Cooled Liquid Rocket Thrust Chamber

F24

Senior Project Scope of Work

Prepared for: Cal Poly Space Systems

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Abstract

Cooling may affect the thrust output of a small-scale rocket. Little research has been published about small-scale rocket performance. The thrust produced likely varies as the amount of cooling varies. To facilitate testing this hypothesis, we will be designing and building a liquid rocket rated for at least 25 lb_f of thrust. Our objective is to build in parallel with Cal Poly Space Systems, who will build a rocket rated for at least 25 lb_f of thrust without cooling. Our challenge is to integrate compatibility, so that cooling may be tested on both designs. The result is an analysis of the effects of cooling on thrust output of small-scale rocket engines.

Table of Contents

Table of Tables

Table of Figures

1 Introduction

Cal Poly Space Systems is a club on campus that designs rockets. The club has mainly designed solid rockets and is trying to grow and design liquid rockets. This has not been done in the past and we are going to help.

We will design and build a liquid rocket rated for at least 25 lb_f of thrust. This design will run parallel with another team of Cal Poly Space Systems students who are designing a liquid rocket which has the same goal thrust. We will design our rocket with cooling and the Cal Poly Space Systems (CPSS) rocket will not have a cooling system. The goal of this report is to identify the criteria that we will base our design on and our project plan. This report will be split into different sections with each section and description listed below:

Background

The background section will describe the current research that we have done on the topic

Project Scope

The project scope section describes what we are going to design based on the stakeholders' needs and what we plan to deliver

Objectives

The objectives give specific measurable goals that we hope our design to achieve.

Project Management

The project management section outlines our process and our preliminary design schedule.

This report is to define the goal of our final design. We do not expect significant changes in the scope of the project, but we do expect to get more information or requirements as the project continues. These changes come from the project running parallel to other projects, specifically the design of the test stand. This is not in the scope of our project, but we expect it to influence our design and process so any updates to project will be included in future deliverables. The timeline for these will be in the project management section.

2 Background

2.1 Stakeholders & Needs

We met on Wednesday, October 5th, 2022, with our sponsors Cal Poly Aerospace professor and expert in hypersonic airbreathing propulsion Dr. Deturris, and CPSS president Mr. Bornhorst. After introductions, we began our meeting discussing the various objectives of our project to gain a better understanding of our task and our sponsor's needs and requirements. We established that our objectives are to design and build a reusable liquid rocket rated for at least 25 lb_f of thrust with the added aspect of cooling. Our design will feature interchangeable injectors to give insights as to how different amounts of cooling will affect the rocket's thrust output. As discussed in the introduction, our design will be running in parallel with the CPSS who will also be designing and building a liquid rocket rated for at least 25 lb_f of thrust, their design will not include cooling. Although we will not be performing the hot fire test, our design will be utilizing CPSS' test stand, fuel source, and electrical components. Because we will be utilizing CPSS' test stand, our final product must be designed around their fuel and oxidizer inputs, mounting points, and ignition connections with their test stand. The complete list of our stakeholders wants, and needs, can be found in section 3, the project scope. Mr. Bornhorst provided us with an additional request to comprehensively document the design process so CPSS will be able to keep a reference and improve the design. Mr. Bornhorst also briefed us with a synopsis of CPSS' project timeline, informing us they plan on hot fire testing in the spring.

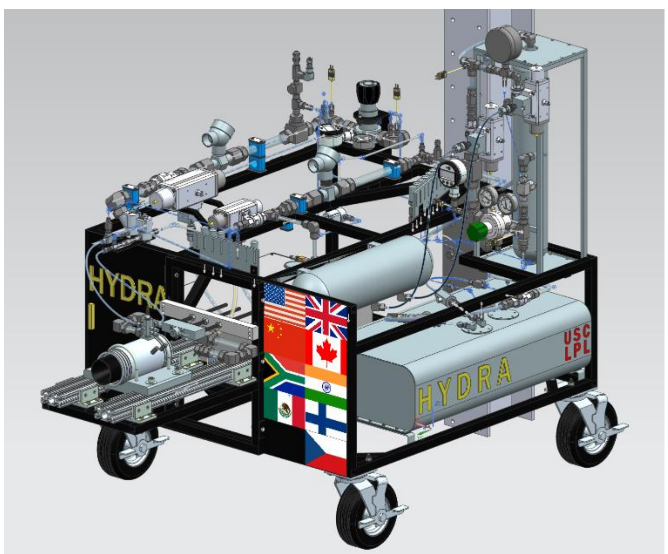

Figure 2.1-1 Example test stand configuration similar to what CPSS will provide [2]

On Tuesday, October 11th, 2022, we met with the CPSS Ground Support team to get briefed on their progress. They have not yet purchased the trailer, which will be the base of the test stand, and as such are not yet able to give us many specific constraints. However, we did learn that the maximum thrust the trailer will be specified to is 1,500 lb_f. The trailer will most likely be made of mild steel, with the frame being set up for a vertical fire. There will be a load cell, to calculate the thrust produced by the rocket, attached to a vertical plate near the end of the trailer. To ensure safety, CPSS will be using a factor of safety around 2.5. Our design will also utilize high factors of safety, where possible, to minimize risks of injury. One component that needs more research is the ignition system, which is a combination of the test stand and engine teams. We were informed that by the end of this quarter, the test stand should be designed, and we will be able to make our design compatible with the CPSS test stand.

2.2 Literature Review

As the field of rocketry has been established for half a century, quite a few technical reports have been authored on the topic. The seminal text on the subject is *Rocket Propulsion Elements* [3], [4] by George Sutton, a leading American rocket scientist from the 1940s to 2020. This work, originally published in 1949 and updated 9 times, is often cited and explains the general systems used in a rocket engine, what variations exist, their pros and cons, and basic design guidelines. Another general guide used is *Modern engineering for design of Liquid-Propellant Rocket Engines* [5] by Huzel. We have used these two sources to frame our research and create a list of likely components that our project will require and their associated technical challenges. The components are the injector, combustion chamber, cooling system, and nozzle. These will be covered in more detail in section 2.4 Technical Challenges.

In addition to general texts, we also utilized more specific reports that pertained to one subsystem. These reports are more useful as design guides than the general texts as they go into detail regarding what parameters to change and their effect on the subsystem. *Liquid Rocket Injectors* [6] produced by NASA, *Review on Film Cooling of Liquid Rocket* [7] by Shine, and *Nozzle Design, Converging-Diverging (CD) Nozzle* [8] produced by NASA all provide specific instructions regarding how to design the component. Additionally, they have more detailed explanations on the pros and cons of certain designs.

