.

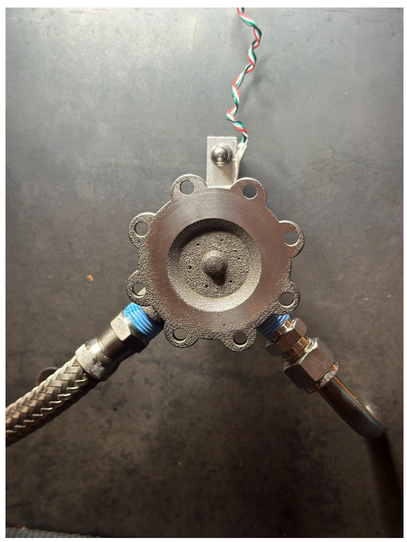




[8] Our gaseous oxygen exit velocity is ~150m/s. Previously, we were treating the GOX as incompressible, but at these speeds (mach .455), GOX may exhibit compressibility. It's unknown how much of an impact this had on density at the point of impingement, if at all. But we did not take this into account when calculating momentum flux ratio. There is a chance that we were overestimating the momentum flux of GOX. This would then result in an overpowered fuel jet not atomizing upon contact with the GOX sheet, ultimately leading to poor combustion in the chamber.

[9] Shankara Narayanan, P., Arun S., and Siddhartha Arul. Experimental Analysis on Pintle Fuel Injector for Rocket Engine. Bachelor's thesis, Hindustan Institute of Technology and Science, May 2022. https://hindustanuniv.ac.in/assets/naac/CA/1_3_4/696_Arun__S.pdf.



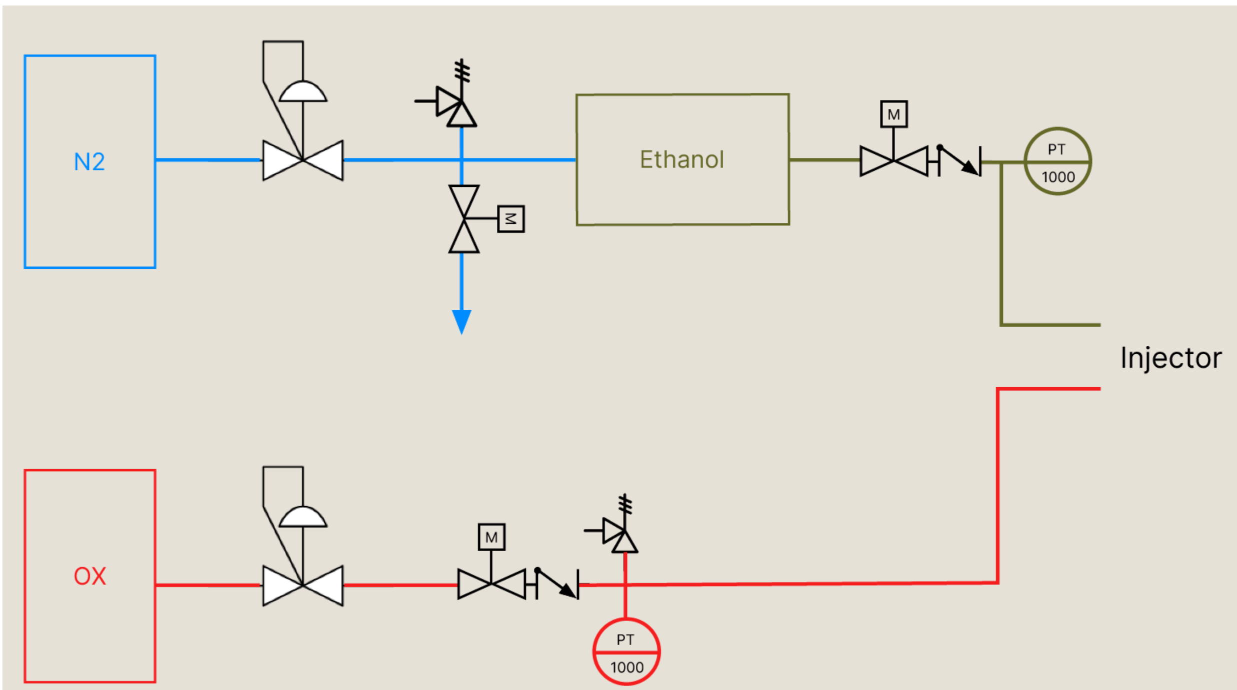
Plumbing Design

Given the simplicity of our design, we decided to use a pressure feed system. The ethanol tank would be pressurized using nitrogen and GOX comes pressurized.

