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LUME-X1 : A Pressure Fed Liquid Rocket Engine

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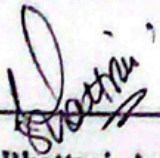
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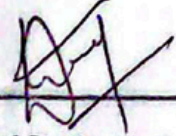
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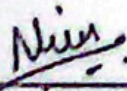
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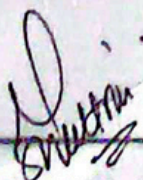
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ABSTRACT

This report presents a comprehensive approach to the experimental and numerical study of pressure fed liquid rocket engine. The engine was designed to use ethanol as fuel, pressure fed by GN₂ and GOX as oxidizers. The engine was optimized to increase specific impulse and overall performance as compared to solid rocket engine. A performance model is used to design liquid engine based on mass flow rate, chamber pressure and thrust, incorporating thermochemical and chemical equilibrium data of a propellant from NASA's CEA. The predicted performance was compared and validated through hot fire testing. Additionally, CFD simulations provide detail into the supersonic flow behavior within the nozzle. The experimental results across multiple tests demonstrate that the actual thrust fell below designed thrust of 130N, giving average combustion efficiency of 0.93 with peak Isp of 213 s. This work will provide a foundation for further advancements from solid rocket propulsion system to liquid rocket propulsion system and with future improvements focused on proper optimization and launch of liquid rocket engine vehicles.

Keywords: Liquid Rocket Engine, Propellant Composition, Injector, Thermochemical Data, Hot Fire Testing, Performance Evaluation.

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LIST OF ABBREVIATIONS

AFT	Adiabatic Flame Temperature
ANSYS	Analysis System
ARP	Aerospace Recommend Practice
CAD	Computer Aided Design
CATIA	Computer-Aided Three-Dimensional Interactive Application
CD	Convergent-Divergent
CEA	Chemical Equilibrium Application
CFD	Computational Fluid Dynamics
COTS	Commercial Off-the-self
CV	Check valve
FVM	Finite Volume Method
GN₂	Gaseous Nitrogen
GO_x	Gaseous Oxygen
GUI	Graphical User Interface
LRE	Liquid Rocket Engine
LOX	Liquid oxygen
LPREs	Liquid Propellant Rocket Engines
MBV	Manual Ball Valve
MEOP	Maximum Expected Operating Pressure
MS	Mild steel
MV	Manual Valve
NASA	National Aeronautics and Space Administration
OpenFOAM	Open Source Field Operation and Manipulation
P&ID	Piping & Instrumentation Diagram
PABV	Pneumatic Actuated Ball Valve
PG	Pressure Gauge
PR	Pressure Regulator
PT	Pressure Transducer
PVC	Poly Vinyl Chloride
RP-1	Refined Petroleum-1

RV	Relief Valve
SAE	Society of Automative Engineers
SS	Stainless Steel

LIST OF SYMBOLS

A	Area of orifice
Ac	Combustion chamber area
Ae	Exit area
At	Area of nozzle throat
B	Burst Factor
C*	Characteristics velocity
C₂H₅OH	Ethanol
Cd	Discharge coefficient
Cf	Coefficient of thrust
CO₂	Carbon Dioxide
Cp	Heat capacity at constant pressure
CR	Contraction Ratio
c	Speed of sound
Dc	Chamber Diameter
ε	Expansion Ratio
F	Thrust
Ft	Nominal thrust
f	Frequency of modes
f_H	Hewit frequency
G	Gibbs free energy
H	Enthalpy
h_g	Convective heat transfer coefficient
Isp	Specific impulse
K	Kelvin
k	Specific heat ratio
L*	Characteristic length of combustion
M	Mach number
<i>ṁ</i>	Mass flow rate of discharge
<i>ṁ_f</i>	Fuel mass flow rate
<i>ṁ_o</i>	Oxidizer mass flow rate

$\mathfrak{M}$	Exhaust molecular mass
N	Number of orifice
N₂O	Nitrous oxide
O/F	Oxidizer-to-Fuel ratio
O₂	Oxygen
P	Pressure
P_c	Chamber pressure
P_e	Exit pressure
P_{MEOP}	Maximum expected operating pressure
P_o	Stagnation pressure
Pr	Prandtl number
P_t	Pressure at nozzle throat
Q	Volumetric discharge
q	Heat flux
R	Gas constant
Re	Reynolds number
r	O/F ratio
S	Entropy
S_G	Specific gravity
T_c	Combustion chamber temperature
v_e	Exhaust velocity
V_c	Chamber volume
W_t	Propellant mass flow rate
ρ	Density
η	Efficiency
β	Convergent Half angle
α	Divergent Half angle
σ_y	Yield strength
σ_{tear}	Tearing strength
μ	Dynamic Viscosity

CHAPTER 1: INTRODUCTION

1.1 Background

Liquid Propellant Rocket Engines (LPREs) in general consist of fuel and oxidizer, both of which are stored in separate tanks. These tanks then supply each to the combustion chamber through pressurization or pumps. The combustion chamber consists of an injector that atomizes the fuel and oxidizer, mixes them, and then aids in high-pressure combustion. In case of gaseous oxygen (GOx), the injector only atomizes fuel and aids in mixing.

The hot gases that are the result of combustion are expanded by passing through a Converging-Diverging (CD)/deLaval nozzle, which in turn produces thrust. The chamber pressure for this kind of system ranges from 10 bar (pressure-fed) to 200 bar (pump-fed). They offer a high specific impulse (isp) around 230-250 s for GOx and ethanol with high throttle ability, which is their main advantage over solid fuels, making them a better choice.

The history of LPREs begins with Robert H. Goddard's work in early 1920s. He on March 16, 1926 launched the world's first liquid-fueled rocket at Auburn, Massachusetts called Nell using gasoline and oxygen in pump-fed combustion chamber. The rocket traveled 2.5 seconds at a speed of 60 mph, reaching an altitude of 41 feet and landing 184 feet away. Post World War II efforts scaled thrust to hundreds of kilonewtons, such as in the V-2 and RD-107, making rapid advances in turbomachinery, regenerative cooling, and staged combustion cycles.

Recently, small-scale LPRE development has become an emerging topic as it is the main basis for future advancement in high-power launch vehicles and sounding rockets. The pressure-fed or electric pump-fed cycles have less system complexity, low cost and are able to produce thrust around 1 kN, which is ideal for small static tests and performance analysis. Full scale hot fire testing of these engines requires huge infrastructure and budget which is not ideal for research purposes.

1.2 Problem Statement

Previously several tests were performed on solid rocket motors, even though the thrust provided by them is enough, it was clear that they have low specific impulse and almost no controllability. Thus, the need for liquid propellant rocket engines arises. Currently, there is an unavailability of a liquid propellant test facility or a documented experimental program to validate the theoretical and computational designs of liquid engines. This prevents students and researchers from hands-on development and iterative correction of the propulsion system, injectors, combustion chamber, and feed system in a cost-effective, safe and controlled manner. To develop LPRE technology, significant static ground tests are required to establish the concept and develop reliable systems. Large-scale testing is very expensive, which is why a dedicated small-scale test facility is important. Hence, developing a small-scale liquid engine complete with pressure-fed system, injectors, instruments, and safety is essential to bridge the gap between analytical models and practical validation, thus it enables the local expertise in future high-thrust system development.

1.3 Objectives

Main objective

Build and test of pressure fed liquid rocket engine using gaseous oxygen and Liquid ethanol.

Specific Objectives

- i. Design and fabricate the combustion chamber, nozzle, injector, and pressure-fed system for proper atomization, stable combustion, and supersonic flow.
- ii. Perform Cold flow test and hot-fire test of engine for performance validation.
- iii. Implement sensors for data acquisition.

1.4 Applications

- i. Experimental verification: Theoretical and design concepts can be visualized and verified.
- ii. Foundation for a full-scale liquid rocket: The small-scale model developed can be scaled up to make a high-performance space vehicle.
- iii. Research: The components and system developed can be used as a basis for future improvements to various technologies and methods.
- iv. Sounding rockets: Propulsion system of small rockets for carrying, micro satellites, and sensors for atmospheric study.

1.5 Feasibility analysis

1.5.1 Economic feasibility

The construction of LPREs requires a high initial investment as a result of the need for tanks to store oxidizer, fuels, and develop complex system around them. However, once the heavy components are build and purchased, the cost per test decreases very significantly. The developed technologies and equipments can be reused for future experiments and advancements, through multiple research groups. Overall, the modest initial investment and low operating cost per test make it economically viable.

1.5.2 Technical feasibility

The technical resources required for LRE are well defined and known. Key equipment necessary for the system such as tanks and plumbings are easily available. The injector and combustion chamber required for the setup can be easily manufactured using machines such as lathe or milling. Hence, this shows the strong technical feasibility of the project.

1.5.3 Operational feasibility

The test setup will consist of a thrust stand and tanks supplying fuel along with oxygen to the combustion chamber mounted on the stand. Hence, all setup can be considered

bench setup. Since only the static testing of the motor will be performed, its operation will not violate any rules.

1.6 System requirements

1.6.1 Material Requirements

The material requirements comprise tanks for propellant storage and pressurization; fluid control, safety, and measurement hardware including valves, regulators, a load cell, and thermocouples; and fabrication tools such as a lathe, milling machine, 3D printer, and laser cutter for constructing the engine and thrust stand from mild and stainless steel.

Table 1.1: Material Requirements

Materials	Application
Oxygen Tank	Storing Oxidizer
Nitrogen Tank	Pressurizing
Ethanol Tank	Fuel
Industrial Regulators	Controlling Tank Flow
Piping and Tubing	Convey Fluid
Manual Valve	On/Off Isolation
Pressure Relief Valve	Remove Excess Pressure
Check Valve	Prevent Backflow
Pressure Gauge	Visual Pressure Indication
Thermocouple	Measure Wall Temperature
Load Cell	Thrust Measurement
HX711	Amplifier
3D Printer	Components Fabrication
Laser Cutter	Cutting
Linear Bearing	Thrust Stand
Welding	Flange System
Lathe, Milling, CNC	Components Machining
Mild Steel, Stainless Steel	Engine and Thrust Stand

1.6.2 Software requirement

The system's software requirements encompass a suite of specialized applications supporting the entire design, analysis, and operation life cycle. This includes CAD software such as Fusion 360, CATIA V5, and Blender for detailed component and assembly design. Analysis and simulation leverage ANSYS and OpenFOAM for computational fluid dynamics, alongside NASA CEA for thermodynamic modeling and RPA for preliminary

engine performance calculations. For control systems and data acquisition, programming environments including the Arduino IDE, VS Code, and Spyder are utilized. Finally, Bamboo Studio is employed for slicing and managing 3D printing operations during component fabrication.

Table 1.2: Software Requirements

Software	Application
Fusion 360, CATIA V5 and Blender	CAD Design
ANSYS and Openfoam	CFD Simulation
Arduino IDE, VS Code and Spyder	Programming
Bamboo Studio	3d Printing
NASA CEA, Rocket CEA	Chemical Equilibrium Analysis
RPA	Preliminary Engine Design

CHAPTER 2: LITERATURE REVIEW

Liquid rocket engines utilize liquid propellants that are delivered under high pressure from propellant tanks into a combustion chamber where controlled chemical reactions generate thrust. Due to their superior specific impulse, thrust controllability, high efficiency and restartability, liquid rocket engines have been extensively studied. A comprehensive design process for a low-thrust, pressure-fed cycle engine, along with numerical simulation of a convergent-divergent nozzle for supersonic flow in ambient conditions, has been established [1].

The injector is one of the most critical component of a liquid rocket engine, as its primary function is to atomize and mix the propellant with oxidizer before combustion for uniform and stable reaction within the chamber. The injector pushes the propellant through small orifices, effectively increasing their flow velocity and momentum until the propellant streams collide, at which point the impact atomizes the streams into fine droplets that vaporize and combust. The geometry of the injector, including orifice diameter, impingement angle and element type, directly governs the atomization quality, combustion efficiency and stability of the engine. Cold flow testing methods for different injector designs across multiple propellant combinations including hypergolic propellants, have been investigated to characterize injector performance prior to hot fire testing [2].

The design of feed system has been an important aspect for performance, efficiency and safety operation of liquid rocket engine as it is responsible for consistently delivering propellants to the injector assembly at required pressure and mass flow rates throughout the duration of burn. A well designed feed system can deliver oxidizer and fuel to injector assembly under various operating conditions. A preliminary design of a propellant feed system utilizing a gas pressure-fed cycle, in which GN2 was employed as a pressurant gas to deliver fuel to the combustion chamber, has been presented and serves as a relevant reference for the current work [3].

Reliable ignition is important for a motor performance. For a liquid rocket engine the ignition system ranges from a pyro igniter to torch igniter. Pyro igniter consists of a

small propellant in a form of pyrocharge to initiate the ignition. Generally pyro igniter is used to ignite solid rocket motors, but it can also be used to ignite bi-propellant liquid rocket engines or hybrid as well. A spark plug may also be option for ignition but it has the timing problem, which may cause a hard start if there is a reasonable amount of mixture when the spark is created or, on the contrary, the spark may occur when there isn't enough propellant to burn. Additionally, a spark plug requires a high voltage source. Consequently, glow plugs have been successfully implemented as igniters for similar small-scale engine configurations[4]. Torch igniters are utilized for other propellant combinations, such as RP-1/LOX, carbon monoxide/GOX, methane/GOX, and aluminum/GOX, providing versatile and robust ignition capability[5] .

To prevent casing failure due to hot combustion gases, thermal liners are used. Common materials include phenolic resin and ablative composites. Thermal protection systems are evaluated for their heat soak capacity, erosion rate, and thermal conductivity. Proper structural design of a motor/engine casing is essential to operate safely under MEOP condition. Comprehensive guides have been documented on design methodologies for pressure vessels in rocketry cover aspects such as material selection, safety factors, wall thickness calculations and failure modes [6]. These methods are particularly relevant for designing cylindrical pressure vessels for propellant storage and engine casings, as well as retention methods for injector pieces and nozzles.

Since, the operation of the engine includes high pressure feed system, proper installation of high pressure tubing and fittings is crucial for proper operation and ensure reliable fuel and oxidizer delivery. Detailed procedures for the assembly and maintenance of tube fittings under extreme operating conditions are available and were consulted during the design of the feed system in this project [7]. Prior research has involved the design, fabrication, and testing of liquid engines utilizing combinations of kerosene and atmospheric oxygen [8]. Furthermore, steps for the design and manufacturing of small liquid rocket engines using GOX with LPG or kerosene, designed to operate at chamber pressures of 80-100 psi via air compressor feeding, has also been reported and provides useful comparative data for the present engine design[9].

Technical reports provide additional details on the working of liquid rocket engines and

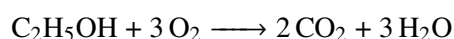
their feed systems, including the use of phenolic resin as an ablative material for chamber wall protection [10]. Experiments with consumer-grade polyester resin (density 1.26 g/cm^3) have shown it to be an effective thermal liner for reducing heat rates in engine casings [11]. Additionally, simplified one-dimensional transient heat transfer models for thrust chambers, combining modified Bartz formulas with slab-based heat conduction approaches, have been proposed and experimentally validated for LOX/Methane heat sink chambers [12] .

CHAPTER 3: THEORETICAL BACKGROUND

This section represents the general theoretical aspect while designing a liquid rocket engine. For this design with the aim of simplifying the process, the system described is assumed to be ideal. Valid assumption in the ideal case is described as taking an example, the working fluid is assumed to match with ideal gas law and to have a homogeneous composition. Likewise, the expansion of a gas through a thrust chamber is considered to be isentropic process [13].

3.1 Combustion and Propellants

The rocket motor operates on a basic conservation on a energy principle. It converts a chemical energy stored in the form of oxidizer, fuel and additives in the propellant to heat energy which is initiated by the ignition source. And ultimately the generated heat energy is converted to kinetic energy by the nozzle providing the ultimate thrust to the system. The reaction in a combustion of a propellant is exothermic i.e a heat is released through the reaction of the reactant present inside the combustion chamber. In general the combustion is represented by a chemical reaction. For example: a following reaction shows the combustion of stoichiometric ratio Ethanol and Oxygen.

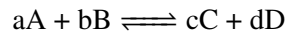


The combustion reaction is initiated by some source of energy which may be in form of igniter. The igniter consists of a small amount of a highly reactive charge as compared to propellant which provides activation energy for the combustion. The reaction in the combustion of oxidizer and fuel is spontaneous. Spontaneity of the reaction is given by the Gibbs Free Energy change ΔG which is also known as a chemical potential. For a chemical reaction:

$$\Delta G^\circ = \sum G^\circ_{\text{products}} - \sum G^\circ_{\text{reactants}} = \Delta H - T\Delta S \quad (3.1)$$

Since the combustion reaction is spontaneous, ΔH , or enthalpy change is negative, corresponding to liberation of heat through the reaction. Generally some propellant are stable at room temperature although they have high negative ΔG value which doesn't

make that propellant to react at that temperature. This is due to the fact that the propellant which are more stable at room temperature (25°C) must be provided with sufficient energy to convert them in reactive, higher energy excited state called an activation energy. In excited state, a reaction will occur to form the anticipated products, with the liberation of considerable energy, all the energy that was required to reach the excited state plus considerably more, which will be the net heat of reaction. The liquid engine works by burning a liquid propellant which includes fuel and oxidizer in required OF ratio which is then accelerated through a nozzle in high exhaust velocity to obtain the required thrust. Liquid rocket engines can use and combust a variety of oxidizer and fuel combinations in which some of them are hazardous, toxic and expensive. To determine the chemical equilibrium i.e whether the chemical reaction will proceed or not there is a constant term definition which is called an equilibrium constant (K_p). This value is dependent upon the temperature in which the species exist. For a reversible reaction taking as:



The chemical equilibrium can be defined as:

$$K_\phi = \frac{y_C^c \cdot y_D^d}{y_A^a \cdot y_B^b} \left(\frac{P}{P_0} \right)^{(c+d-a-b)} \quad (3.2)$$

where, y_X represents the mole fraction of species X, and a , b , c , and d are the stoichiometric coefficients for the reactants and products, respectively. P is the total system pressure, while P_0 denotes the reference pressure. The exponent $(c + d - a - b)$ corresponds to the net change in the number of moles of gaseous species during the reaction. It is to be noted that the product existing in a higher temperature is different from those existing in a lower temperature. At a higher temperature (above 3000K) dissociation of a product occurs in which they break into smaller monoatomic constituents. For current propellant combination of ethanol and GOX this is also the case.

From the law of conservation of energy, with the assumption of adiabatic combustion and no change in kinetic or potential energy occurs, the enthalpy of reaction (H_R)

must be equals to enthalpy of products (H_P).

$$\sum_x n_i [h_f + \Delta h]_i = \sum_p n_e [h_f + \Delta h]_e \quad (3.3)$$

The enthalpy of a reaction through which Adaibatic Flame Temperature (AFT) is calculated for a combustion reaction of fuel and oxidizer can be determined using standard enthalpy of formation.

3.2 Thrust Chamber

The thrust generated by a system can be expressed as:

$$F = \dot{m} v_e + (p_e - p_a) A_e \quad (3.4)$$

where, $\dot{m}$ is the total mass flow rate of a propellant in $kg s^{-1}$, v_e is the effective exhaust velocity, p is the pressure in pascal and A is cross-sectional area in m^2 , the subscripts e and a represent nozzle exit and ambient condition respectively. The ambient pressure can be calculated as [14]:

$$P = 101325 \left(1 - 2.25577 \times 10^{-5} \cdot h \right)^{5.25588} \quad (3.5)$$

where, h is an altitude from sea level.

The total propellant mass flow rate in a combustion chamber which includes the oxidizer and fuel mass flow rate hold true for the following relation:

$$\dot{m} = \dot{m}_o + \dot{m}_f \quad (3.6)$$

and

$$\dot{m}_f = \frac{\dot{m}}{r + 1} \quad (3.7)$$

where r is the OF ratio and is given by:

$$r = \frac{\dot{m}_o}{\dot{m}_f} \quad (3.8)$$

The total propellant mass flow rate is a crucial aspect to control the chamber pressure in a thrust chamber. The total mass flow rate can be related by:

$$\dot{m} = \frac{P_c A_t}{c^*} \quad (3.9)$$

Considering the above equations, the thrust can be represented as from the continuity equations:

$$F = A^* P_0 \sqrt{\frac{2k^2}{k-1} \left(\frac{2}{k+1} \right)^{\frac{k+1}{k-1}} \left[1 - \left(\frac{P_e}{P_0} \right)^{\frac{k-1}{k}} \right]} + A_e (P_e - P_a) \quad (3.10)$$

This expression considers the following condition:

- Choked flow condition at the throat $M=1$
- Critical throat velocity i.e $v=a$
- Equation of state $P = \rho RT$

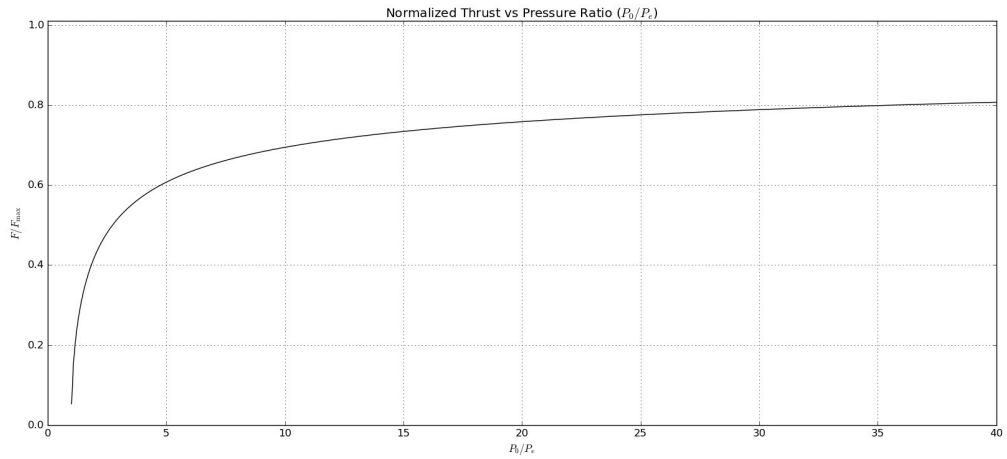


Figure 3.1: Thrust Ratio vs Pressure Ratio.

Figure [3.1](#) shows the plot of thrust ratio F/F_{max} and pressure ratio P_0/P_e , where F_{max} is the thrust that would have been obtained in infinite pressure ratio i.e expanding into vacuum with $P_e \sim 0$.

The above equation can be modified into a important term which provide an expression for how much thrust is amplified by divergent section and in terms with chamber pressure and throat area.

$$F = C_f A_t P_c \quad (3.11)$$

where, C_f is coefficient of thrust, A_t is throat area, and P_c is chamber pressure.

3.3 Nozzle Theory

The nozzle is a component of a rocket engine that converts the heat energy generated by the combustion of a propellant to kinetic energy by exhausting the hot gas product at high velocity. The nozzle used in rocket motor is typical supersonic nozzle. The familiar rocket nozzle, also known as a convergent-divergent, or deLaval nozzle, accomplishes this remarkable feat by simple geometry. In other words, it does this by varying the cross-sectional area (or diameter) in an exacting form. The analysis of a nozzle involves a state one dimensional compressible fluid Flow. For an experimental rocket engine a typical conical convergent-divergent nozzle is used.