2.3 Competitor Analysis

Most rockets that exist are of a much larger scale than what is called for by our customer. The space shuttle main engine for instance has a thrust 20,500 times our goal [9]. So, we will focus on academic student designs and small satellite propulsion systems. The studied academic projects were from Purdue University [10], University of Akron [11], and Arizona State University [12], and the studied small satellite propulsion systems were from Dawn Aerospace [13] and Stellar Explorations [14]. Using Purdue's rocket as a representative example, it had a thrust of 1125 lb_f, did not have film cooling, and was mostly manufacturable in a university shop. [10] The rocket's thrust, visible in Figure 2.3-1, was almost 2 orders of magnitude higher than our own goal of 25 lb_f so some of the design decisions made in this rocket may not apply to our rocket. The combustion chamber had a diameter of around 8 in and employed a pentile injector system. Stellar Exploration's bi-prop engine produced thrust of 1 lb_f, had a diameter less than 2 in and used a simple hole injector. Stellar Exploration's engine and the similar Dawn Aerospace engine are shown in Figure 2.3-2 and Figure 2.3-3 respectively. Therefore, we will need to exam the driving factors between these two representative designs and make our decision. Unfortunately, there were no rocket engines within 2 orders of magnitude of our goal that employed film cooling, and adjustable film cooling does not exist at all. Therefore, we will consult design guides made for much larger engines such as the RS-25 used as the space shuttle main engine.

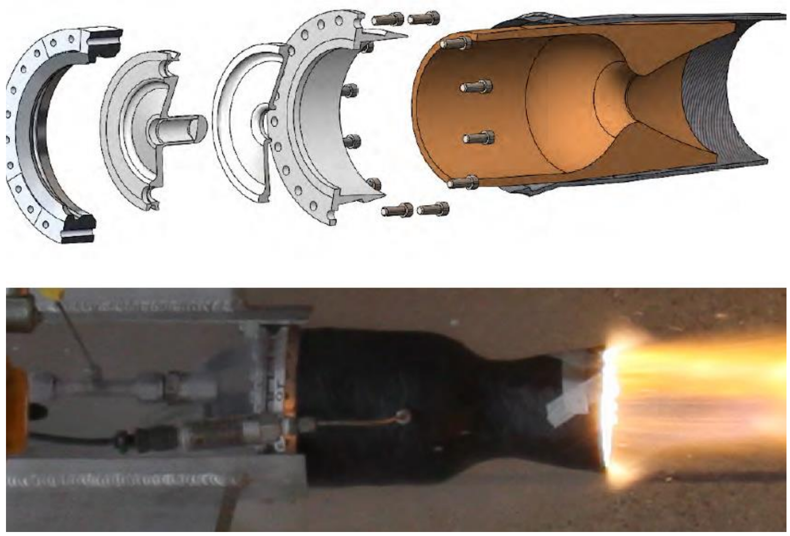

Figure 2.3-1 Purdue rocket engine CAD asem. and test fire [10]

Figure 2.3-2 Stellar Exploration 1 lb_f engine

Figure 2.3-3 Dawn Aerospace 5 lb_f engine

2.4 Technical Challenges

A liquid rocket engine is composed of three main components, the injector, the combustion chamber, and the nozzle. The injector injects fuel into the combustion chamber. Depending on the fuel, an oxidizer also needs to be injected into the combustion chamber. The combustion chamber is where fuel is burned and where the energy comes from to create thrust. The nozzle directs the exhaust from the combustion to create thrust. Each component has to work together to ensure that the rocket works properly and creates thrust. This section describes some of the challenges that our design will have to overcome to work properly. These challenges were found using textbooks and technical reports.

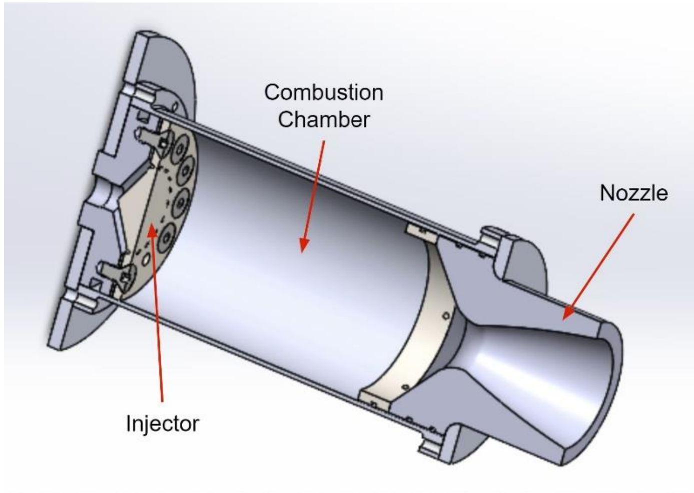

Figure 2.4-1 Example layout of rocket components (sans cooling system) [15]

2.4.1 *Injector*

Combusting propellants is one of the primary functions of a rocket engine. To do this, propellants must be accepted from the tanks they are stored in, guided to the combustion chamber, mixed thoroughly, and ignited. The primary way this is accomplished is the injector assembly. The injector assembly traditionally accomplishes this with four subparts: propellent inlets, manifold, injector holes, and igniter.

The propellent inlets is perhaps the simplest of the four subcomponents. It will need separate inlets for the fuel and oxidizer. The inlets will need to be standard couplings such that the thrust chamber can interface with the feed system which is beyond the scope of this project. At this stage, there must be a way to change the amount of fuel diverted to the film cooling.

The manifold is the system that guides the propellants from the inlets to the combustion chamber. It is extremely important that the propellants do not contact each other because that would disturb the local ratio of propellants and could lead to combustion instability and possible an explosion. Therefore, it is prudent to seal the oxidizer and fuel channels from each other, possibly using a series of O-rings. Given the goal to construct this using Cal Poly machine tools, the 3d geometry required to route these propellants without contact is challenging. One possible method as shown in Figure 2.4-2 uses stacks of 2.5d machined plates that are combined to create a 3d series of passages. However, as the propellants flow though these passages there are losses in pressure, as well as difference in pressure in relation to position as seen in Figure 2.4-3. This will require either mitigating methods to smooth out the pressure differential or the injector placement will need to account for the differences in pressure.

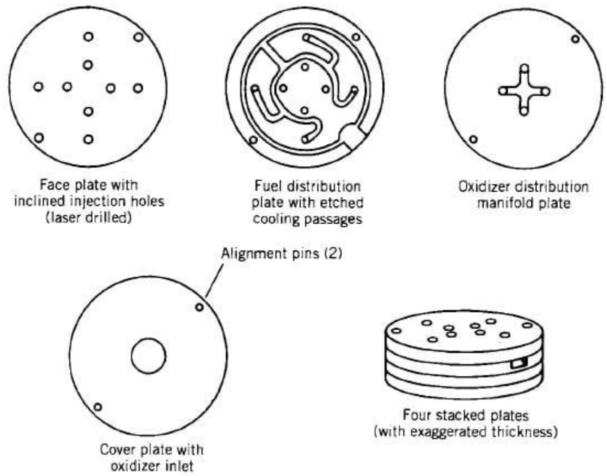

Figure 2.4-2 Stacked manifold design

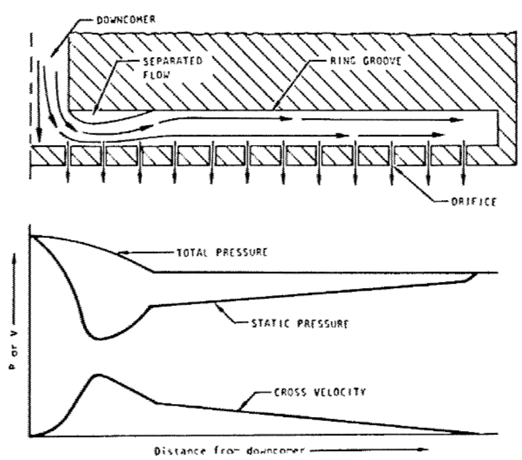

Figure 16. – Variation of pressure and velocity in a constant-area ring groove.

