



TRIBHUVAN UNIVERSITY
INSTITUTE OF ENGINEERING
PULCHOWK CAMPUS

B-04-BAS-2019/24
BUILD AND TEST A SMALL SCALE LIQUID ROCKET ENGINE

By:

Bijay Paudel (076BAS008)
Dhaneshwar Sah (076BAS010)
Dipak Das (076BAS011)
Pritam Sah Teli (076BAS030)

A PROJECT REPORT SUBMITTED TO THE DEPARTMENT OF MECHANICAL
AND AEROSPACE ENGINEERING IN PARTIAL FULFILLMENT OF THE
REQUIREMENT FOR THE BACHELOR'S DEGREE IN MECHANICAL AND
AEROSPACE ENGINEERING

DEPARTMENT OF MECHANICAL AND AEROSPACE ENGINEERING
LALITPUR, NEPAL

MARCH, 2024

COPYRIGHT

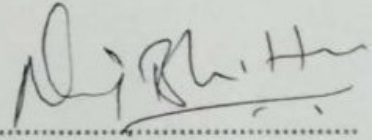
The authors have agreed that the library, Department of Mechanical and Aerospace Engineering, Pulchowk Campus, Institute of Engineering may make this report freely available for inspection. Moreover, the authors have agreed that permission for extensive copying of this project report for scholarly purpose may be granted by the professor(s) who supervised the project work recorded herein or, in their absence, by the Head of the Department wherein the project report was done. It is understood that the recognition will be given to the authors of this report and to the Department of Mechanical and Aerospace Engineering, Pulchowk Campus, Institute of Engineering in any use of the material of this project report. Copying or publication or the other use of this report for financial gain without approval of the Department of Mechanical and Aerospace Engineering, Pulchowk Campus and author's written permission is prohibited.

Request for permission to copy or to make any other use of the material in this report in whole or in part should be addressed to:

Head of Department
Department of Mechanical and Aerospace Engineering
Pulchowk Campus, Institute of Engineering
Pulchowk, Lalitpur
Nepal

TRIBHUVAN UNIVERSITY
INSTITUTE OF ENGINEERING,
PULCHOWK CAMPUS
DEPARTMENT OF MECHANICAL AND AEROSPACE ENGINEERING

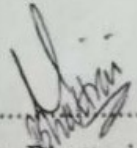
The undersigned certify that they have read, and recommended to the Institute of Engineering for acceptance, a project report entitled "BUILD AND TEST A SMALL SCALE LIQUID ROCKET ENGINE" submitted by Bijay Paudel, Dhaneshwar Sah, Dipak Das, Pritam Sah Teli in partial fulfillment of the requirements for the Degree of Bachelor in Aerospace Engineering.



Supervisor Asst. Prof. Nawaraj Bhattarai
Department of Mechanical and Aerospace
Engineering



External Examiner, Er. Akin Chettri
Mechanical Engineer



Asst. Prof. Sudip Bhattarai (Ph.D.)
Head of Department
Department of Mechanical and Aerospace Engineering

Date : 12th March 2024

ABSTRACT

A liquid rocket engine is a type of propulsion system that uses liquid propellants, typically a fuel and an oxidizer, stored separately in tanks and then mixed and burned in a combustion chamber to produce thrust. Liquid rocket engines are widely used in various space exploration missions, satellite launches, and military applications due to their high performance and controllability. In this paper the components of a liquid rocket engine i.e nozzle, combustion chamber and fuel injector were designed and tested. Two fuels LPG and kerosene were combusted with gaseous oxygen to generate thrust. The components were simulated in ANSYS to test their performance. Pressure fed system was adopted to pressurize kerosene and LPG didn't needed to be pressurized as it was already stored under pressure. Multiple test results were recorded and exhaust plume was analyzed. The thrust of magnitude 12N and 30N was measured in two tests and in other tests only exhaust plume was analyzed. Both the tests had to stopped midway due to chamber and injector failure due to high temperature produced in chamber. The yellow flame was observed in the exhaust plume which is caused by the glow of hot unburnt carbon particles produced due to incomplete combustion. So, further works on materials, manufacturing process and cooling is required.

Keywords: Liquid Rocket Engine, Nozzle , Combustion Chamber ,Fuel Injector , Sizing ,Simulation

ACKNOWLEDGEMENT

First and foremost, we would like to express our sincere gratitude towards Nawaraj Bhattarai (Ph.D.) , our project supervisors for his constant guidance, inspiration and precious encouragement. Without his invaluable supervision and suggestions, it would not have been a journey as smooth as it has been for us.

We would like to express our sincere gratitude towards Sudip Bhattarai (Ph.D.), Mr.Ashish Karki and Mr.Akin Chettri for providing their unwavering support and assistance throughout the project. Our sincere appreciation to Mr.Kamal Darlami (Head of Incubation, Innovation and Entrepreneurship Center (IIEC)) for letting us to use the resources for our project.

We would also like to thank Team SRB Pulchowk members Mr.Mridul Neupane , Mr. Nischal Shrestha and Mr.Prashant Giri for their support in the project.We would also like to thank Kabir Maharjan and Bal Mukunda Shrestha for their guidance on lathe operations .We would also like to thank all of our friends who have directly and indirectly helped us in doing this project.

We would like to thank the Department of Mechanical and Aerospace Engineering, Institute of Engineering, Pulchowk Campus for providing us opportunity of collaborative undertaking which has helped us to implement the knowledge gained over these years as major project for fourth year, and develop a major project of our own that has greatly enhanced our knowledge and provided us a new experience of teamwork.

Finally, we extend heartfelt gratitude to our families, whose unwavering support has served as a continuous source of inspiration throughout our endeavors. We welcome and value any feedback, be it constructive criticism or suggestions, with sincere appreciation and acknowledgment.

TABLE OF CONTENTS

TITLE PAGE	i
COPYRIGHT	ii
ABSTRACT	iv
ACKNOWLEDGEMENT	v
TABLE OF CONTENTS	vii
LIST OF FIGURES	ix
LIST OF TABLES	x
1 INTRODUCTION	1
1.1 Background	1
1.1.1 Liquid rocket engine	1
1.1.2 Rocket Exhaust plume types	2
1.2 Problem Statement	4
1.3 Objectives	4
1.3.1 Main Objectives	4
1.3.2 Specific Objectives	5
1.4 Scope of the project	5
1.5 System Requirements	5
1.5.1 Hardware Requirements	5
1.5.2 Software Requirements	5
2 LITERATURE REVIEW	6
3 METHODOLOGY	9
3.1 Sizing of the engine	10
3.2 CAD model of test stand	18
3.3 Manufacturing and components fittings	19
3.3.1 Nozzle	22
3.3.2 Injector	23
3.4 Safety procedures	26
4 RESULT AND DISCUSSION	27
4.1 Test and exhaust plume analysis	27

4.1.1	LPG and gaseous oxygen	27
4.1.2	Kerosene and gaseous oxygen	28
4.1.3	Alternate injector test	31
4.2	Simulation	33
4.2.1	Nozzle simulation	33
4.3	Limitations	36
4.4	Problems Faced	36
4.5	Budget Analysis	37
5	CONCLUSION AND FUTURE ENHANCEMENT	38
5.1	Conclusion	38
5.2	Scope for Future Enhancement	38
	REFERENCES	39
	APPENDICES	i

List of Figures

1.1	Parts of Liquid Rocket Engine[1]	1
1.2	Rocket Propellant and their exhaust color[2]	3
3.1	Work plan	9
3.2	Frequently used geometrical shapes for combustion chambers [3]	14
3.3	Experimental formula for chamber CSA estimation, A_{ch} [4]	14
3.4	CAD Design of proposed static test stand	18
3.5	Block diagram of LPG and GOX combustion	19
3.6	Test setup of the rocket engine	20
3.7	Thrust stand (Courtesy of SRB ,IOE Pulchowk)	20
3.8	Manufacturing of injector turning process	21
3.9	Injector Drilling Process	22
3.10	Manufactured Nozzle	22
3.11	Manufactured Nozzle	22
3.12	Injector Assembly)	24
3.13	Injector manifold	24
3.14	Oxygen Injector	25
3.15	Headset and Fire Extinguisher	26
4.1	Yellow flame(LPG + GOX)	27
4.2	Blue flame (LPG + GOX)	28
4.3	Thrust versus time plot	28
4.4	Combustion chamber failure	29

4.5	Combustion flame through the nozzle	29
4.6	Thrust versus time plot	30
4.7	Charring of the combustion chamber after the test	30
4.8	Chamber and oxygen injector failure	31
4.9	Combustion flame	31
4.10	Injector test with water	32
4.11	Pressure Contour	34
4.12	Temperature Contour	35
4.13	Velocity contour	35
A1	NASA CEA Interface	i
A2	NASA CEA interface	ii
A3	Injector manifold	iii
A4	Injector plate	iv
A5	Fuel Nozzle	v
A6	Injector Assembly	vi
A7	Assembly Cross-section	vii
A8	Nozzle	viii
A9	CAD model of Nozzle	ix

List of Tables

3.1	Calculated performance of some propellants [5]	11
3.2	General data on some storable liquid rocket propellants [5]	11
3.3	Characteristics length [6]	13
4.1	Solver Properties	33
4.2	Mesh Properties	33
4.3	Material Properties	33
4.4	Boundary Conditions of the Simulation	33
4.5	References Values	34
4.6	Solver Settings	34
4.7	Calculated Values at the Nozzle Exit	35
4.8	Budget Analysis	37

LIST OF ABBREVIATIONS

LRE	Liquid Rocket Engine
SRM	Solid Rocket Motor
ISRU	In-Situ Resource Utilization
LPG	Liquefied Petroleum Gas
ANSYS	Analysis System
CATIA	Computer Aided Three-Dimensional Interactive Application
MATLAB	Matrix Laboratory
NASA	National Aeronautics and Space Administration
CEA	Chemical Equilibrium Applications
CD	Convergent Divergent
CFD	Computational Fluid Dynamic
CAD	Computer Aided Design
CSA	Curved Surface Area
MS	Mild Steel
PPE	Personal Protective Equipment
GOX	Gaseous Oxygen
LOX	Liquid Oxygen
RP	Rocket Propellant
SRB	Solid Rocket Booster
Isp	Specific Impulse

LIST OF SYMBOLS

H_2O	<i>Water</i>
O_2	<i>Oxygen</i>
CO	<i>Carbon Monoxide</i>
CO_2	<i>Carbon Dioxide</i>
N_2O_4	<i>Nitrogen Tetroxide</i>
N_2	<i>Nitrogen gas</i>
N_2O	<i>Nitrous oxide</i>
O/F	<i>Oxygen to Fuel ratio</i>
C	<i>Degree Celsius</i>
F	<i>Degree Fahrenheit</i>
C_2H_5OH	<i>Ethanol</i>
Al	<i>Aluminium</i>
Ni	<i>Nickel</i>
Mn	<i>Manganese</i>

CHAPTER 1: INTRODUCTION

1.1. Background

1.1.1. Liquid rocket engine

Liquid rocket engines (LREs) represent a prevalent and versatile propulsion system widely employed in modern rocketry. Typically adopting a bi-propellant setup, LREs store fuel and oxidizer in separate tanks, a design that allows for precise control over the combustion process. The propellants can be injected into the combustion chamber either through high-pressure gases or via pumps, facilitating efficient and controlled thrust generation.

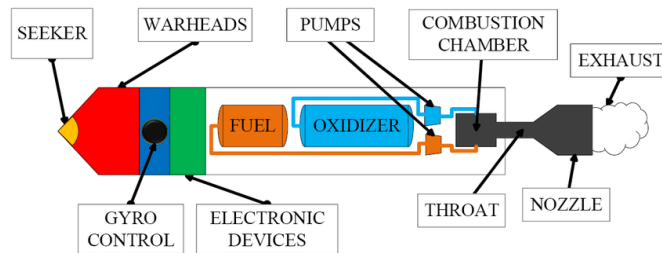


Figure 1.1: Parts of Liquid Rocket Engine[1]

Another notable advantage of LREs is their testability prior to use. Unlike SRMs, which rely heavily on stringent manufacturing processes to ensure reliability, LREs can undergo rigorous testing and validation, contributing to enhanced operational safety and performance predictability. These attributes make LREs a preferred choice for a wide range of space missions, from satellite launches to crewed space exploration endeavors.

In smaller engines, an alternative to using pumps for propellant delivery involves storing an inert gas under high pressure. This gas serves to force the propellants into the combustion chamber. While this method may result in a higher mass ratio, it offers increased reliability, making it a popular choice for satellite propulsion systems focused on orbit maintenance. This simplicity in design and operation contributes to the widespread use of such engines in space applications.