Both tanks would have regulators. Amateurs coming from or reading about N2O may be tempted to skip the oxygen regulator, but this is entirely infeasible due to the required mass flow rate of GOX and the enormous volume in which one would need to store GOX at a lower pressure. Oxygen regulators are extremely expensive (we bought ours from Aqua Environment for ~$330), but a necessary cost.

We decided to equip the system with two PRVs and check valves for safety and use HalfCat servo-actuated ball valves instead of solenoids as a cost-cutting measure. We also added two pressure transducers on the end of each line for basic telemetry.

For projects with a larger budget and longer timeline, we recommend adding pressure transducers (PTs) before pressurization valves on both lines and on the injector and engine.



The P&ID of Gina, our rocket engine.

Flow Simulations

We conducted simple flow simulations following the pressure drop equations for compressible and incompressible fluids.

$$Cv = \frac{Q}{962}\sqrt{\frac{SG*T}{(P_1^{\,2} - P_2^{\,2})}}$$

Flow calculation for compressible gas over an orifice under sub-critical flow P1 < 2P2.

$$Cv = Q\sqrt{\frac{SG}{\Delta P}}$$

Flow equation for incompressible fluid over an orifice.

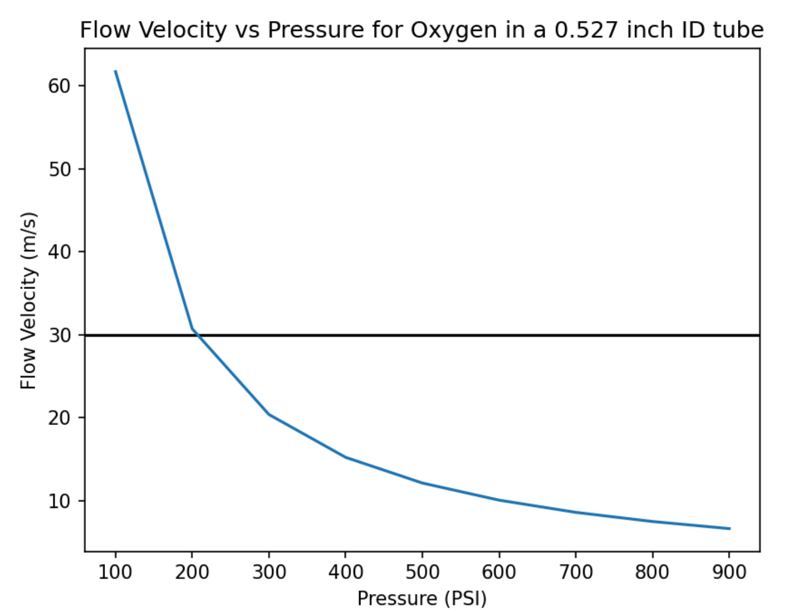
Our pressure drop simulations used gradient descent to minimize a simple error function, working backward to find the needed inlet pressure of the system.

```
Running simulation with parameters:
{'fluid': Ethanol,
 'fluid_purity': 100,
 'fluid_type': <FluidType.LIQUID: 'liquid'>,
 'SG': 0.787,
 'desired_outlet_pressure': 322.305555556,
 'T': np.float64(531.67),
 'mass_flow_rate': 0.0655906655,
 'valves': [{'name': 'Ball Valve 1', 'Cv': 1.5},
            {'name': 'Ball Valve 2', 'Cv': 1.5}],
 'initial_reg_pressure_guess': 1000,
 'iterations': 1000,
 'learning_rate': 0.1,
 'density_change_over_valves': True}
Ball Valve 1 Inlet P:  323.5188849001605 Outlet P:  322.9122230489817 dP:  0.606661851178842
Ball Valve 2 Inlet P:  322.9122230489817 Outlet P:  322.30555555600023 dP:  0.606667492981444
Final outlet pressure:  322.30555555600023
```

A sample output from a simulation run with two valves.