Following Equations shows an isentropic relation between the stagnation properties:

$$\frac{T_0}{T} = \left(1 + \frac{k-1}{2} M^2\right) \quad (3.12)$$

$$\frac{P_0}{P} = \left(1 + \frac{k-1}{2} M^2\right)^{\frac{k}{k-1}} \quad (3.13)$$

$$\frac{\rho_0}{\rho} = \left(1 + \frac{k-1}{2} M^2\right)^{\frac{1}{k-1}} \quad (3.14)$$

The Mach number M is a dimensionless flow parameter and is used to locally define the ratio of the flow velocity v to the local acoustic velocity a :

$$M = \frac{v}{a} = \frac{v}{\sqrt{kRT}} \quad (3.15)$$

The following equation gives an important relation between the area ratio and the mach

number as:

$$\frac{A}{A^*} = \frac{1}{M} \left[\frac{2}{k+1} \left(1 + \frac{k-1}{2} M^2 \right) \right]^{\frac{k+1}{2(k-1)}} \quad (3.16)$$

When plotting a graph between the area ratio and the mach number a following graph is obtained which shows the variation of mach number along the nozzle axis:

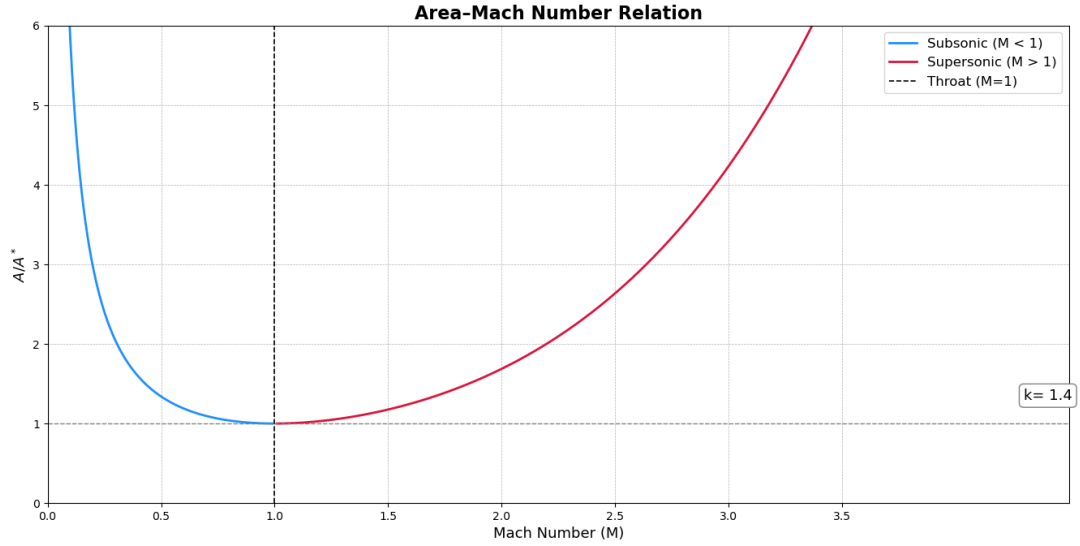


Figure 3.2: Variation of mach number along with area ratio.

The Figure 3.2 clearly shows that a converging-diverging passage with a section of minimum area is required to accelerate the flow from subsonic to supersonic speed. Supersonic flow is only attained in the divergent section of the nozzle which shows the importance of having a divergent section in the nozzle as without it a flow would never be greater than the sonic velocity.

The effective exhaust velocity through the nozzle is given by the following equation:

$$v_e = \sqrt{2 T_o \left(\frac{R'}{\mathfrak{M}} \right) \left(\frac{k}{k-1} \right) \left[1 - \left(\frac{P_e}{P_o} \right)^{\frac{k-1}{k}} \right]} \quad (3.17)$$

A better understanding of the nozzle behavior may be obtained by looking closely to this equation. It may be seen that: -3pt

1. Maximum exhaust velocity is obtained when exhausting into a vacuum ($P_e = 0$). This is the so-called infinite pressure ratio, P_0/P_e .
2. Increasing the chamber pressure does not significantly increase the exhaust velocity.
3. A higher combustion temperature T_0 and a lower effective molecular weight $\mathfrak{M}$ are both significantly and equally beneficial, being proportional and inversely proportional to the square root power, respectively.

The $T_0/\mathfrak{M}$ ratio plays an important role in optimizing the propellant mixture ratio. The another important parameter is Thrust coefficient C_f , which determines the amplification of thrust due to gas expansion in the nozzle as compared to the thrust that would be exerted if the chamber pressure acted over the throat area only and is given by:

$$C_f = \sqrt{\frac{2k^2}{k-1} \left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}} \left[1 - \left(\frac{P_e}{P_0}\right)^{\frac{k-1}{k}}\right]} + \frac{(P_e - P_a)A_e}{P_0 A^*} \quad (3.18)$$

The expansion ratio of a nozzle, defined as the ratio of the exit area A_e to the throat area A_t , plays a critical role in determining the flow characteristics at the nozzle exit. It can be related to the pressure ratio $\frac{P_e}{P_0}$ (exit pressure to chamber pressure) using isentropic flow relations.

$$\frac{A_t}{A_e} = \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}} \left(\frac{P_e}{P_0}\right)^{\frac{1}{k}} \sqrt{\left(\frac{k+1}{k-1}\right) \left[1 - \left(\frac{P_e}{P_0}\right)^{\frac{k-1}{k}}\right]} \quad (3.19)$$

This is known as the nozzle design condition and it helps determine the required exit area A_e for a given chamber pressure and ambient pressure.

CHAPTER 4: METHODOLOGY

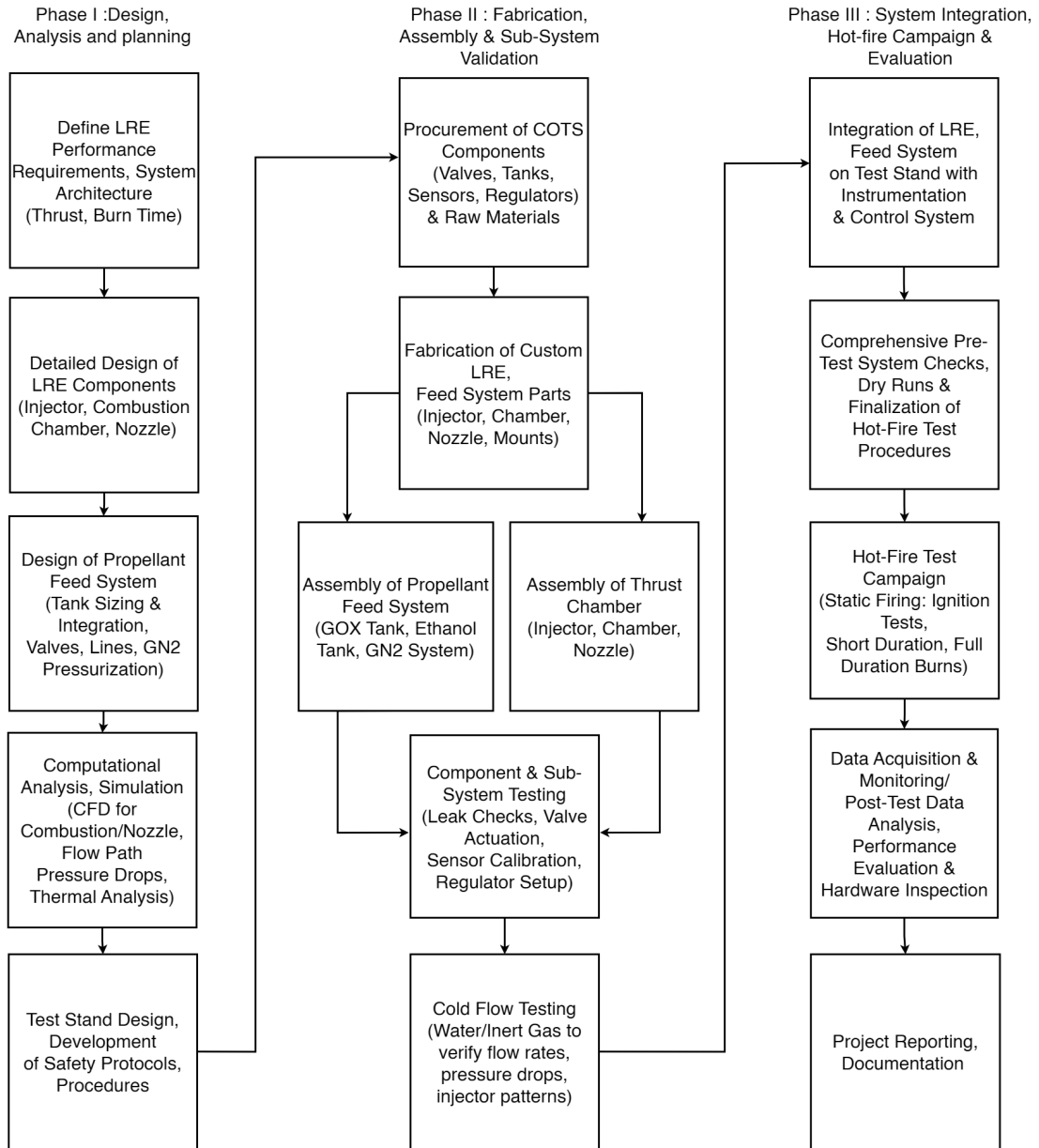


Figure 4.1: Methodology

4.1 Initial System Design

4.1.1 Chamber Pressure and Mixture Ratio

The first step is to determine the theoretical performance to be expected from a propellant combination. For a current combination, ethanol was chosen as fuel for its availability, modest specific impulse, burns cooler than kerosene and produce less soot as compared to long chain hydrocarbon and gaseous oxygen as an oxidizer. The ethanol/ethyl alcohol

may be found in a dilution state as well. Approx 85-95% dilution (by mass) with water can also be used which lowers the adiabatic combustion chamber temperature. This reduces some I_{sp} but may reduce some complexity such as regenerative and film cooling technique may not be required for short burn time. Ablative cooling can be effective in this case for short burn time. Industrial Gaseous Oxygen is chosen as an initial oxidizer for its ease of availability, non-cryogenic nature unlike in Liquid Oxygen (LOX), and modest I_{sp} with an ethanol and kerosene combination. There is another oxidizer in option which is nitrous oxide N_2O which is known for its having self-pressurizing characteristics.

Using NASA's CEA [15], an open-source thermochemical analysis tool provided by Glenn Research Center, important and critical properties describing propellant combustion characteristics can be obtained. A Python Wrapper named RocketCEA is applied which makes direct calls to the NASA FORTRAN CEA code in "rocket" mode to calculate I_{sp} , C_{star} , T_C etc. and provides tools to help determine useful mixture ratio range, optimum MR and more [16]. By utilizing that wrapper numerical python scripts were written to iterate through various combustion chamber pressure and mixture ratio MR and used to parse and graph the CEA output. All dependent properties like characteristics velocity C^* , Adiabatic Flame Temperature T_C , exhaust molar mass $\mathfrak{M}$ and specific heat ratio k is permuted based on the assumption of shifting equilibrium flow and at the operational ambient pressure.

For current engine design a mixture ratio of 1.4 for Ethanol/GOX was chosen and studied mainly in the interest of combustion temperature around 3000 K and a theoretical approx C^* of 1700 m/s, given a chamber pressure of 150 psi. This chamber pressure was chosen primarily to minimize the system cost and maintain the capacity of the available pressure regulator and the feed system. However the actual OF ratio may vary during the real manufactured injector geometry.

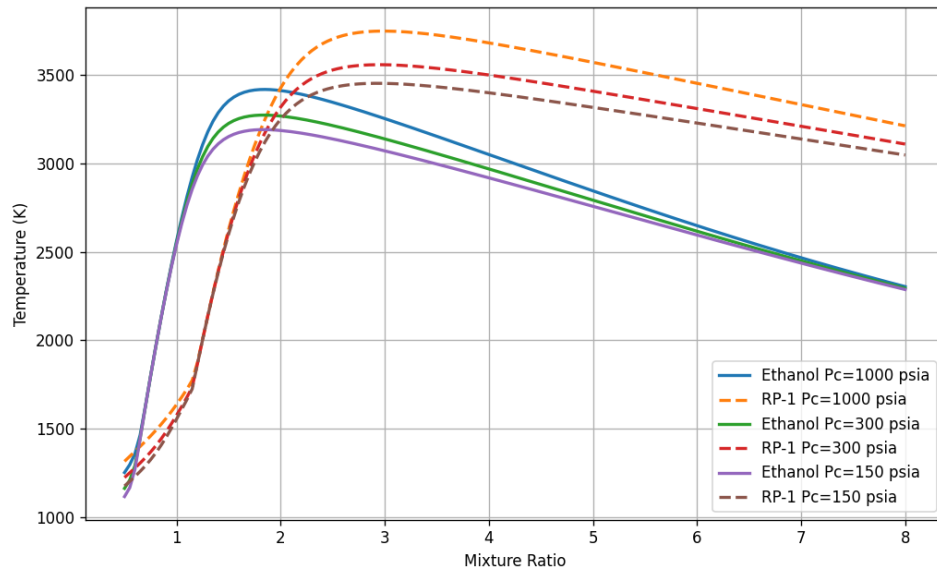


Figure 4.2: Theoretical Combustion Temperature T_c vs OF Ratio for Ethanol and Kerosene

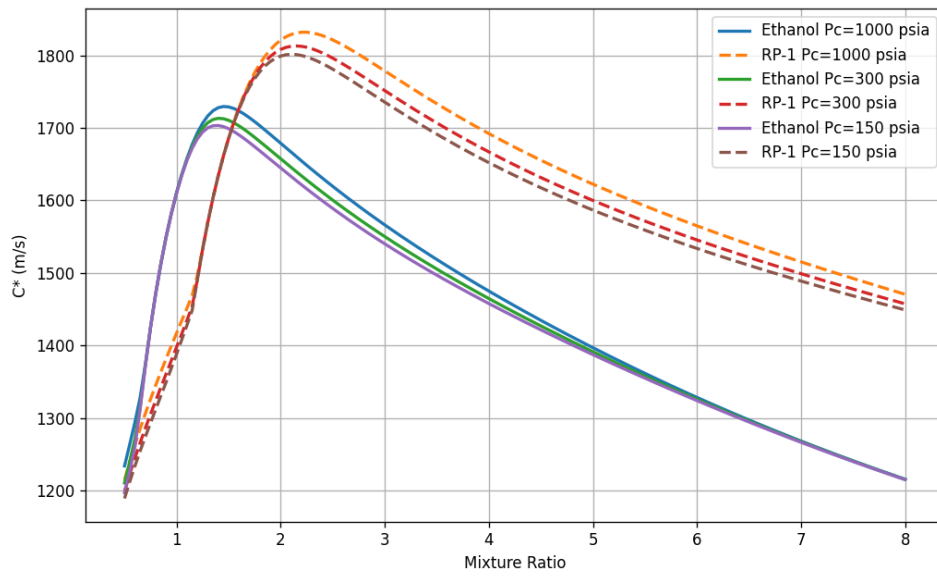


Figure 4.3: Theoretical C^* Efficiency O/F Ratio for Ethanol and Kerosene

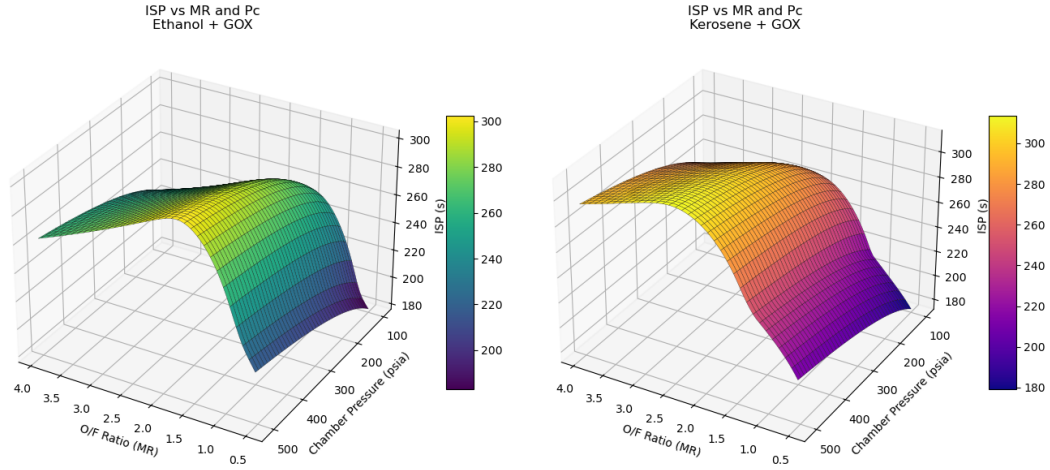


Figure 4.4: Theoretical I_{sp} vs Chamber Pressure vs O/F Ratio

For ease of plotting and analysis, same procedure can be implemented to obtain a different thermochemical result from RPA software. The Figure:4.5 and Figure:4.6 shows the typical mole fraction of a combustion product at different station of chamber and performance profile in different OF ratio for current propellant respectively.

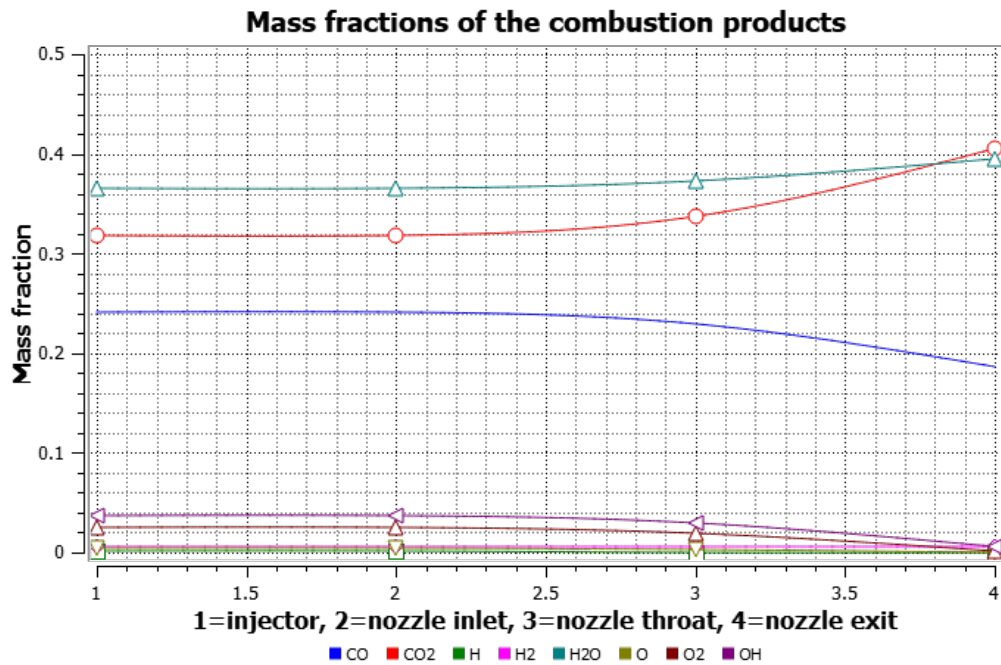


Figure 4.5: Mole fraction at different station of thrust chamber

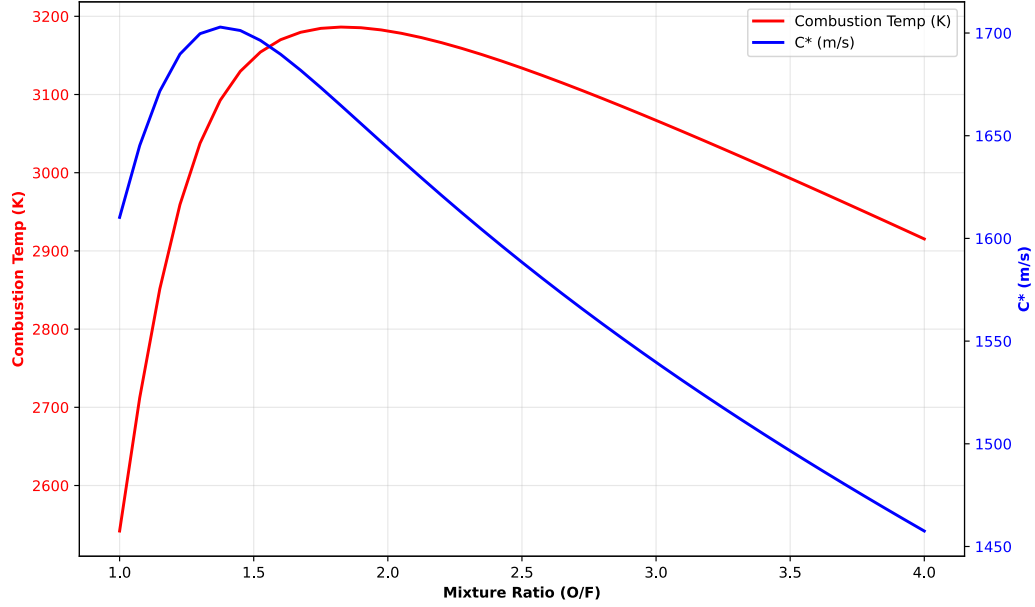


Figure 4.6: C^* and Combustion Temperature(T) for GOX/Ethanol(90%)

4.2 Engine System Design

The following section represents the thrust chamber design process for bi-propellant liquid rocket engine to obtain a desired objectives.

4.2.1 Chamber and Nozzle Sizing

Table 4.1: Summary of design parameters.