Liquid-propellant systems differ from solid systems by carrying their propellants in separate tanks external to the combustion chamber. Typically, these systems utilize liquid fuel and oxidizer, which are transferred into the engine via pumps. These pumps ele-

vate the pressure above the engine's operating threshold, ensuring efficient atomization and rapid mixing of the propellants within the combustion chamber. Liquid-propellant engines boast several advantages over solid systems, including higher attainable effective exhaust velocities (v_e), greater mass fractions (propellant mass divided by mass of inert components), and the ability to control operating levels during flight, including throttle-ability, stop-and-restart capability, and emergency shutdown.

1.1.2. Rocket Exhaust plume types

Liquid rocket engines (LREs) can use various types of fuels, each with its own unique properties and exhaust plume characteristics. Here's an explanation of different fuels commonly used in LREs and their exhaust plume properties:

Liquid Hydrogen

Liquid hydrogen is highly cryogenic and has a low density. It is stored and transported at very low temperatures (around 20 K) to maintain its liquid state. The combustion of liquid hydrogen with liquid oxygen produces water vapor (H_2O) as the primary exhaust product. The exhaust plume of liquid hydrogen engines is almost transparent and colorless. It produces a very hot and high-velocity plume, which may appear faintly bluish due to ionization. The plume contains mainly water vapor, which quickly dissipates in the atmosphere.

RP-1 (Refined Kerosene):

RP-1 is a refined form of kerosene, which is less cryogenic compared to liquid hydrogen. It is a hydrocarbon fuel with a higher density and higher boiling point. Combustion of RP-1 with liquid oxygen produces carbon dioxide (CO_2) and water vapor (H_2O) as the primary exhaust products. The exhaust plume of RP-1 engines is typically dark and sooty, due to the incomplete combustion of hydrocarbons. It may also contain carbon monoxide (CO) and unburned hydrocarbons. The plume appears bright at the nozzle exit due to the combustion luminosity.

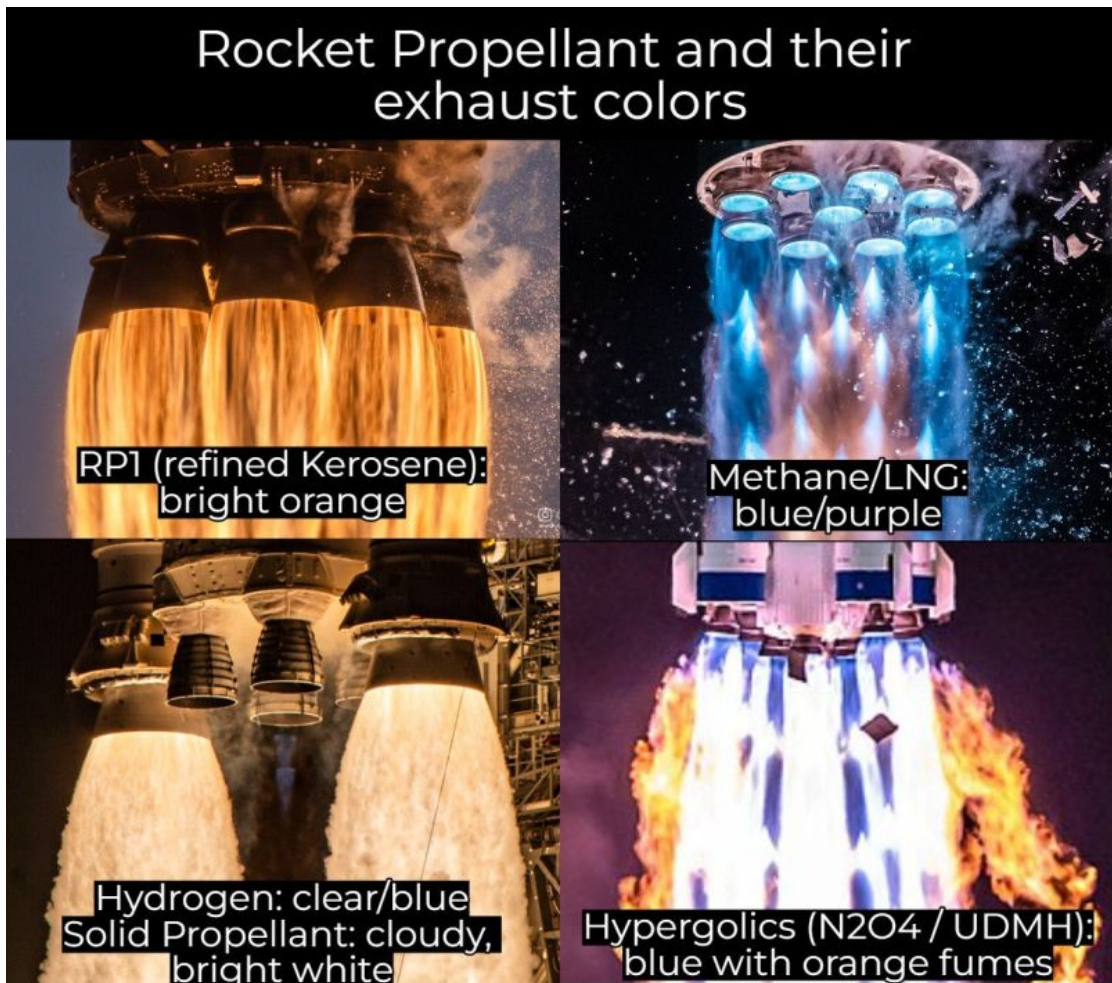


Figure 1.2: Rocket Propellant and their exhaust color[2]

Liquid Methane

Methane is another hydrocarbon fuel that is gaining popularity due to its cleaner combustion compared to RP-1 and its potential for in-situ resource utilization (ISRU) on other planets. Combustion of methane with liquid oxygen produces carbon dioxide (CO₂) and water vapor (H₂O) as the primary exhaust products. The exhaust plume of liquid methane engines is cleaner compared to RP-1 engines, with fewer soot particles. It may still contain some carbon monoxide (CO) and unburned methane. The plume appears less dark and sooty compared to RP-1 engines.

Hydrazine

Hydrazine is a hypergolic propellant, meaning it spontaneously ignites upon contact with an oxidizer. It is toxic and requires careful handling. Combustion of hydrazine with an oxidizer (often nitrogen tetroxide, N₂O₄) produces nitrogen gas (N₂) and water vapor (H₂O) as the primary exhaust products. The exhaust plume of hydrazine engines is

typically colorless or slightly yellowish. It produces a relatively low-temperature plume compared to hydrocarbon-based fuels.

Each of these fuels has its advantages and disadvantages, and the choice of fuel depends on factors such as specific impulse requirements, mission objectives, engine design constraints, and environmental considerations.

1.2. Problem Statement

In this age of space exploration, liquid rocket have become a effective and reliable source of propulsion. Liquid rocket engine development necessitates a large investment of time, money, and resources, including access to specialized materials and machinery, highly specialized knowledge, and advanced engineering and production skills. In spite of above challenges, it is important to further the growth of liquid propulsion system [7]. With Nepal's current level of economic and technological development, it is improbable that the nation now possesses the requisite infrastructure and resources to start working on the construction of liquid rocket engines. It is important to note that Nepal has expressed interest in space technology and exploration. Nepal achieved a major milestone in its space program in 2019 with the successful launch of its first satellite, NepaliSat-1, from the United States. Even though the satellite didn't use a liquid rockets. [7]

Lots of research papers are published on combustion simulation ,testing. Also combustion in a major area of study in turbo jet, ramjet, liquid rocket engine and solid rocket boosters. So, in our project we want to perform combustion testing using LPG and Kerosene as fuel and gaseous oxygen as oxidizer and study the combustion phenomenon at different mixture ratios.

1.3. Objectives

1.3.1. Main Objectives

The main objective of this project is to build and test the working of a small scaled liquid rocket engine.

1.3.2. Specific Objectives

1. To mathematically model , design and manufacture nozzle,combustion chamber and fuel injector.
2. To measure pressure and calculate mass flow rates at different points on the flow.
3. To measure the thrust produced by the engine.

1.4. Scope of the project

This project involve mathematical modelling,designing and fabrication of the different components of the liquid rocket engine using locally available materials.Hence insight on the working,designing and governing equations of different components can be gained.This project helps to understand the construction and working principle of the liquid rocket engine.

1.5. System Requirements

1.5.1. Hardware Requirements

A computer for design,simulation and modelling.Valves,gas regulator,pipes and hollow and solid shafts.

1.5.2. Software Requirements

ANSYS:Used for combustion simulation.

CATIA: CATIA was used to Create 3D models of stand,nozzle,injectors and systems,including mechanical parts and generative shape creation.Assemble components into a finished product,manage and manipulate the assembly.

MATLAB: MATLAB was used for writing algorithm for components design and mathematical calculation.

NASA CEA: The combustion simulation of kerosene and gaseous oxygen can be performed.

CHAPTER 2: LITERATURE REVIEW

In the late 19th century, the focus has shifted from current perspective for engines to the development of new concept, exploring the areas of propellant, Nozzle and combustion chamber designs and ignition methods. Hagemanne(1998), who discusses the development of new nozzle designs to increase performance and Law(2012), who presents ideas of next generation fuels for propulsion use, including ethanol are presenting work to benefit the space exploration community by advancing the current technology. Due to low cost and availability, high energy density, easy storage and handling, kerosene is also utilize as rocket propellant[8].

Gottmann et al (2015) empirically investigated ethanol as fuel in liquid rocket engine. Ethanol is green propellant like- N₂O, kerosene etc which is considered to be environment friendly. During the world war, Germans has done stimulation using ethanol/water and liquid oxygen as propellant in different concentration through computer program for estimation of chemical equilibrium compositions. The simulations show that when the concentration of ethanol and oxygen in the mixture is less than 1:3 the combustion is flawed. Increase in the oxidizer reduces the combustion efficiency, therefore estimated quantity of O/F ratio is 1:6. The simulations showed that the performance of the propellant decreases with increasing the watercontent in the fuel[9].

Jayaprakash, P., Dhinarakaran, D., and Das, D. (2022) published their work on the design and analysis of a CD nozzle . They performed CFD analysis of nozzle for static temperature , pressure and velocity analysis. The values of CFD and Theoretical methods were slightly different because the CFD software considers the factors like boundary layer effects, shock waves, velocity component and more. From the result it was clear that the one-dimensional nozzle analysis is enough to find out the nozzle performance therefore giving clarity for the selection of the nozzle[10].

Florián Andrés, Urrego P.J. Alejandro(2015) designed and manufactured a liquid(LOX and Gasoline) rocket motor with stratospheric apogee. After the bench test, the behavior and characteristics of the designed and constructed rocket motor were found within the expected ranges: thrust, burnt time and temperatures. The combustion chamber, tanks and other structural and hydraulic elements behaved in safely and properly way during the operating time, which supported its correct selection and design process[11].

Michal Málek(2021) worked on the design of small liquid propellants rocket engine. In the paper he used different CAD software to design a small liquid propellant rocket en-

engine and did CFD analysis of the nozzle. He designed a 3D-modelled and numerically verified a small experimental liquid fuel rocket engine with regeneratively cooled nozzle. According to performed numerical simulations it is safe to declare, that the engine is designed correctly and if built, should perform as desired[12].

Alejandro aljure Osorio(2007) published a paper on construction of a model liquid fueled rocket engine .He manufactured thrust chamber , nozzle , fuel injector ,cooling jacket, fuel tanks and support structure. Three variants of the combustion chamber's architecture have been studied. After the experimental research, the conclusion was that none of them fully reached the target; however, significant improvements have been achieved compared with the starting point[13]. In the exploration of rocket-engine transient regimes dating back to 1977, [12] focused on the implications of sudden changes in operational conditions on engine performance and stability. Key factors highlighted by the author included the rate of change in engine operating conditions, the mechanical and thermal response of engine components, and the dynamics of the combustion process. Numerical simulation emerged as a primary technique for studying transient regimes in rocket engines as discussed. Additionally, [12] delved into the assessment of experimental techniques for investigating rocket-engine transient regimes, which experimental setups offer valuable insights into the behavior of engine components during transient regimes, including the response of the fuel injection system, combustion chamber, and nozzle to changes in operational conditions. According to [11], simulating transient combustion in rocket engines requires integration of fluid dynamics, thermodynamics, and combustion science. Accurate modeling of combustion, heat transfer, and turbulence is crucial for predicting engine performance and stability under varying conditions. Although computational work plays a crucial role in the initial design stages of rocket engines, the majority of engine designs are ultimately informed by past successful experimental research. The design process can be quite diverse, leading to certain decisions being based on a combination of engineering intuition and the knowledge gained from previously successful designs[Young. O blood]. Various studies have explored kerosene-based rocket engine performance, highlighting optimal injector designs [5], improved combustion stability with swirling injectors [6], and kerosene's efficacy in hybrid rocket engines for high thrust and specific impulse [7]. The study [5], also shows that CFD modelling provides insights into the complex internal dynamics of rocket combustion chambers, which are challenging and costly to observe through laboratory experiments.

