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# Design, Build and Test Small Scale Liquid Rocket Engine

by

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A PROJECT REPORT
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**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 "Design Build and Test Small Scale Liquid Rocket Engine" submitted by Krishna Kant Mishra, Prajwal Gautam and Prakash Yadav in partial fulfillment of the requirements for the degree of Bachelor of Aerospace Engineering.

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# ABSTRACT

The project entitled, "Design, Build and Test Small Scale Liquid Rocket Engine" reports about the liquid rocket engine which is a device in which liquid propellant is used to produce thrust. In this project at least 200N of thrust is aimed to be generated. This project aims to generate thrust by using liquid propellant which is the combination of oxidizer and fuel. This project is an experimental study of the amount of thrust produced by the liquid rocket engine and design of various components needed to achieve the project target. The combustion chamber, nozzle, cylinder and fuel selection and design are focused and thus the target of the project is being achieved. This project helps us implement our knowledge and understanding of propulsion system and make us practically study its behavior which will surely be helpful for enhancing our knowledge and understanding of the propulsion phenomenon.

In this project kerosene as fuel and oxygen as oxidizer is used. Initially atmospheric air as oxidizer is used which is supplied into chamber with the help of compressor at a specific pressure. Kerosene stove as a fuel tank is used in the project. The fitting of components is done using GI pipe of 1 inch diameter. Ignition to the combustion chamber is done with the help of electric coil that is put from outside into the combustion chamber. The chemical reaction takes place inside the combustion chamber. A nozzle that is best suited for our liquid rocket engine is designed and manufactured. When the fuel and oxygen burn inside the combustion chamber, chemical reaction occurs and flame passes through the nozzle which increases the momentum of the flame and in reaction to that momentum, the thrust is produced which we expect around 200 N.

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| | |
|---|---|
| $U_e$ | Exhaust Velocity |
| $C_p$ | Heat capacity of constant pressure |
| $T_c$ | Chamber Temperature |
| $W_t$ | Propellant mass flow rate |
| $P_t$ | Pressure at throat |
| $T_t$ | Temperature at throat |
| $A_t$ | Area of throat |
| M | Mach no. |
| $A_c$ | Area of combustion chamber |
| $L^*$ | Characteristic length |
| $I_{sp}$ | Specific impulse |
| W | mass flow rate of discharge |
| A | Area of orifice |
| $\Delta$P | pressure drop across orifice. |
| $C_d$ | Orifice discharge coefficient |
| R | gas constant |

# CHAPTER ONE: INTRODUCTION

A liquid rocket engine is a particular kind of rocket engine that produces thrust using liquid propellants. The fundamental idea underlying a liquid rocket engine is to ignite a mixture of two liquids typically a fuel and an oxidizer in a combustion chamber. Hot gases are released through a nozzle at the back of the engine when the fuel and oxidizer burn. This results in a reaction force that drives the rocket forward.

In addition to military missiles and other aerospace uses, liquid rocket engines are frequently utilized in space launch vehicles. Although they tend to be more complicated than solid rocket engines, they have several benefits, including as the ability to throttle the engine and shut it down if necessary and the use of a wider variety of propellants for higher performance and versatility.

Liquid rocket engines come in a variety of forms, including pressure-fed engines, pump-fed engines, and staged combustion engines. Each has pros and cons of its own. It takes specialized knowledge and proficiency in fields like fluid dynamics, thermodynamics, materials science, and propulsion systems to design and construct a liquid rocket engine, which is a difficult and complex task.

## 1.1 Background

There is currently little to no presence or knowledge of liquid rocket engine technology in Nepal. 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. Extensive testing, research and development, and regulatory compliance are frequently involved.

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 rocket engine, it nevertheless marks a substantial improvement in Nepal's space capabilities and might open the door for future advancements.