Figure 2.4-3 Pressure vs. radial position along injector [6]

The mixing of propellants is a key aspect of proper combustion. The design ratio of oxidizer to fuel, which is usually close but not exactly the stoichiometric ratio, is important to maintain. If certain regions of the chamber have a relatively high or low ratio, these areas will combust differently and cause hot spots that may damage the engine and possible explosions. To prevent this, it is important to ensure proper mixing. The simplest method to do this is a showerhead design where the oxidizer and fuel exit the injector in axial streams which mix due to diffusion and turbulence. This method is no longer used in large scale engines due to its inefficiency and requirement of long combustion chambers for proper mixing. The most common method is to use impinging injector offices. This involves directing propellants jets to hit each other and the resulting fans of dispersed propellants will mix. There are multiple versions of impinging orifices that vary the number of holes and whether “like” or “unlike” propellant jets hit each other as shown in Figure 2.4-4. [4] [6]

Table III. – Comparison of Typical Injector Elements for Gas/Liquid Injection

ELEMENT DESIGNATION	ELEMENT CONFIGURATION (FLOW DIRECTION)	ADVANTAGES	DISADVANTAGES	DESIGN CORRELATIONS*	ENGINE APPLICATION
UNLIKE TRIPLET (2 ON 1) ●○●	LIQ GAS LIQ	Excellent atomization Good overall mixing Resultant spray direction is axial	Wall compatibility is good only when fuel is used in outer orifices Sensitive to design tolerances Performance sensitive to continuous throttling	Mixing (ref. 12) Atomization (ref. 12)	None known
UNLIKE PENTAD (4 ON 1)	LIQ GAS LIQ	Excellent atomization Good overall mixing Applicable to extremely high or low mixture ratios or density ratios Well characterized	Difficult to manifold Wall compatibility problems Performance sensitive to throttling Tends to produce high heat flux to injector face	Mixing (ref. 12) Atomization (ref. 12)	None known
CONCENTRIC TUBE (WITH SWIRLER)	GAS LIQ RECESS	Excellent mixing and atomization Low pressure drop Proven dependability	Difficult to fabricate if annulus gap is very small (<0.06 in.) Tends to become unstable when throttled	Generalized (ref. 13)	J-2, RL-10, M-1 Russians use this element extensively
CONCENTRIC TUBE (WITHOUT SWIRLER)	GAS LIQ	Very good wall compatibility Low pressure drop Excellent atomization	Same as above	Mixing (refs. 5 and 14) Atomization (refs. 5 and 14)	
LIKE DOUBLET (1 ON 1) ● ● OR ○ ○	LIQ OR GAS	Easy to manifold Excellent for deliberate control of spray for wall temperature Good mixing Very stable element	Requires increased axial distances to mix the fuel and oxidizer Sensitive to design tolerances	None	None known

*For circular orifices unless otherwise noted.

Figure 2.4-4 Table of injector types [6]

The final subcomponent is the igniter whose purpose is to ignite the mixed cloud of gaseous propellant. Igniters can either be reusable or expendable. If an igniter is expendable, it acts like a high-tech match that needs to have a lit flame inside the rocket engine until the gasses ignite and expel it. This presents issues of preventing the nozzle from damage as the igniter is expelled as well as preventing the igniter from becoming a projectile. If the igniter is reusable, it will have to withstand the high temperatures of combustion and the also not present a leak path for the high-pressure gas in the combustion chamber.

2.4.2 Combustion Chamber

The combustion chamber is where the fuel is burned. The temperature that is reached in this process is enough to melt the walls of the combustion chamber. This leads to the first challenge in the design of the combustion chamber, cooling. The combustion chamber needs to be cooled or else it will fail. The main ways of cooling are steady-state and unsteady. Thermal equilibrium is reached in steady-state methods. Unsteady methods do not reach thermal equilibrium and are not sustainable for when the engine needs to be burned for longer periods of time.

Steady state cooling is achieved in two ways, regenerative cooling, and radiation cooling. Regenerative cooling is done by circulating fuel around the combustion chamber before it is burned. They absorbs the energy that is released. This technique is mostly used in larger rockets as it makes manufacturing harder and might not be needed in smaller applications. Radiation cooling uses material that can withstand higher temperatures to radiate heat away from the engine.

Unsteady methods are when the combustion chamber is designed to absorb the energy that is released. This method limits the time that fuel can burn but is much simpler to construct. The material choice and the wall thickness are the main factors that tell how long the fuel can burn in this method.

There are some methods that are used in combination with both methods, film cooling and ablative materials. Film cooling is when the injector injects fuel onto the walls of the combustion chamber and creates a film that will absorb heat. Ablative cooling methods use a material that is placed inside the chamber that burns up, slowing the heat transfer to the wall. This makes the engine single use because this material needs to be in the chamber for it to work.

The combustion chamber also needs to be able to withstand the stresses that come with being under pressure. The design needs to be strong enough to not fail while being under thermal load and the stress from being pressurized. This is done with material choice and dimensions. The size of the chamber relates the amount of heat transferred to walls the exit velocity of the gas. Larger chamber volume leads to lower velocities and less heat transfer. The dimensions also must allow for the combustion to be stable. A smaller volume chamber will require higher pressure to achieve complete combustion.

[4]

2.4.3 Nozzle

The nozzle of a rocket is where the exhaust gases produced by the burning propellants expand and accelerate, leaving the rocket at hypersonic velocities. Thrust, as described by Newton's third law of motion, is produced through the momentum created due to the expulsion of mass from the rocket. The amount of thrust produced is dependent on the mass flow rate through the engine, the exit velocity of the flow, and the pressure at the exit of the engine. The values of these flow variables are determined by the design of the rocket nozzle.

A nozzle is relatively simple, consisting of a specially shaped tube through which hot gases flow. The most common nozzle used in rockets today, is the convergent-divergent nozzle. The convergent section, which is adjacent to the combustion chamber, is followed by the throat and a fixed divergent section. In this type of configuration, the hot exhaust leaves the combustion chamber and converges down to the minimum area, or throat, of the nozzle. The throat size is chosen to choke the flow and determine the mass flow rate through the system.

The flow in the throat is sonic, which means that it will be subjected to high temperatures and must be designed to withstand those high temperatures. Although the flow expands and accelerates to a supersonic Mach number, which causes the temperature to decrease from the throat to the exit, the divergent section of the nozzle will also need to be designed to withstand high temperatures. This will be done through material choice and dimensioning.