We were also concerned about the flow velocity of GOX through the line due to its potential to create heat and combust with any debris or materials. NASA recommends GOX be kept at <30m/s, while anecdotes from various amateurs suggest 50-60 m/s is safe, depending on line cleanliness.

Visualizing the flow velocity at various pressures provided us with the peace of mind that we were not going to see a GOX fire when pressurizing lines, manually or remotely.



V (m/s) vs. P (psi) of Oxygen in a 0.527 inch ID tube.

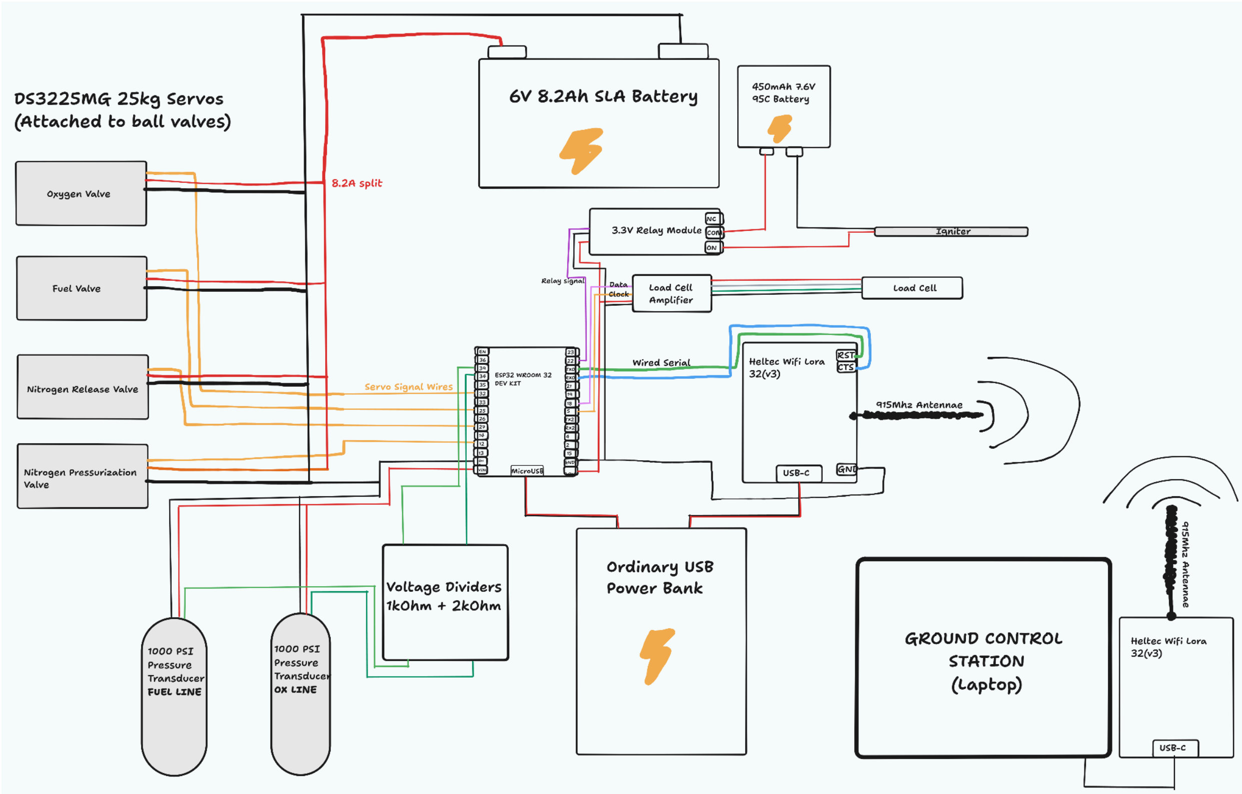
All scripts can be found here:
https://github.com/danielv012/GINA/tree/main/Simulations%20and%20Calculations