Name	Value	Unit
Propellant (Fuel)	90%.Ethanol (C_2H_5OH)	N/A
Propellant (Oxidizer)	Gaseous Oxygen (O_2)	N/A
O/F, Oxidizer/Fuel Ratio	-	N/A
F_t , Nominal Thrust	-	N
C^* , Characteristics Velocity	-	m/s
$\dot{m}$, Total Mass Flow Rate	-	kg/s
P_c , Chamber Static Pressure	-	Pa
P_e , Ambient Pressure	-	Pa
T_c , Chamber Static Temperature	-	K
$\mathfrak{M}$, Exhaust Molecular Mass	-	kg/kmol
k , Specific Heat Ratio	-	N/A

The nozzle sizing process followed a standard procedure through series of isentropic relation parameter calculation. So, assuming fully isentropic flow (adiabatic and reversible) in a supersonic nozzle with choked flow condition at the throat, an ideal convergent-

divergent nozzle can be characterized. The first parameter includes the calculation of the area ratio:

$$\epsilon = \frac{A_e}{A_t} = \left[\left(\frac{k+1}{2} \right)^{\frac{1}{k-1}} \left(\frac{P_e}{P_0} \right)^{1/k} \sqrt{\frac{k+1}{k-1} \left(1 - \left(\frac{P_e}{P_0} \right)^{(k-1)/k} \right)} \right]^{-1} \quad (4.1)$$

where A_t is the cross-sectional area of the throat, A_e is the cross-sectional area at nozzle exit, and P_0 and P_e is the chamber pressure and exit pressure respectively. From the above area relation an optimum expansion ratio can be determined for a given pressure ratio. The obtained expansion ratio is used to estimate the optimal nozzle exit diameter D_e :

$$A_t = \text{throat area}, \quad A_e = \epsilon_{\text{avg}} A_t, \quad D_e = \sqrt{\frac{4A_e}{\pi}} \quad (4.2)$$

To find the Mach number at the nozzle exit, the isentropic area-Mach relation is solved numerically using `fsolve()` from `scipy` using `python` library:

$$\epsilon = \frac{1}{M} \left[\left(\frac{2}{k+1} \right) \left(1 + \frac{k-1}{2} M^2 \right) \right]^{\frac{k+1}{2(k-1)}} \quad (4.3)$$

This relation gives a solution in two region, one in subsonic and another in supersonic region in which a supersonic solution was chosen. Also a function was defined to determine the static pressure at the nozzle exit, calculated using the isentropic relation:

$$P_e = \frac{P_0}{\left(1 + \frac{k-1}{2} M^2 \right)^{k/(k-1)}} \quad (4.4)$$

A condition is to predict the flow separation across the nozzle known as Summerfield's Criterion for Flow Separation. According to the criterion that in an over-expansion to 40 percent of the surrounding pressure, the flow separates from the wall [17] and is given as:

$$P_e/P_a < 0.4$$

The thrust coefficient C_f which is an important parameter for a nozzle performance calculation defined in simple term as a how much thrust is amplified from the thrust that would have been obtained only from chamber pressure acting on a throat section. The

ideal C_f is computed as:

$$C_f = \sqrt{\frac{2k^2}{k-1} \left(\frac{2}{k+1} \right)^{\frac{k+1}{k-1}} \left(1 - \left(\frac{P_e}{P_0} \right)^{\frac{k-1}{k}} \right)} + \frac{(P_e - P_a)A_e}{P_0 A_t} \quad (4.5)$$

Considering a correction factor η for different losses in the nozzle section as described above then the C_{freal} is given as:

$$C_{freal} = \eta_{nozzle} C_f \quad (4.6)$$

This value is calculated in each iteration to determine the variation during the operation of a motor. Now, the thrust can be defined in terms of chamber pressure, throat area and coefficient of thrust as:

$$F = C_f A_t P_0 \quad (4.7)$$

The two inputs: the convergent half-angle (β) and the divergent half-angle (α), both in degrees is used to compute the geometric profile of the nozzle using the specified inlet, throat, and exit diameters. The lengths of the convergent and divergent sections are calculated using basic trigonometric relationships:

$$L_{convergent} = \frac{r_{inlet} - r_{throat}}{\tan(\beta)}, \quad L_{divergent} = \frac{r_{exit} - r_{throat}}{\tan(\alpha)}$$

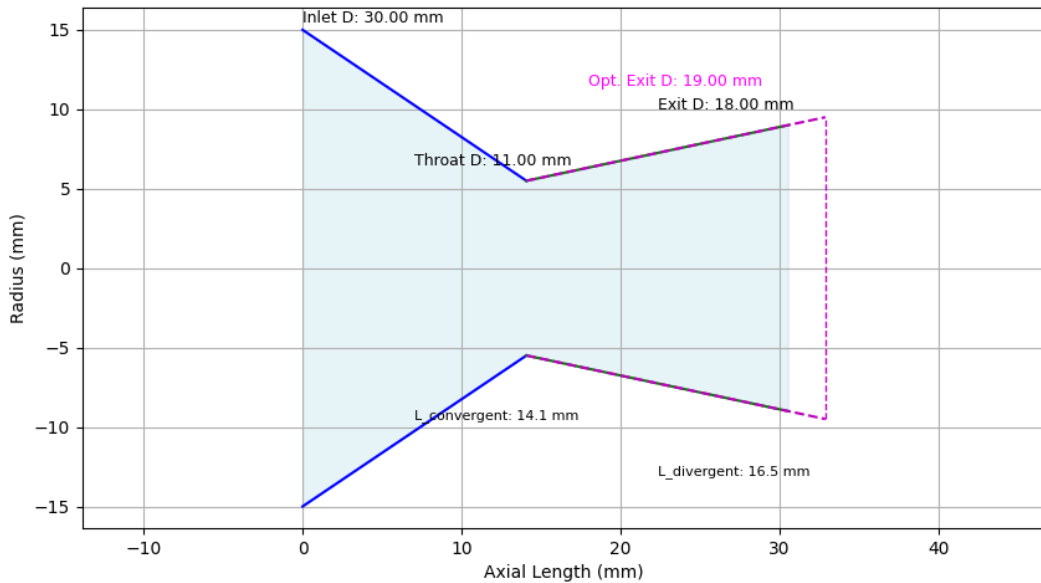


Figure 4.7: Nozzle Designer

The following table represents the summary of a nozzle design parameters:

Table 4.2: Characteristic length (L^*) ranges for various propellant combinations

Propellant Combination	L^* (cm)
Nitric Acid / Hydrazine-base Fuel	76–89
Nitrogen Tetroxide / Hydrazine-base Fuel	76–89
Hydrogen Peroxide / RP-1 (including catalyst bed)	152–178
Liquid Oxygen / RP-1	102–127
Liquid Oxygen / Ammonia	76–102
Liquid Oxygen / Liquid Hydrogen (GH_2 injection)	56–71
Liquid Oxygen / Liquid Hydrogen (LH_2 injection)	76–102
Liquid Fluorine / Liquid Hydrogen (GH_2 injection)	56–66
Liquid Fluorine / Liquid Hydrogen (LH_2 injection)	64–76
Liquid Fluorine / Hydrazine	61–71
Chlorine Trifluoride / Hydrazine-base Fuel	51–89

Finally the physical dimension of a chamber can be determined using a various relation. The characteristics chamber length L^* , a parameter for characterizing a necessary chamber volume for adequate mixing and combustion of a propellants, must be carefully considered. Ideally, L^* is purely a function of the chemistry of the propellant combination and is based on the previously successful engine design. Typical L^* values for various propellants are shown in the table below. With throat area and minimum required L^* established, the chamber volume can be calculated.

Table 4.3: Characteristic Length (L^*) for Various Propellant Combinations [18]

Propellant Combination	L^* (cm)
Nitric Acid / Hydrazine-base Fuel	76–89
Nitrogen Tetroxide / Hydrazine-base Fuel	76–89
Hydrogen Peroxide / RP-1 (including catalyst bed)	152–178
Liquid Oxygen / RP-1	102–127
Liquid Oxygen / Ammonia	76–102
Liquid Oxygen / Liquid Hydrogen (GH_2 injection)	56–71
Liquid Oxygen / Liquid Hydrogen (LH_2 injection)	76–102
Liquid Fluorine / Liquid Hydrogen (GH_2 injection)	56–66
Liquid Fluorine / Liquid Hydrogen (LH_2 injection)	64–76
Liquid Fluorine / Hydrazine	61–71
Chlorine Trifluoride / Hydrazine-base Fuel	51–89

For an ethanol based system a low-range value of $L^* = 120\text{cm}$ was chosen. Mathematically L^* can be defined as:

$$L^* = \frac{V_c}{A_t} = \frac{\pi r_c^2 L_c}{A_t} \quad (4.8)$$

where L_c is the length of chamber and r_c radius of a chamber. Due to availability and o-ring sizing a chamber radius r_c of 15mm was chosen. Then based on L^* value the length of a combustion chamber can be obtained as:

$$L_c = \frac{A_t L^*}{\pi r_c^2} \quad (4.9)$$

To calculate the volume of the combustion chamber Eq. 4.10 is used. As shown in Table 4.3 the value of L^* typically ranges from 0.8m to 3m. For the combination of hydrocarbon and oxygen the typical L^* ranges from 1.02m to 1.3m.

$$V_c = \left(\frac{\pi D_c^2}{4} \right) \left(L_c + \left(\frac{D_c - D_t}{2 \tan \beta} \right) \left(1 + \left(\frac{2}{D_c} \right) \sqrt{\frac{A_t}{\pi}} + \frac{4A_t}{\pi D_c^2} \right) \right) \quad (4.10)$$

Eq 4.10 was plotted several times for different values of D_c against a range of L_c values and compared with the calculated value. The range of D_c used was selected based on the suggestion that chamber diameter should be two to five times the nozzle throat diameter so the injector will have usable face area.

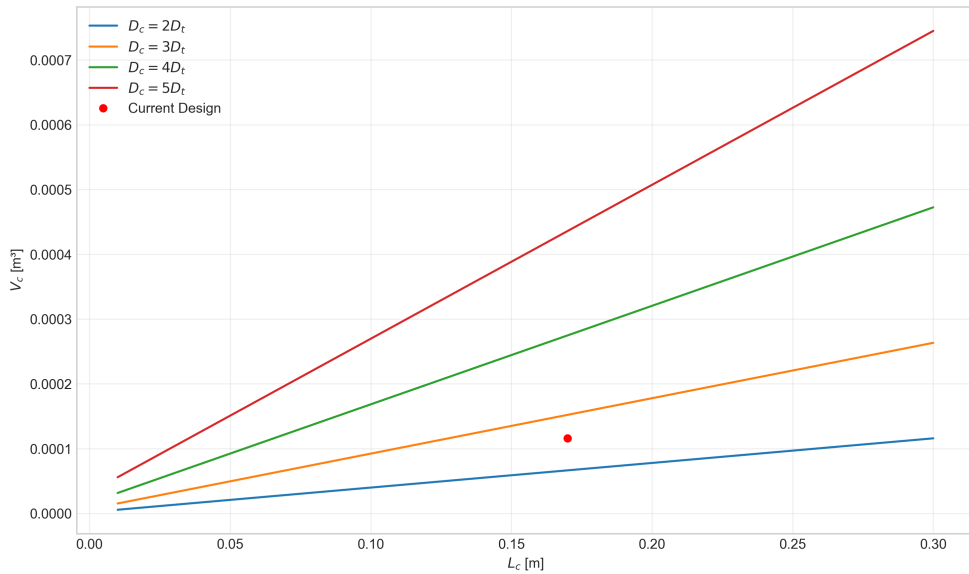


Figure 4.8: Chamber Volume

Table 4.4: Rocket engine performance and geometric parameters

Name	Value	Unit
$\dot{m}$, Total mass flow rate	-	kg/s
$\dot{m}_f$, Fuel mass flow rate	-	kg/s
$\dot{m}_o$, Oxidizer mass flow rate	-	kg/s
v_e , Exhaust velocity	-	m/s
η_{nozzle} , Nozzle efficiency	-	—
I_{sp} , Specific impulse	-	s
C_F , Thrust coefficient	-	—
L^* , Characteristic length	-	mm
CR, Contraction ratio	-	—
L_c , Chamber length	-	mm
A_c , Chamber area	-	mm ²
D_c , Chamber diameter	-	mm

4.2.2 Injector Sizing

The injector has two main objective in rocket engine: the proper mixing (atomization) of the upstream incoming propellant streams to provide maximum combustion efficiency and to prevent and minimize the combustion instabilities. Some injector uses film cooling to reduce the thermal load on a chamber and nozzle but for a current design the system is solely left on a heat capacity of the chamber and thin fiberglass reinforced phenolic resin ablative layer to maintain wall integrity. For current design, the injector will have two manifolds, which are two machined grooves. Each manifold will have number of orifices for one of either the fuel or oxidizer, which is where the propellants pass through from the feed system to the combustion chamber.

For a current design, an Unlike doublet impinging type injector is chosen where there are pairs of fuel and oxidizer orifices which are angled to collide with each other. The orifices of the injector will be sized such a way that the desired mass flow rate of the oxidizer and fuel enter the combustion chamber. The thrust is dependent on total mass flow rate and the ratio between the fuel and oxidizer mass flow rates is dependent on the desired O/F ratio needed for a desired combustion. The mass flow rate through an injection orifice is given below:

$$\dot{m} = Q\rho = C_d A \sqrt{2\rho\Delta p} \quad (4.11)$$

where C_d is a dimensionless discharge coefficient which is experimentally, A is the area of orifice, and Δp is the pressure drop across the injector. As for a preliminary design the value of C_d ranges from 0.6 to 0.65 for a sharp edge orifice [13]. Injector flow velocity is obtained as:

$$v = Q/A = C_d \sqrt{2\Delta p/\rho} \quad (4.12)$$

The value of C_d was determined experimentally after manufacturing an injector with the help of cold flow testing and obtained value will be back substitute to get required mass flow rate of propellants. It has been suggested that the ratio of orifice length to orifice diameter should be between 4 and 10 i.e $4 < L/D < 10$ [19]. It is found that smaller orifice size provides better mixing in all scenarios, limited to orifice diameters < 0.762 mm saw insignificant improvement in atomization and mixing [20]. Due to manufacturing constraint the hole size of an orifice will be limited to the diameter of 1mm. So, using the information presented above a design parameter of an overall injector pressure drop of 25% of a chamber pressure is used. And based on that an orifice area and diameter for oxidizer and fuel is determined. For individual orifice diameter calculation the following relation is used:

$$d = 2 \sqrt{\frac{\dot{m}}{C_d N \pi \sqrt{2\rho\Delta p}}} \quad (4.13)$$

where N is the number of orifice in injector. For a current design the number of oxidizer and fuel orifice chosen is 2 due to manufacturing point of view and ease of atomization. The angle of impingement λ is defined as the angle between two propellant jet coming from an orifice. The value of 60 is chosen since it is a most common impingement angle and prevents significant backlash into an injector face and also produces best atomization characteristics. It is found the impingement distance, known as the distance from the injector plate to the point where the oxidizer and fuel streams collide of the oxidizer and fuel stream should be between 3 to 7 times the diameter of orifice (i.e $3 < \frac{L_i}{D} < 7$). The L_i/D ratio of 5 is chosen, and free-stream jet length is determined. The angle between the resulting stream and normal line of an injector plate is determined as:

$$\tan \beta = \frac{\dot{m}_o v_o \sin \alpha_1 - \dot{m}_f v_f \sin \alpha_2}{\dot{m}_o v_o \cos \alpha_1 + \dot{m}_f v_f \cos \alpha_2} \quad (4.14)$$

The resulting angle towards the centerline is considered as positive while is desired to have a resulting angle of zero for better efficiency a positive angle helps to minimize the heat flux from propellant combustion. The orifice distance, distance between two fuel and oxidizer orifice is determined using an impingement distance relation:

$$d_o = d_i(\tan \alpha_f + \tan \alpha_o) \quad (4.15)$$

Table 4.5: Injector Parameters

Parameter	Oxidizer	Fuel
Density (ρ)	– kg m ⁻³	– kg m ⁻³
Mass flow rate ($\dot{m}$)	– kg s ⁻¹	– kg s ⁻¹
Discharge coefficient (C_d)	–	–
Number of orifices (n)	–	–
Orifice Area (A)	– mm ²	– mm ²
Individual orifice area (a)	– mm ²	– mm ²
Impingement angle (λ)	–	–
Impingement distance (L_i)	– mm	
Resultant Angle (β)	– towards centerline	
Orifice Distance (d_o)	– mm	
Orifice plate thickness (L_o)	– mm	
Injector element type	–	

4.2.3 Pressure Vessel Design

A proper design of a pressure vessel is necessary for proper storage and desired supply of the propellant. For current analysis, a pressure burst factor and bolted closer design is studied. The tank that was used for storing a propellant was initially designed and used for solid rocket motor casing. The tank is designed to store the liquid ethanol and is maintained at operating feed pressure using gaseous nitrogen as the pressurant. Based on material availability, the following material properties were chosen. The properties were obtained from [21] as specified in MIL-HDBK-5G.

The burst factor of a pressure vessel is obtained from historical test and using polynomial coefficients a, b, c, d, e obtained from linear regression, the burst factor B is calculated

Table 4.6: Material properties of the tank (Steel, stainless)

Property	Value (SI)
Material	Stainless Steel
Specification	SAE 301, 1/4 hard, tubing
Typical Usage	General purpose
Yield Strength	517 MPa
Ultimate Tensile Strength	862 MPa
Ultimate Shear Strength	468 MPa
Yield Bearing Strength	930 MPa
Young's Modulus (E)	186.3 GPa
Poisson's Ratio	0.27

by:

$$B = a\beta^4 + b\beta^3 + c\beta^2 + d\beta + e \quad (4.16)$$

The casing design pressure P_d was derived from thin-wall pressure vessel theory:

$$P_d = \frac{2t\sigma_y}{D_{out}S_d} \quad (4.17)$$

The casing burst pressure P_u is given by:

$$P_u = \frac{2t\sigma_y B}{D_{out}} \quad (4.18)$$

The burst safety factor S_u is the ratio of burst pressure to design pressure:

$$S_u = \frac{P_u}{P_d} \quad (4.19)$$

Table 4.7: Propellant Tank Design Output (Steel, Stainless)

Property	Value
Yield Strength	517 MPa
Young's Modulus	186.3 GPa
Poisson's Ratio	0.27
Outer Diameter	76.2 mm
Wall Thickness	1.6 mm
Design Pressure	1574.50 psi (≈ 10.85 MPa)
Burst Pressure	5463.47 psi (≈ 37.68 MPa)
Design Safety Factor	2
Burst Safety Factor	3.47

Since the propellant tank will be using a bolted closer design an analysis is carried out for different bolted closer failure. Currently flange system is not modeled. The procedure for the calculation of bolt shear stress, tear out stress and casing tensile stress has been studied [6]. The shear stresses in each bolt due to internal tank pressure is given as:

$$\sigma_{\text{bolt, shear}} = \frac{\frac{\pi}{4}(D_{\text{out}} - 2t)^2 P_{\text{MEOP}}}{N \frac{\pi}{4} d_{\text{bolt, minor}}^2} \quad (4.20)$$

where D_{out} is the outer diameter of the casing (mm), t is the wall thickness (mm), P_{MEOP} is the Maximum Expected Operating Pressure (MPa), N is the number of bolts, and $d_{\text{bolt, minor}}$ is the minor diameter of the bolt (mm). Factor of safety in bolt shear can be given by:

$$FS_{\text{bolt, shear}} = \frac{0.75 \sigma_{\text{yield, bolt}}}{\sigma_{\text{bolt, shear}}} \quad (4.21)$$

The bolt tear out is the failure occurs when the bolt connecting the casing tear out from casing edge. This failure occurs when the bolt connection is very close to casing edge. Force acting on a single bolt due to chamber pressure can be estimated as:

$$F_{\text{bolt}} = \frac{\frac{\pi}{4}(D_{\text{out}} - 2t)^2 P_{\text{MEOP}}}{N} \quad (4.22)$$

The equation for calculating the tear-out stress was then implemented as:

$$\sigma_{\text{tear}} = \frac{F_{\text{bolt}}}{2t(E - \frac{d_{\text{bolt, major}}}{2})} \quad (4.23)$$

where E is the edge distance (mm), and $d_{\text{bolt, major}}$ is the major diameter of the bolt (mm). Hence, the safety Factor in tear given as:

$$FS_{\text{bolt, tear}} = \frac{\tau_{\text{ultimate, Casing}}}{\sigma_{\text{tear}}} \quad (4.24)$$

When there is large number of holes a portion of a casing will likely to stretch beyond its limit then there will be casing tensile failure. It is also likely to occur when the casing wall is thin. The stresses and safety factor can be calculated as:

$$\sigma_{\text{tensile}} = \frac{\frac{\pi}{4}(D_{\text{out}} - 2t)^2 \cdot P_{\text{MEOP}}}{[(D_{\text{out}} - t)\pi - N d_{\text{bolt, major}} t]} \quad (4.25)$$

$$FS_{\text{tensile}} = \frac{\sigma_{\text{yield, casing}}}{\sigma_{\text{tensile}}} \quad (4.26)$$

Bearing failure occurs when force from the bolts pushing against the edges of their holes causes the casing material to fail in compression. It is most likely to occur when there is small number of bolts of small diameter, or when the casing wall is relatively thin.

$$\sigma_{\text{bearing}} = \frac{F_{\text{bolt}}}{d_{\text{bolt, major}} t} \quad (4.27)$$

$$FS_{\text{bearing}} = \frac{\sigma_{\text{bearing, yield}}}{\sigma_{\text{bearing}}} \quad (4.28)$$

The following table represents the bolt material that will be used and it is to be noted that this bolt material has been successfully used previously for solid rocket motor design.

Table 4.8: Bolt Material: Alloy Steel (ASME B18.3.1 ≤ M16)

Property	Value
Yield Strength	1170 MPa
Tensile Strength	1300 MPa
Shear Strength	780 MPa
Number of Bolts	8
Major Diameter	5 mm
Minor Diameter	4.2 mm
Edge Distance	13 mm ($\geq 2 \times$ Major Dia.)
Operating Pressure	300 psi (≈ 2.06 MPa)

Table 4.9: Bolt Joint Factor of Safety

Failure Mode	FoS
Shear	11.23
Tear	14.56
Tensile	10.18
Bearing	6.88

4.3 Combustion Instability

Although high frequency instability doesn't occur in a small diameter chamber. Combustion stability was evaluated using the high-frequency instability methodology im-

plemented in RocketIsp, based on the Hewitt stability criterion [22]. High-frequency instabilities arise from coupling between the combustion process and the chamber acoustic modes and are the most potentially damaging to a liquid rocket engines.

The following equations are used to calculate the frequencies of the different modes [23] as:

$$f_{1T} = (1.8412 c)/(2\pi R_c) \quad (4.29)$$

$$f_{2T} = (3.0542 c)/(2\pi R_c) \quad (4.30)$$

$$f_{1R} = (3.8317 c)/(2\pi R_c) \quad (4.31)$$

$$f_{2R} = (7.0156 c)/(2\pi R_c) \quad (4.32)$$

Injector instability was analysed using the Hewitt criterion, which relates injector geometry and injection velocity of the least volatile propellant to chamber acoustics. Assuming a Strouhal number of 0.2, the injector disturbance frequency is given by

$$f_H = 0.2 \frac{U}{d}, \quad (4.33)$$

where U is the injector exit velocity and d is the injector orifice diameter. For stable operation, according to Hewitt criterion the following condition must be satisfied:

$$f_H < f_{crit}. \quad (4.34)$$

Also it is to be noted that since the combustion takes place in a gas phase i.e both oxidizer and fuel must be vaporized which in our design a least volatile propellant is ethanol that vaporizes last and therefor controls combustion rate and the potential combustion coupling with chamber acoustics. The calculated acoustic frequencies and injector stability parameters for the present design are summarized in Table 4.10.

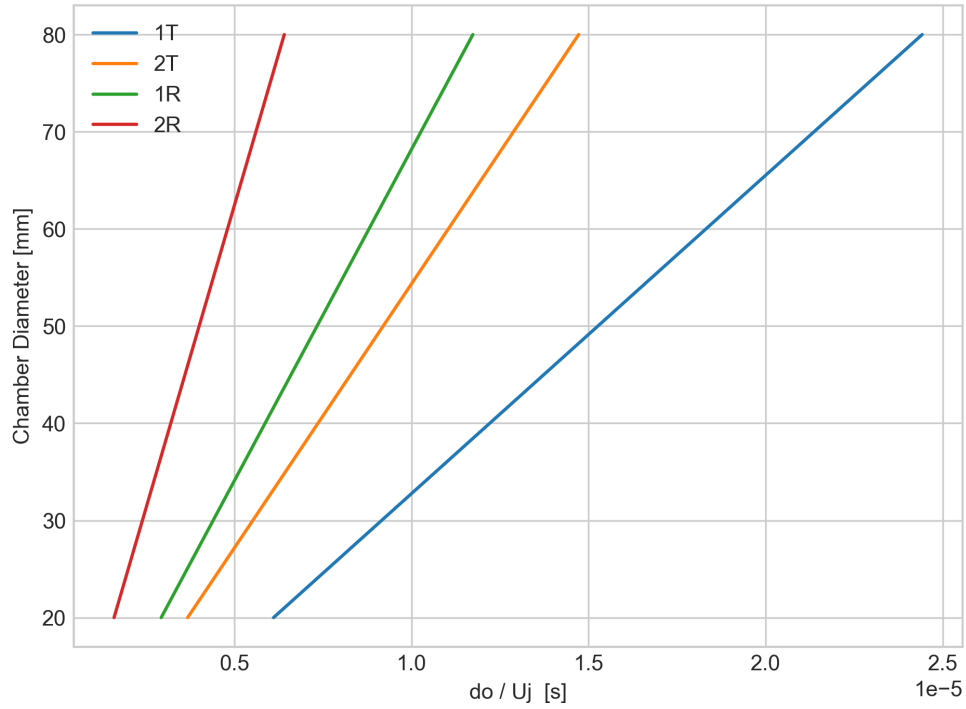


Figure 4.9: Hewitt Stability

Table 4.10: Combustion stability analysis results

Parameter	Value	Unit
Throat radius, r_t	-	mm
Contraction ratio, CR	-	-
Injector velocity (oxidizer)	-	m/s
Injector velocity (fuel)	-	m/s
Injector diameter (oxidizer)	-	mm
Injector diameter (fuel)	-	mm
First tangential mode, f_{1T}	30,037	Hz
Critical frequency, f_{crit}	30,037	Hz
Hewitt frequency, f_H	1,937	Hz

Since the Hewitt frequency is significantly lower than the critical chamber acoustic frequency, the engine is expected to operate in a stable regime in terms of high-frequency combustion instabilities.

4.4 Material Selection and Manufacturing

The next steps after sizing injector, thrust chamber and nozzle will be the selection of materials to be manufactured. As the primary material for an engine system the materials

that are suitable for liquid rocket engine: Copper, Low carbon steel and Stainless Steel. The high thermal conductivity material more specifically copper is the area of option for uncloosed chamber but due to cost constraint and unavailability it is less focused. Then, the Mild Steel is chosen which was previously used in manufacturing of nozzle and bulkhead of a solid rocket motor. Although low carbon steel couldn't be used in area of combustion in contact with GOX as it has tendency to oxidized, all the engine components were fabricated from the low carbon steel which includes chamber, injector and nozzle due to it's availability and manufacturibility.

As for an initial firing of engine the combustion chamber was designed to use ablative materials. This material will acts as a liner between the inner wall of a cylindrical combustion chamber to dissipate heat from a combustion of propellants. There are many materials available that acts as ablative materials like Epoxy, Epoxy/Hydrated Magnesium Sulfate, Fiberglass, Fiberglass Reinforced Phenolic Resin, etc From the various report it is recommended to use Phenolic based fiberglass liner. PVC has also been seen as a cheap alternative and effective ablative materials. All the options will be finalized based on ease of synthesis, toxicity, availability.