Nozzle design will be based off mass flow rate, fuel ratio, and several other variables. Calculations and analysis using Sutton as a reference will be supplemented with computer simulations through the software Rocket Propulsion Analysis (RPA). More analysis is needed before a nozzle is selected.

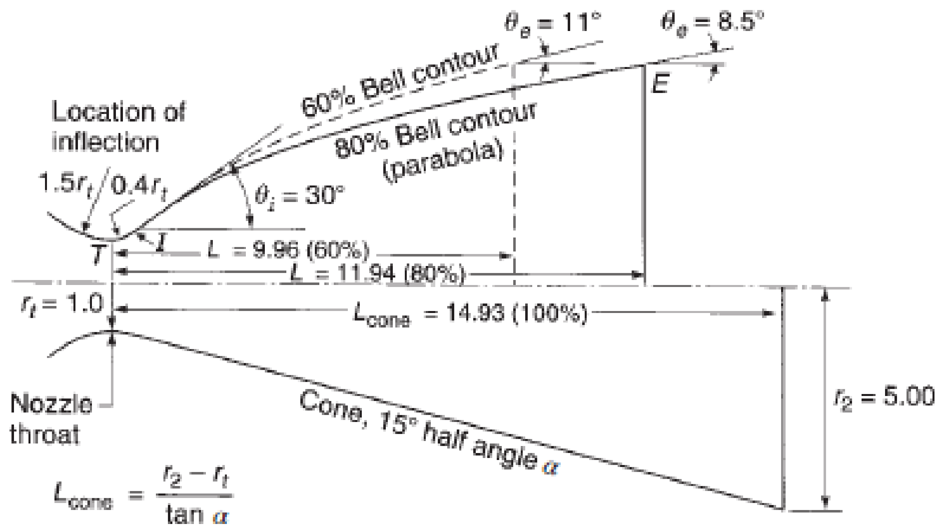

Figure 2.4-5 Convergent divergent nozzle [8]

[4] [8]

2.4.4 Cooling

During the combustion processes a lot of heat is produced and this heat can weaken the combustion chamber or nozzle walls and cause failure. The cooling system must work in synthesis with the previously discussed injector, combustion chamber, and nozzle. We aim to implement a film cooling system that will use excess fuel to cool the combustion chamber walls. Shown in Figure 2.4- is a simplified example of how the film cooling can be implemented. Most of the fuel and oxidizer combine and combust to provide thrust, but a portion of the fuel is redirected to the combustion chamber walls. Having the liquid fuel on the wall requires the heat from the combustion to provide energy to the fuel to go through a phase change before heating the actual wall material.

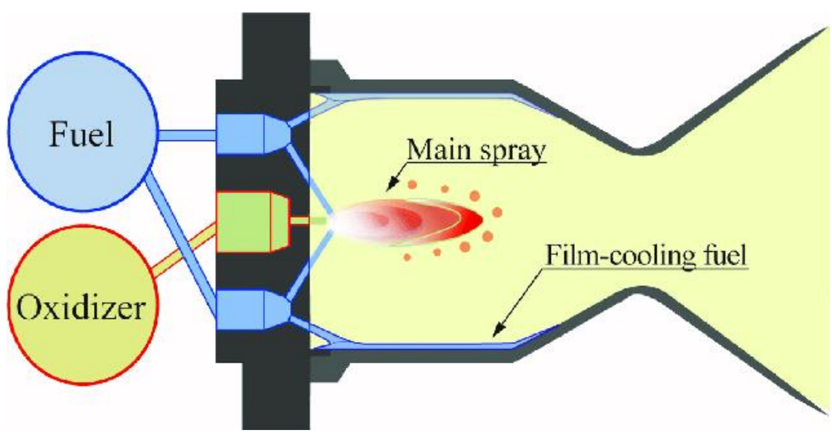

Figure 2.4-6. Simple cross section of film cooled thrust chamber [16]

Film Cooling has been performed with both liquid fuels and gaseous fuels, more studies have been done with gaseous fuels, but the process of the phase change makes the liquid fuels a better option. The cooling is also heavily influenced by the method of injection. This has to do with the angle, size of hole and many other considerations per Shine [7].

2.5 Patents

As mentioned in the current solutions and the technical challenges, many of the current rockets are on a much bigger scale than what will be made in this project. The patents found in our research describe solutions that are most likely out of reach of our project scope but are a good reference.

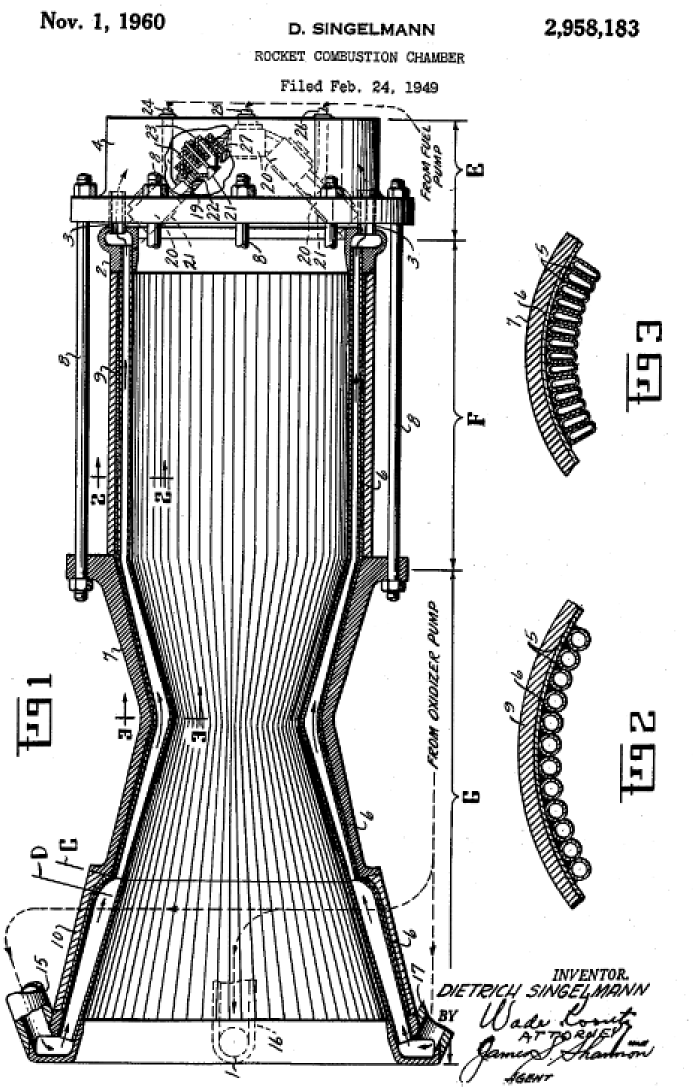

Figure 2.5-1: Patent for a combustion chamber with regenerative cooling

Figure # is a patent for a combustion chamber and nozzle with regenerative cooling. The tubes show in the right of figure # show how the fuel is circulated around the chamber to be used for cooling. This patent describes the geometry of the tubes that are circulating the engine. The tubes need to be flattened at different sections of the chamber because of the converging diverging nozzle. [17]

Other relevant patents describe the material that the rocket engine is made of. The internal temperature can reach up to 6500° F. One invention can create of nozzle out tungsten. Tungsten can withstand the internal temperatures but is extremely hard to machine. Using a plasma arc, the tungsten can be sprayed onto the contour of the nozzle and the layers can be built up to create the shape of the nozzle. This allows for a changing ratio to be used where the inner layer of the nozzle can be mostly tungsten and the percentage can change as more layers are applied. The nozzle is then encapsulated in a steel shell to keep everything in place [18].