Electronics

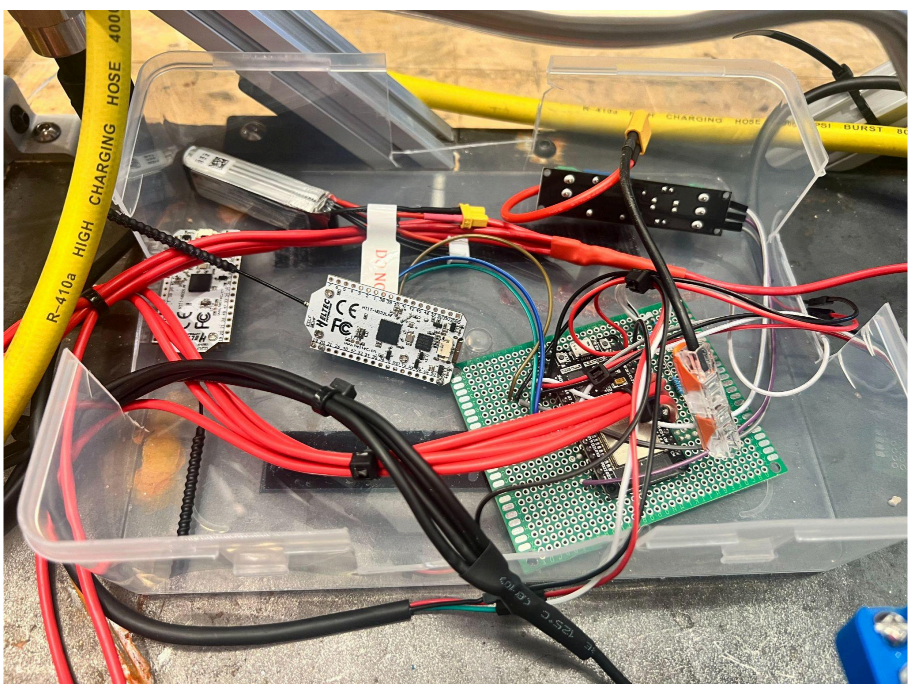
To control our servo-actuated ball valves and monitor telemetry, we opted for an ESP32 Dev Kit board that Kevin had lying around. We initially thought we could also use the ESP32's WiFi module and a potentially upgraded antenna for remote communication with a laptop. But later we chose to use radio to eliminate any range/connection/latency issues. The radio configuration will be discussed in the next section.

All connections to the ESP32 MCU were soldered onto a perfboard. Next time, a custom-designed PCB would be helpful, but it was nice to be able to resolder connections if necessary. Voltage dividers were used on the 5V signals from the pressure transducers since the ADCs on the ESP32 were limited to 3.3V. A USB power bank I had in my backpack was used to power both the ESP32 and the LoRa home board. A separate 7.6V LiPO battery was used to load the igniter, and a 6V 8.2Ah SLA Battery was used to power all servos.

As mentioned previously, we used servos to actuate our ball valves. Solenoid valves were out of budget. We followed the steps by HalfCat Rocketry, and they worked great. We just had to limit the operating range of each servo–one of our original servos burned out trying to reach an impossible position once it was tied down to the valve.

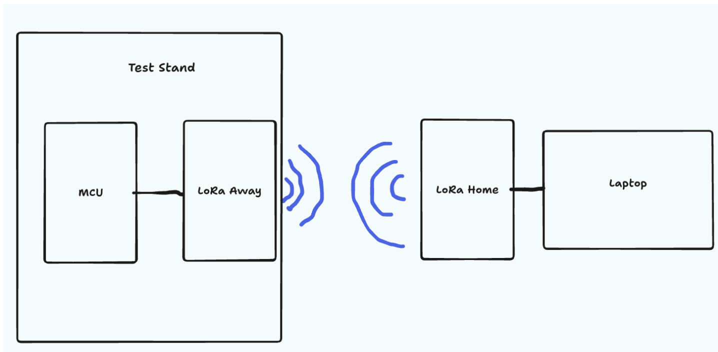


Complete Wiring Schematic



Remote System & Software ([GitHub](#))

To communicate securely and over long-range, we opted for 2x WiFi LoRA 32(V3) ESP32 boards. One board (home) would communicate with our Ground Control System/Control Panel over serial via USB-C to a laptop. The second board (away) would sit on the test stand and communicate over wired serial to the ESP32 MCU using UART. Both boards on the *test stand* are powered by an ordinary USB power bank.