However in the final engine configuration, the use of an ablative liner was not implemented due to the small combustion chamber diameter, which gives strict geometric constraints and made liner placement impractical without significantly reducing the effective chamber volume.

The converging diverging nozzle especially at throat which experience a highest heat flux will be made up of Mild Steel. Although mild steel is not a good option but due to ease of manufacturing and availability it is go to options. There is also a graphite material which is great materials for heat dissipation due to its molecular structure, and has been used in many experimental and large scale launch vehicles. Hence, if the graphite is available it will be used as an nozzle materials.

The selection of propellant feed lines was carried out with particular emphasis on operational safety and pressure compatibility. For the oxidizer (GOX) line, a steel braided flexible hose was selected. This choice was mainly due the risk of flashback and

ignition associated with high pressure gaseous oxygen. Steel-braided hoses provides resistance to ignition, abrasion, and mechanical damage compared to polymer hoses, making them suitable for oxygen service feed system applications.

For the fuel line, a high-pressure hydraulic hose was used. Since the fuel used is less reactive than oxygen and does not present the same ignition potential, a hydraulic hose rated for the operating pressure provided a better option while maintaining adequate safety margins.

The injector design underwent a modification during the manufacturing phase. The initial design was based on a triplet injector configuration, consisting of two fuel injection orifices arranged around a central oxidizer orifice. However, during fabrication, drilling two spaced fuel holes with the required small diameter was challenging due to tooling limitations and drill bit issues at small scales. To overcome this manufacturing limitation, the injector design was simplified to a doublet configuration, employing a single fuel injection orifice which is larger in diameter and a single oxidizer orifice. This modification reduced fabrication limitation while maintaining atomization and mixing characteristics.

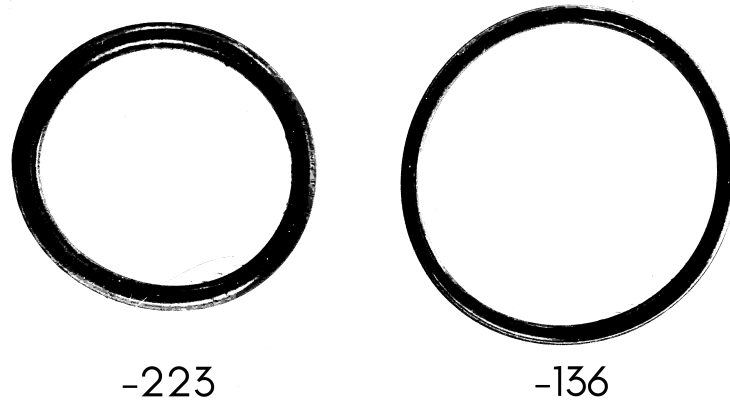
Another selection criteria includes the selection of perfect O-rings. O-rings are the seals that prevent fluid flow between two adjacent metallic plates. They work by absorbing heat and responding to the released pressure by expanding to fill a groove gap. The Buna Nitrile O-rings will be used for an injector and nozzle assembly. For a groove and gland design, an ARP 1232 (Aerospace Recommend Practice) procedure is used. This include the sizing of various series of o-ring which are -010, -100, -200, -300 series. The Parker's O-Ring Handbook provides guideline for handling and assembling o-rings. For the gland depth and gland width the Table 4.12 represents the design crieteria based on standard ARP Practice.

Table 4.11: O-Ring Dimensions (Size 136-223)

Size	Nominal ID	± Tol. ID	Nominal OD	Min. OD	Max. OD
136	1.98	0.017	2.19	2.17	2.210
223	1.61	0.015	1.88	1.87	1.9

Table 4.12: O-Ring Gland Depth and Width

O-ring Series	D, GLAND DEPTH		G, GLAND WIDTH	
	Min.	Max.	Min.	Max.
010	0.050	0.052	0.080	0.085
100	0.080	0.084	0.120	0.125
200	0.109	0.113	0.165	0.170
300	0.168	0.172	0.240	0.245

**Figure 4.10:** Different series O-ring

4.5 Pressure Drop Across the System (Δp_{system})

When designing a feed system for a liquid rocket engine it is necessary determine the tank volume and pressure. This pressure and volume will drives everything downstream of it and in fact it helps in operation of an entire system. To set the minimum propellant tank pressure it is necessary to determine the pressure drop across the plumbing system. The main object is to prevent the backflow of the propellant due to negative pressure gradient at the end of burn time. Theoretically the pressure drop in the system is sum of pressure drop across injector, valve and head loss. Mathematically, this can be represented as:

$$\Delta p_{sytem} = \Delta p_{injector} + \Delta p_{valve} + \Delta p_{head} \quad (4.35)$$

As described earlier, the pressure drop across the injector $\Delta p_{injector}$ is designed for about 15-25% of chamber pressure which leads to $\Delta p_{injector}$ of 37.5psi. During the flow through pipe due to pipe roughness, length and other factor there will be some extra loss in pressure in pipe. A new term is implemented named head loss and valve

loss which occurs due to viscous effects, sudden expansion/contraction and due to obstruction.

The head loss term, h_f , is implemented to account for viscous effects in flow equations that otherwise assume inviscid flow. The term head loss represents the additional pressure loss in a pipe system due to fluid friction, sudden changes in pipe geometry, and flow obstructions such as bends, elbows, and valves. To take an account on head loss in pipe flow calculations, the Darcy-Weisbach equation is commonly used. This equation is suitable for flow in ducts of any cross-section and is applicable for all Reynolds numbers.

$$h_f = f \frac{L}{D} \frac{V^2}{2g} \quad (4.36)$$

where f represents the non-dimensional Darcy friction factor which is the function of pipe roughness and resistance. A friction factor can also be expressed as in alternate form [24]:

$$\frac{8\tau}{\rho V^2} = f = F(Re, \frac{\epsilon}{d}) \quad (4.37)$$

Then the pressure drop due to head loss can be given as $\Delta p = h_f \rho g$.

Table 4.13: Piping Design Parameters

Parameter	Value	Unit
Oxidizer	Gaseous Oxygen (at 298 K)	N/A
Fuel	Ethanol(90%)	N/A
μ_o , Dynamic viscosity (oxidizer)	2.22×10^{-5}	$\text{kg m}^{-1} \text{s}^{-1}$
μ_f , Dynamic viscosity (fuel)	0.001232	$\text{kg m}^{-1} \text{s}^{-1}$
ρ_o , Density (oxidizer)	19.44	kg m^{-3}
ρ_f , Density (fuel)	789	kg m^{-3}
L , Pipe length	-	m
d , Pipe diameter	-	mm
$\dot{m}_o$, Oxidizer mass flow rate	-	kg s^{-1}
$\dot{m}_f$, Fuel mass flow rate	-	kg s^{-1}
Pipe material	304 stainless steel, EPDM	N/A
ϵ , Roughness	0.003	m

Table 4.13 presents the piping and feed system parameter for current engine design.

An explicit relation can be used to predict the friction factor for a given pipe using

Eq 4.38.

$$\frac{1}{\sqrt{f}} \approx -1.8 \log_{10} \left(\frac{6.9}{Re} + \left(\frac{\varepsilon}{3.7D} \right)^{1.11} \right) \quad (4.38)$$

The another parameter that determines pressure drop across the system is the drop in the geometric obstruction like valve, elbows, adapters etc. To obtain the pressure drop in those obstruction, a relation between pressure drop and volume flow rate is used.

$$C_v = Q \sqrt{\frac{S_G}{\Delta P}} \quad (4.39)$$

Table 4.14: Check Valve Design Parameters (Oxidizer Side)

Parameter	Symbol	Value	Unit
Specific Gravity (Ox)	SG_{ox}	0.02	—
Volumetric Flowrate	Q	28.38	gal min ⁻¹
Flow Coefficient	C_v	5.00	—
Flow Pressure Drop	ΔP_{flow}	0.63	psi
Cracking Pressure	P_{crack}	5.00	psi
Total Valve ΔP	ΔP_{valve}	5.63	psi

Table 4.15: System Pressure Losses

Component / Segment	Pressure Drop ΔP (psi)
Main Tank to Ball Valve (Pipe)	12.49
Ball Valve to Injector (Pipe)	4.08
Check Valve (Flow + Cracking)	5.63
Elbows / Fittings (Estimated)	20
Total System ΔP_{ox}	42.2

4.6 Operational Sequence

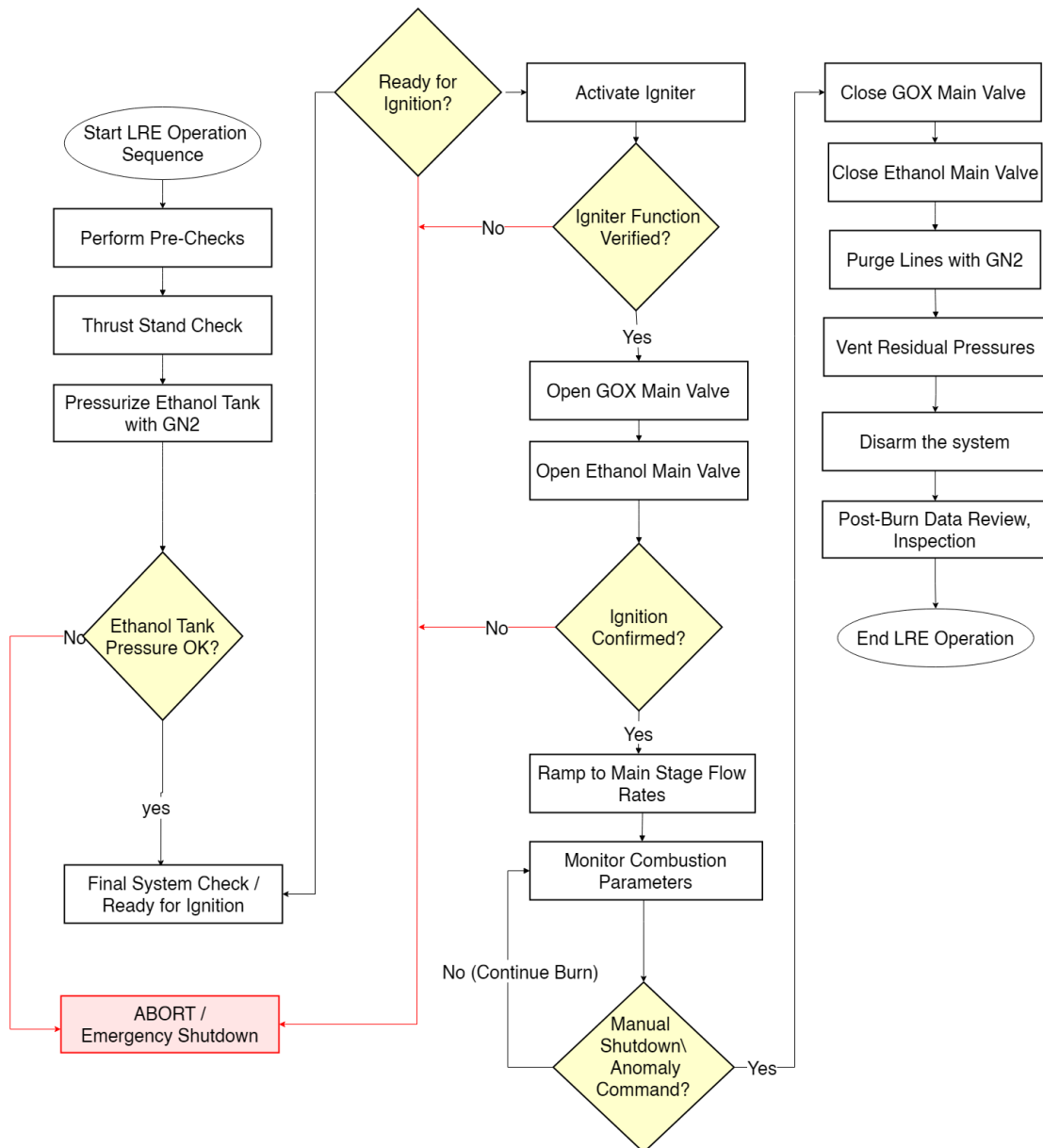


Figure 4.11: Operational sequence

4.7 Feed System Design

This phase includes the design of feed system, which is responsible for carrying a propellant from respective tank to engine. This process is heavily tied with ground operations, software analysis, safety procedure and engine design. The first steps in the design procedure of a engine feed and plumbing system includes choosing the working engine cycle, which describes the method in which the pressurization can be achieved such that propellants flow from a positive pressure gradient from the tank to

engine during operation. This may include the use of turbo-pumps, expander cycle, combustion tap-offs, inert gas etc. Due to its cost effectiveness and simplicity the pressurization of tank via inert gas is chosen for this project (also known as pressure fed cycle).

In terms of pressure fed cycle there are two methods in which a propellant tank can be pressurized. This includes pressure regulated and pressure blowdown systems (Unregulated). In pressure regulated system there is a separate pressure tank containing the pressurizing gas which pass through a regulator and into the main propellant tanks. This regulator helps to maintain the constant pressure across a burn period even the gas in the tank is expanding. With the design a typical constant thrust and pressure is expected to be obtained during the firing operation. In pressure blowdown system, the pressurant is stored in a main propellant tanks along with the propellants. The main demerit of this system is having a properties of dynamic and decaying pressure. It is obvious that, the blowdown system is chosen for its simplicity in operation and design. In a current engine design only ethanol will be pressurized using pressurant Nitrogen while GOX in a gas phase can expand during a operation.

In terms of choosing a pressurizing gas it is quite straightforward. As said earlier the pressurizing gas should be inert in nature, hence for this the only available options is Argon, Helium and Nitrogen. Based on availability and cost effectiveness and suitable with ethanol GN₂ is chosen as pressurant for propellant feed system. With determining the engine cycle and pressurant a general outline of a plumbing system is obtained. For the feed system representation, a piping and instrumentation diagram (P&ID) is defined which is a flow diagram shows the interaction and connection of feed system parts throughout the entire plumbing system. The following P&ID diagram represents the feed system assembly. This design process will be revised based on the other design aspects that may occur.

As shown in the below diagram, the feed system can be divided into mainly two parts: GOx feed system and Ethanol feed system pressurized by GN₂ and in between then it consists of a assembly for main propellant ball valve MPBV. Different type of valve will be used for feed system design.

The different working parts of the feed system can be explained further. MV-1 is used to pass the high pressure GN2 gas to pressure regulator PR-2 to pressurize Ethanol tank. At that time the pneumatic actuated ball valve MV-4 is closed. For the initial fuel tank pressurization into desired tank pressure, then MV-4 will be opened. A check valve CV-4 and pressure relief valve RV-1 is provided to prevent backflow and depressurize the tank if it exceeds the threshold pressure respectively. A pressure gauge PG-8 is provided to monitor a ethanol tank pressure. Downstream to an ethanol tank, fuel drain and fuel fill system which consists of MV-5 and MV-6 is provided which helps to fill the tank with fuel and in case of excess fuel it also helps to drain the required fuel. Downstream of the section a manual ball valve MBV-1 is provide which lets the fuel to pass to main propellant ball valve MPBV-2. By opening MPBV-2 pneumatically during firing operation of engine the pressurized fuel is passed to injector assembly through check valve CV-1. A pressure transducer PT-1 is placed to monitor the pressure.

From the perspective of oxygen system, manual valve MV-7 control the flow from high pressure oxygen tank to pressure regulator PR-1. Then the high pressure GOx is regulated to a desired pressure and is passed through a GOx feed line. A regulated blowdown GOx system which is directly fed from gox tank is used to deliver the required mass flow rate. After the line is pressurized to required pressure, then the system is ready for firing. A pressure relief valve RV-2 is also provided for safety reason. The pressure gauge PG-7 monitor the GOx buffer tank pressure. The manual ball valve MBV-2 control the flow from buffer tank to main propellant valve MPBV-1. During operation of engine MPBV-1 is operated to pass the GOx to injector and in combustion chamber. Like in fuel system check valve CV-2 is provided to prevent the backflow due to negative pressure gradient.

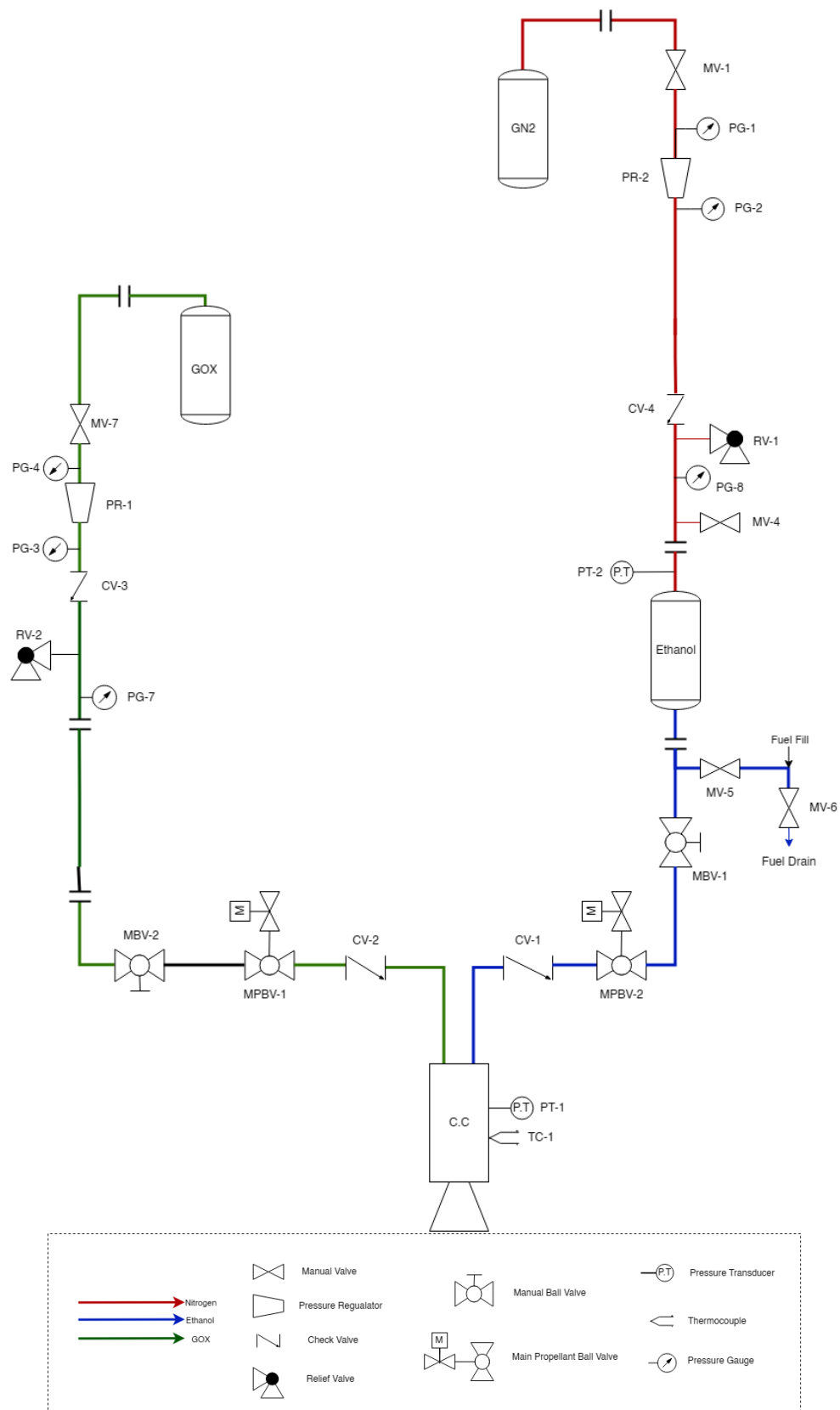


Figure 4.12: Full system piping and instrumentation diagram (P&ID).

4.8 System Design

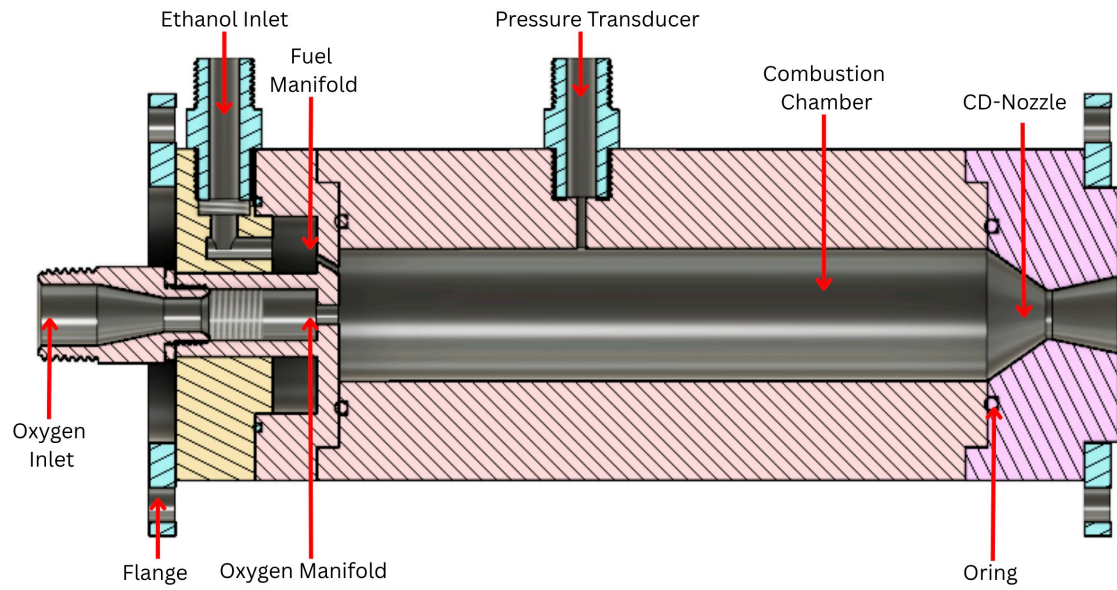


Figure 4.13: Cross-section of Thrust Chamber

4.8.1 Feed system and Thrust chamber assembly

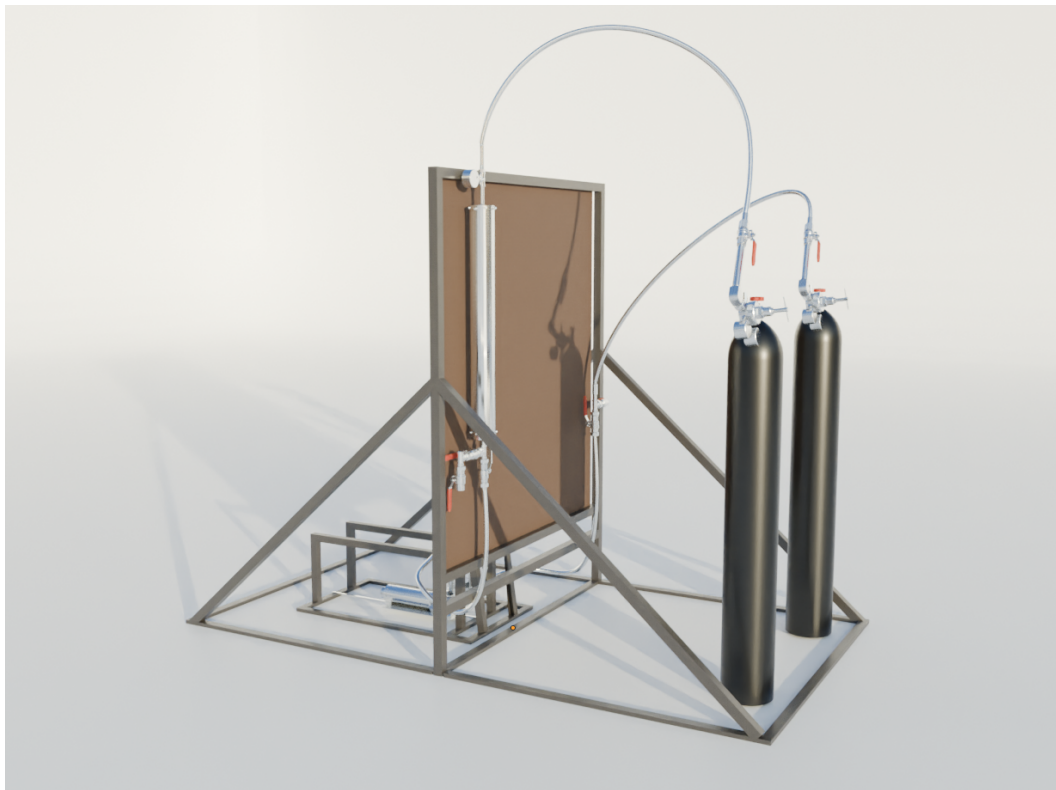


Figure 4.14: Feed system and Thrust chamber assembly (Design)