The careful selection of propellants and precise calculation of mixture ratios are pivotal factors in both the design and operation of liquid-propellant rocket engines. According to Huzel and Huang, a thorough understanding of the diverse range of propellants, en-

compassing hypergolic, storable, and cryogenic options, is essential. Their work offers an in-depth examination of these propellants, highlighting their distinct characteristics, performance capabilities, handling considerations, and compatibility with engine materials. Meanwhile, Mishra's exploration of rocket propulsion principles emphasizes the critical role of stoichiometry and mixture ratio calculations. By elucidating the chemical reactions involved in combustion, Mishra underscores the significance of attaining the precise stoichiometric ratio for achieving efficient combustion and optimal thrust generation.

In their seminal work, Huzel and Huang (1967) provide comprehensive insights into the modern engineering practises for designing liquid-propellant rocket engines, including detailed discussion on components. Krzycki's practical guidebook(2018) on design, building and testing liquid-fuel rocket for amateur offers invaluable hands-on-knowledge applicable to component's fabrication. By detailing, manufacturing process and assembly techniques, Krzycki bridges the gap between theoretical concept and practical implementation, facilitating the realization of components. Additionally, investigation conducted by Gill and Nurick(1967, NASA) have contributed significantly to the understanding of the injector selection and performance under varying operating condition. The practical guidebook (Krzycki's) provides valuable insights on the design small-liquid fuel rocket and corresponding basic systems.

CHAPTER 3: METHODOLOGY

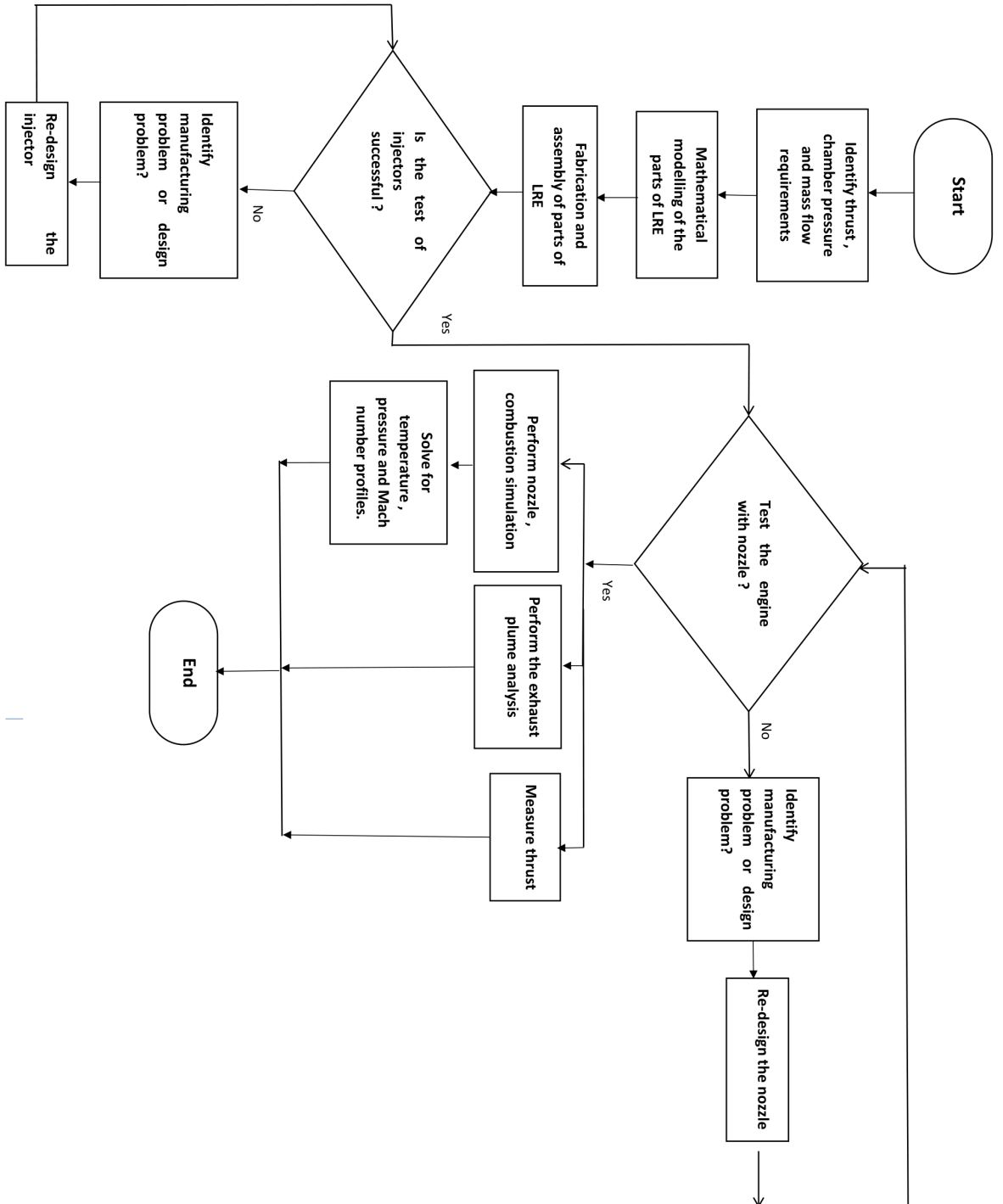


Figure 3.1: Work plan

1. First the requirements for which the engine is build is needed to be listed out. The requirements can be mass flow rate , chamber pressure and thrust depending upon the mission profile of the rocket.
2. Using those requirements the sizing of the engine is to be done where the dimensions of chamber,injector and nozzle are calculated.
3. After the design the components needs to be manufactured according to the dimensions.Some common machines like lathe,drill machine , grinder etc can be used.
4. Then the test of both injectors need to be carried out.Water can be used to test the performance of injectors and the atomization process should be observed.
5. If the test of injector is inaccurate then the injectors needs to re- designed or re-manufactured.
6. After the test of injector,then engine can be tested with nozzle installed.One can carry tests until s/he gets the desired result.If any of the components undergoes failure then the components needs to be re-designed or re-manufactured according to the need.
7. The test can be performed in a thrust stand or other data acquisition system can be installed in the system to measure required data.
8. After the test of engine the data measured can be processed and the exhaust plume can also be analyzed.

3.1. Sizing of the engine

1. Choose propellant combination

The very first step is the appropriate selection of propellants. In context with the available resources, availability, and economic criteria, Kerosene (RP-1) and LPG (Liquefied petroleum gas) are used as fuel along with gaseous oxygen as an oxidizer.

Stoichiometry of Propellant Combustion

In the chemical reaction, mass balance describing exactly how much oxidizer must be supplied for complete combustion of certain amounts of fuel is generally termed as stoichiometry . In simple, the ratio of oxidizer to fuel that is sufficient to burn fuel, leading to complete products of combustion. Taking Kerosene as fuel,

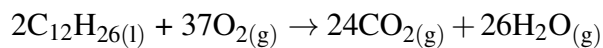
Table 3.1: Calculated performance of some propellants [5]

Propellant Combination (O/F)	Pressure(Mpa)	Mixture ratio(r)	Temp(°C)	Isp(sec)
Liquid Oxygen & gasoline	2.068	2.5	3020	242
Gaseous Oxygen & gasoline	2.068	2.5	3170	261
Gaseous Oxygen & gasoline	3.45	2.5	3240	279
Liquid Oxygen & JP-4(Jet Fuel)	3.45	2.2	3250	255
Liquid Oxygen & methyl alcohol	2.068	1.25	2860	238
Gaseous oxygen & methyl alcohol	2.068	1.2	2880	248
Liquid Oxygen & Hydrogen	3.45	3.5	2480	363
Red fuming nitric acid & JP-4	3.45	4.1	2840	238

Table 3.2: General data on some storable liquid rocket propellants [5]

Propellants	Gaseous(o2)	Kerosene	Ethanol	Lpg
Formula	O2	Mil-Spec-F25576b	C2H5OH	(xc3h8+yc4h10)
Mol. Wt.(gm/mol)	32	165-195	41.25	44.10 To 58.12
Freezing Pt.(Of)	-362	-47 To -64	-189	-216.9
Boiling Pt. (Of)	-297.33	342 To 507	172	-43.8 To 31.1
Density(gm/cc)	0.0014229	0.8 At 68 °f	0.81 At 60 °f	0.0018
Stability	Good	Auto Ignition 470°f	Good	Stable
Handling Hazard	High	Flammable	Flammable	Flammable
Storability	Good	Good	Good; 130°f	Good
Compatibility	Al, Steel	Al, Steel, Ni Alloy	Al, Steel, telfon	Al, Steel, Mn

its combustion with gaseous oxygen is represented by the following balanced chemical equation.



Clearly, every 2 moles of kerosene consumes 37 moles of gaseous oxygen producing 24 moles of carbon dioxide and 26 moles of water are produced. Thus, the ratio of kerosene to gaseous oxygen in the mixture is 2:37. It represents the stoichiometric ratio required for combustion under ideal conditions. In practical, it may differ due to additives, operating conditions.

2. Specify mission requirements

Depending on mission requirements two of the three main characteristics parameters which include combustion chamber pressure , thrust required and mass flow rate of the propellants can be chosen and choosing two will constraint the third.[14]

From mission requirement combustion chamber pressure and thrust required are known parameters . Now , using these two parameters other parameters for the engine can be calculated.

The chamber pressure is chosen to be 80 Psi and thrust required to be 125N.This known parameters are used in sizing of the engine.

3. NASA CEA to find optimum O/F ratio ,Specific impulse and Combustion chamber temperature

To perform chemical equilibrium and analysis NASA CEA can be used. Kerosene and gaseous oxygen reaction can be simulated at different O/F ranging from 1 to 3 at interval of 0.1 at 80 psi. Then design O/F ratio of 1.875, with chamber temperature of 3081.15 K and specific impulse of 201.5291 m/sec.[5]

4. Calculate required mass flow rate of fuel and oxidizers

We know ,

$$\text{Mass flow rate} = \frac{\text{Thrust Required}}{\text{Specific Impulse}} \quad (3.1)$$

Hence from thrust required of 125N and specific impulse of 201.5291 sec ,the mass flow rate can be calculated to be around 62.8gm/sec. Let OF ratio be denoted by r .i.e r = 1.875.

$$\dot{m} = \dot{m}_f + \dot{m}_o \quad (3.2)$$

$$\dot{m}_f = \frac{\dot{m}}{r + 1} \quad (3.3)$$

Thus the mass flow rate of fuel is 21.8 gm/sec and mass flow rate of oxidizer is 40.9 gm/sec.

5. Calculate the required nozzle throat area for the choked flow

$$A^* = \frac{\dot{m}}{P_0} * \sqrt{\frac{T_0 R}{\gamma}} * \left(1 + \frac{\gamma - 1}{2}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \quad (3.4)$$

Substitute $\dot{m} = 0.0628 \text{ kg/sec}$, $P_0 = 80 \text{ Psi}$, $T_0 = 3081.15 \text{ K}$, $R = 415.7$, $\gamma = 1.1583$ in the above equation we get $A^* = 201.1536 \text{ mm}^2$

Solving for the diameter of throat we get $D^* = 16.0036 \text{ mm}$

6) Calculate the exit Mach number and expansion ratio of the nozzle required

From NASA CEA we get , exit mach number to be 1.798 and $A_e/A_t = 1.5661$. We can get the value for exit Mach and expansion ratio using formula such that : [5]

$$M_e = \sqrt{\frac{2}{\gamma - 1} \left[\left(\frac{P_e}{P_0} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]} \quad (3.5)$$

$$\frac{A}{A^*} = \left(\frac{1}{M} \right) \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \left[1 + \frac{\gamma - 1}{2} M^2 \right]^{\frac{\gamma + 1}{2(\gamma - 1)}} \quad (3.6)$$

Solving the above equations we get , $M_e = 1.8145$ and $A_e/A^* = 1.5898$. We can take any of the value but we take calculated value.

Hence the nozzle exit diameter $D^e = 20.178mm$.