Material choice needs to balance strength and temperature resistance. The problem is that this limits many high temperature resistant materials because of their lack of strength. A ceramic, for example, can withstand very temperatures but is extremely brittle so it is not possible to be used for a combustion chamber. We found a patent that uses a fiber reinforced ceramic to create combustion chambers. The fiber reinforcement can withstand 800° C. The temperature that it can withstand is not extremely high, but it can be manufactured in a way that builds a shell, so cooling is easier. [19]

The relevant patents were useful information for helping us learn more about liquid rockets, but due to the scale and the resources available to us they will most likely not be helpful for the scope of our project.

3 Project Scope

3.1 Boundary Sketch

The boundary sketch in Figure 3.1- depicts the area of the complete design our team is responsible for. The complete project involves using the engine being designed with the test stand being constructed by the Cal Poly Space Systems (CPSS) Ground Support Equipment (GSE) team. We are responsible for designing the thrust producing portion of design and allowing for connection to the frame and plumbing of the test stand.

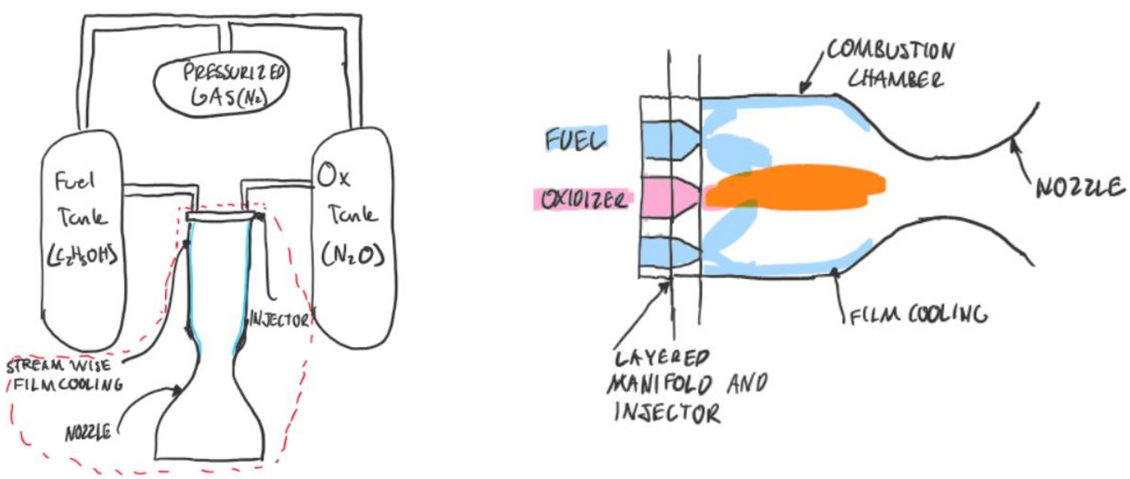

Figure 3.1-1. Boundary sketch (left) and lower-level boundary sketch of thrust chamber (right)

3.2 Stakeholders' Wants & Needs

The stakeholders being the Cal Poly Space Systems club and associated faculty and students as discussed in section 2, had outline a few needs and wants that they expected from the project. These needs and wants are listed below and are refined into specification in section 4. The needs and wants are listed as follows:

- Minimum 25 lb_f of thrust
- Adjustable film cooling (optional)
- Compatible with CPSS ground support equipment (GSE)
- Well documented design process
- Places to attach thermocouples
- Manufacturable with Cal Poly resources
- Nitrous and Ethanol bipropellant
- Operate vertically
- ~5 second burn time
- Remain within $1200 budget
- Reusable

3.3 Functional Decomposition

The design should be capable of the functions depicted in Figure 3.3-. The primary function of our design is to produce thrust. To produce thrust in the manner our customer's desire we will need to integrate our design with their test stand, facilitate the combustion of the propellants, accelerate the exhaust of the combustion, maintain structural integrity of the design, and provide cooling to the design. These secondary functions of our design will be the driving principles when coming up with creative solutions to our problem.