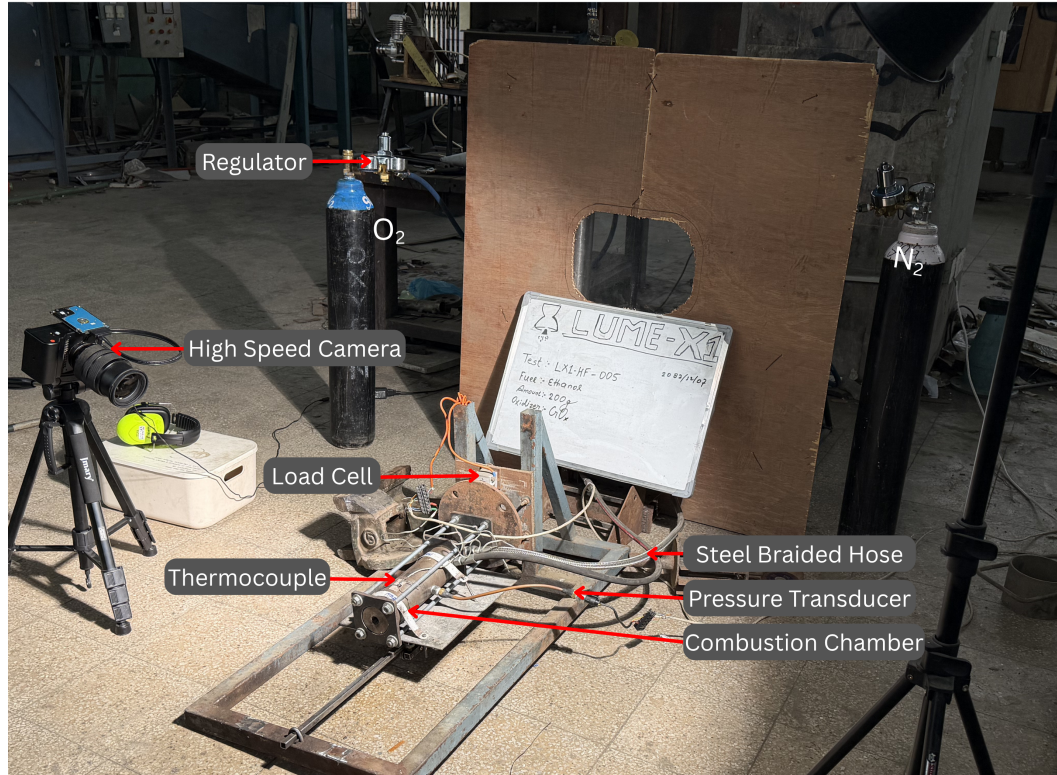


Figure 4.15: Feed system and Thrust chamber assembly (Fabricated)

4.9 Heat Transfer Analysis of Uncooled Chamber

One of the first steps in designing a thrust chamber cooling system is evaluating the heat transfer from the hot combustion gases to the chamber walls, commonly referred to as the gas-side heat transfer. Due to the high velocity of the gases along the chamber walls, heat is greatly transferred by forced convection, which involves the transport of thermal energy caused by the relative motion of the fluid. Before reaching the wall, the heat must pass through a thin layer of nearly stationary gas adjacent to the surface, known as the boundary layer. The complex convective heat transfer process can be approximated using the following correlation:

$$q = h_g(T_g - T_w), \quad (4.40)$$

where h_g is the convective heat transfer coefficient, T_g is the gas temperature, and T_w is the wall temperature at the inner surface. The Bartz correlation accounts for the Mach number variation and nozzle area ratio along the engine:

$$h_g = 0.026 \frac{\mu^{0.2} C_p}{Pr^{0.6}} \left(\frac{P_c}{c^*} \right)^{0.8} \left(\frac{D_t}{R_c} \right)^{0.1} \left(\frac{1}{A/A_t} \right)^{0.9} \sigma \quad (4.41)$$

with σ correcting for compressibility effects and Mach number variations along the axis.

$$\sigma = \left[0.5 \frac{T_w}{T_c} \left(1 + \frac{k-1}{2} M^2 \right) + 0.5 \right]^{-0.68} \left(1 + \frac{k-1}{2} M^2 \right)^{-0.12} \quad (4.42)$$

The convective heat flux is not same along the chamber and nozzle, it reaches a maximum near the throat due to the combined effects of high gas velocity, elevated pressure, and increased turbulence in this region. The constriction at the throat accelerates the flow and thins the boundary layer, which intensifies the convective heat transfer to the wall. Consequently, the throat is often the most thermally critical location in the chamber.

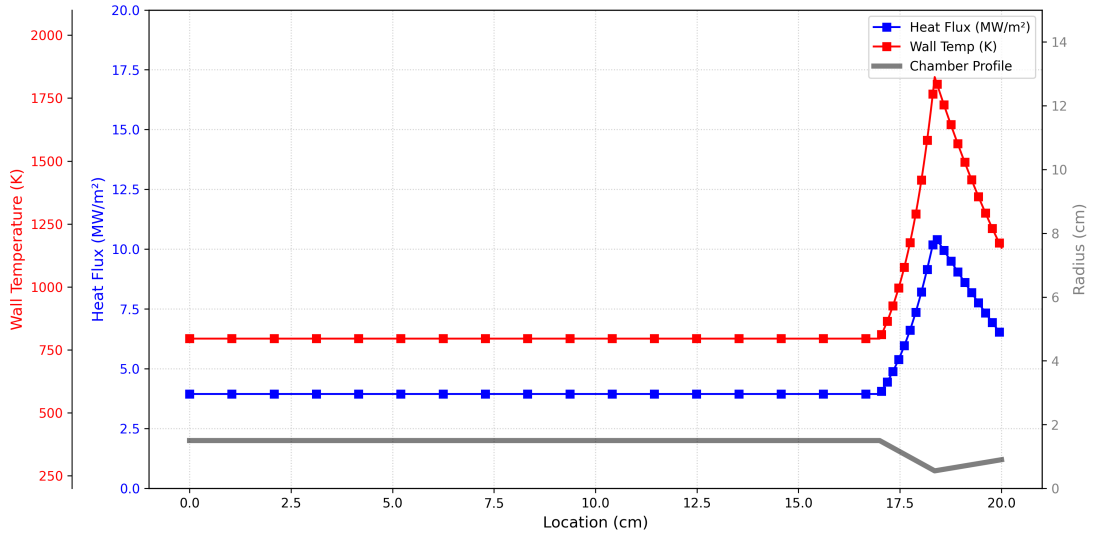


Figure 4.16: Heat Tranfer in Uncooled Chamber

4.10 Numerical Simulation of Supersonic Flow in a Nozzle

CFD is a computational tool that enables to obtain approximate solutions of the reality of the physical phenomenon, if it is appropriately considered the geometry of the computational domain, the refinement of the mesh, the turbulence model, setting the errors during the iteration stage, as well as other variable control parameters. This part is intended to continue with the analysis of the flow in the engine's nozzle, with the purpose

of determining the behavior of the flow in the throat and divergent sections for initial test. In order to achieve the given objective, the supersonic flow field was simulated to obtain the field of pressure, Mach number, velocity and temperature. Another goal is also to compare the nozzle performance with a hot fire testing.

4.10.1 Mathematical Fundamentals

For compressible flow it is taken into account a pressure and of temperatures ratio as a function of the Mach number, which is the dominant parameter, and are expressed as:

$$\frac{p_0}{p} = \left(1 + \frac{k-1}{2}M^2\right)^{\frac{k}{k-1}} \quad (4.43)$$

$$\frac{T_0}{T} = \left(1 + \frac{k-1}{2}M^2\right) \quad (4.44)$$

Where, p_0 is the total pressure, T_0 the total temperature, k the ratio of specific heats and M the Mach number. The supersonic Mach number at the outlet of the nozzle for a combustion gas flow and without shock at the outlet of the divergent section is determined with:

$$\frac{A}{A^*} = \frac{1}{M} \left[\frac{2}{k+1} \left(1 + \frac{k-1}{2}M^2\right) \right]^{\frac{k+1}{2(k-1)}} \quad (4.45)$$

where A/A^* is the expansion ratio, A is the area at the outlet of the nozzle and A^* is the area of the throat. The considerations of the Mach number are the following: $M < 0.3$ for incompressible flow; $0.3 < M < 0.8$ for subsonic flow; $0.8 < M < 1.2$ for transonic flow; $1.2 < M < 5$ for supersonic flow; $M > 5$ for hypersonic flow; and $M = 1$ for the flow with sonic velocity

In current analysis, the model of the Reynolds-averaged Simulation (RAS) equations are employed for simulating the flow field of compressible flow as described below:

1. Continuity equation (mass conservation):

$$\nabla \cdot (\rho \mathbf{u}_i) = 0 \quad (4.46)$$

2. Momentum equation:

$$\nabla \cdot (\rho \mathbf{u}_i \mathbf{u}_j) = -\nabla p + \nabla \cdot \boldsymbol{\tau} + \nabla \cdot (-\rho \overline{\mathbf{u}_i \mathbf{u}_j}) \quad (4.47)$$

where $\boldsymbol{\tau}$ is the viscous stress tensor and the Reynolds stress term $-\rho \overline{\mathbf{u}_i \mathbf{u}_j}$ is modeled using the k - ε turbulence model.

3. Energy equation:

$$\nabla \cdot [\mathbf{u}_i (\rho E + p)] = \nabla \cdot (k_{\text{eff}} \nabla T + \boldsymbol{\tau}_{\text{eff}} \cdot \mathbf{u}_i) \quad (4.48)$$

where E is the total energy per unit mass, T is the temperature, k_{eff} is the effective thermal conductivity, and $\boldsymbol{\tau}_{\text{eff}}$ is the effective stress tensor including turbulence effects.

4. Equation of state:

$$p = \rho RT \quad (4.49)$$

assuming an ideal gas, where R is the specific gas constant.

According to Sutherland's law, the dynamic viscosity μ of a gas as a function of temperature T is given by:

$$\mu(T) = \mu_0 \left(\frac{T}{T_0} \right)^{\frac{3}{2}} \cdot \frac{T_0 + S}{T + S} \quad (4.50)$$

where $\mu(T)$ is the dynamic viscosity at temperature T , μ_0 is the reference dynamic viscosity at a reference temperature T_0 , and S is Sutherland's constant. For air, typical values are $\mu_0 = 1.716 \times 10^{-5} \text{ kg/(m.s)}$, $T_0 = 273.15 \text{ K}$, and $S = 110.56 \text{ K}$. In OpenFOAM it is expressed as:

$$\mu = A_s \frac{\sqrt{T}}{1 + T_s/T} \quad (4.51)$$

So the values for a thermophysical properties would be $A_s = 1.46 \times 10^{-6} \text{ kg} \cdot \text{m}^{-1} \cdot \text{s}^{-1} \cdot \text{K}^{-0.5}$ and $T_s = 110.56 \text{ K}$.

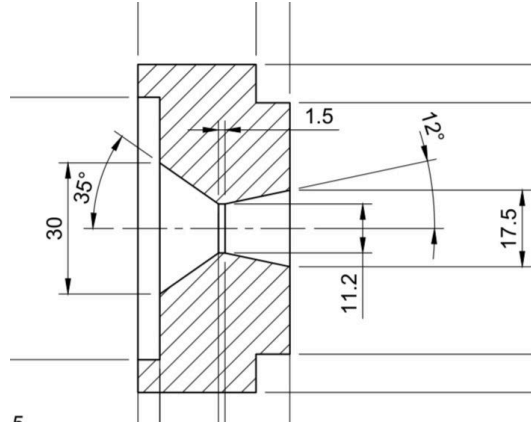


Figure 4.17: Focused Area of Domain for Study.

4.10.2 Computational Domain

The geometry consists of a typical convergent divergent nozzle with an divergent half-angle of 12° including its main dimension and corresponding units in millimeter (mm): Throat diameter: $D_t = 11.2\text{mm}$ and with an expansion ratio of $A_e/A_t = 2.67$.

The 2D computational domain consists of nozzle and atmosphere. The initial and boundary conditions are given as:

- In the combustion chamber, the desired absolute total pressure is taken as 1034213.6 Pa; and the total temperature as 3000 K.
- The flow velocity in the radial direction is zero. In the walls, the velocity is zero due to the no-slip condition.
- There is no heat loss through the wall of nozzle.

Propellant consisting of Ethanol and GOX at an OF ratio of 1.4 are considered as ideal gases. As parameters, it is fixed the specific heat ratio $k = 1.13$, $\mathfrak{M} = 21.88\text{ kg/kmol}$ and $R = 379.8\text{ J/(kg} \cdot \text{K)}$. It is noted that the data of pressure and temperature applying as a boundary condition is obtained from the chemical equilibrium analysis. For the current numerical simulations, the gases burnt as a result of the combustion. The required specific heat for the thermophysical properties is determined as

$$c_p = \frac{kR}{k-1}.$$

4.10.3 Computational Method

The simulation of compressible flow was performed using an explicit, transient, density-based method suitable for transonic and supersonic flows. The standard $k-\omega$ SST turbulence model was used to account for turbulent effects. The computational domain was considered two-dimensional and axis-symmetric about the x-axis, appropriate for nozzle geometry. The mesh was composed of structured hexahedral cells and passed all mesh quality checks, aspect ratio and skewness limits. This setup was used to analyze the flow evolution, especially the Mach number, velocity, temperature and pressure fields.

4.10.4 Post Processing

The results of flow field for input pressure load of 1034213.6 Pa, temperature of 3000 K, ambient pressure at the outlet of the nozzle of 85494.9 Pa and temperature of 300 K is presented. The variations of the Mach number, Temperature, Pressure and Velocity, shows the regions of the domain where it reached the maximum and minimum values.

The flow exiting a from a nozzle is under-expanded and it is shown in an atmosphere how the shock wave occurred and the reason of supersonic stream where these wave are present is known as plume. All variables (velocity, Mach, pressure, temperature) show oscillations after, which are indicative of shock diamonds or shock waves and expansion fans, typical in under-expanded supersonic jets. The fluctuations due to the shock wave are shown. After first the shock, the flow is still supersonic with presence of dampen fluctuations. Mach number plot shows an increase from subsonic to supersonic, peaking about Mach 2, and then oscillating. It can be observed that the nozzle is operating in a supersonic regime, with signs of shock-induced compression and expansion, characteristic of under-expanded flow.

Analyzing the behavior of the flow from another perspective, throughout the nozzle section it can be observed that the exit Mach is around 1.8. At the throat region and the flow take transition from subsonic to supersonic from throat section showing a choked flow condition.

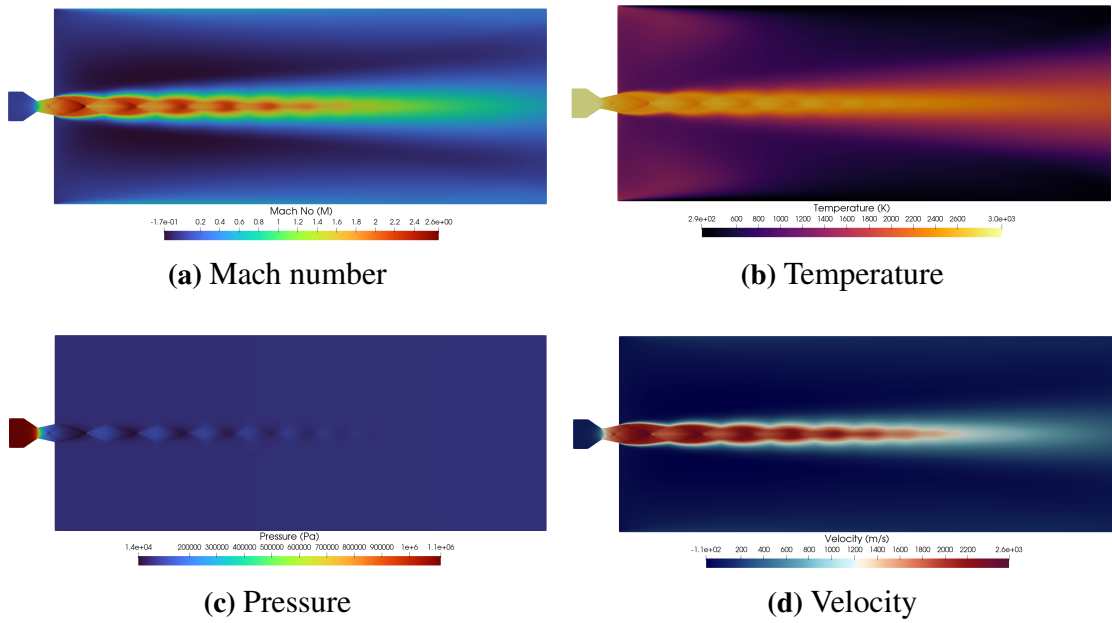


Figure 4.18: Flow field variables: (a) Mach number, (b) Temperature, (c) Pressure, (d) Velocity.

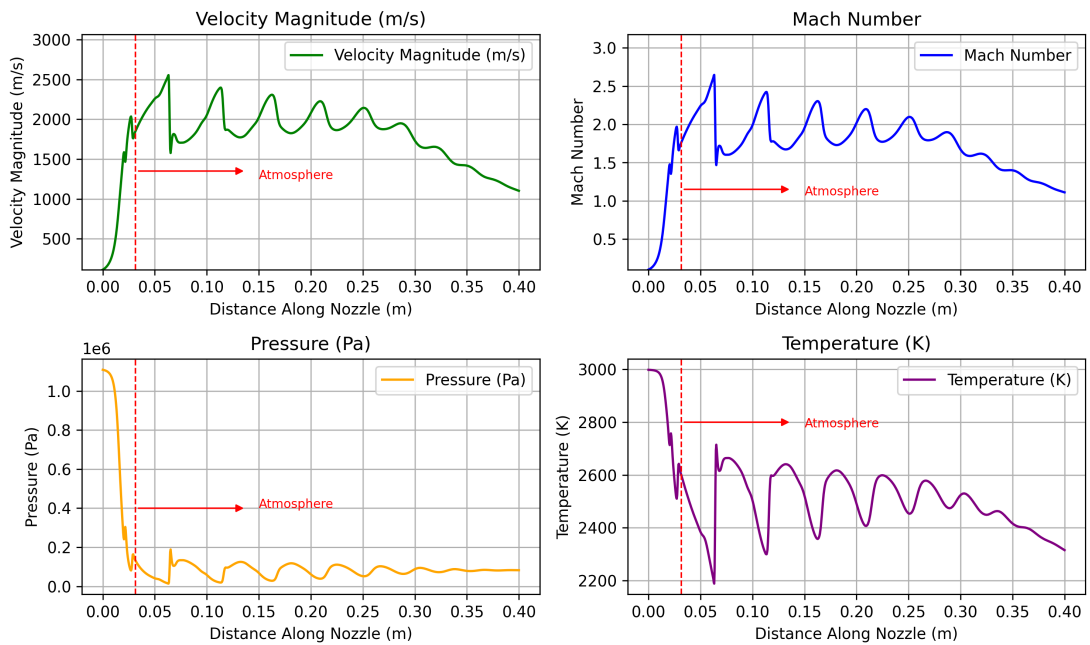


Figure 4.19: Profile evaluated in axial symmetry.

From the results of the numerical simulations of the flow it is found that at the end of a divergent section the flow reaches to supersonic about mach 2.0. Looking at a profile in the atmosphere domain it reaches to an estimated peak value of Mach 2 and flow velocity of 2000 m/s before occurrence of first shock. In the axial symmetry part of the

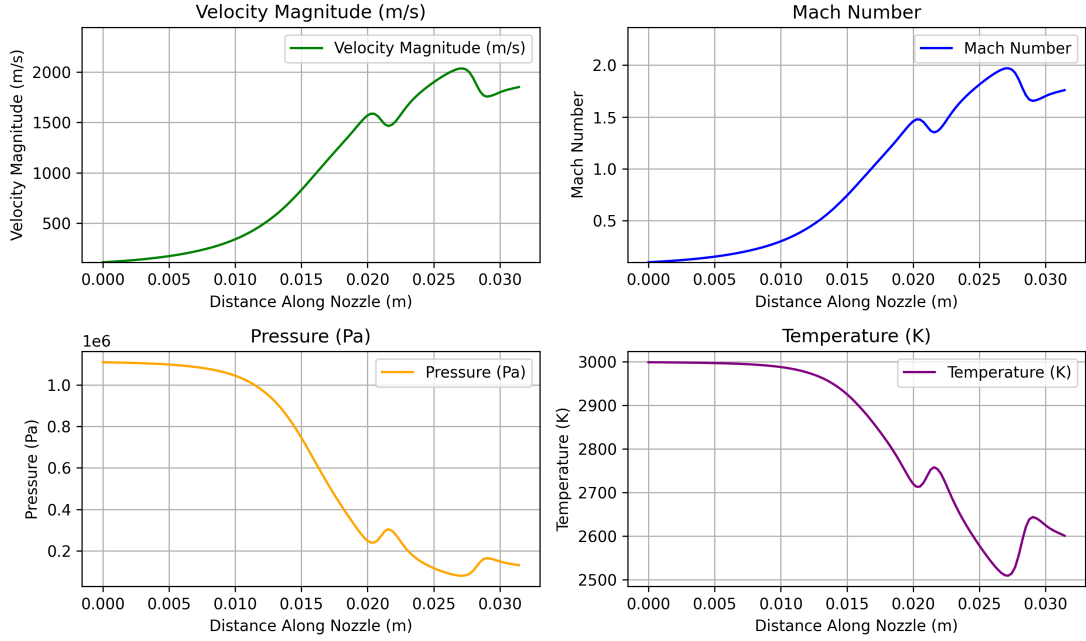


Figure 4.20: Profile evaluated in the X axis, in the nozzle section.

nozzle in the throat, a flow region is taking transition from subsonic to supersonic at the throat section, concluding a choked flow.

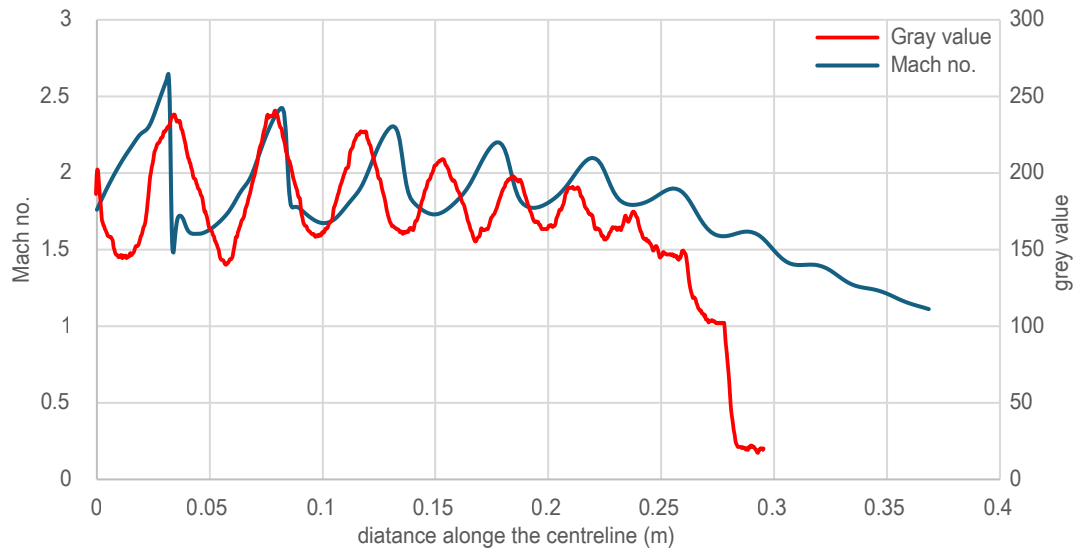


Figure 4.21: Correlation between simulated Mach number and experimental grayscale values along the exhaust centerline.