6. Find chamber volume

The customary method of establishing the L^* of a new thrust chamber design largely relies on past experience with similar propellants and engine size. Under a given set of operating conditions ,such as type of propellant , mixture ratio, chamber pressure, injector design, and chamber geometry ,the value of the minimum required L^* can only be evaluated by actual firings of experimental thrust chambers .Typical L^* values for various propellants are shown in the table below.[5]

Table 3.3: Characteristics length [6]

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 (GH2 injection)	56-71
Liquid oxygen/liquid hydrogen (LH2 injection)	76-102
Liquid fluorine/liquid hydrogen (GH2 injection)	56-66
Liquid fluorine/liquid hydrogen (LH2 injection)	64-76
Liquid fluorine/hydrazine	61-71
Chlorine trifluoride/hydrazine-base fuel	51-89

Since we didn't find data for gaseous oxygen and RP -1 fuel so for our case we choose maximum value of Liquid oxygen/RP-1 propellant i.e $L^* = 127 cm$ Now , we know

$$V_{ch} = L^* A^* \quad (3.7)$$

We then calculate chamber volume V_{ch} to be $255.4651 cm^3$.

7. Find the diameter and length of the combustion chamber

Now we have got the volume of chamber required we need to estimate its area and length .

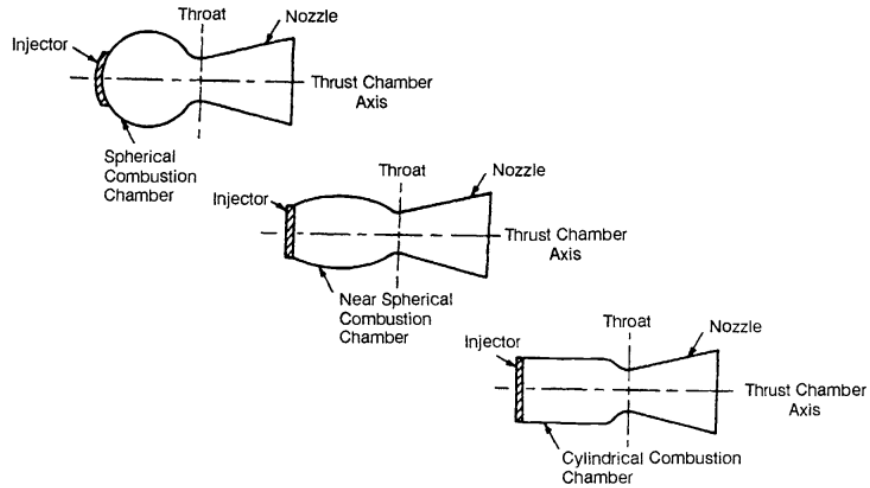


Figure 3.2: Frequently used geometrical shapes for combustion chambers [3]

Lets use a cylindrical combustion chamber and now to determine CSA and length of the chamber we use a empirical formula as

$$\frac{A_c}{A^*} = 8D_*^{-0.6} + 1.25 \quad (3.8)$$

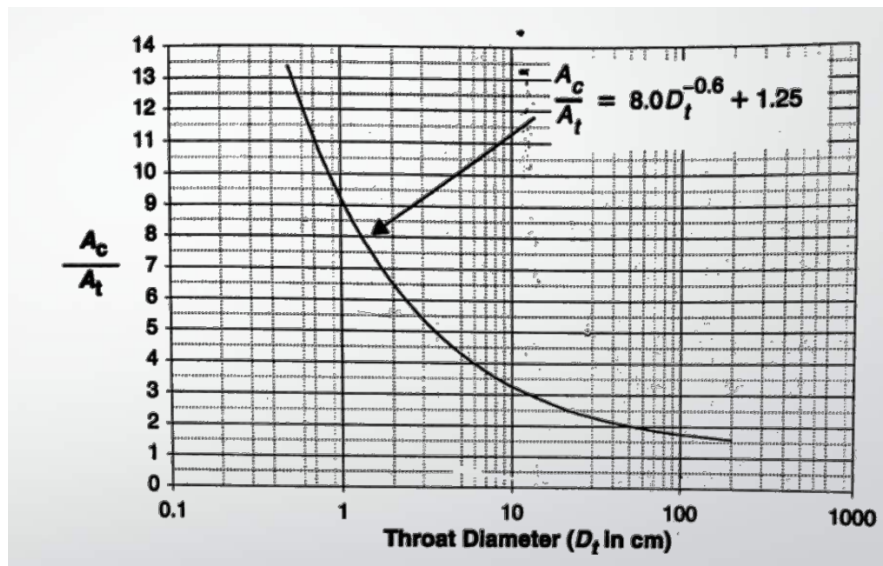


Figure 3.3: Experimental formula for chamber CSA estimation, A_{ch} [4]

Substituting the value of A^* and D^* in above equation we get $A_{ch} = 2164.8 \text{ mm}^2$. Hence the diameter of chamber is 52.5 mm and length of combustion chamber becomes 118.0112 mm .[5]

8. Thickness of the combustion chamber

The combustion chamber must be able to withstand the internal pressure of the hot combustion gases. The combustion chamber wall thickness must be sufficient for welding or brazing purposes. Since the chamber will be a cylindrical shell, the thickness of wall is given by

$$t_w = \frac{PD}{2S} \quad (3.9)$$

Where P is the pressure in the combustion chamber , D is the mean diameter of the cylinder, S is the working stress of the chamber material. Since the material we intend to use for combustion chamber is mild steel so the working allowable stress for mild steel is 29007.5 Psi (200 MPa).

We calculate the combustion chamber thickness to be around 3.5 mm (over estimated value).[5]

9. Injector design

Injectors are crucial components in the liquid rocket engine which is responsible for precisely mixing fuel and oxidizer in the combustion chamber to create stable and efficient combustion. Its design plays a crucial role in achieving desire stable combustion. The injection area of the injector can be calculated using the following equation:

$$\dot{m} = A \cdot C_d \cdot \sqrt{2 \cdot \rho \cdot \Delta P}$$

where:

- A is the injection area,
- C_d is the discharge coefficient,
- ρ is the fluid density, and
- ΔP is the pressure differential across the injector.

Taking C_d to be 0.75 , ρ_{ox} to be 9.1184 kg/m^3 (at 100psi), ΔP to be 16Psi we can calculate the oxidizer injector area to be ,

$$A_{ox} = 38.4845 \text{ mm}^2$$

Taking C_d to be 0.75 , ρ_{fuel} to be 820 kg/m^3 , ΔP to be 16Psi we can calculate the fuel injector area to be ,

$$A_{fuel} = 2.1644 \text{ mm}^2$$

The injection diameter (D) can be determined using the formula:

$$D = \sqrt{\frac{4A}{\pi * N}}$$

where N is the number of orifices used in an injection head.

For fuel we will take 3 injection elements , and we get injection diameter to be ,

$$D_i \sim 1mm$$

For oxidizer we will take 15 injection elements , and we get injection diameter to be ,

$$D_i \sim 2mm$$

Properties of pipes to transport fluid Since kerosene and gaseous oxygen are storable propellants , we can estimate the injection velocity to be around 30 m/s .

$$\dot{m} = \rho * A * V$$

For fuel with mass flow rate 0.0218 kg/sec , density 820 kg/m³ and velocity 30 m/s ,we calculate diameter of pipe to be 2 mm.

For gaseous oxygen with mass flow rate 0.0409 kg/sec , density 9.1184 kg/m³ and velocity 180m/s ,we calculate diameter of pipe to be 7mm .

The pressure drop (P) across a pipe can be calculated using Darcy's equation for pressure drop in a pipe. The general form of the equation is:

$$\Delta P = f \frac{L}{D} \frac{\rho V^2}{2}$$

where - ΔP is the pressure drop across the pipe, - f is the Darcy friction factor, - L is the length of the pipe, - D is the diameter of the pipe, - ρ is the density of the fluid, - V is the velocity of the fluid.

Calculating pressure drop across pipe we found the pressure drop to be negligible . So

, we ignore in our calculations.

3.2. CAD model of test stand

The structure is intended to be fastened in position so that the engine's (maximum thrust of 200 lbf) is transferred to the ground or the rear of the structure. The wall at the back of the construction provides stability. The fuel tank, nitrogen and oxidizer tanks, all fittings, bench setup, combustion chamber, and nozzle are all part of the model. The nitrogen tank is used to compress fuel. CATIA software is used to design the CAD model. This made it easier for us to see our engine's realistic true model. Below is a display of the first suggested model's CAD design:

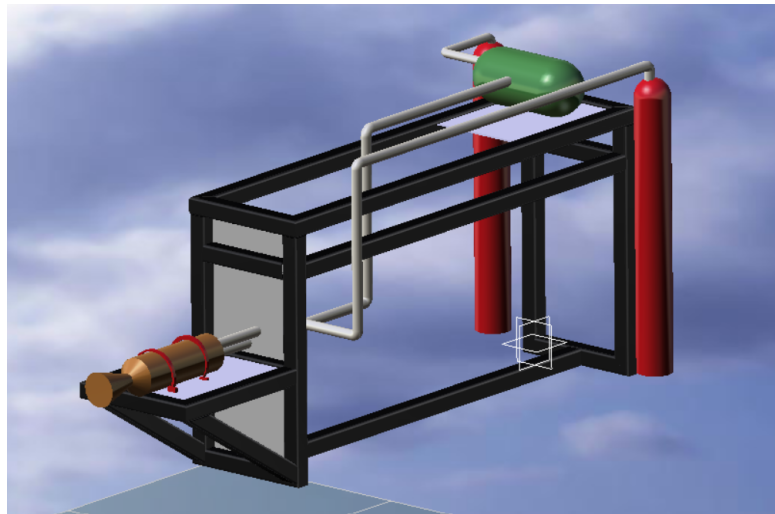


Figure 3.4: CAD Design of proposed static test stand

3.3. Manufacturing and components fittings

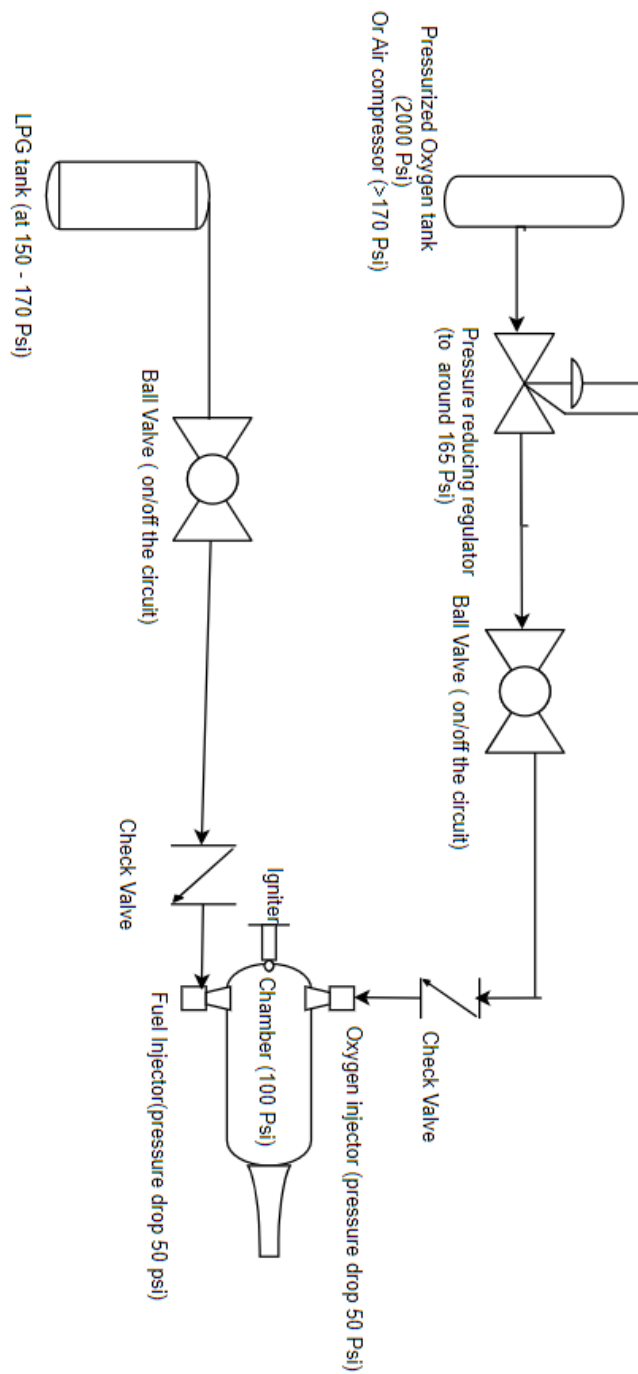


Figure 3.5: Block diagram of LPG and GOX combustion

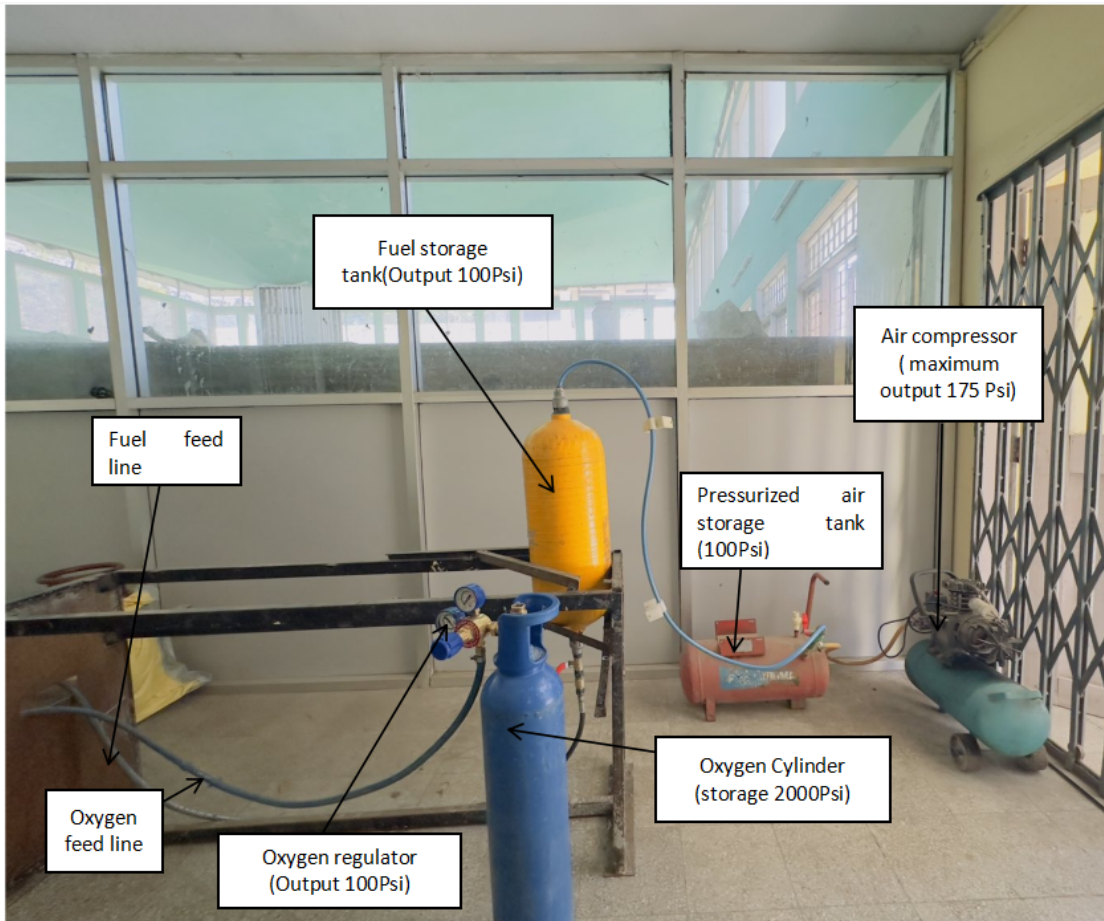


Figure 3.6: Test setup of the rocket engine



Figure 3.7: Thrust stand (Courtesy of SRB ,IOE Pulchowk)

1. Turning

The cylindrical workpiece is first securely mounted onto the spindle in a lathe before the turning process commences. The revolving workpiece is then engaged

with a suitable cutting tool that has been chosen and positioned. The requirements for material and surface polish determine the spindle speed and feed rates. The cutting tool gradually eliminates material from the workpiece as it rotates, forming it into the appropriate cylindrical shape. For better surface finishes and tighter tolerances, finishing passes might be used. Tool life and surface quality are improved by coolant or lubrication, which helps with heat dissipation and chip evacuation. A quality inspection verifies that the workpiece satisfies requirements. After finishing, the machined workpiece is taken out and could go through additional procedures. Maintaining the lathe and its cutting tools on a regular basis guarantees their continuing optimal functioning.

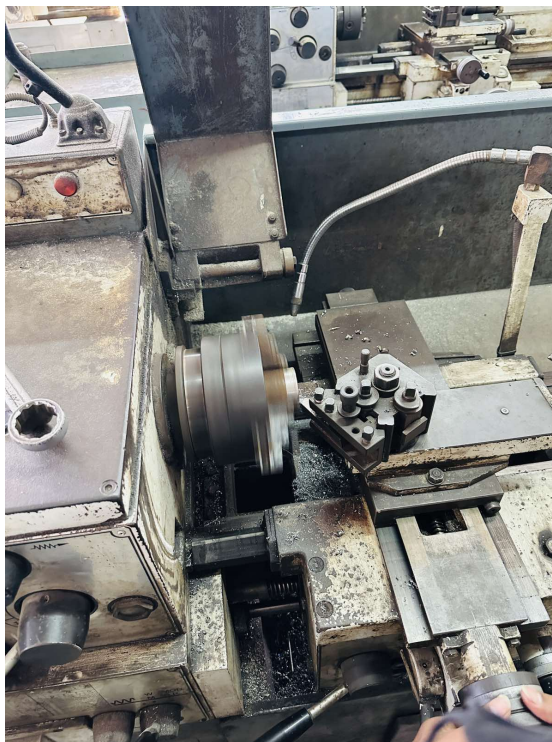


Figure 3.8: Manufacturing of injector turning process

2. Drilling

Mounting the workpiece firmly onto the spindle and positioning the drill bit on the tailstock are the two steps in the drilling process on a lathe. The drill bit is aligned with the intended drilling spot and the spindle speed and feed rate of the lathe are adjusted. The machinist regulates feed pressure while the drill enters the workpiece with the spindle engaged. Chip removal is accomplished by using cutting fluid or coolant. Using measuring instruments, depth control verifies precise hole depths, which are then inspected for quality. The machined workpiece is taken out for additional processing after it is finished. Maintaining a machine's functionality over time requires regular maintenance.



Figure 3.9: Injector Drilling Process

3.3.1. Nozzle

Initially, drilled a shaft that was 130mm long using a 14mm drill bit. Then, drilled the same shaft using a 16mm drill bit. After that, used a boring tool to bore internally from one side at a 30-degree angle of length 50mm. Then, on the other side of the shaft, bored internally at a 12-degree angle, leaving a remaining length of 80mm.



Figure 3.10: Manufactured Nozzle



Figure 3.11: Manufactured Nozzle

3.3.2. Injector

Injectors are pivotal components within a liquid rocket engine, tasked with precisely blending fuel and oxidizer within the combustion chamber to facilitate stable and efficient combustion. Their design significantly influences the attainment of stable combustion. The injector assembly comprises an injector plate and an injector manifold. The manifold directs the liquid fuel and gaseous oxygen separately to the injector plate. The injector plate, in turn, directs the fuel and oxidizer into the combustion chamber. A detailed diagram of the injector plate is provided in the appendix.[15]

The injector manifold is machined from a 60 mm diameter and 5 cm long MS shaft. The shaft undergoes facing, turning, and grooving operations on a lathe. To connect with the combustion chamber, turning operations are conducted until the inner diameter matches that of the combustion chamber, which is 53 mm. Two quarter connector connections are machined using a quarter tap, type, and an 11.5 mm drill bit for the propellant passage, onto which female socket connectors are later attached to connect propellant pipes. O-rings are installed to optimize the flow channel of the propellant and prevent leakage. Sealing between injector plate and manifold was also accomplished using o-ring seals of diameter 15 mm and 40 mm, having thickness 3 mm. [15]

The injector plate is fabricated from a 5 mm MS sheet using a plasma cutter with a 52 mm diameter disk. The injector plate is designed with a spray pattern, featuring 16 ports, each with a diameter of 2 mm, surrounding a central fuel port with a diameter of 3 mm. The oxidizer port holes are angularly drilled at an angle of approximately 18 degrees. An oxidizer inlet is also machined to accommodate the manifold and injector plate, while also serving as a port for fuel injection.[15]



Figure 3.12: Injector Assembly)

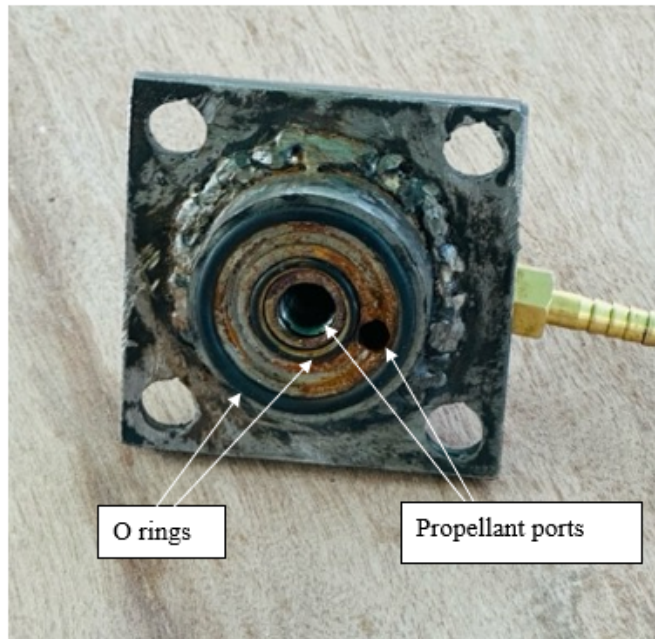


Figure 3.13: Injector manifold

Oxygen Injector

Below figure shows a injector with 15 holes of 2mm for oxygen .The designed pressure drop on the injector was 20 Psi. After the test the parts of injector was burned.



Figure 3.14: Oxygen Injector

3.4. Safety procedures

1. Inspections are conducted regularly to ensure proper functioning of safety valves.
2. Fuel lines are regularly checked for leaks and maintained according to protocol.
3. Emergency shutdown procedures are established and practiced if required.
4. Precautionary measures are taken to prevent ignition sources from coming into contact with fuel or oxidizer.
5. Protective gear is worn by personnel during handling and maintenance tasks.
6. Hazardous materials are stored and handled according to safety regulations and guidelines.
7. made sure we had on the appropriate Personal Protective Equipment, including gloves, safety glasses, earplugs and flame-resistant clothes.
8. It was made sure that there was enough ventilation in the testing area to avoid dangerous gases or fumes building up. To maintain air quality, think about utilizing ventilation fans or exhaust systems.
9. The fire extinguishers and other firefighting supplies were made accessible easily.



Figure 3.15: Headset and Fire Extinguisher

CHAPTER 4: RESULT AND DISCUSSION

4.1. Test and exhaust plume analysis

4.1.1. LPG and gaseous oxygen

First LPG was combusted with atmospheric air and yellow flame was observed. The pressure in the chamber was around 15 Psi . A yellow flame usually indicates incomplete combustion. When there's insufficient oxygen supply or poor mixing of fuel and air, combustion becomes incomplete. In such cases, carbon particles (soot) may be produced due to incomplete oxidation of the fuel, and these particles emit yellow light when heated.

In the second test also LPG was combusted with the compressed oxygen (send at 20 psi) and blue flame was observed at ideal mixture ratio however the blue flame was not sustained for long period of time .A blue flame is indicative of complete combustion. When LPG combusts fully with oxygen, it produces a blue flame. This blue color is due to the efficient burning of the gas, resulting in a high-temperature flame. The blue flame signifies that the carbon in the LPG is being converted completely into carbon dioxide (CO_2), and the hydrogen is being converted into water vapor (H_2O).The blue flame is desirable not only for maximizing energy output but also for minimizing harmful emissions such as carbon monoxide (CO) and particulate matter.

Several other tests were also performed at higher pressure of oxygen at around 50Psi but the velocity of oxygen was to high to maintain the combustion .Also , flame burn out was observed at 70-80 Psi .So the limiting pressure was assumed to be around 50 Psi.The use of the check valve in the tank also prevented gases backflow in the LPG tank.



Figure 4.1: Yellow flame(LPG + GOX)



Figure 4.2: Blue flame (LPG + GOX)

4.1.2. Kerosene and gaseous oxygen

In many liquid rockets kerosene is used as fuel so multiple tests were performed with kerosene. Two of the tests where thrust was measured are listed below.

First test

The maximum thrust of 13N was observed which was less than the designed thrust .

Also due to excessive heating the chamber failed (melted) from the rear. It was realized that the thickness of plate used to cover the chamber was not sufficient and new thicker plate for the next test was used. The test was stopped in mid (incomplete combustion of fuel) and the flow from nozzle can be observed in figure.