Liquid rocket engine is a device in which liquid propellant is used to produce thrust. In this project we aim to generate at least 200N of thrust. This project aims to generate thrust by using liquid propellant which is the combination of oxidizer and fuel. This project is an experimental study of the amount of thrust produced by the liquid rocket engine and design of various components needed to achieve the project target. The combustion chamber, nozzle, cylinder and fuel selection and design is focused and thus the target of the project is achieved. This project helps us implement our knowledge and understanding of propulsion system and make us practically study its behavior which will surely be helpful for enhancing our knowledge and understanding of the propulsion phenomenon.

Several types of liquid rocket engines exist. They are:

1. Pressure-fed engine: The propellants in this sort of engine are pushed into the combustion chamber by high-pressure tanks. Although this is an easy-to-use and dependable design, the thrust and duration are constrained.

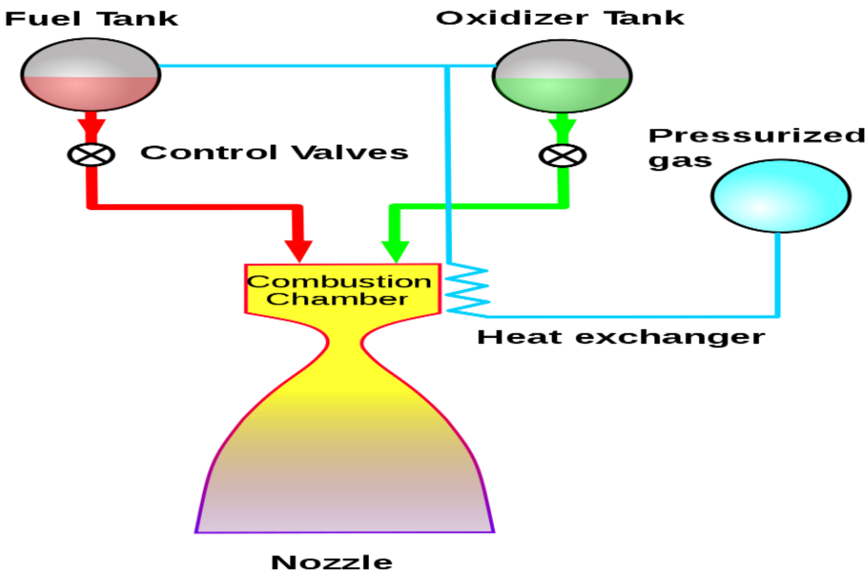



*Figure 1: Diagram of pressure fed engine.*

*(Source: https://en.wikipedia.org/wiki/Pressure-fed_engine)*

2. Pump fed engine: Pumps are used in this sort of engine to push the propellants into the combustion chamber. A turbine, an electric motor, or another device may be used to power the pumps. Compared to pressure-fed engines, pumps can produce more thrust and have longer burn durations.