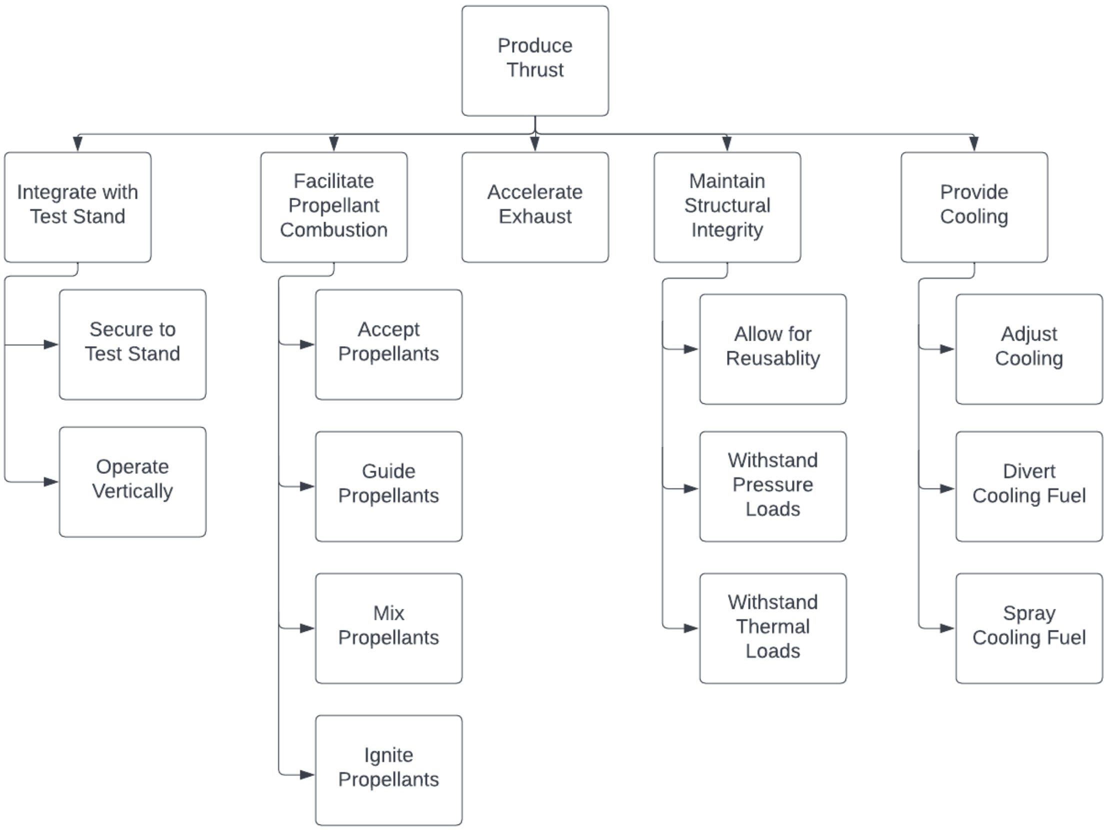

Figure 3.3-1. Function decomposition

3.4 Expected Deliverables

We plan to deliver the protypes that we develop that fulfils all the function requirements shown in Figure 3.3- and meet all the requirements in Table 4-1. The prototype will be capable of performing a hot fire test with integration to the CPSS test stand. Along with the hardware we plan to provide CPSS with all the CAD, CAM, and CAE files that would be necessary to reproduce the prototype. We plan to provide the club with all the design decisions that were made and the calculations and analysis that back them up. Finally, we will provide CPSS with all results from the test that will be conducted. All deliverables will be properly organized and placed in the CPSS SharePoint for all members to access and learn from for the future of liquid propulsion in the club. Documentation will be delivered in the format of this document, the Scope of Work, the preliminary design report, critical design report, and the final design report.

4 Objectives

4.1 Problem statement

CPSS needs a well-documented small liquid rocket engine with cooling to be used as reference to create larger rockets in the future.

4.2 QFD Process

During the QFD process we collected the requirements we got from our research and sponsor interview and implemented them into the house of quality spreadsheet in appendix A. We weighed each requirement based on the different customers for our product. Most of the customers had similar weights on each requirement. From these requirements we produced 9 tests to conduct to prove we meet the requirements described by our customers. These 9 tests are the basis for our engineering specifications described in Table 4-1. After appropriately weighing the customer requirements and their relations to the test and engineering specifications the technical importance rating of our specifications was out putted and gives us a good understanding of which specifications are the most important to focus on.

4.3 Engineering Specifications

Proceeding the QFD processes a series of engineering specifications were created. Each specification has a target, risk and compliance summarized in Table 4-1. These specifications are subject to change depending on the input of our sponsor or coach, or changes in the scope of the project.

Table 4-1. Summary of engineering specifications and tolerances

Spec. #	Specification Description	Requirement or Target	Tolerance	Risk*	Compliance**
1	Thrust	25 lb_f	±10	L	A
2	Max temperature	FoS of 2.0 ult	N/A	H	A, T
3	Fit to Test Stand	5'x3'x2'	N/A	M	I, T
4	Components manufactured at Cal Poly	All	N/A	H	I
5	Pressure Capacity	2.0 Ult	N/A	M	A, T
6	No Welded/Permanent joints	0	N/A	M	I
7	Cost	$1200		L	I
8	Burn time	5 seconds	±2	L	A, S
9	% of Fuel for film cooling	15%	±10	H	A, T

* Risk of meeting specification: (H) High, (M) Medium, (L) Low

** Compliance Methods: (A) Analysis, (I) Inspection, (S) Similar to Existing, (T) Test

4.3.1 Descriptions of All Specifications

1. **Thrust:** The rocket should produce 25 lb_f of thrust
2. **Max temperature:** The maximum temperature reached by the combustion chamber material should not diminish the strength of the material to the point of failure with a factor of safety of 2.0

3. **Within bound box:** The entire design will remain within the bounding box of 5'x3'x2' specified by the CPSS group support team.
4. **Components manufactured at Cal Poly:** All custom components designed by our team will be manufactured in Cal Poly machine shops. This is not to say O-rings, fasteners, or other standardized components.
5. **Pressure capacity:** The combustion chamber shall be capable of holding the expected pressure extrapolated from analysis with a factor of safety of 2. The margins of safety will be calculated in accordance with NASA-STD-5012.
6. **No Welded/Permanent joints:** The design shall contain no welds or any other manor of permanent joints, instead all joints shall be connected using fasteners and shall be analyzed per NASA-STD-5020.
7. **Cost:** The cost of the design including purchasing of raw materials, manufacturing, and testing of all prototypes will remain below the allotted balance of $1200.
8. **Burn time:** The engine is expected to exhibit a burn time of roughly 5 seconds.
9. **Percent of fuel for film cooling:** The amount of fuel used for film cooling should be able to vary from 5% to 25%.

4.3.2 High risk specifications

The high-risk specifications are the max temperature, components manufactured at Cal Poly, and percent of fuel for film cooling as shown in Table 4-1. These three specifications are marked as high risk as we believe they will be the most difficult specifications to meet. First, we believe the max temperature specification will be difficult to meet because this will heavily depend on the effectiveness of the film cooling and time of the injection and ignition. This process will involve the cohesiveness of several complex designs and will be difficult to measure the effectiveness. The second specification we flagged as high risk is the manufacturing of parts at Cal Poly. We believe this will be difficult because the parts involved in the injector are quite complex and will more than likely require the use of a CNC mill. The parts composing the combustion chamber and nozzle will require delicate use of a lathe. The combination of these complex manufacturing techniques is why we believe the manufacturing of our design at Cal Poly will be difficult. The final specification we deemed as high risk is percent of fuel for film cooling. We believe this will be one of the most difficult designs on our rocket engine because it will require the interchangeability of the injector to compensate for differing amounts of requested film cooling.

5 Project Management

The overall design process will begin after the scope of the work is agreed on. The next deliverable will be the preliminary design review. This will explain the direction that we plan to go in any design justifications made. This part of the project will include a concept protype. We plan to have multiple concept protypes for each component to allow for configurations to optimize performance for any future club needs. The next step will be to create a structural prototype. This will include a more functional protype. This prototype will be more for understanding how all the rocket components fit together. The protype will be presented in the Critical Design Review. We will then provide our final protype and report on all design decisions.

This is a very general project plan and more specific details will be given in later reports once we can make more design decisions. The included Gannt chart in the appendix gives specific dates for our project plan. A summary of these dates is included in Table 5-1.

Table 5-1. Dates of key milestones and descriptions

Date	Milestone	Description
11/15/2022	Preliminary Design Review	Concept protypes with a description of our design direction
2/14/2023	Critical Design Review	Detail of our Final design
3/16/2023	Manufacturing Review	Plan of our manufacturing process
5/23/2023	Design Verification Plan	Test of prototypes/design
6/9/2023	Final Design Review	Results of design

This next deliverable is the preliminary design review. This will include a concept of a final design based on the functional decomposition in Figure 3-2. We have broken the design down into five different components. We currently have assigned team members to each component and have a plan to create a preliminary design for each function. Each component has a very similar plan that includes brainstorming, preliminary analysis, and key findings/decisions. The preliminary design review will detail key findings and our planned design direction. Our major tasks to get to the preliminary design review is to create a concept protype that accomplishes each function and a justification on how it fits the function. This will not be our final design decision as we will still have other decisions and will include other alternatives in the preliminary design review.