Simple diagram

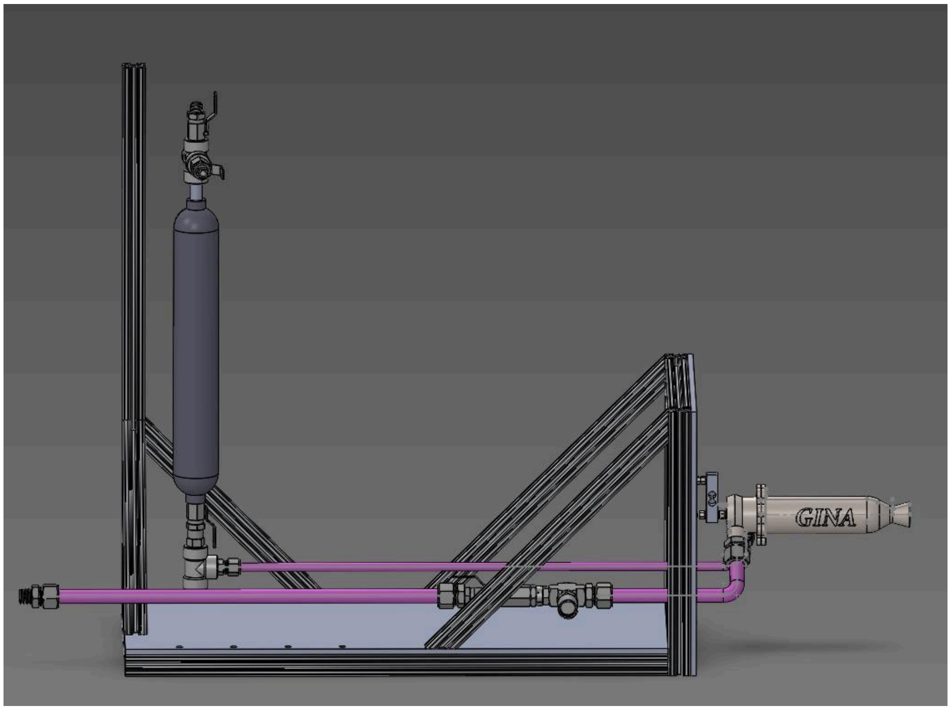
Radio packets were sent over 915 Mhz. A simple transmission and acknowledgment protocol was programmed to ensure command packets were received and processed accordingly. We were having trouble with radio traffic, which included commands, telemetry, acknowledgements, and heartbeats. So we also optimized radio packet timing and size. Safing was also added, such as closing all valves if a ping hasn't been received for X seconds. More can be done to add a blocking feature for the ignition sequence, or better error transmission and processing.

We also programmed a simple GUI for the laptop that displayed serial input/output, telemetry graphs (line pressure, load cell reading), and sequence/valve buttons.

Control Panel GUI

Test Stand

This was a simple build with aluminum extrusions and rolled steel plates for mounting and added weight. Our initial plan for both propellant lines was to route the tubing through appropriately sized holes in the front steel plate. Unfortunately, our copper tubes on the oxygen line were too difficult to bend and align with the injector, so we replaced the last section of copper tubing with a stainless steel hose (we should've done this for both lines from the beginning due to ease of use). Our fuel tank is seen standing vertically.



Test stand CAD model. (Servos, tanks, electronics, regulators not pictured).

Safety & Operation

Procedures were written and reviewed beforehand for the firing day. These included materials, priming, ignition sequence, contingencies, and clean-up.

Results

Fire 1: Partial combustion
Fire 2: Failure, no ignition
Fire 3: Partial combustion (IPA), improvised ignition
Fire 4: Failure, no ignition
Fire 5: Partial combustion (IPA, improvised ignition
Fire 6: Failure, no ignition

Failure Points

After deliberating and analyzing data from our six firing attempts, we have identified three (3) possible errors that alone or combined could've resulted in the partial combustion, or lack of complete combustion, we observed. Our initial flow test (pictured below) would also prove to be useless, as we were using a broken oxygen gauge and possibly a broken fuel gauge at this point. Our supplied nitrogen pressure during the flow test would've been 200+psi less than expected, which would've shown no mixing/impingement.