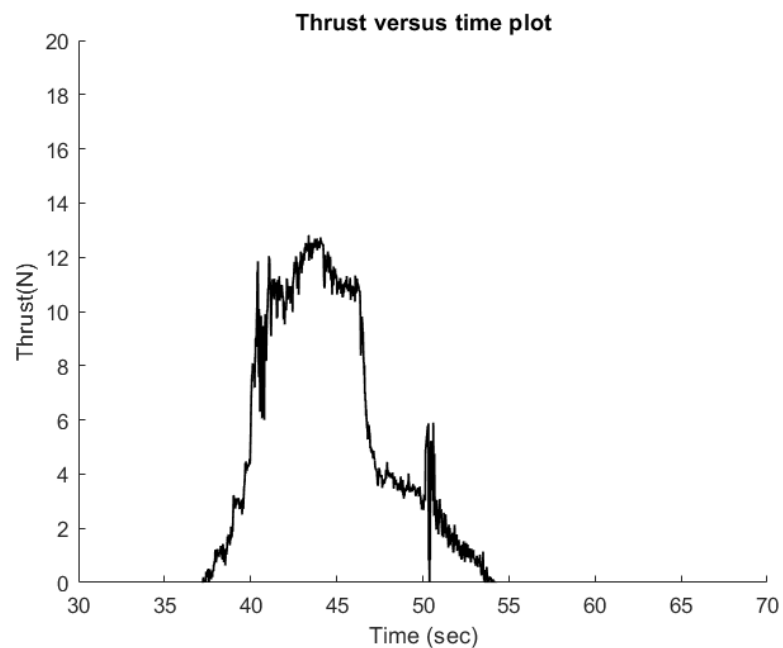


Figure 4.3: Thrust versus time plot

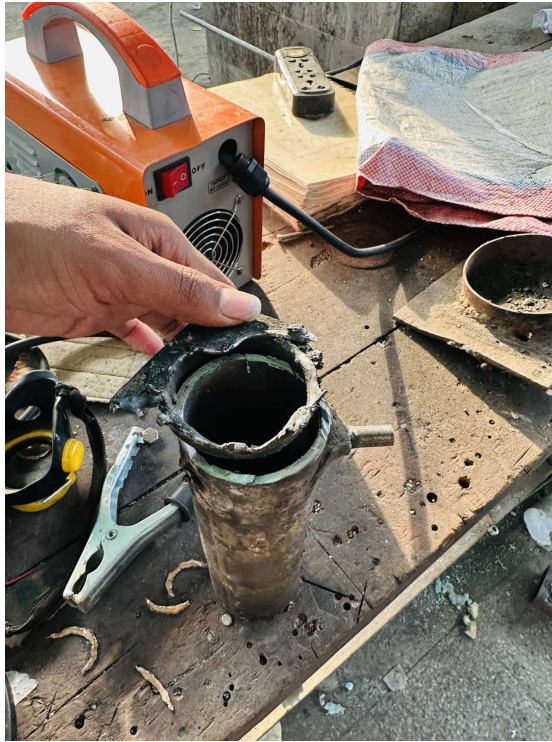


Figure 4.4: Combustion chamber failure



Figure 4.5: Combustion flame through the nozzle

Second test

For the next test oxygen injector was re -designed with 25 holes of 2mm diameter. The maximum thrust measured was 30N. Also , the test had to be stopped in the mid due to failure of the chamber and oxygen injector as shown.

The charring was observed in the chamber wall. It is a chemical process that occurs when solids are partially burned at high temperatures. The process removes hydrogen and oxygen from the solid, leaving behind a black carbon material called char.[16]

Also, the color of a rocket's exhaust depends on the fuel used. For example, rocket

engines that use liquid oxygen as an oxidizer produce a pale blue flame. Engines that use a hydrocarbon fuel like kerosene produce a yellow-orange flame.[17]

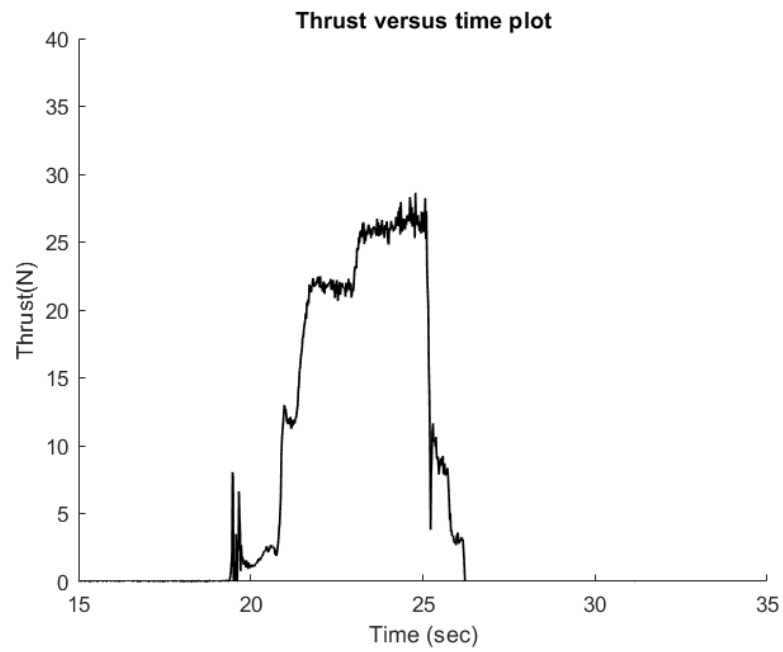


Figure 4.6: Thrust versus time plot



Figure 4.7: Charring of the combustion chamber after the test



Figure 4.8: Chamber and oxygen injector failure



Figure 4.9: Combustion flame

4.1.3. Alternate injector test

Another injector with same design parameters was also manufactured but it was not used in the test with kerosene. The test of another injector with water was done and the atomization of water was observed. The fuel will be passed through the middle hole and the oxygen gas from the side holes.



Figure 4.10: Injector test with water

4.2. Simulation

4.2.1. Nozzle simulation

Table 4.1: Solver Properties

Solver Models	
Type	Density-Based Implicit
Space	Axisymmetric
Viscous	k-epsilon (Realizable)
Heat	Enabled

Table 4.2: Mesh Properties

Mesh Size	
Cells	12600
Faces	25430
Nodes	12831

Table 4.3: Material Properties

Fluid (Air)	
Density	ideal gas
Cp (Specific Heat)	3273.5 J/(kg K)
Thermal Conductivity	0.2103 W/(m K)
Viscosity	sutherland
Molecular Weight	10.7916 kg/kmol
Solid (Aluminum)	
Density	2719 kg/m ³
Cp (Specific Heat)	871 J/(kg K)
Thermal Conductivity	202.4 W/(m K)

Table 4.4: Boundary Conditions of the Simulation

Boundary Conditions	
Inlet	Pressure Inlet (555028 Pa)
Outlet	101325 Pa (Gauge Pressure)

Table 4.5: References Values

	Reference Values
Density	3.526623 kg/m ³
Enthalpy	2891450 J/kg
Pressure	551581 Pa
Temperature	3000K
Viscosity	3.853034e-05 kg/(m s)
Ratio of Specific Heats	1.1583

Table 4.6: Solver Settings

Steady	
Discretization Scheme	Second Order Upwind
Solver	Implicit

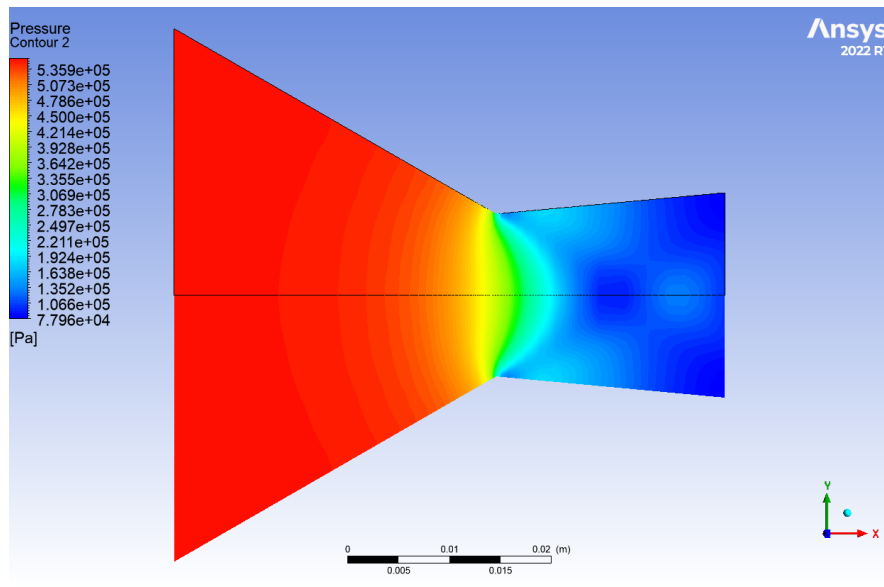


Figure 4.11: Pressure Contour

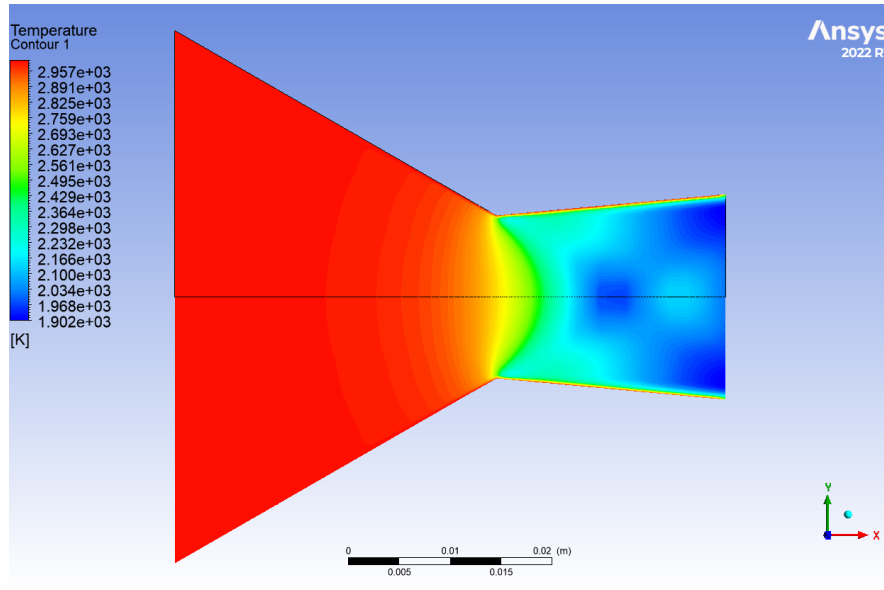


Figure 4.12: Temperature Contour

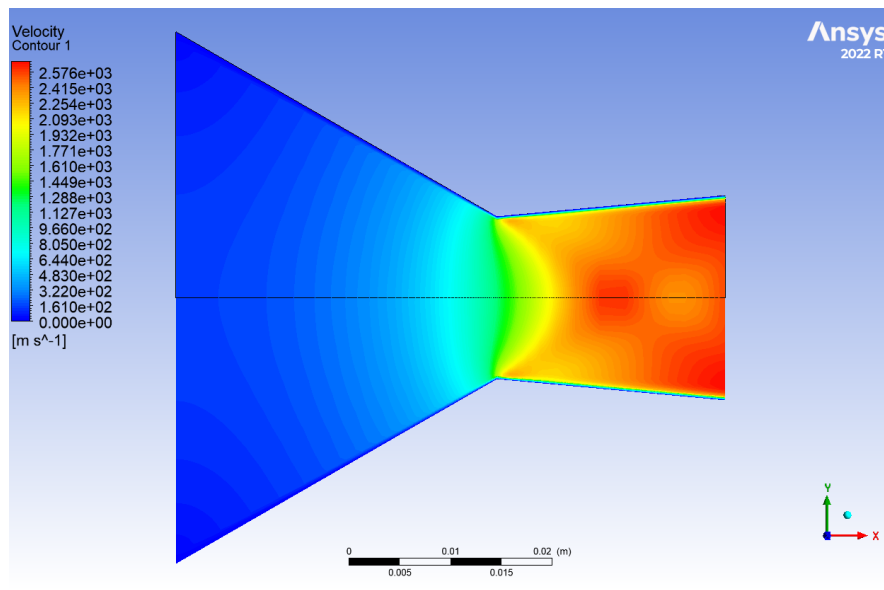


Figure 4.13: Velocity contour

Table 4.7: Calculated Values at the Nozzle Exit

	Calculated Values
P_o	101325 Pa
P_e	77960 Pa
V_e	2093 m/s
D_e	20.178 mm
\dot{m}	62.8E-03
T	123.968 N

4.3. Limitations

- (a) The engine wasn't designed for cryogenic propellants as handling and storing cryogenic propellants present logistical challenges and increase operational complexity.
- (b) No thrust vectoring mechanism was installed in the system.
- (c) No structure support was designed for the engine.
- (d) Weight optimization for the engine wasn't performed and project was relied on easily available parts.

4.4. Problems Faced

Some of challenges faced during the project are summarized below:

- (a) The pipes weren't available at required dimensions of 4mm for fuel and had to use 8mm diameter pipe.
- (b) The designed - hole of the oxygen and fuel injector had to be 1mm and 2mm respectively . It was really a trouble drilling such small holes.
- (c) Its was difficult to perform operations on lathe because of lack of experience and guidance.
- (d) Cooling mechanism had to be used to make components safer and sustainable.