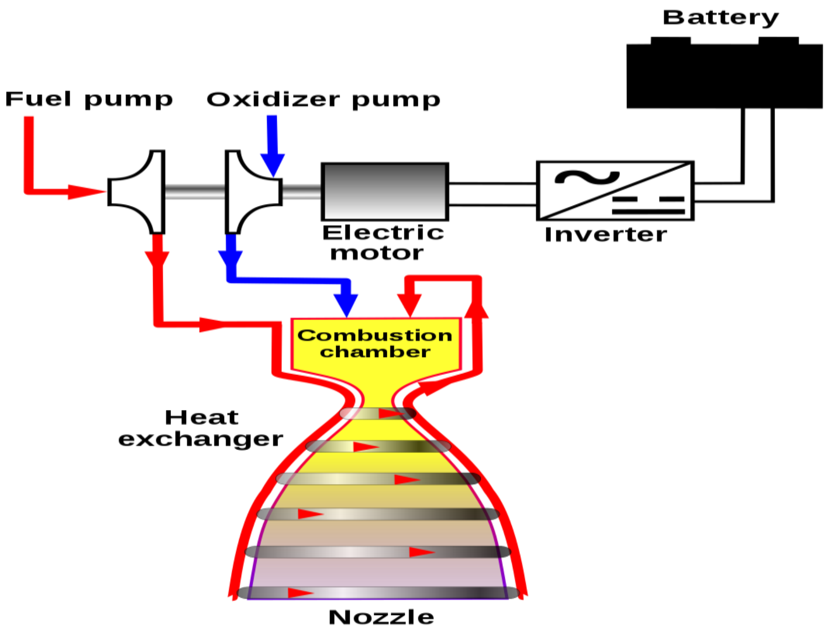



*Figure 2: Diagram of pump fed engine*

*(Source: https://en.wikipedia.org/wiki/pump-fed engine)*

3. Gas-generator engine: This kind of engine produces hot gases from a little number of propellants, which are then utilized to turn a turbine that drives the pumps. While complex and expensive, this design is highly efficient and capable of producing a lot of thrust.

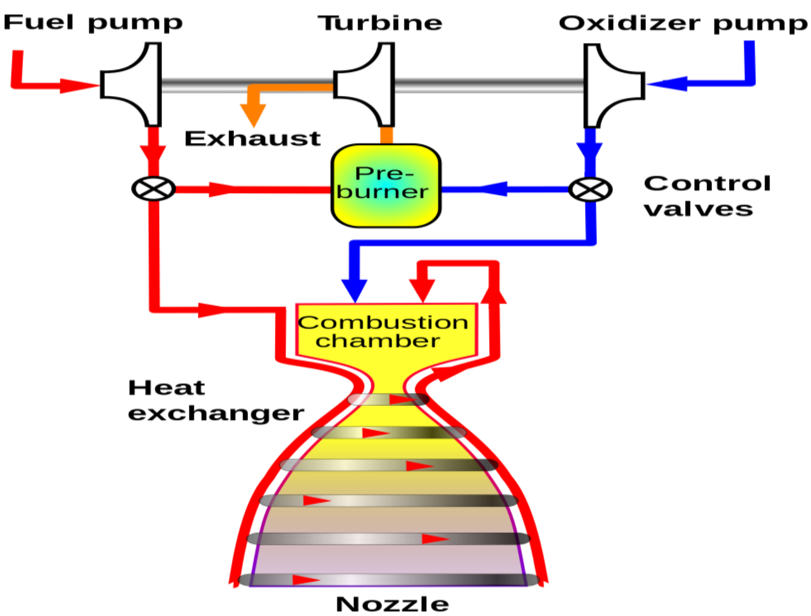



*Figure 3: Diagram of gas generator engine*

*(Source: https://en.wikipedia.org/wiki/Gas-generator_cycle)*

4. Staged combustion engine: The propellants are consumed in stages in this kind of engine, with some of the hot gases being utilized to drive the pumps and the remainder being used for thrusting. Although this design is exceedingly efficient and capable of producing a lot of thrust, it is also intricate and challenging to construct.