The graph [4.21](#) compares the Mach number from simulation (blue) with the grayscale intensity from the experimental exhaust image (red) along the nozzle centerline. The

oscillatory pattern in both curves indicates the presence of shock cells (Mach diamonds) in the supersonic jet with peaks aligning closely showing that higher gray intensity corresponds to stronger shock structures. This agreement suggests that the experimental visualization is effectively capturing the same flow features predicted by the simulation. Further downstream (around 0.31 m) the gray value drops sharply indicating the breakdown of visible shock structures due to mixing and dissipation while the Mach number decreases more gradually as the flow expands and loses energy.

4.11 Fabricated Components and Assembly

The fabrication and assembly of the liquid propellant rocket engine have progressed through critical design for precision machining phases. The injector assembly, combustion chamber, and convergent divergent nozzle were fabricated from mild steel utilizing conventional machining operations (turning, milling, drilling) with high precision. This ensured dimensional accuracy within the specified tolerances of the designed geometry, particularly in critical flow regions such as the injector orifices, chamber throat, and nozzle expansion section. The injector plate, a face shower type unlike doublet elements, was fabricated from mild-steel considering orifice tolerances and flatness. The propellant storage system was fabricated from stainless steel to ensure structural integrity and corrosion resistance under pressurized conditions. The feed system constructed from stainless-steel corrugated pipe fittings for the oxidizer line, providing flexibility and vibration tolerance while maintaining leak-tight performance. For the fuel line, hydraulic hose fittings were employed to accommodate pressure variations and dynamic loading, thereby ensuring reliable propellant delivery to the injector manifold. The oxidizer delivery line employs stainless-steel corrugated pipe fittings, selected for their high fatigue resistance, flexibility under thermal cycling, and ability to absorb vibrational loads without compromising structural integrity.

4.11.1 Injector

The injector employs a hybrid element design, primarily featuring a face-shower type pattern across the plate, but also incorporating at least one specialized doublet impingement element for critical mixing. This doublet element orifices, each with an area of

1.77 mm², set at a 60° impingement angle and spaced 9.7 mm apart on the 6.0 mm thick plate. Their jets are designed to collide at a point 10.76 mm from the plate, creating a resultant spray angled 8.8° toward the chamber's centerline for stable combustion. In contrast, the dominant face-shower pattern, represented by the larger 15.9 mm² orifice, is designed for broader distribution. The injector plate was machined from mild steel, maintaining exact orifice tolerances, individual areas, and plate flatness was paramount to ensure the designed spray patterns, flow rates, and resultant mixing efficiency were achieved for stable engine operation.

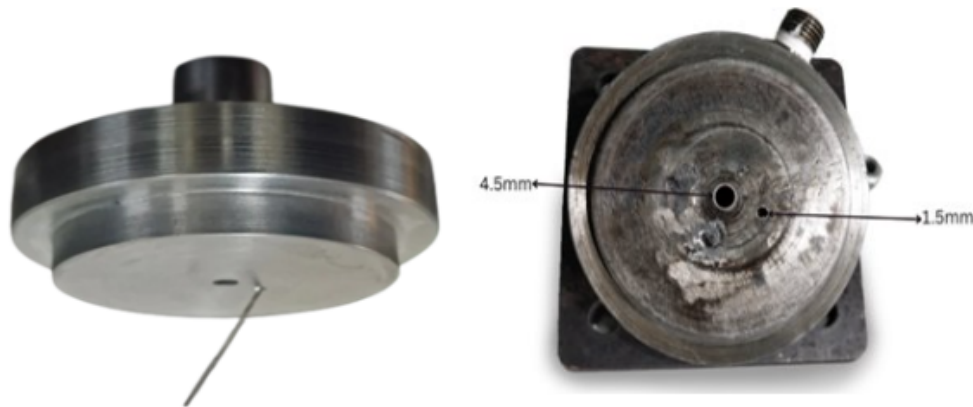


Figure 4.22: Injector

4.11.2 Combustion Chamber



Figure 4.23: Combustion chamber

The combustion chamber is a thick-walled cylindrical pressure vessel fabricated from mild steel via high-precision lathe turning and milling operations. The component features an inner diameter (ID) of 30 mm and an external diameter (OD) of 75 mm, yielding a substantial wall thickness of 22.5 mm to ensure structural integrity under

extreme thermal and pressure loads. Lathe turning was employed to achieve the precise external and internal cylindrical profiles and to establish the critical inner bore geometry, including the chamber's internal contour. Milling operations were subsequently used to machine mounting interfaces, injector and nozzle flange mating surfaces, and any external features required for integration into the engine assembly. This hybrid machining approach guaranteed strict dimensional tolerances for both the internal flow path and external interfaces, ensuring reliable pressure containment and stable combustion dynamics.

4.11.3 Nozzle

The convergent-divergent nozzle, fabricated from mild steel, features a critical throat diameter of 11 mm and an exit diameter of 18 mm. The component was machined from a solid billet via lathe turning to achieve the axi-symmetric internal profile and external contour, including the convergent and divergent sections. This precision machining ensured exact dimensional fidelity at the throat and proper contouring of the supersonic expansion region, both essential for achieving the targeted pressure ratio and specific impulse. The substantial material thickness around the high-heat-flux throat region provides the necessary thermal and structural margin for the duration of the firing.

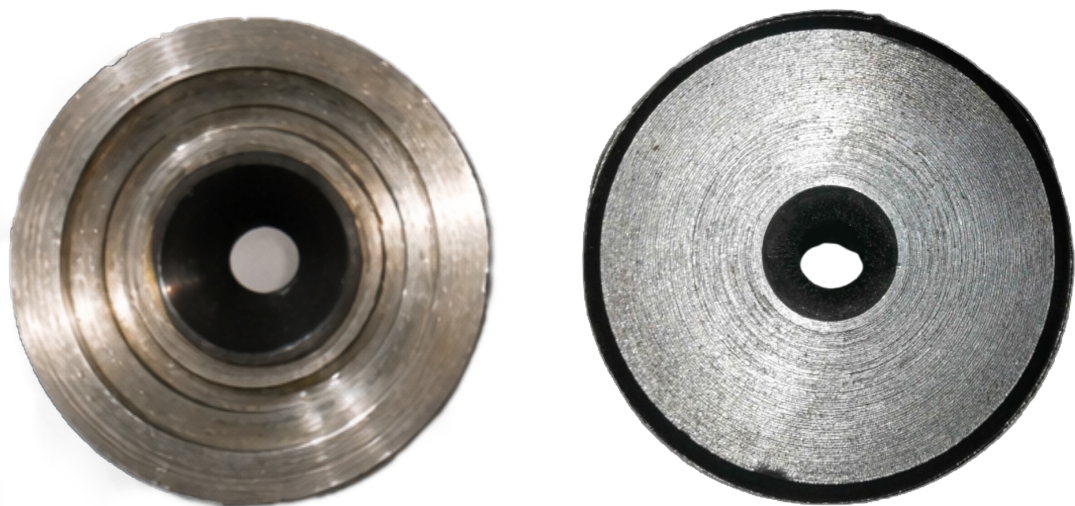


Figure 4.24: Nozzle

4.11.4 Flange system

The flange system, in conjunction with sealing elements, ensures the structural and fluidic integrity of the engine assembly. The system utilizes three standard 106 mm x 106 mm x 6 mm flanges positioned at the nozzle exit, the injector manifold, and the load-cell interface. These components are structurally integrated into a single system using M10 threaded rods and M10 hex nuts, which clamp the entire injector-combustion chamber-nozzle stack. This creates a unified load path that transmits thrust directly to the top-mounted load cell flange.

Critical static sealing system is engineered according to ARP 1232 Aerospace Recommended Practice, with Buna-N (Nitrile) O-rings selected for their compatibility and resilience under operational thermal and pressure loads. While assembly and handling followed the Parker O-Ring Handbook to prevent damage during installation. This approach ensures reliable static sealing through thermal expansion and under dynamic pressure during engine operation. O-ring seals the high-pressure interface between the injector, combustion chamber, and nozzle. This combination of a bolted flange structure and elastomeric seals ensures robust pressure containment, precise component alignment and reliable thrust vector transfer to load cell for performing static fire testing.

4.11.5 Propellant Tank and Feed System

The Propellant tank and feed system are engineered for the reliable storage and delivery of propellants under pressure. The Propellant tank is fabricated from stainless steel to ensure structural integrity and corrosion resistance under high-pressure. It serves as a pressurized reservoir that mitigates pressure fluctuations and provides a stable propellant supply to the engine during operation.

The feed system is constructed using specialized fittings for each propellant line to address distinct operational requirements. The oxidizer line employs stainless steel corrugated pipe fittings, selected for their high fatigue resistance, flexibility under thermal cycling, and ability to absorb vibrational loads while maintaining a leak-tight seal. For the fuel line, hydraulic hose fittings are utilized to accommodate pressure variations and dynamic loading, thereby ensuring consistent and reliable propellant delivery to the

injector manifold. This dual-material, functionally optimized approach ensures robust performance across all operational phases of the propulsion system.



Figure 4.25: Feed system

4.12 Injector Cold Flow

Cold flow testing includes a range of experimental activities conducted by flowing a fluid through the propulsion system in the absence of combustion. The specific objectives of cold flow testing may vary and include leak detection, pressure drop measurement, flow visualization, and verification of injector performance.

Leak testing is typically performed on the fully assembled system and is especially due to sealing challenges. Water is commonly used for this which eliminates the safety hazards associated with oxidizer exposure.

When evaluating pressure losses within the feed system, for a given mass flow rate it is generally unnecessary to measure the pressure drop across every individual component. Instead, pressure measurements are usually taken at key locations such as the propellant tank outlet, injector manifold inlet, and combustion. These measurements are often required during normal operation as well. If an unexpectedly large pressure drop is observed, additional diagnostic testing may be conducted to identify the responsible

component.

During initial development, water is frequently used as the working fluid for injector testing. This approach enables direct observation of flow behavior and spray characteristics while minimizing risk and cost. Water is widely available, non-hazardous, and supported by extensive empirical data, allowing experimental results to be extrapolated with reasonable accuracy to the intended propellants.

Overall, implementation of cold flow testing depends on the specific objectives of the development. Since cold flow testing refers broadly to any non-combusting flow experiment, inert and low-cost fluids are typically preferred, particularly during iterative design phases, provided their physical properties reasonably approximate those of the final propellants.

In our current testing the purpose is to calibrate the discharge coefficient C_d of the propellant orifice. Calibration of the discharge coefficient (C_d) was carried out to ensure desired control of propellant mass flow rate, mixture ratio and burn time enforced. The discharge coefficient for the fuel flow system was experimentally determined through a completed water cold flow testing. The results of this testing were subsequently used to validate the flow model and refine mass flow predictions for engine operation.

As described the primary objective of the injector is to produce a designed mass flow rate into the combustion chamber. Equally significant is the consideration of fluid velocity and properties during the injection process, as well as the surface tension of the fluid. To study these parameters, it is common utilizing dimensionless numbers. Specifically, the mass flow rate, fluid viscosity μ , and velocity U are connected through the utilization of the Reynolds number, which provides a characterization of the fluid flow dynamics and helps gain the potential for turbulent or laminar flow within the injector.

$$Re = \frac{\rho U D}{\mu} = \frac{4\dot{m}}{\pi D \mu} \quad (4.52)$$

Turbulence does promote a better atomization by introducing instabilities in the flow

that tend to break up the liquid into smaller droplets. The discharge coefficient strongly depends on the turbulence regime, becoming constant at high values of Reynolds number as shown in Figure:4.26.

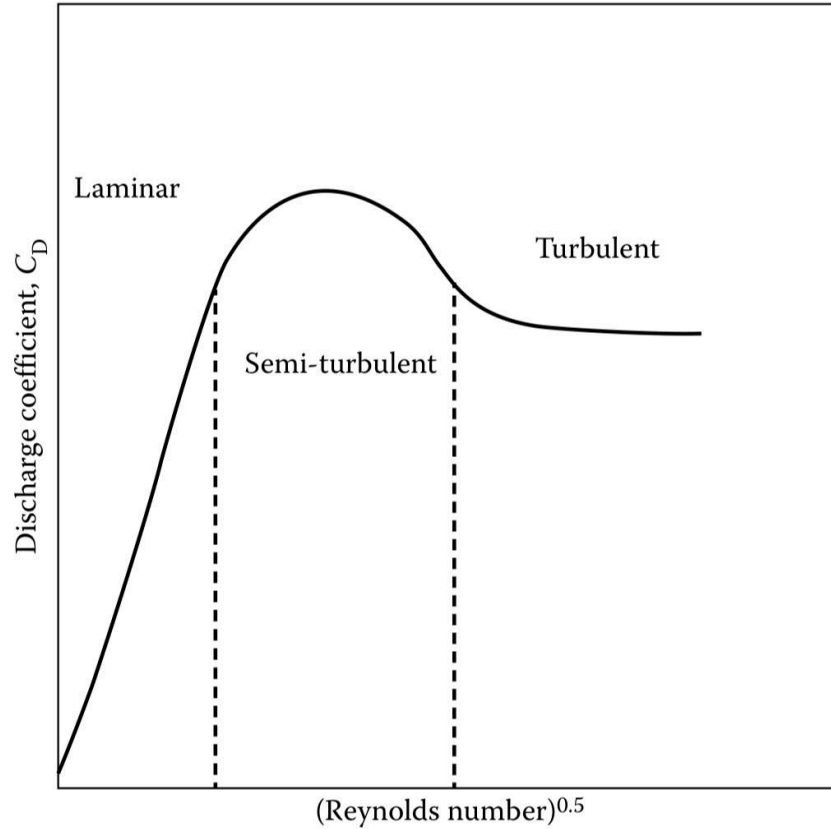


Figure 4.26: Discharge Coefficient and Reynolds number [25]

During the cold flow test campaign firstly the 3D printed injector was designed and tested to test the effectiveness of the impinging pattern and number of orifice. For the current test configuration and injector design two pattern was used:4-on-1 Impinging Injector Design, 2-on-1 Impinging Injector Design and 1-on-1 Impinging Injector Design. After the printed injector is tested then the fabrication of the real injector component is done. Manufactured injector cold flow is shown in Figure:4.27 and Figure:4.28.



Figure 4.27: Cold flow testing of 1 on 1 impingement injector at 50 psi



Figure 4.28: Cold flow testing of 1 on 1 impingement injector at 150 psi

4.13 Ablation Material

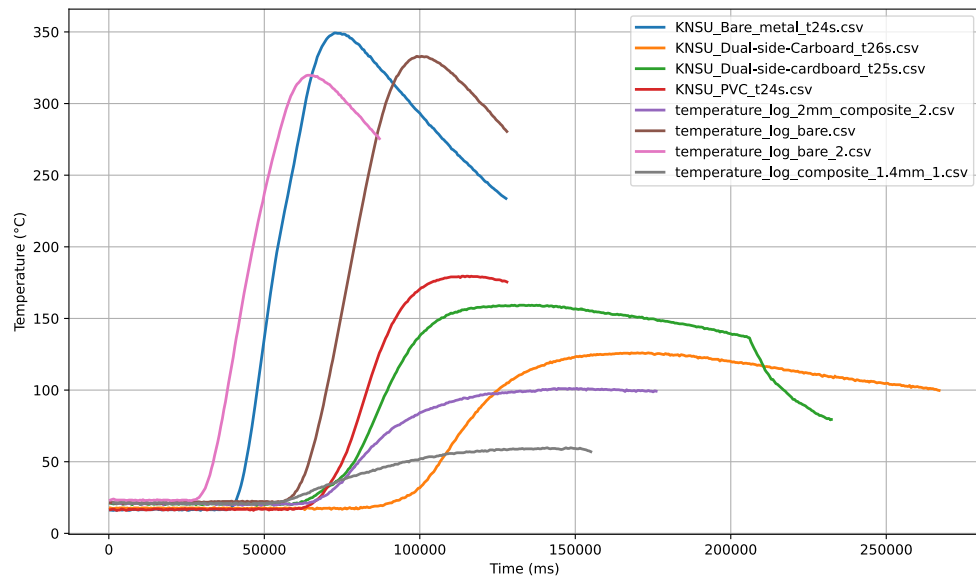


Figure 4.29: KNSU Torch Liner Test

During the initial design phase of the project, an ablative liner was selected as the primary cooling technique for the rocket engine. However, for the current hot fire test campaign, the engine was tested without the use of any ablative material. Instead, a general temperature characterization test was conducted. A solid propellant KNSU, which burn at approximately 1600 K, was used as the heat and combustion source. A thermocouple was placed on the metallic chamber wall, which had a thickness of 1.6 mm, to record the wall temperature. Various ablative materials were tested separately based on the measured thermal data. As shown in the Figure 4.29 the composite liner shows the significant temperature insulation properties as compared to other material like resin coated cardboard and PVC.

4.14 Pyrogen Based Ignition System

The selected propellants are non-hypergolic and therefore require an external ignition source. A simple and reliable pyrogen-based ignition system was selected for current test system. The igniter uses a potassium nitrate and sucrose (KNSU) pyrogen composition to provide a high-temperature source for ignition.

The pyrogen is made in a compact ignition cartridge and is inserted near the injector face of the inside the combustion chamber. Ignition is achieved by electrically initiating the pyrogen using a current supply. Upon activation, the pyrogen rapidly burns and generates a short duration, high-temperature flame, which provides reliable ignition of the propellants during engine startup. During the ignition sequence, a small pre-flow of oxidizer is first established in the chamber. The igniter is then energized, after which the main fuel flow is introduced. Once steady combustion is achieved, the pyrogen is fully consumed and the engine continues operation on the primary propellants.

This pyrogen-based ignition method was selected due to its simplicity, low cost, and high reliability, making it well suited for small-scale liquid rocket engine applications.

CHAPTER 5: RESULT AND DISCUSSION

5.1 Output

5.1.1 Injector Cold Flow

The results obtained from the cold-flow test campaign are summarized in Tables 5.1 and 5.2, corresponding to the ethanol and oxygen injector orifices, respectively. This experimental campaign was done to evaluate the performance of the injectors under respective feed pressures and to found the mass flow rates and discharge coefficients.

For each test case, the mass flow rate was determined from the measured change in fluid mass over a defined time interval, while the nominal discharge coefficient was evaluated using Eq. 4.11. The tests covered a range of feed pressures in order to assess the characteristics of the injector performance to pressure variations and to verify consistency with the design assumptions. In parallel, observations of the spray formation were conducted to ensure that the intended injection behavior was achieved within the selected operating criteria.

The ethanol injector exhibited a relatively stable discharge coefficient across the tested pressure range, indicating a turbulent flow regime within the orifice. Minor variations in the discharge coefficient were observed and are attributed to experimental uncertainties and slight deviations in flow conditions. Overall, the results confirm a near linear relationship between mass flow rate and pressure drop once the discharge coefficient is characterized.

For the oxygen injector, a wider deviation in the discharge coefficient values was observed. This behavior is primarily associated with variations in the test configuration, including changes in effective orifice L/D ratio and the use of compressor feeding in selected tests. Despite this deviation, the measured discharge coefficients remain within the expected range for sharp-edged orifices operating in transitional to turbulent regimes. Tests conducted at higher pressure drops showed improved repeatability, consistent with fully developed turbulent flow.

Based on the experimental results, an optimal feeding pressure range was identified for both injectors that ensures the desired spray characteristics and stable mass flow rates. Once the discharge coefficient C_d is determined, a direct relation between pressure drop and mass flow rate can be applied for injector sizing and mass flow control. It is noted that the discharge coefficient is not strictly constant in the laminar regime as shown Fig however, the injector is designed to operate in turbulent conditions. Also since the oxygen orifice is designed for the flow of gas phase which in reality operates on higher Reynold number.

Table 5.1: Ethanol Orifice Flow Test Results

Test	Pressure [psi]	Start Weight [g]	Final Weight [g]	Time [s]	Flow Rate [kg/s]	C_d (nominal)
1	58	64	365	9.50	0.032	0.634
2	58	62	605	19.25	0.028	0.564
3	58	63	340	9.85	0.028	0.563
4	58	61	488	15.00	0.028	0.569
5	87	62	431	10.76	0.034	0.560

Table 5.2: Oxygen Orifice Flow Test Results

Test	Pressure [psi]	Start Weight [g]	Final Weight [g]	Time [s]	Flow Rate [kg/s]	C_d (nominal)
1	58	100	300	20.65	0.009	0.422
2	29	100	190	17.50	0.005	0.317
3	58	62	1149	7.89	0.138	0.306
4	38	63	770	5.41	0.131	0.359
5	42	63	783	4.50	0.160	0.418
6	48	63	945	5.80	0.152	0.372
7	48	63	945	5.80	0.152	0.371
8	50	71	1293	8.80	0.139	0.332
9	24	62	675	7.20	0.085	0.294
10	58	60	900	7.00	0.120	0.267
11	140	63	1460	6.00	0.233	0.333
12	49	63	1743	12.10	0.139	0.336
13	48	65	1816	14.50	0.121	0.295
14	48	62	1650	14.98	0.106	0.259
15	172	1650	2121	2.50	0.188	0.243

5.1.2 Injector Atmospheric Hot fire Test



Figure 5.1: Starting of the flame in atmosphere



Figure 5.2: Ending of the flame in atmosphere

The hot fire testing of the injector was performed outside of the combustion chamber in atmosphere to verify the injector spray pattern and proper burning of the ethanol. At the start of the combustion it was visualized that the flame liftoff distance was high and the atmospheric back pressure was not able to hold the flame because of which the Flame front was not able to move upstream and it kept moving downstream. Due to

flame front moving downstream the flow was not able to sustain in atmosphere. This phenomena occurred as the injector was designed to work at high pressure inside the combustion chamber where the walls of the combustion chamber provide extra heat to the flow which helps in sustaining the flow.

The operating parameter so obtained for the injector atmosphere hot fire test is shown in Table 5.3. Also from the observation of the plume color it is shown the characteristics of the incomplete combustion as shown in Figure 5.1. This is due to the fact that for the design ΔP of an injector that is tested on atmosphere correlated with the reduction of density at the lower feed pressure which causes significantly reduction in OF ratio.

Table 5.3: Injector Flow at atmospheric condition

Oxygen Flow in Choked Orifice		
Pressure Ratio	0.22	-
Injection Pressure	459880.49	Pa
Gas Constant	259.8	J/kg.K
Injection Temperature	293	K
Heat Ratio (k)	1.4	-
Mass Flow Rate	0.009	kg/s
OF ratio	0.45	

5.1.3 Hot Fire Test

The subsequent three hot fire tests were done in the same engine with a similar feed system configuration. However, as the current campaign measured only for wall temperature and thrust, the process for full system validation remains limited at this stage. As described Ethanol was used in these tests in order to understand the engine's behaviour, its stability and confirm test operations consistency. The calculated specific impulse (Isp) following tests provides a baseline for the engines's efficiency, but it is important to note that this value is currently calculated from the designed mass flow rate rather than experimental measurements. Since, Isp is a ratio of the total impulse to the weight of the propellant consumed, any variation between the theoretical propellant mass flow rate and the actual combustion process can lead to deviation. Five multiple tests are presented below:

Test-1

The first test was done with the designed feed pressure and in the assembled chamber. The provided graph in Figure 5.3 illustrates the performance of an engine during a static fire test, specifically taking experimental Thrust (measured in Newtons) and Estimated Chamber Pressure (measured in psi) over time. The engine exhibits a steady burn profile, characterized by a rapid rise to peak performance followed by a steady plateau that lasts for the majority of the burn which confirm the constant feed pressure upstream to an injector. According to the data, the engine reached a maximum thrust of 99.61 N while maintaining an estimated chamber pressure around 160 psi. Since chamber pressure isn't measured directly within chamber the pressure profile is obtained from the relation of theoretical C^* and obtained Isp the estimated chamber pressure values may exhibit deviation from the actual operating conditions. The combustion process gives with a sharp tail down phase after approximately 2.4 seconds, resulting in a total Burn Time of 2.68 s.