6 Conclusion

This year we hope to provide Cal Poly Space Systems with a bipropellant rocket engine that produces 25 pounds of thrust. We expect this rocket engine to be cooled using film cooling that can be adjusted. The rocket engine will easily mount to the CPSS test stand and to its plumbing. This document aims to outline how we understand the scope of this project and what sorts of requirements we expect to meet. This document also outlines briefly what sorts of design challenges we expect to face, so we can jump on them as soon as we can. We want to reiterate some of the key points from this document such as our engineering specifications outlined in Table 4-1, the needs and wants of our stakeholders outlined in section 3, and the timeline we aim to meet throughout this project outline on our Gantt Chart in appendix BB. The next deliverable that can be expected from us is the preliminary design report which can be expected on November 17th. Finally, we would like to ask if you agree with the scope we have outlined in this document and would encourage any feedback from you.

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Appendices

A. QFD House of Quality

Correlations	
Positive	+
Negative	−
No Correlation	

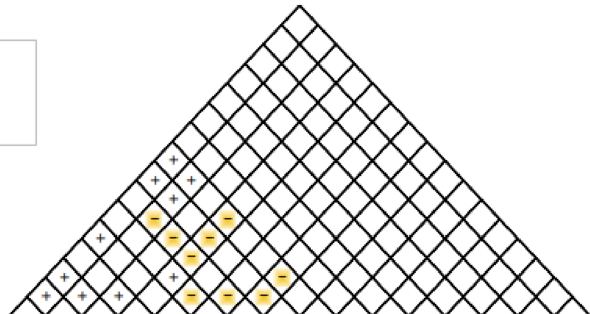
Relationships	
Strong	●
Moderate	○
Weak	▽

Direction of Improvement	
Maximize	▲
Target	◇
Minimize	▼

QFD House of Quality

Project: Thrust or Bust

Revision Date: 10/11/2022



WHO: Customers

Row #	Weight Chart	Relative Weight	Cal Poly Space Systems	Matthew Bornhorst	Dr. DeTurris	CPSS New Members	Maximum Relationship	WHAT: Customer Requirements (Needs/Wants)
1		6%	4	3	8	2	9	25lbf of Thrust
2		6%	3	5	9	2	9	Adjustable Film Cooling
3		9%	8	8	8	3	9	Compatible with CPSS GSE
4		12%	7	7	5	8	9	Operate Horizonatally and Vertically
5		9%	9	9	9	2	9	Ability to measure combustion chambe
6		16%	10	10	10	10	9	Manufacturability
7		6%	4	5	7	2	9	Must be Safe
8		9%	6	8	10	2	9	Reconfigurable
9		8%	9	9	5	2	9	Ethonal/Nitrous Oxide Bipropellant
10		8%	9	9	9	0	9	low cost
11		8%	9	9	9	0	9	Reusable
12		2%	2	2	2	0	9	5 Second Burn Time
13		0%						
14		0%						
15		0%						
16		0%						

HOW: Engineering Specifications (Tests)

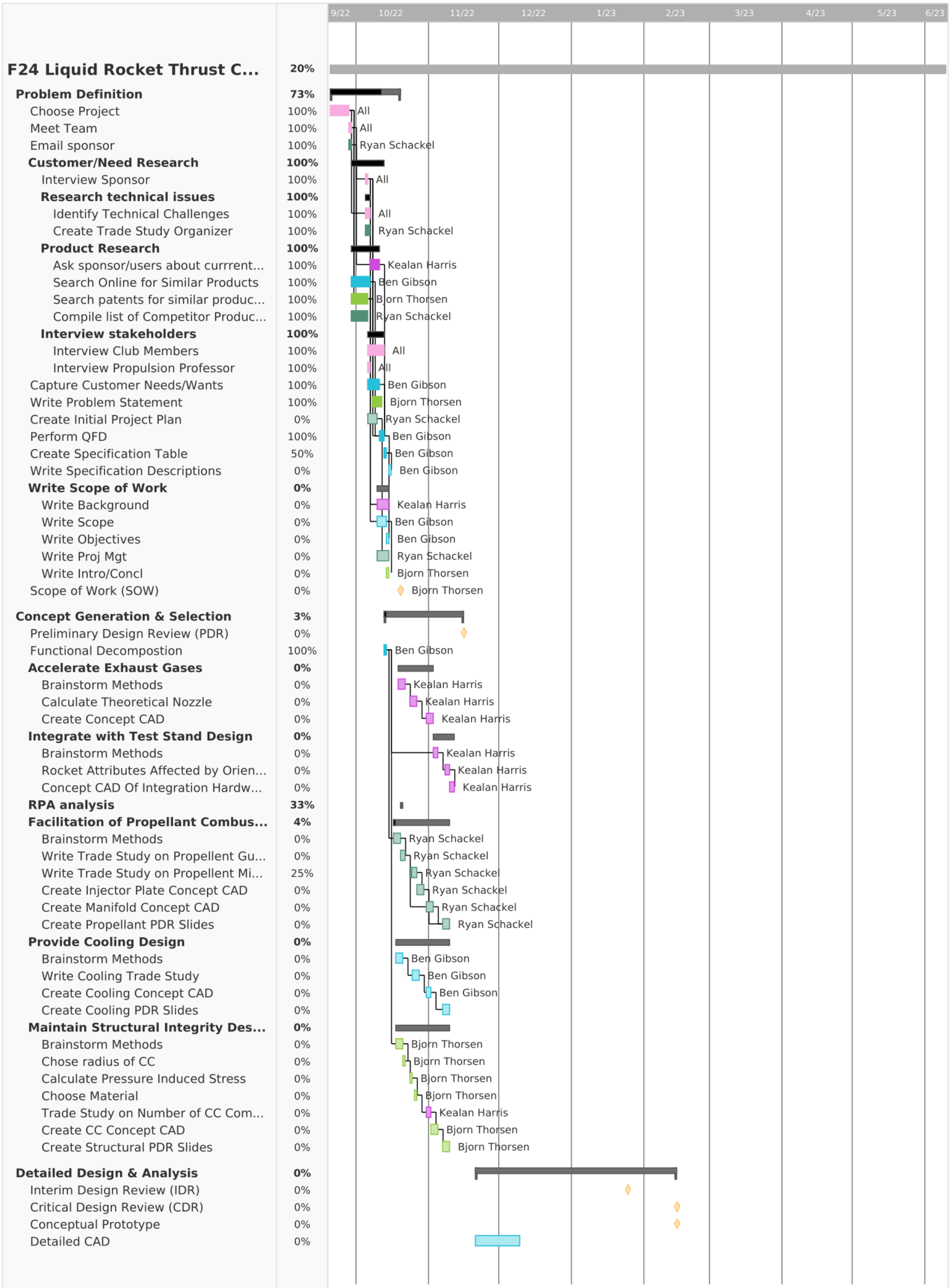
Column #	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Direction of Improvement	◇	▼	◇	◇	◇	▼	▼	▼	▲	◇						
Row #	Thrust	Combustion Chamber Temperat	Fit to Test Stand	Number of Parts Machined at Ca	Pressure Capacity	# of Permanent Joints	Cost Analysis	Minimum % of fuel for film cooli	Maximum % of fuel for film cool	Burn Time						
1	●															
2	○	●						●	●							
3	○		●													
4	○		●													
5	○	●	○													
6			○	●		●	○									
7	●	●			●											
8			▽	○		●										
9	○		●													
10							●									
11	○			●	●											
12	○									●						
13																
14																
15																
16																