1. **Weak ignition source.**
2. **Weak propellant mixing as a result of overestimation of GOX mass flux.**
3. **Injector geometry and combustion chamber compatibility (sheet angle, impingement point, combustion/recirculation zones)**

While on test day we did observe a broken 600 psi gauge on our GOX regulator outlet, our three successful fires (1, 3, and 5) were fuel-rich, balanced, and ox-rich respectively. We believe this rules out a possible fourth error of improper propellant ratio and impact of broken gauges.

Weak ignition source

For our ignition, we used a switch-activated wire igniter that, upon completing a circuit with 9 amps, black propellant would ignite at the end of the coiled wire and hold a visible flame for around 5 seconds. As seen in the schematic and mentioned in the software section, this igniter was triggered during an ignition sequence and given one second to light before propellants would enter the chamber. This wire was not secured onto or in the chamber by any means. We simply pushed the black cap as far as we could into the chamber, which is where it ignited. This could've caused two problems: inadequate ignition (snuffed and/or blown out immediately), or ignition at the back of the chamber on the injector face. This could be the reason why we see a glowing ring at the back of the chamber, as we have partial combustion near the face of the injector, but none circulating in the center of the chamber.

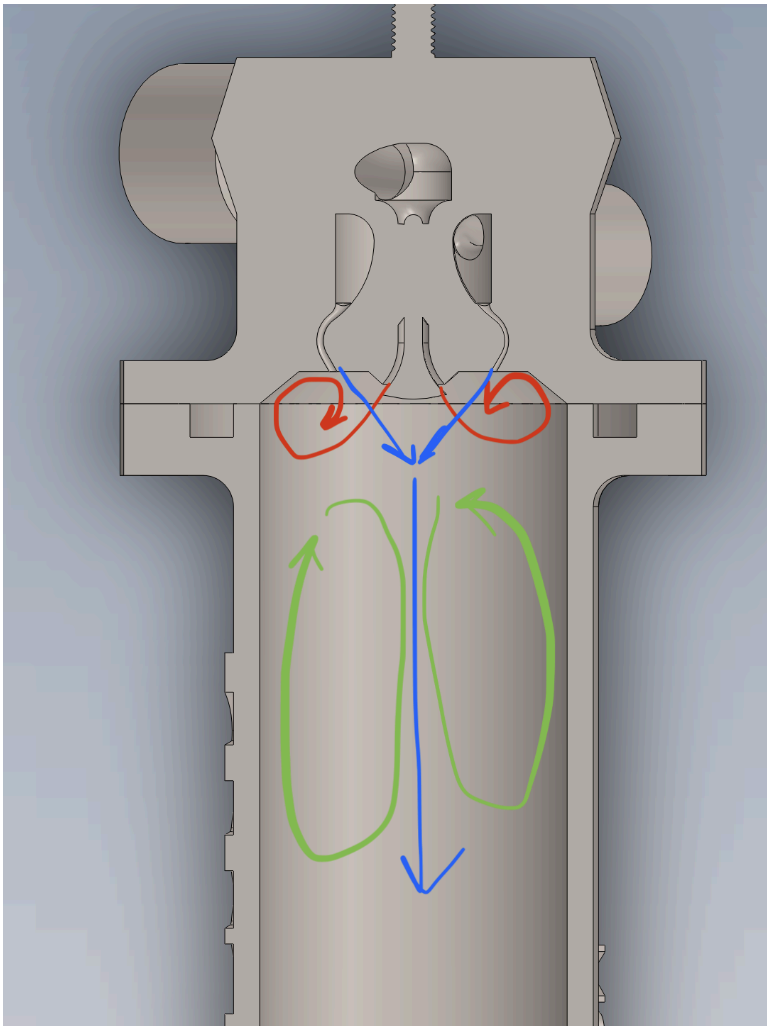
We ran out of these igniters for fires 3-6 and instead, used paper soaked in IPA, which we then lit with an igniter. Often, the flame would be snuffed out on the inside of the chamber, but exiting propellants could catch the flame. We see this in our videos as flames trailing the exiting propellant. This method was not sufficient at all and was only used because we had no other way to fire the engine.

Weak propellant mixing

We did not take into consideration that as oxygen reaches speeds above 100 m/s (calculated ~150 m/s), compressibility effects begin to take shape. At this speed, gaseous oxygen density may decrease, which leads to decreased momentum, and thus fuel is overpowering during impingement (fuel jets push right through the radial oxygen sheet and no mixing occurs).