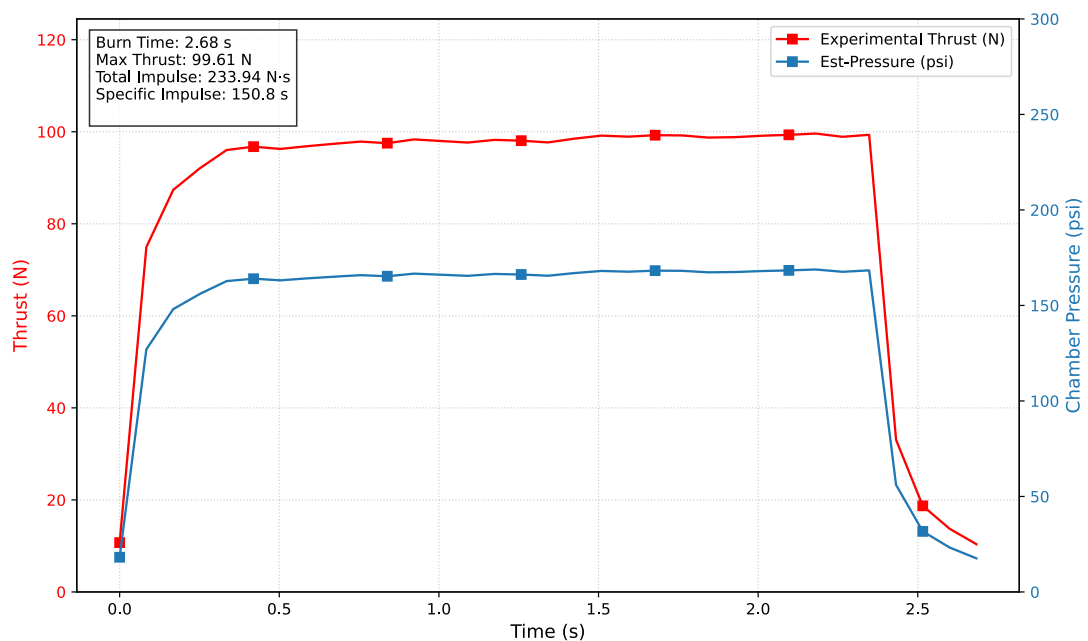


Figure 5.3: Performance Profile First Test

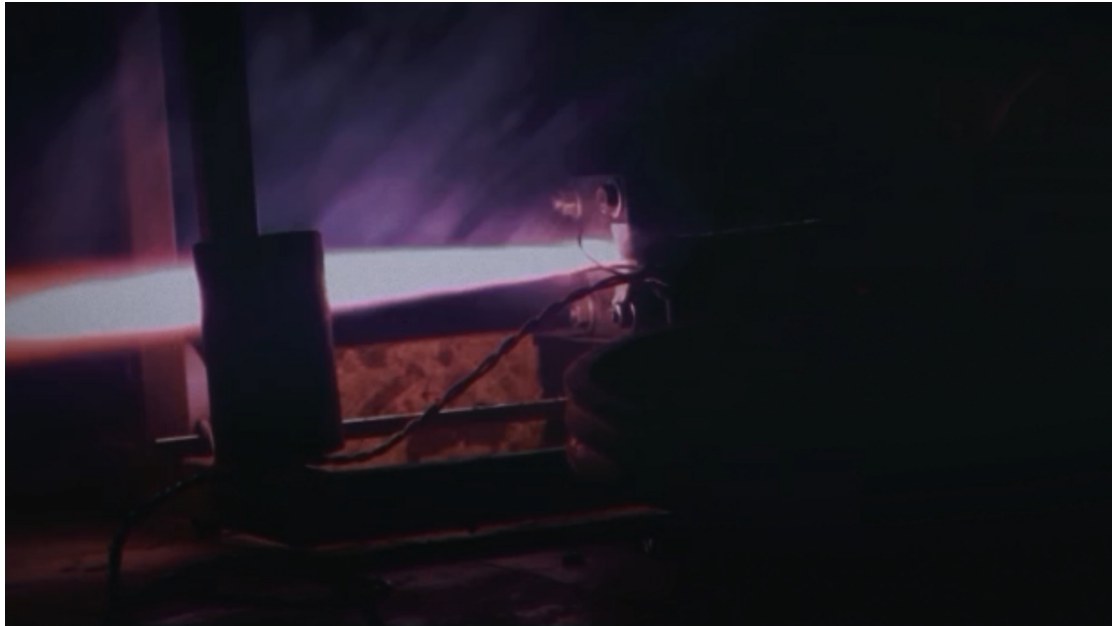


Figure 5.4: Exhaust of First Hot-Fire

Test-2

Similarly, the second test was performed using the same feed pressure and propellant combination as the previous test. Figure 5.5 presents the engine performance during the static fire test showing the measured thrust profile along with the corresponding estimated chamber pressure profile. As observed in the previous test, the thrust profile indicates a steady and stable burn. During this test, the engine achieved a peak thrust of approximately 121.51N while maintaining an estimated chamber pressure of around 155psi.

As shown in the thrust profile, a slight oscillation is observed during the tail-down phase. This behavior is primarily attributed to the rapid termination of the fuel supply which results in transient flow instabilities and pressure fluctuations within the combustion chamber during shutdown.

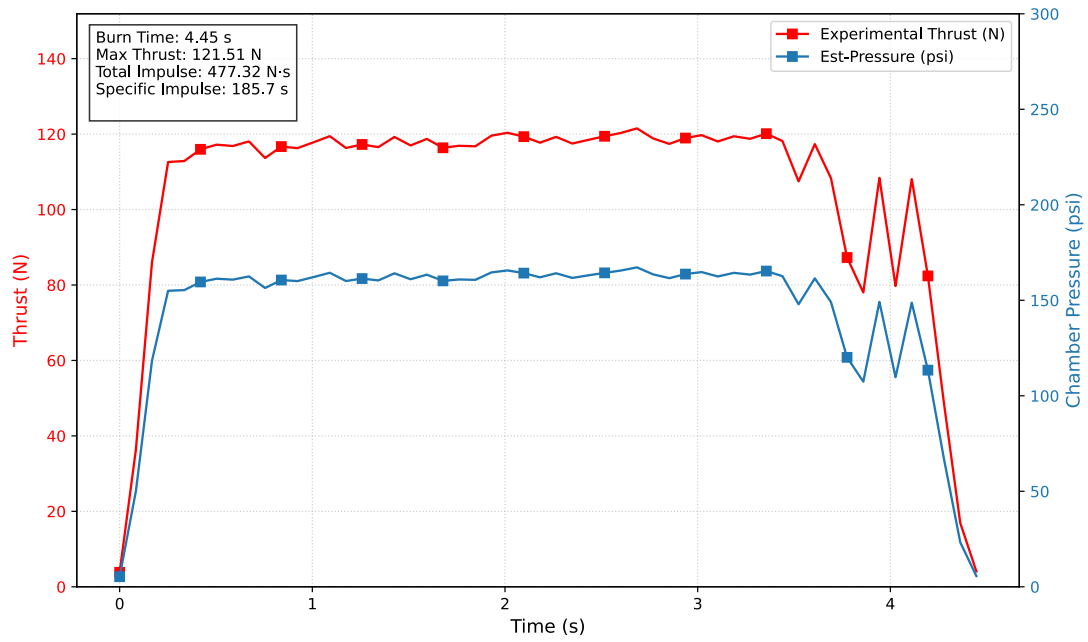


Figure 5.5: Performance Profile Second Test

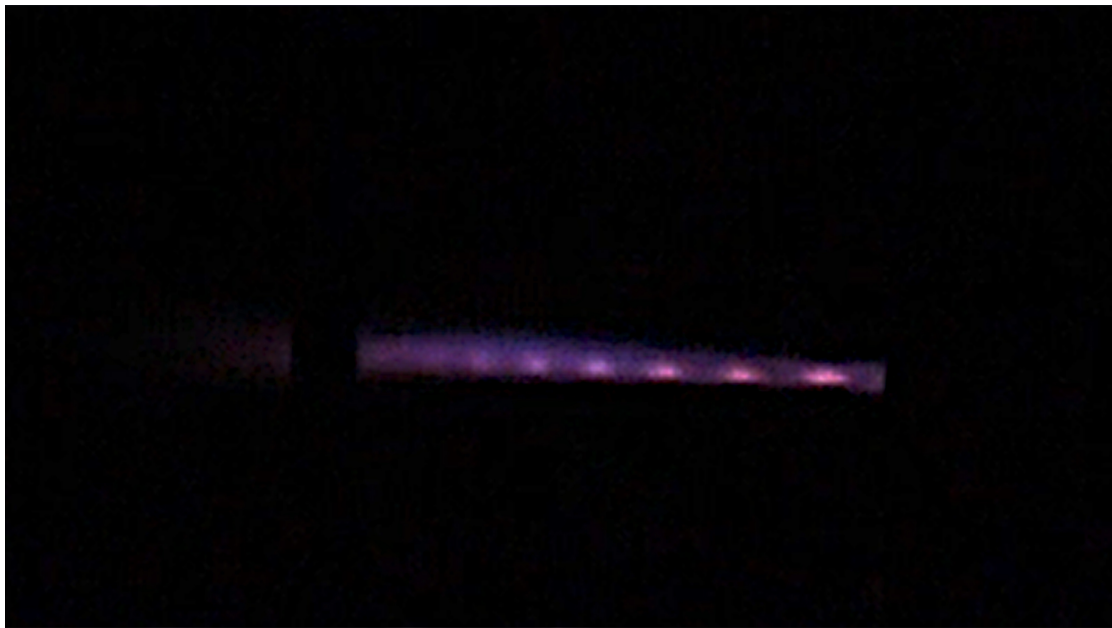


Figure 5.6: Exhaust of Second Hot-Fire

Test-3

Likewise, the third test was performed using the same feed pressure and propellant combination as the previous tests. Figure 5.8 illustrates the engine performance during the static fire test showing the measured thrust profile along with the corresponding estimated chamber pressure profile. As seen in the previous test, the engine exhibited

a steady and stable burn. However, the peak thrust achieved during this test was approximately 111N, while the estimated chamber pressure remained around 150 psi.

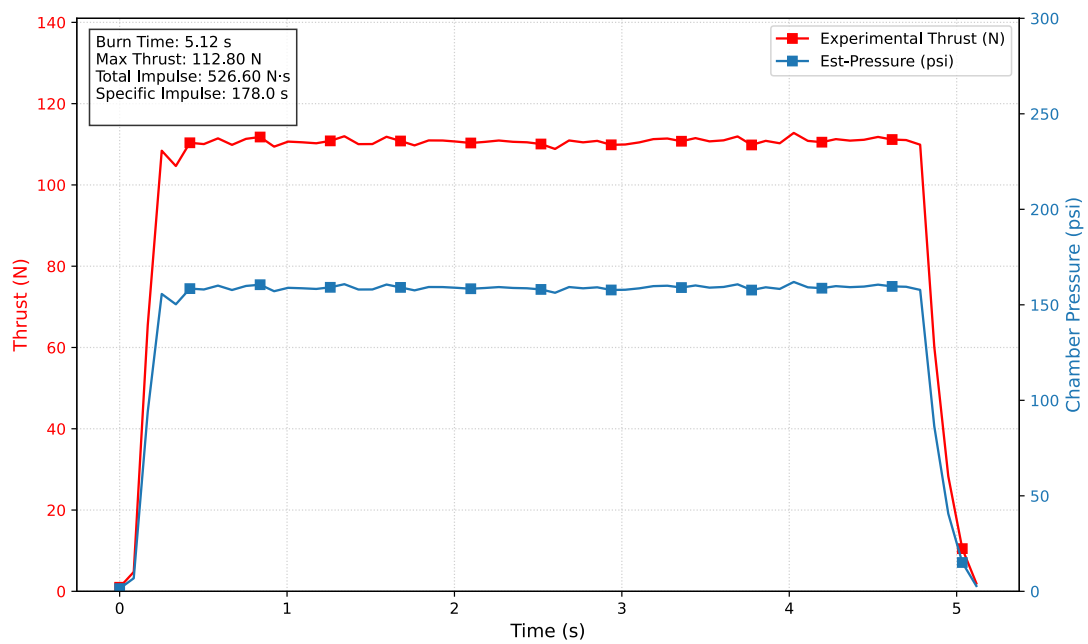


Figure 5.7: Performance Profile for Second Test

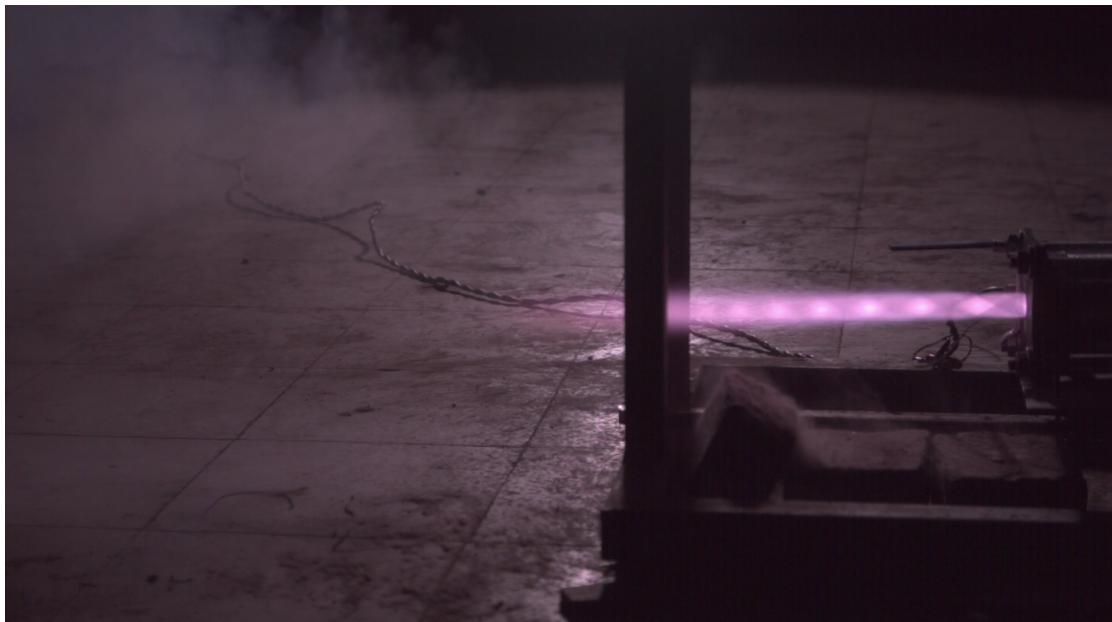


Figure 5.8: Exhaust of Third Hot-Fire

As observed in the exhaust plume during hot firing, distinct Mach diamonds were visible. These shock structures are formed due to the expansion and compression of the exhaust gases as they pass through the converging–diverging nozzle. The nozzle accelerates the flow to supersonic velocities and due to slight mismatch between the nozzle exit pressure and ambient pressure, the exhaust undergoes alternating expansion and compression resulting in the observed Mach diamond pattern.

Test-4

Likewise, the fourth test was conducted using the same feed pressure and propellant combination as the previous tests. Figure 5.10 illustrates the engine performance during the static fire test, showing the measured thrust profile along with the corresponding chamber pressure variation over time. As observed the engine demonstrated a stable and sustained burn throughout the test duration. Unlike the other previous test this test include the installation pressure transducer to measure the chamber pressure. This help to obtain the delivered c^* efficiency of the engine. The delivered c^* is obtained using the Eq 5.1.

$$c^* = \frac{A_t \Delta t}{m_p} \sum_{i=1}^n P_i \quad (5.1)$$

After obtaining the delivered c^* it is compared with the ideal value obtained from the chemical equilibrium analysis tool using Eq. 5.2 which helps as a benchmark of propellant combustion efficiency.

$$\eta_c = \frac{c_{\text{delivered}}^*}{c_{\text{ideal}}^*} \quad (5.2)$$

Likewise with the help of the pressure data the delivered thrust coefficient is also obtained using Eq 3.11. This factor quantifies the degree to which a engine's thrust force is amplified as a result of gas expansion through the nozzle divergent cone.

The thrust rapidly increased during ignition and reached a steady state value of approximately 90 N while the chamber pressure stabilized around 148 psi. Minor fluctuations were observed in both thrust and pressure signals, which can be attributed to mea-

surement noise and slight flow instabilities. Towards the end of the burn, both thrust and chamber pressure exhibited a sharp decline indicating propellant depletion and shutdown.

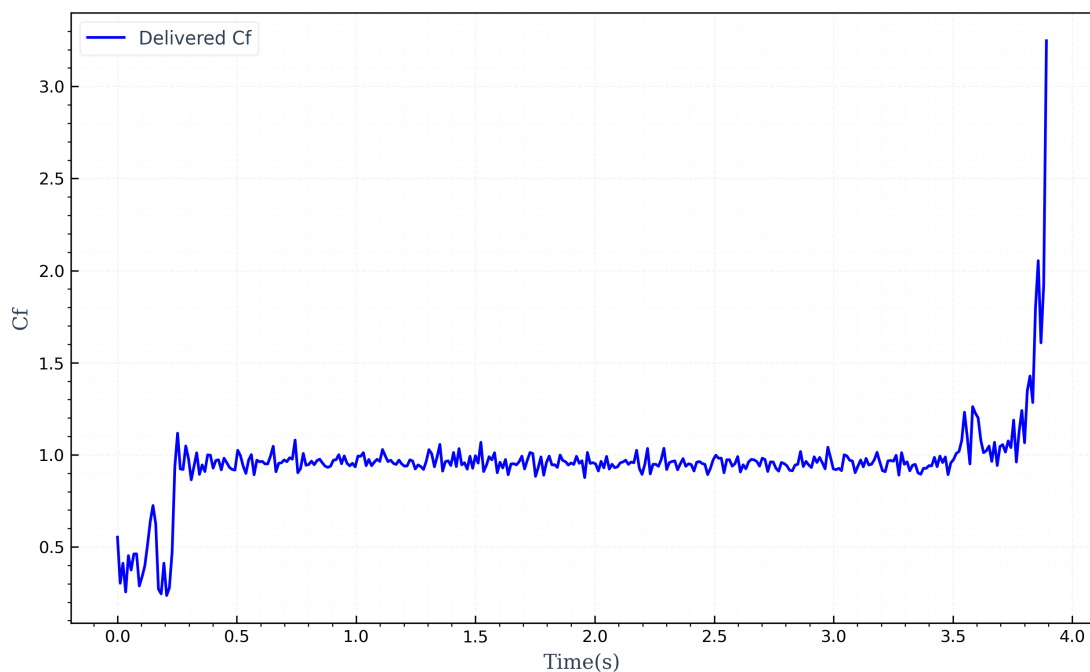


Figure 5.9: Delivered Thrust Coefficient

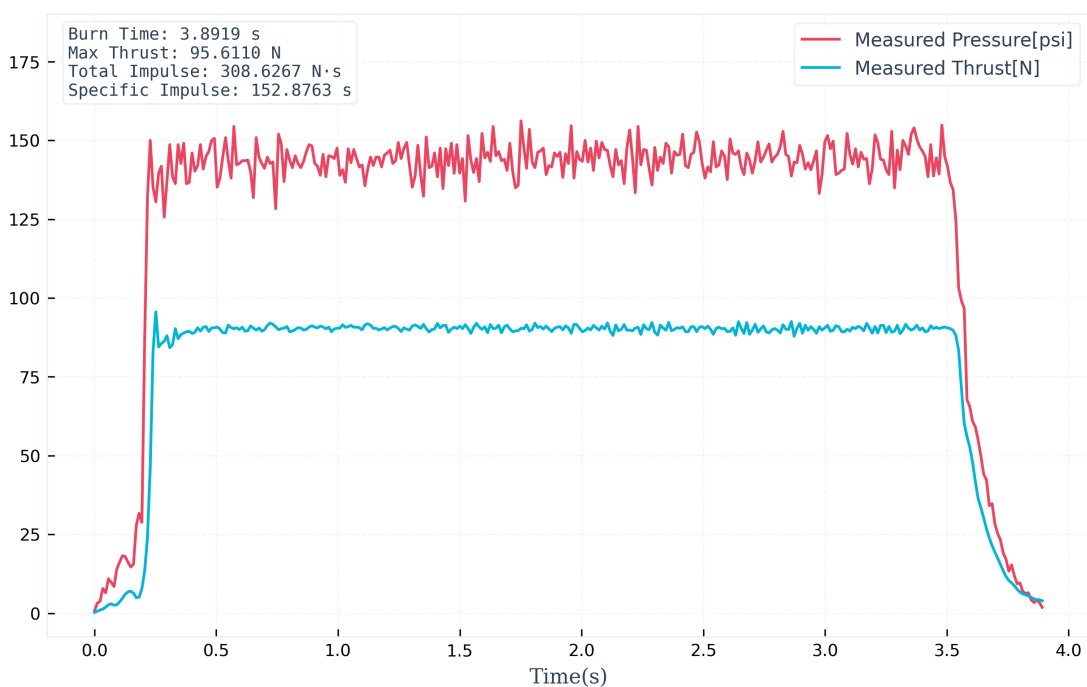


Figure 5.10: Plotted rocket engine performance for Test 4. Chamber pressure and thrust are plotted as a function of time. Average steady- state chamber pressure was 146 psi.

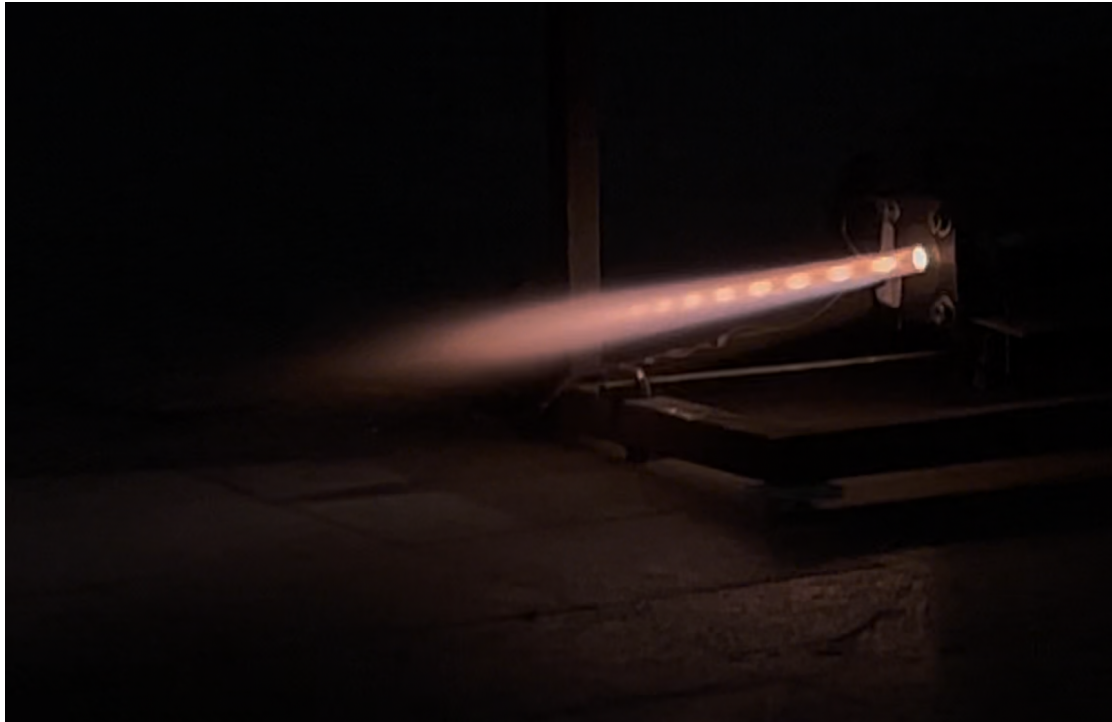


Figure 5.11: Exhaust of 4th hot-fire

Figure 5.9 presents the variation of the delivered thrust coefficient (C_f) with time. The coefficient remains relatively constant at approximately 0.95–1.0 during the steady-state phase indicating consistent nozzle performance and efficient expansion. However near the end of the burn, a noticeable spike in C_f is observed which is likely due to transient flow behavior and pressure drop during shutdown. The experimental engine performance and efficiency is summarized in Table 5.5.