4.5. Budget Analysis

The project came to a total cost of Rs 15,095. The following table displays the project's total cost:

Table 4.8: Budget Analysis

S.N.	Name of the components	Quantity	Price(NRs)	Remark	Source
1	Oxygen	-	1,000		Bhajuratna
2	Fire Extinguisher	1	-	Available	Incubation
3	Pipe and Fittings	-	6,570		Krishna Hardware
4	Cotton	-	50		nearby local shop
5	Shaft	2	1,000		Teku
6	Valve	5	1,200		Krishna Hardware
7	Load Cell	-	-	Available	Incubation
8	Pressure Regulators	1	3,000		Bhajuratna
9	Water tape and M Seal	5	680		Nearby local shop
10	Fuel (Kerosene-5L)	-	795		Devi Oil Store
11	Fuel Storage Cylinder	-	-	Available	Incubation
12	Connectors	7	800		Krishna Hardware
	Total		15,095		

CHAPTER 5: CONCLUSION AND FUTURE ENHANCEMENT

5.1. Conclusion

Multiple tests with LPG and kerosene as fuel were performed. Yellow and blue flame were observed with LPG and no thrust was produced during combustion. The chamber, injector and nozzle were designed for 80Psi chamber pressure and 125N thrust using kerosene as fuel. In all of the test performed with kerosene desired thrust was not achieved and test was stopped in the mid. The major reasons of failure include material selection, cooling process and manufacturing process. Using mild steel at such high temperature was unsuitable as chamber and injector melted in the tests. Also, operating at such high temperature requires cooling of chamber and nozzle for sustained burn time.

5.2. Scope for Future Enhancement

- (a) The tests can be made more controlled. The mass flow rate , chamber pressure should be controlled for the desired thrust.
- (b) Safety valves needs to be used in every feed system.
- (c) Also, other propellants like methanol, ethanol can also be explored.
- (d) Other manufacturing processes needs to be explored .
- (e) Cooling mechanism for combustion chamber and nozzle can be a area of study.
- (f) A reason why liquid rocket are preferred over solid are because of thrust vector control. Controlling the direction of thrust can also be a area of research.

References

- [1] The parts of liquid rocket engine. [Online]. Available: https://www.researchgate.net/figure/The-parts-of-liquid-rocket-engine_fig1_320628712
- [2] karol sarnek on linkedin: rocketpropulsion rocketscience rocketlaunch propellants... — 28 comments. [Online]. Available: https://www.linkedin.com/posts/karol-sarnek_rocketpropulsion-rocketscience-rocketlaunch-activity-7057043153983336448-dxjL
- [3] D. Mishra, *Fundamentals of Rocket Propulsion*. CRC Press, New York, 2017.
- [4] Rocket engine sizing. [Online]. Available: <https://www.youtube.com/watch?v=b9NcdDhhZio>
- [5] L. KRZYCKI, *How to design, build and test small liquid-fuel rocket engines*. ROCKETLAB, 2018.
- [6] [Online]. Available: https://www.researchgate.net/figure/Combustion-chamber-characteristic-length-L_tbl4_342152372
- [7] Design, build and test small scale liquid rocket engine. [Online]. Available: <https://elibrary.tucl.edu.np/handle/123456789/17744>
- [8] G. Hagemann, H. Immich, T. Van Nguyen, and G. E. Dumnov, “Advanced rocket nozzles,” *Journal of Propulsion and Power*, vol. 14, no. 5, pp. 620–634, 1998.
- [9] C. Gottman, W. Alves, J. Rocco, K. Iha, and R. Goncalves, “Liquid rocket propellants: Ethanol as fuel,” *Global Journal of Advanced Research*, pp. 109–119, 2015.
- [10] P. Jayaprakash, D. Dhinarakaran, and D. Das, “Design and analysis of a rocket c-d nozzle,” *International journal of health sciences*, 2022.
- [11] F. Andrés, U. P. Alejandro, and R. Fabio, “Design, manufacture, assembly and testing of a liquid (lox and gasoline) rocket motor for a vehicle with stratospheric apogee in colombia: The sua ii engine,” 2015.
- [12] M. Michal, “Návrh malého motoru na kapalné pohonné látky,” Master’s thesis, České vysoké učení technické v Praze. Vypočetní a informační centrum., 2021.
- [13] G. P. Sutton and O. Biblarz, *Rocket propulsion elements*. John Wiley & Sons, 2016.

- [14] Best of web fall 2023 – austin sommars – falcon 9 exhaust plume. [Online]. Available: https://www.flowvis.org/2023/09/08/_trashed-6/
- [15] G. Gill and W. Nurikck, *Liquid Rocket Engine Injectors*. National Aeronautics And Space Administration, 1976.
- [16] H. L. Kim and J. Kim, “Performance comparison of kerosene based rocket engines with various injectors,” *Journal of Propulsion and Power*, pp. 1316–1323, 2015.
- [17] J. J. Hansen, M. Hussein, S. Moehring, M. Stauffer, C. Acosta, J. Daly, N. Chenoju, and P. Imper, “Student design of a bipropellant liquid rocket engine and associated infrastructure,” in *AIAA Propulsion and Energy 2020 Forum*, 2020, p. 3918.

APPENDICES

THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM

COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR

Pin = 80.0 PSIA
CASE = bijaypaudelnepa

	REACTANT	WT FRACTION (SEE NOTE)	ENERGY KJ/KG-MOL	TEMP K
FUEL	RP-1	1.0000000	-24717.700	298.150
OXIDANT	O2	1.0000000	0.000	0.000

O/F= 1.87500 %FUEL= 34.782609 R,EQ.RATIO= 1.816355 PHI,EQ.RATIO= 1.816355

	CHAMBER	THROAT	EXIT
Pinf/P	1.0000	1.7513	5.4421
P, BAR	5.5158	3.1496	1.0135
T, K	3081.15	2879.71	2459.25
RHO, KG/CU M	4.2465-1	2.6215-1	1.0021-1
H, KJ/KG	-615.15	-1315.50	-2569.41
U, KJ/KG	-1914.06	-2516.92	-3580.83
G, KJ/KG	-41066.0	-39121.8	-34855.6
S, KJ/(KG)(K)	13.1285	13.1285	13.1285
M, (1/n)	19.723	19.929	20.217
(dLV/dLP)t	-1.01623	-1.01065	-1.00319
(dLV/dLT)p	1.3136	1.2183	1.0746
Cp, KJ/(KG)(K)	4.7569	4.0487	2.8514
GAMMA _s	1.1583	1.1659	1.1953
SON VEL, M/SEC	1226.6	1183.5	1099.5
MACH NUMBER	0.000	1.000	1.798

PERFORMANCE PARAMETERS

Ae/At	1.0000	1.5661
CSTAR, M/SEC	1777.8	1777.8
CF	0.6657	1.1121
Ivac, M/SEC	2198.6	2488.6
Isp, M/SEC	1183.5	1977.0

Figure A1: NASA CEA Interface

MASS FRACTIONS

*CO	0.61068	0.60610	0.59613
*CO2	0.13575	0.14295	0.15863
*H	0.00205	0.00145	0.00051
HCO	0.00001	0.00000	0.00000
*H2	0.02224	0.02266	0.02380
H2O	0.21266	0.21713	0.21893
*O	0.00136	0.00059	0.00005
*OH	0.01410	0.00861	0.00192
*O2	0.00115	0.00050	0.00004