Injector geometry and chamber sizing

Since we did not have a successful flow test, it is unclear how the flow cone was structured coming out of the injector. However, it is possible that the impingement angle or overpowering of fuel did not lead to recirculation zones in the middle of the chamber. This is pictured below with blue arrows. Also, our chamber diameter to throat ratio was 2.8x, instead of the recommended 3-5x. A wider chamber/nozzle could've benefited our injector geometry. It is also possible that recirculation zones formed near the injector plate, combined with inner ignition, resulted in partial combustion, pictured in red.



Moving Forward

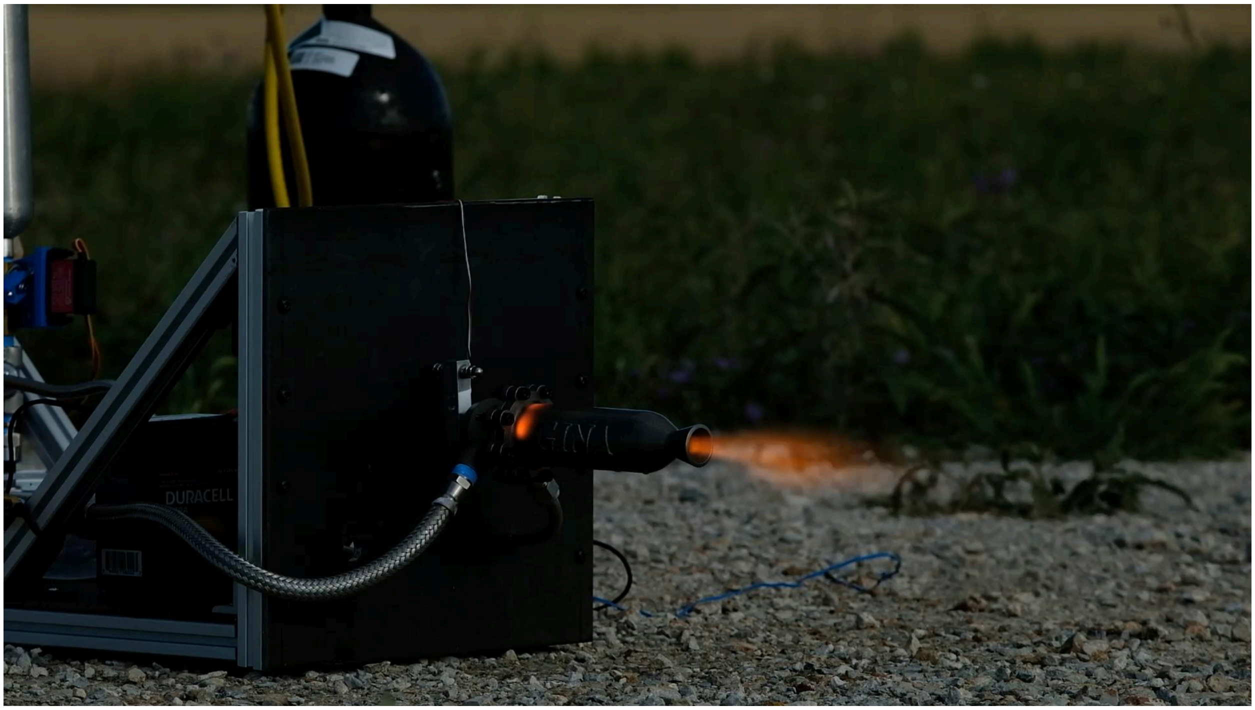
Given the results of the hotfires, a secondary injector flow test must be conducted with a new oxygen outlet gauge. The goal of this test is to confirm three things:

1. Propellant mixing under nominal conditions is thorough.
2. Pressure measured by PTs is nominal.
3. Mass flow rate (adjusted for chamber pressure) is correct.

Depending on the results of the secondary injector flow test, the injector may require a potential redesign to a coaxial shear injector, which is better suited for two-phase flow.

Future design improvements should also include a PT in both the injector and engine, a pyro cartridge igniter built into the injector or engine, PTs on both pressurant lines, a nitrogen purge line into the engine, and/or regenerative cooling.

Gallery


DURACELL


GINA


Natural Play