Table 5.4: Experimental engine performance results summary for Test-4

Parameter	Value	Unit
Delivered Characteristic Velocity ($c_{delivered}^*$)	1573.54	m/s
Combustion Efficiency (η_c)	0.92	-
Specific Impulse (I_{sp})	152.8	s
Delivered Steady State Thrust Coefficient (C_F)	0.96	-

Test-5

Similarly, the fifth test was carried out using the same feed pressure and propellant combination as in the previous tests. Figure 5.13 illustrates the engine performance

during the static fire showing the measured thrust profile along with the corresponding chamber pressure variation over time. The engine exhibited a rapid rise in both thrust and chamber pressure during ignition followed by a relatively stable steady-state operation.

The thrust reached a peak value of 111.12 N and stabilized around 93–95 N during the main burn phase. The chamber pressure initially peaked near 125–130 psi and then settled to an average value of approximately 110–115 psi. Minor oscillations are observed in both signals which are likely due to measurement noise and small-scale flow instabilities within the feed system and combustion chamber.

Towards the end of the burn a sharp drop in both thrust and chamber pressure is observed indicating propellant depletion and engine shutdown. The burn duration was approximately 3.56 seconds with a total impulse of about 317 N·s, reflecting consistent engine performance.

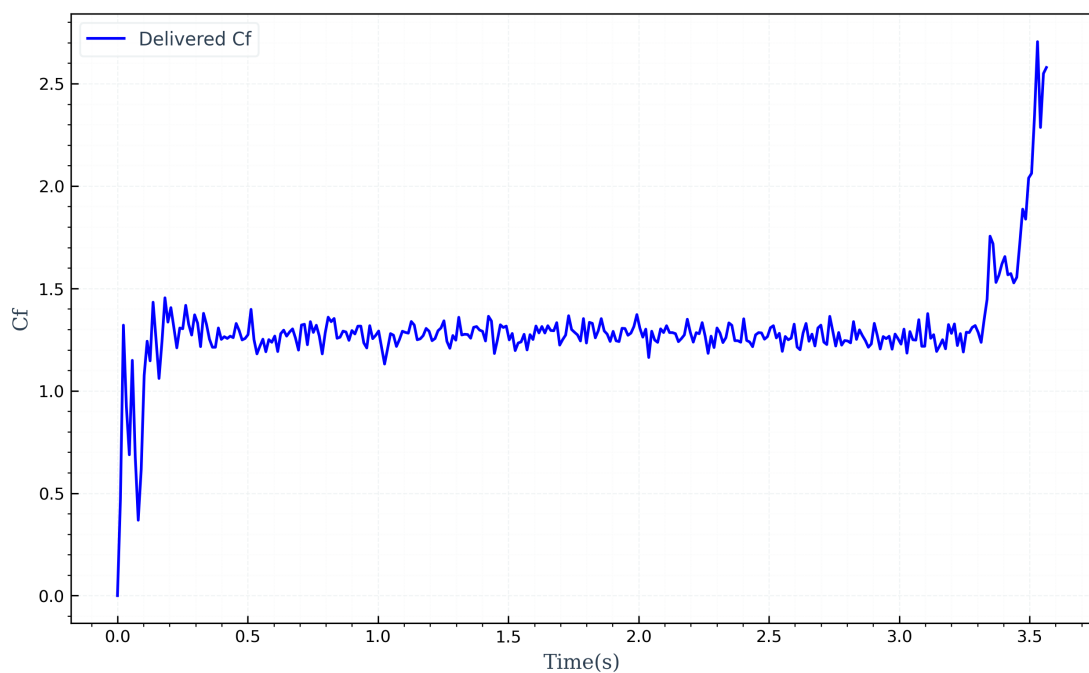


Figure 5.12: Delivered Thrust Coefficient

Figure [5.14](#) shows the variation of the delivered thrust coefficient (C_f) over time. The coefficient stabilizes around 1.25–1.30 during the steady-state phase indicating efficient nozzle performance and improved expansion characteristics compared to previous tests. However, a significant spike in C_f is observed near the shutdown phase, which can be

attributed to transient flow conditions and rapid pressure decay.

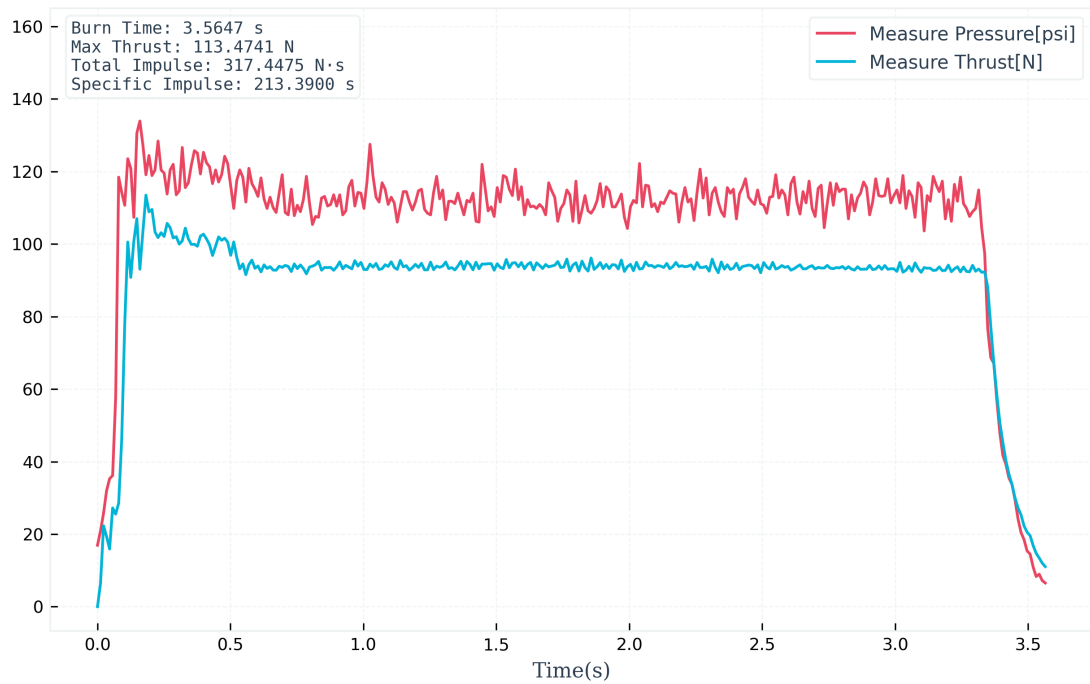


Figure 5.13: Plotted rocket engine performance for Test 5. Chamber pressure and thrust are plotted as a function of time. Average steady-state chamber pressure was 115 psi.

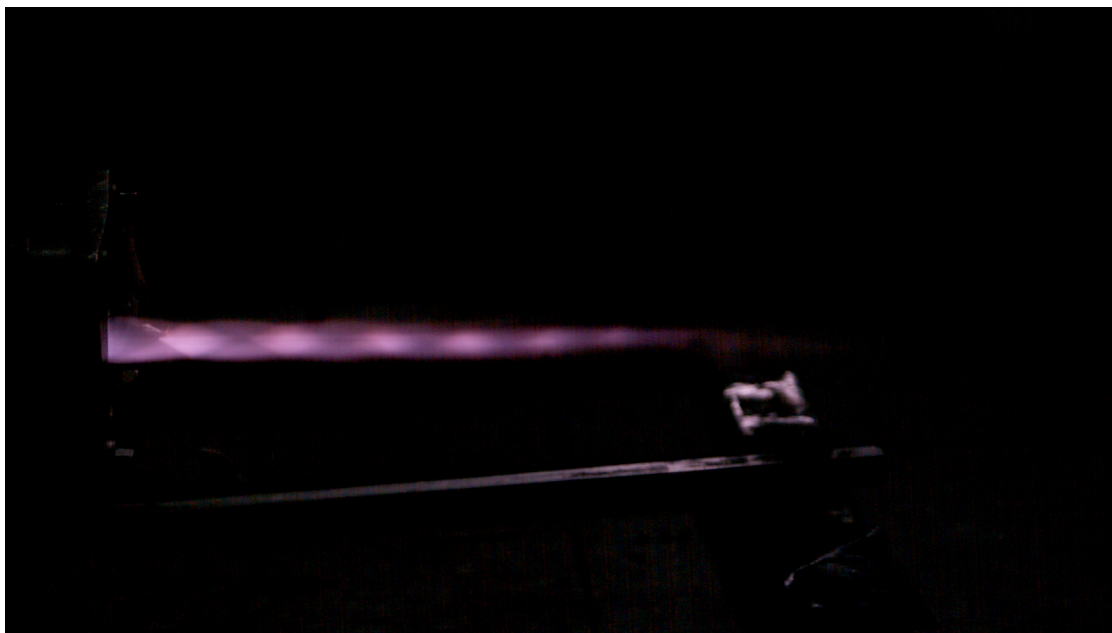


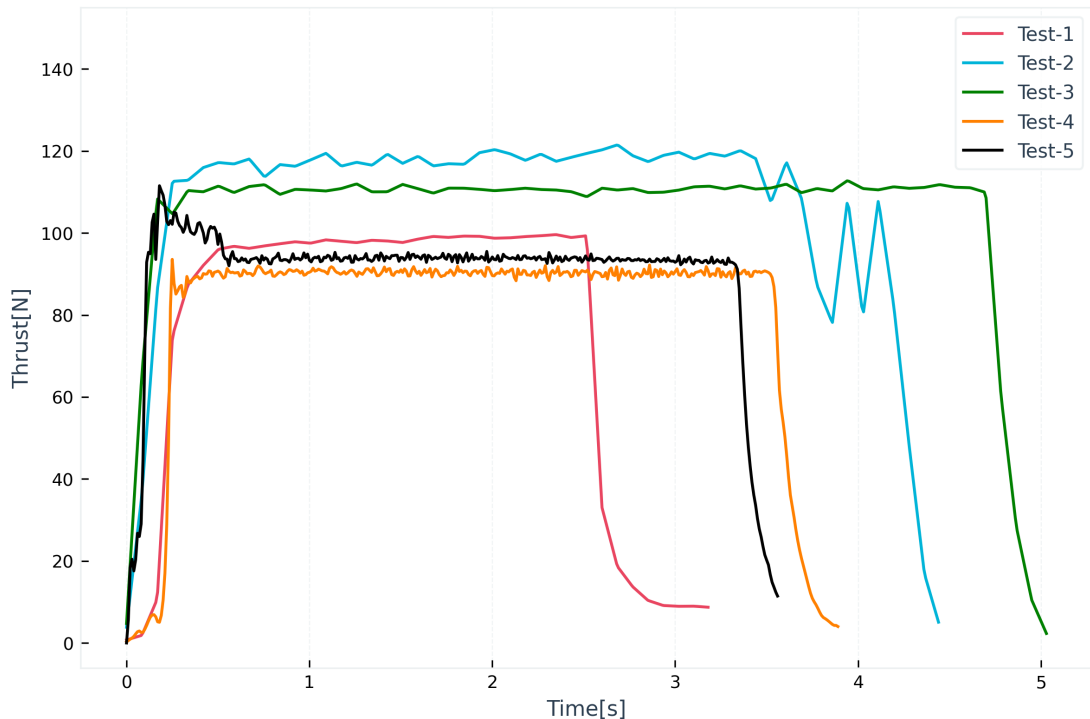
Figure 5.14: Exhaust of 5th hot-fire

Table 5.5: Experimental engine performance results summary for Test-4

Parameter	Value	Unit
Delivered Characteristic Velocity ($c_{delivered}^*$)	1601.13	m/s
Combustion Efficiency (η_c)	0.94	-
Specific Impulse (I_{sp})	213.3	s
Delivered Steady State Thrust Coefficient (C_F)	1.24	-

Collective Test Analysis

The predicted thrust of approximately 130 N was higher than the experimentally measured thrust which peaked around 121 N. This difference is primarily due to non ideal real world effects such as incomplete combustion, injectors performance and nozzle inefficiencies. The model assumes ideal flow conditions and complete combustion whereas the experimental system experiences unavoidable losses that reduce effective chamber pressure and exhaust velocity.

**Figure 5.15:** Designed Thrust vs Thrust Variation in Different Test

Furthermore thrust measurements from five test runs showed variation (100 N, 121

N, 111 N, 93N and 111N). These variations are attributed to transient pressurization behavior, injector flow inconsistencies, thermal conditioning of the engine and minor mixture ratio deviations between tests.

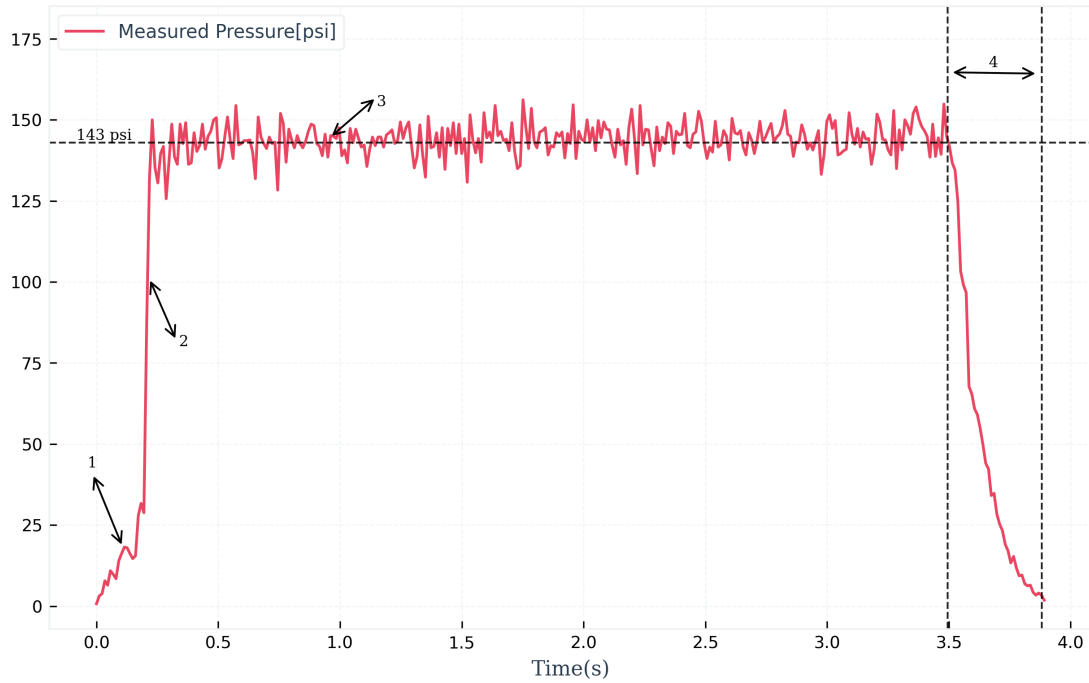


Figure 5.16: Test 4/5 rocket engine operation with plotted chamber pressure. Labeled events and regions include: Igniter initiation (1); Propellant supply valves open, starting propellant flow to engine (2); Steady state region of operation (3); Engine shutdown period (4).

Table 5.6: Hot fire test data analysis

Parameter	Test-1	Test-2	Test-3	Test-4	Test-5
Burn Time (s)	2.68	4.45	5.12	3.89	3.56
Peak Thrust (N)	99.61	121.51	111	95.61	113.47
Steady State Thrust (N)	96	115	110	90	95
Total Impulse (N.s)	233.94	477.32	526.60	308.63	317.45
Specific Impulse I_{sp} (s)	150.8	185.7	178.0	152.8	213.4
Pressure Transducer	—	—	—	Installed	Installed
Chamber Pressure (Psi)	—	—	—	146	115
Characteristic Velocity C^* (m/s)	—	—	—	1573.54	1601.13
Combustion Efficiency η_c	—	—	—	0.92	0.94
Thrust Coefficient C_F	—	—	—	0.96	1.24

The test data presented in Table 5.6 gives the overall performance summary of the engine for five different test runs, including the performance parameter. The results indicates

a general trend of change in parameter from test to test for same set of operating condition. Further the chamber condition obtained from the instrumented test shows that the combustion characteristics are stable with better characteristics velocity and combustion efficiency throughout the last two test series.

5.1.4 Post Firing Observation



Figure 5.17: Nozzle erosion

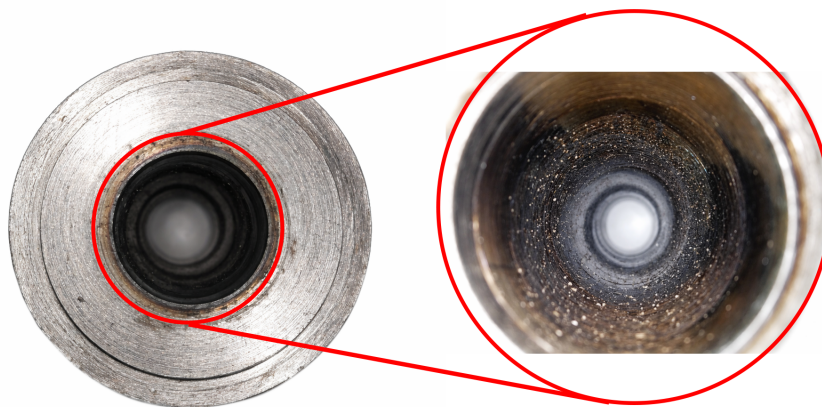


Figure 5.18: Chamber wall erosion

After the hot-fire test, the inner surfaces of the stainless steel nozzle and the combustion chamber exhibited uniform black discoloration. This discoloration is attributed to high-temperature oxidation, soot deposition and surface scaling caused by exposure to

combustion gases at elevated temperatures. No visible cracks, burn-through or localized melting were observed on the nozzle wall.

The observed discoloration indicates that the nozzle material experienced temperatures approaching its thermal limits during the firing duration, which is expected for an uncooled metallic nozzle operating under high chamber temperature conditions.

Post-test inspection also revealed partial thread wear and minor damage on some adapters and threaded connectors. This damage is mainly due to mechanical loading due to thrust forces and vibration during operation and possible galling of brass adapter's threads under high load.

5.2 Limitations

- i. Precision in machining of the injector and its atomization performance.
- ii. Proper pressurization of buffer tanks.
- iii. Implementation of integrated cooling system.
- iv. Constant mass flow rate of fuel and oxidizer.
- v. Proper sealing of the overall system.
- vi. Availability of high pressure reliable plumbing system in nepal.
- vii. Expensive equipments leading to financial burden.

5.3 Problem Faced

During the development of the projects following problems were faced:

- i. The unavailability of the good quality 1mm drill bit along with its collet, led to breaking of various 1mm drill bits while making the injector, eventually forcing us to redesign the injector multiple times.
- ii. Proper reliable data sheet of the plumbing equipment's is not available for components found in nepal, these components themselves are difficult to find.

- iii. Proper sealing of the ethanol pressurizing chamber was difficult as required type of O-Ring was not available.
- iv. As for ignition solid propellant is being used, so if proper care is not taken while opening the valve the igniter would be thrown away from the combustion chamber.
- v. Due to low sampling rate of pressure transducer combustion instabilities that may have been present are not measured, also low sampling rate of temperature sensor(MAX31856) caused the comparison of slopes difficult.
- vi. For the initial tests, no good setup of camera capable of capturing mach diamonds was present.

5.4 Budget Analysis

Table 5.7: Budget analysis

S.N	Component	Quantity	Total Price (Rs)	Used for
1.	Oxygen Tank	1	12,700	Tank system
2.	Nitrogen Tank	1	12,700	
3.	Industrial Regulator	2	12,400	
4.	Key	1	500	
5.	Gas Fillings	—	5,000	
6.	Buffer Tank	—	2,000	
7.	Ethanol	—	3,000	
8.	Ball Valve	6	4,000	Feed system
9.	Pipe and Fittings	—	20,000	
10.	Pressure Gauge	2	2,000	
11.	Nozzle Material	—	3,000	Thrust chamber
12.	SS-Casing	—	2,000	
13.	Machining Tools	—	1,500	
14.	Flange System	—	2,000	
15.	Injector Material	—	1,000	
16.	Thermocouple	3	6,000	Instrumentation
17.	Pressure Transducers	2	7,000	
18.	Silicone oil	1	1,500	
19.	Coiled copper tube	1	2,000	
20.	Micro-controllers	—	8,000	
21.	Servo motors	2	8,000	
22.	Miscellaneous	—	10,000	
TOTAL			1,26,300	

5.5 Safety and Precautions

Throughout the testing phase, special emphasis was placed on safety to minimize the risk of accidents and ensure controlled operation. Fire extinguishers were kept readily accessible and all personnel involved in the test wore appropriate personal protective equipment, including safety goggles. In addition, standard safety procedures and pre-test checklists were followed to ensure preparedness in the event of an unexpected incident.



Figure 5.19: Safety equipments

5.6 Gantt chart

The following Gantt chart shows our work schedule:

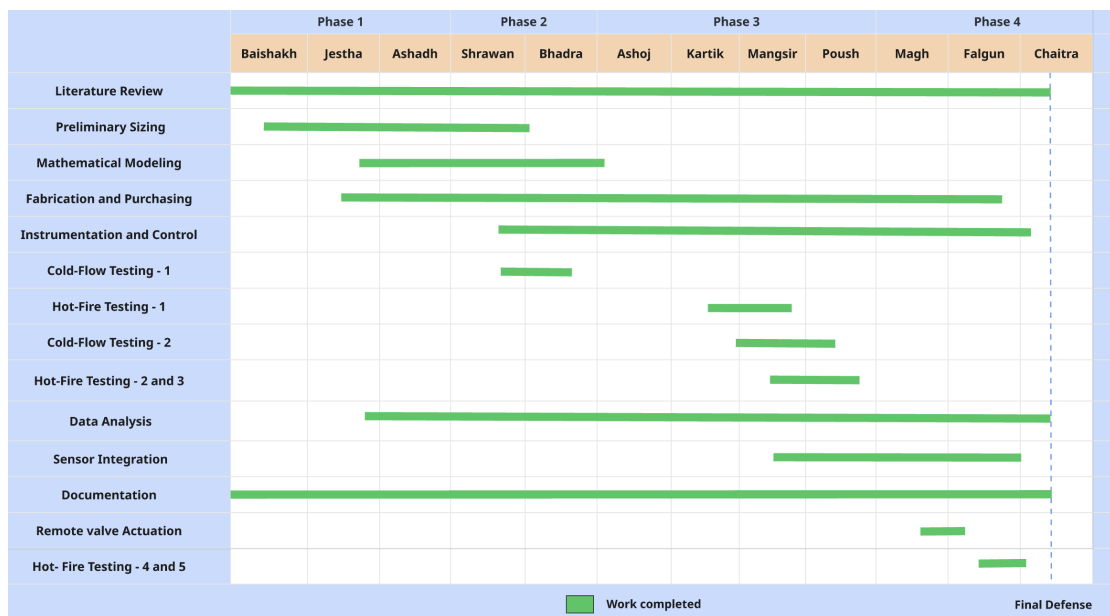


Figure 5.20: Gantt chart

CHAPTER 6: CONCLUSION

6.1 Conclusion

The static test results showed that the system was designed and fabricated properly. The designed combustion pressure was 150 psi. This combustion pressure when measured using a pressure transducer came to an average of 143 psi in fourth hot fire test. The engine was designed to produce the thrust of 130 newtons, however the static test produced a slight decrease in thrust with 121 newtons as maximum thrust in second static test.

The delivered characteristic velocity($c^*_{delivered}$) came to an value of 1601.13 m/s with combustion efficiency (η_c) 0.94 which indicates that the injector is operating at a near optimal mixture ratio with the propellants achieving effective atomization and combustion within the chamber. The specific impulse (ISP) of 210 s with Delivered Steady State Thrust Coefficient (C_f) of 1.24 further confirms that the overall thermodynamics and nozzle performance of the engine is in very close to the theoretical predictions.

In conclusion, design and testing of the engine was successful with the experimental performance parameters falling within acceptable margins of the predicted values, as exact agreement is not expected due to real flow effect. These results will be critical for future enhancement of the liquid rocket propulsion.

6.2 Scope for future Enhancement

This initial system design will be the basis for future improvements, following enhancements could be performed:

- i. Design for higher chamber pressure leading to higher thrust.
- ii. Cooling implemented combustion chamber resulting in long burn times.
- iii. Changing of injector pattern for better atomization.
- iv. Use of cryogenic propellant to make flight ready vehicles engine.

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