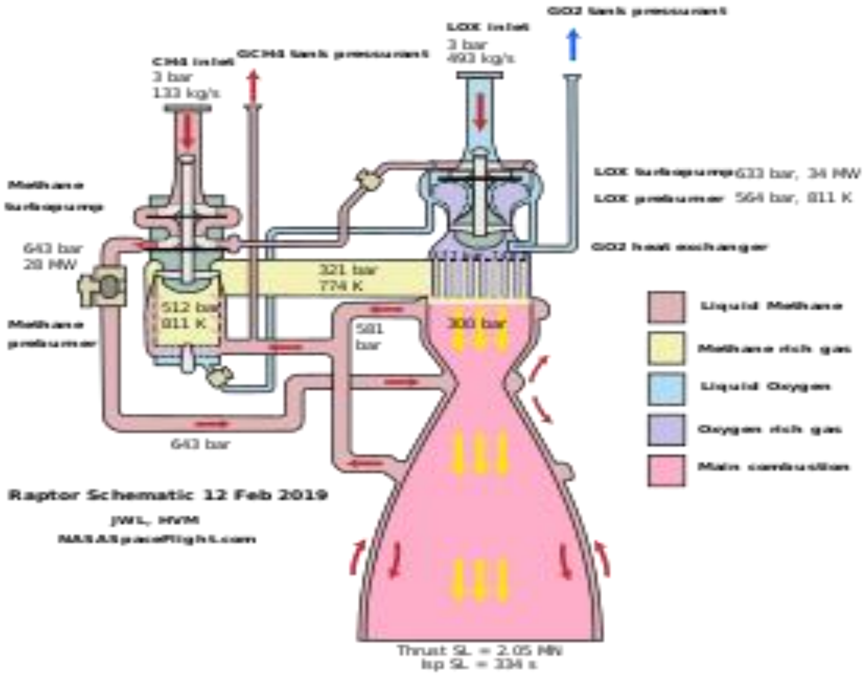



*Figure 4: Diagram of staged combustion engine*

*(Source: https://en.wikipedia.org/wiki/Staged_combustion_cycle)*

Normally, propellant passes through two different types of combustion chambers: the primary combustion chamber and the preburner. A tiny amount of propellant, often fuel-rich, is partially burned in the preburner, and the resulting volume flow is utilized to power the turbopumps that supply the engine with fuel. The other propellant and the gas are then entirely burned together to create thrust once the gas is introduced into the main combustion chamber.

5. Dual-expander engine: The combustion chamber and nozzle of this kind of engine are cooled by two independent loops, enabling increased performance and longer burn periods. Yet building one is equally expensive and complicated.

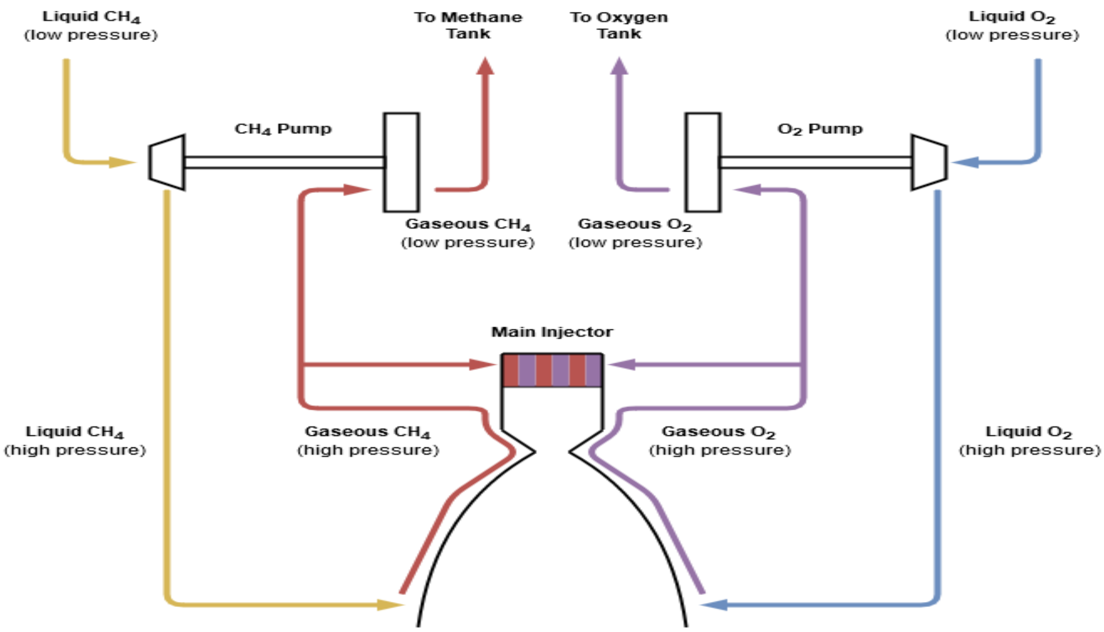



*Figure 5: Diagram of dual expander engine*

*(Source: https://space.stackexchange.com/questions/41165/why-arent-expander-cycle-engines-used-on-lower-stages)*

This type of rocket engine is based upon dual expander cycle. During this cycle, the fuel picks up heat and undergoes a phase change while cooling the engine's combustion chamber. The engine's fuel and oxidizer pumps are driven by the turbine powered by the heated and gaseous fuel, which is then injected into the combustion chamber and consumed.