Figure A2: NASA CEA interface

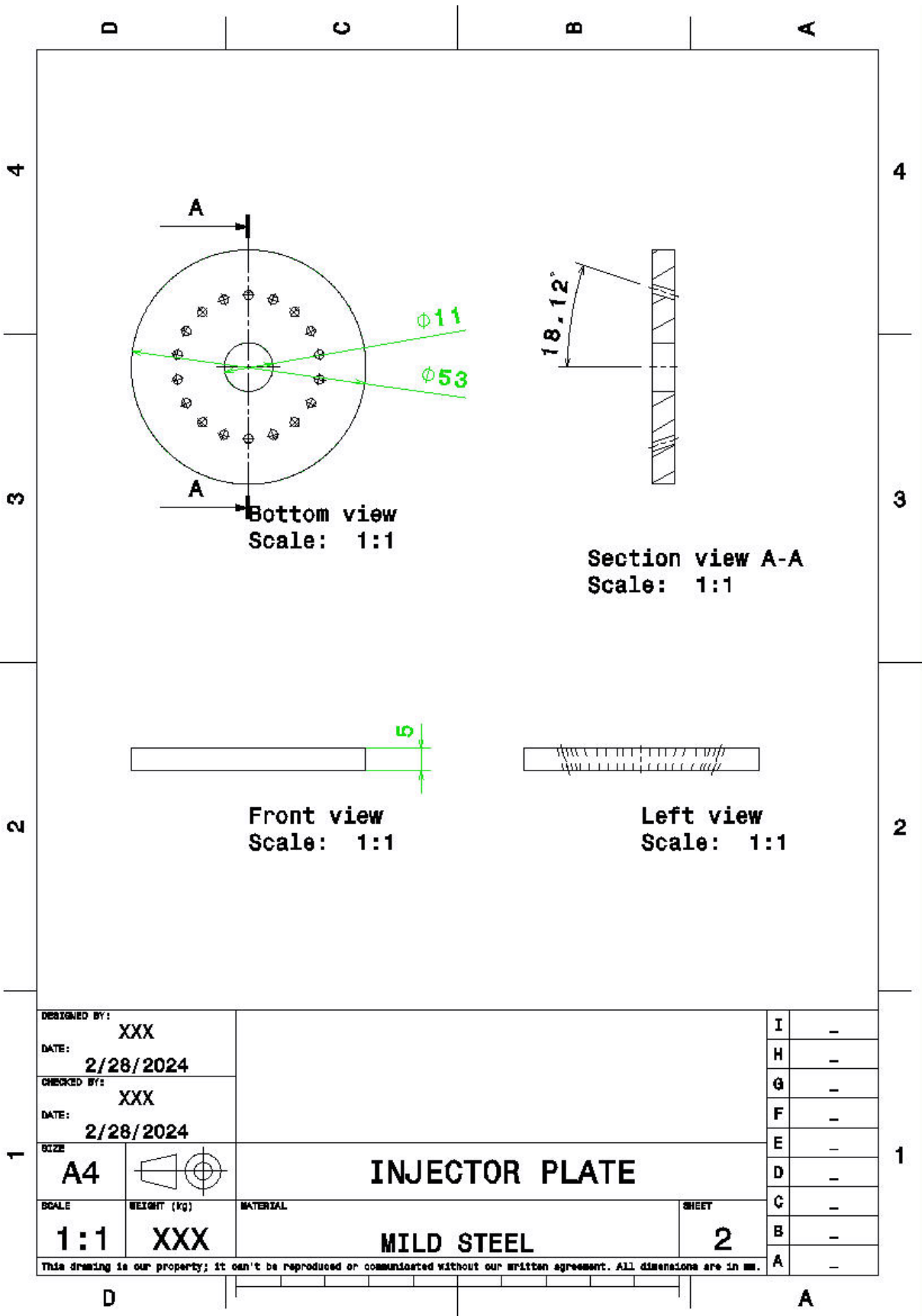


Figure A4: Injector plate

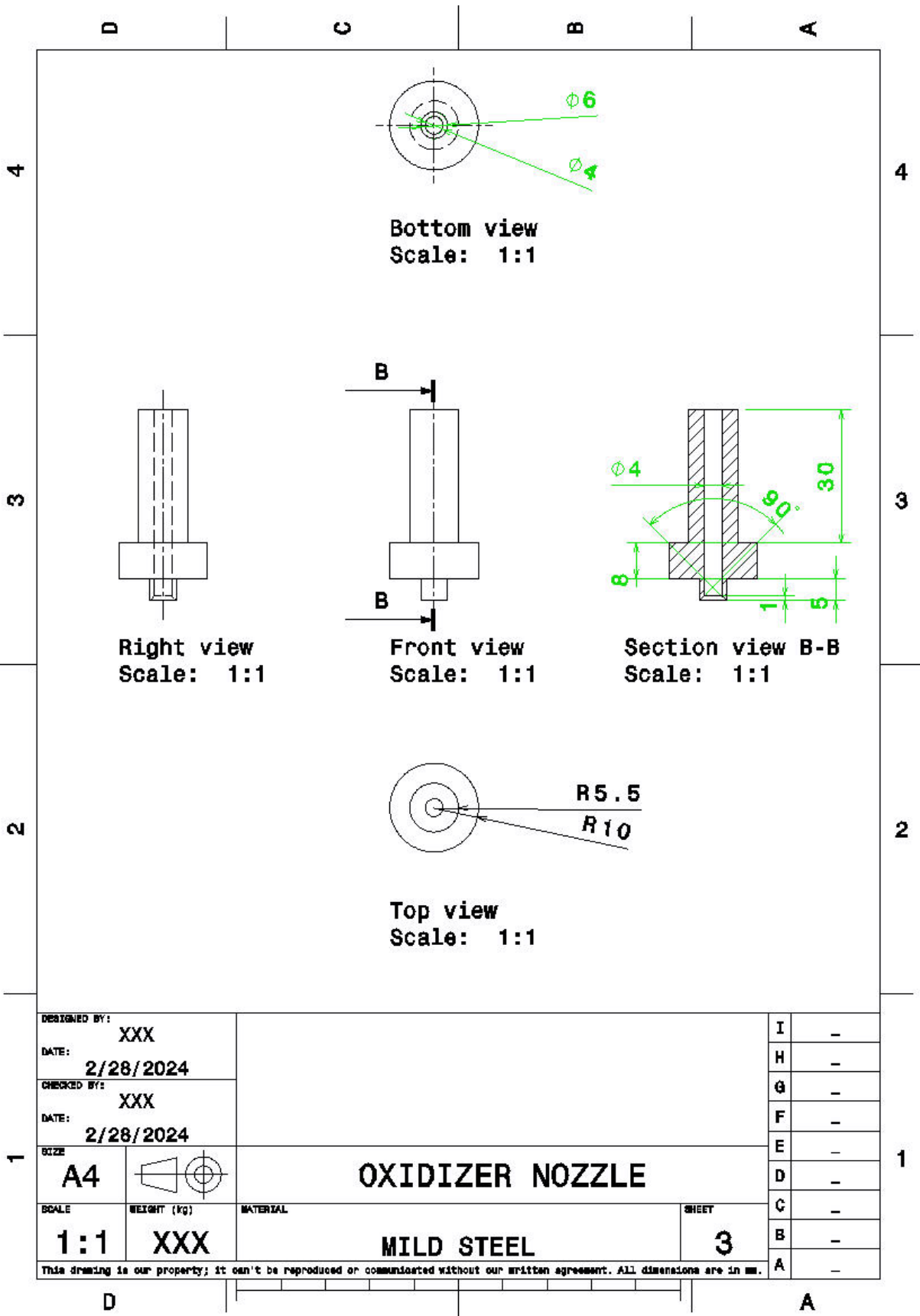


Figure A5: Fuel Nozzle

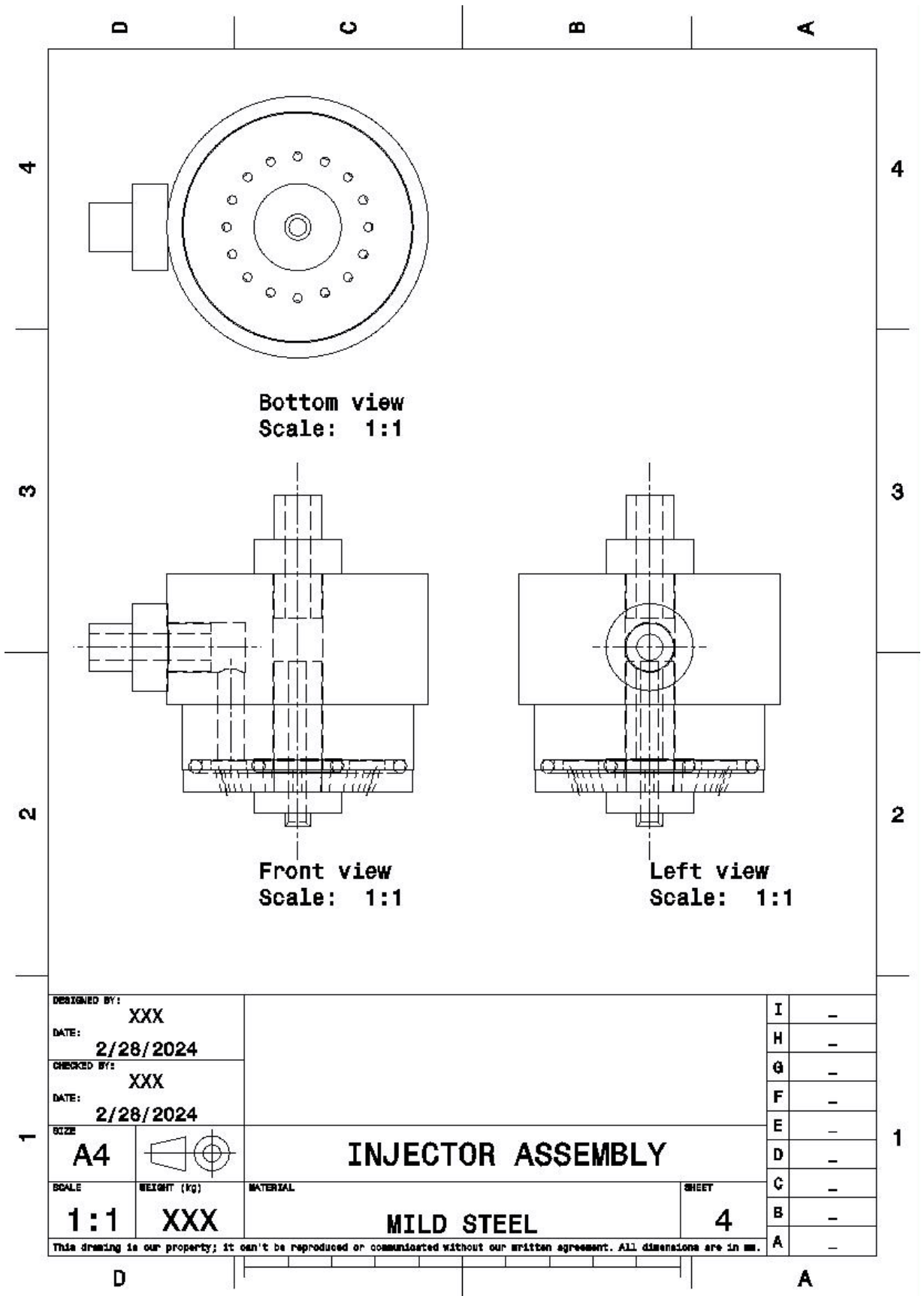
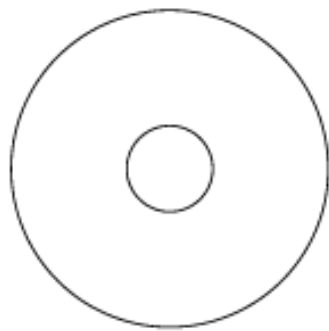
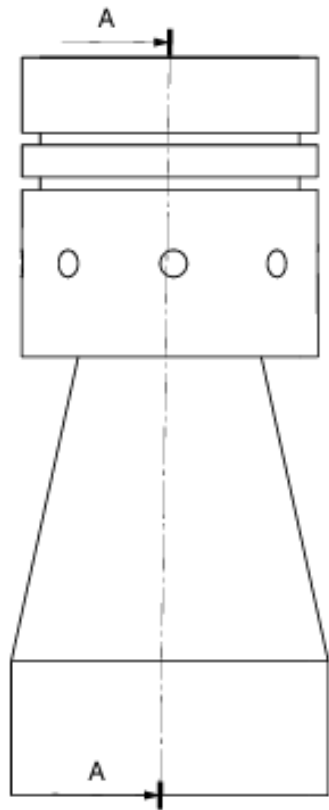


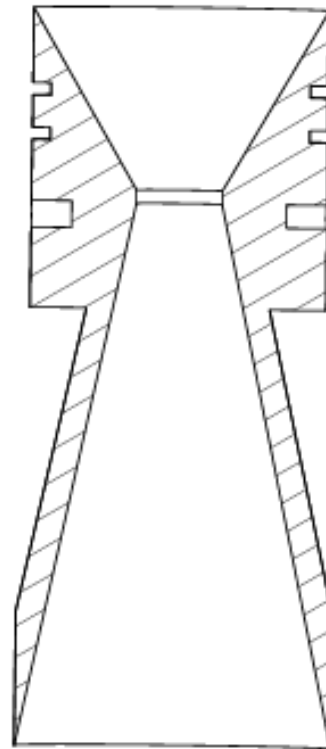
Figure A6: Injector Assembly



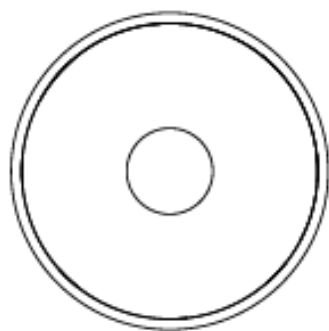
Top View
Scale: 1:1



Front view
Scale: 1:1



Section view A-A
Scale: 1:1



Bottom view
Scale: 1:1

Figure A8: Nozzle

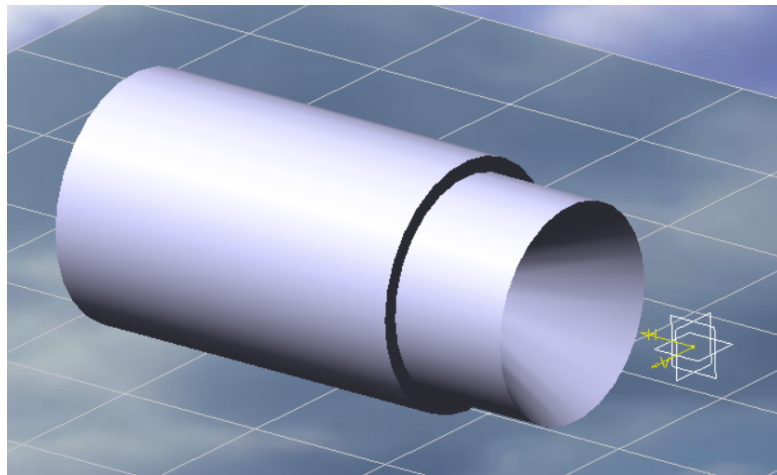


Figure A9: CAD model of Nozzle

BUILD_AND_TEST_A_SMALL_SCALE_LIQUID_ROCKET_ENGI... 13-50.pdf

ORIGINALITY REPORT

15%

SIMILARITY INDEX

PRIMARY SOURCES

1	elibrary.tucl.edu.np Internet	215 words — 3%
2	sites.google.com Internet	157 words — 2%
3	spacha.github.io Internet	69 words — 1%
4	sportdocbox.com Internet	64 words — 1%
5	www.eucass.eu Internet	64 words — 1%
6	Alessandro de Iaco Veris. "Fundamental Concepts of Liquid-Propellant Rocket Engines", Springer Science and Business Media LLC, 2021 Crossref	58 words — 1%
7	sciencescholar.us Internet	54 words — 1%
8	dspace.cvut.cz Internet	47 words — 1%
9	dx.doi.org	

Internet

39 words — 1%

10 www.mdpi.com

Internet

36 words — 1%

11 www.britannica.com

Internet

30 words — < 1%

12 [Diaaeldin Mohamed, Luisa Cabrera, Norman D. Love, Ahsan R. Choudhuri. "High Velocity Oxy Fuel Thermal Spray Gun Design", 52nd Aerospace Sciences Meeting, 2014](#)

Crossref

21 words — < 1%

13 [David H. Huang, Dieter K. Huzel. "Modern Engineering for Design of Liquid-Propellant Rocket Engines", American Institute of Aeronautics and Astronautics \(AIAA\), 1992](#)

Crossref

20 words — < 1%

14 123dok.net

Internet

19 words — < 1%

15 [Steven Goddard, Yufeng Yao. "A CFD investigation of nozzle position effect and turbulent flow phenomena of a hyperloop model", International Journal of Theoretical and Applied Multiscale Mechanics, 2020](#)

Crossref

18 words — < 1%

16 www.irjet.net

Internet

16 words — < 1%

17 acikders.ankara.edu.tr

Internet

9 words — < 1%

18 s3.amazonaws.com

Internet

9 words — < 1%

19 www.meducator.net
Internet

9 words — < 1%

20 Sweeney, Brian A.. "Like-Doublet Injectors: The Effects of Varying the Impingement Distance and an Analysis of the Primary Atomization Zone", The University of Alabama in Huntsville, 2023
ProQuest

8 words — < 1%

21 www.etasgroup.com
Internet

8 words — < 1%

22 Alessandro de Iaco Veris. "Chapter 2 The Thrust Chamber Assembly", Springer Science and Business Media LLC, 2021
Crossref

7 words — < 1%

EXCLUDE QUOTES ON

EXCLUDE SOURCES OFF

EXCLUDE BIBLIOGRAPHY ON

EXCLUDE MATCHES OFF