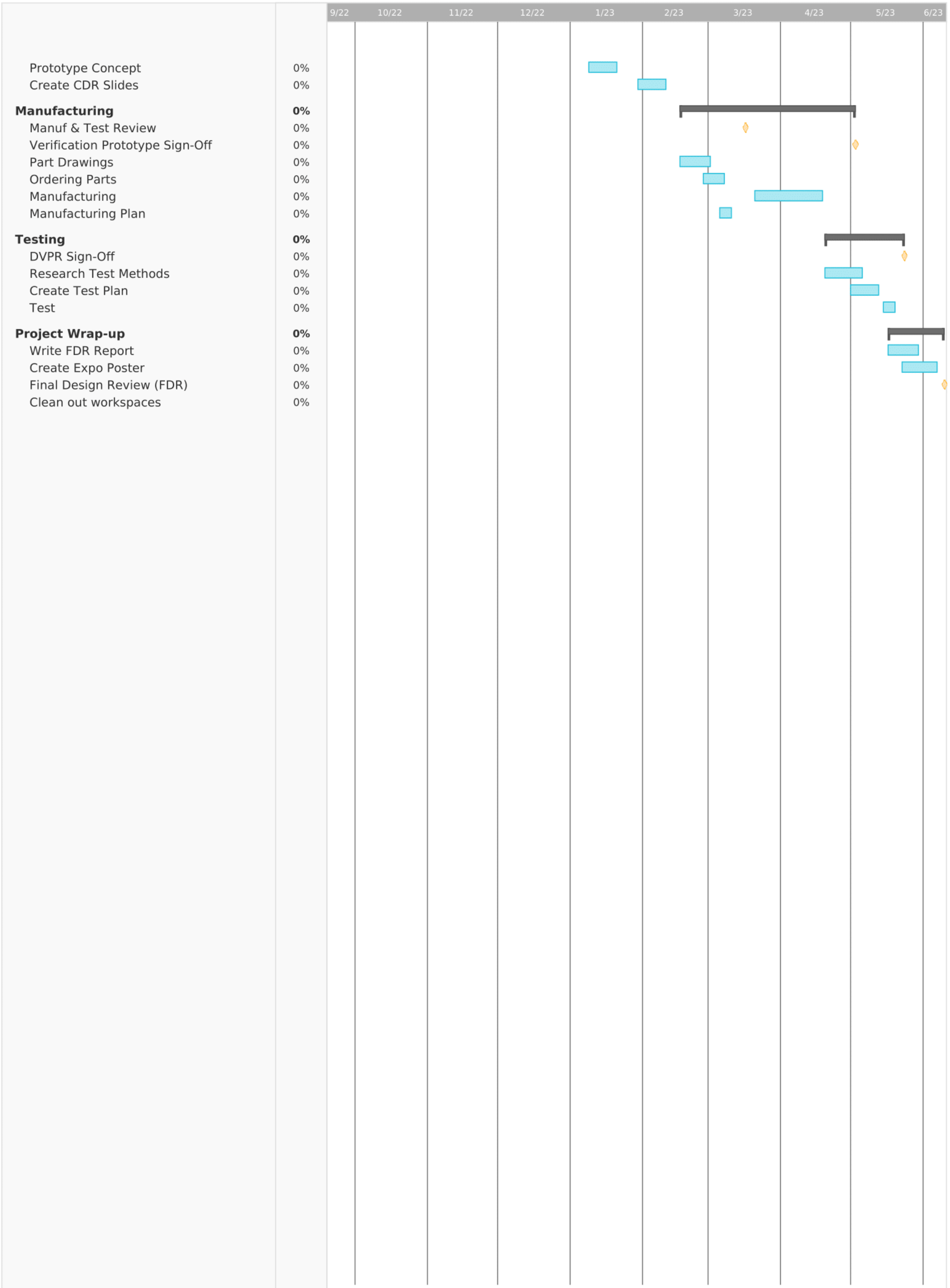
NOW: Curr. Products

Our Current Product: I140W-14	Competitor #1: Purdue	Competitor #2: Akron	Competitor #3: Arizona State	Competitor #4 Dawn b20 thruste	Row #
5	5	4	5	1	1
1	2	1	2	1	2
1	2	2	2	4	3
5	5	5	5	5	4
1	2	4	5	2	5
5	3	4	4	1	6
4	3	3	4	4	7
1	4	2	4	1	8
1	1	5	1	1	9
4	2	2	2	2	10
1	1	3	5	5	11
2	5	5	5	5	12
					13
					14
					15
					16

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
HOW MUCH: Target Values	25 lbf	Max temperature that maintains strength FoS	Within 5'x3'x2' bounding box	All custom Components manufactured at Cal Poly	Combustion Chamber holds pressure with 2.0 FoS Ultimate	0 Welded/ Permanent Joints	<$1200	5%	25%	5 Seconds						
Max Relationship	9	9	9	9	9	9	9	9	9	9						
Technical Importance Rating	273.5	198.6	351.2	245	127.3	224.5	120.9	57.72	57.72	15.93	0	0	0	0	0	0
Relative Weight	16%	12%	21%	15%	8%	13%	7%	3%	3%	1%	0%	0%	0%	0%	0%	0%
Curr. Products: rent Product: I140W-14A (solid motor)	5	5	5	5	1	5	5	1	1	4						
Competitor #1: Purdue	5	5	2	2	3	4	2	2	2	5						
Competitor #2: Akron	5	5	2	4	3	2	4	1	1	4						
Competitor #3: Arizona State	5	5	2	3	3	4	2	1	1	3						
Competitor #4 Dawn b20 thruster	1	5	5	1	4	1	1	1	1	5						
Column #	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16

B. Gantt Chart





9/22
10/22
11/22
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3/23
4/23
5/23
6/23
Prototype Concept 0%
Create CDR Slides 0%
Manufacturing 0%
Manuf & Test Review 0%
Verification Prototype Sign-Off 0%
Part Drawings 0%
Ordering Parts 0%
Manufacturing 0%
Manufacturing Plan 0%
Testing 0%
DVPR Sign-Off 0%
Research Test Methods 0%
Create Test Plan 0%
Test 0%
Project Wrap-up 0%
Write FDR Report 0%
Create Expo Poster 0%
Final Design Review (FDR) 0%
Clean out workspaces 0%