The square-cube law sets a thrust limit on the expander cycle since a phase change is required. The volume of fuel that needs to be heated grows as the cube of the radius when a bell-shaped nozzle is scaled, yet the nozzle surface area needed to heat the fuel increases as the square of the radius. There is therefore insufficient nozzle area after about 300 kN (70,000 lbf) of force to heat enough fuel to power the turbines, and consequently the fuel pumps. Bypass expander cycles allow for higher thrust levels by directing some fuel straight to the main chamber injector instead of the turbine and/or thrust chamber cooling tunnels.

6. Aerospike engine: The adoption of a special nozzle design on this kind of engine enhances performance by lowering atmospheric drag. It has not yet been widely utilized in rockets and is still in the experimental stage. At a broad range of heights, this type of engine retains its aerodynamic efficiency.

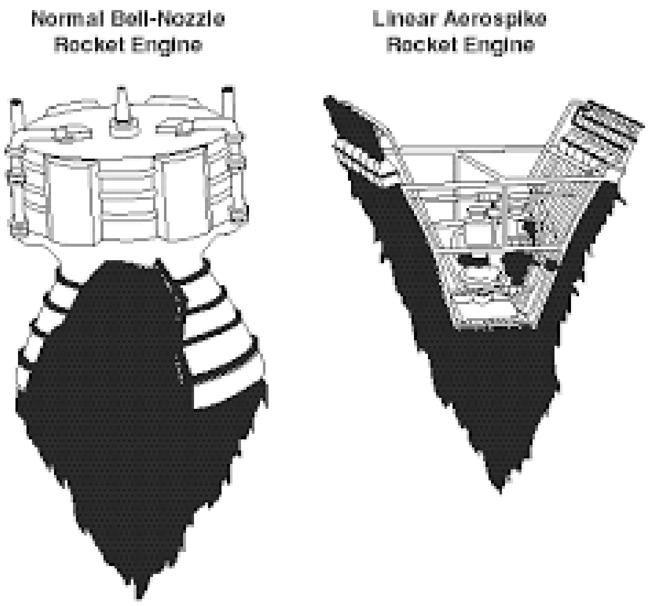



*Figure 6: Diagram of Aerospike engine*

*(Source: https://en.wikipedia.org/wiki/Aerospike_engine)*

## 1.2 Problem Statement

Slowly the interest of Nepalese people is increasing in aerospace industry which can be seen by the recent development like launch of first Nepali satellite Nepalisat-1 in 2019, increase in space organization working in development of satellites and many more. So we collectively have a dream that one day we will have our own space organization and we would have infrastructure to launch any payloads into space by ourselves. For that dream to get true, we need to move slowly one step at a time. In that process, we have thought to do research on liquid propulsion system and fabricate one with the local resources available.

## 1.3 Objectives

### 1.3.1 Main Objective

The main objective of this project is to design, build and test the small-scale liquid rocket engine.

### 1.3.2 Specific Objectives

- To design the small liquid engine capable of producing 200N of thrust.
- To investigate the different characteristic of nozzle and combustion chamber
- To fabricate one small prototype of liquid rocket engine.

## 1.4 Application

This project is a small-scale liquid rocket engine. This project can be used to understand the actual physics and phenomenon about the actual liquid rocket engine that is made in the large scale. Besides this project can be used to understand the characteristics of different materials and their properties which have assisted in the successful construction of the physical model of the project. The amount of thrust that is produced can be analyzed and observed and further study can be done by the upcoming students who are the aspirant in the same sort of project. This project will also put light on different factors like characteristics length of the combustion chamber, materials used for construction of model, maximum temperature reached during combustion, maximum thrust measured which will greatly assist the similar future projects.

## 1.5 Features

Liquid rocket engine refers to the rocket engine in which liquid propellant is used. It involves production of thrust by burning, pressurizing, and finally expanding of propellant to generate massive amount of thrust. Liquid propellant consists of oxidizer and fuel. Both are fed into the combustion chamber through injector in which they burn and chemically react to produce hot and pressurized gases. The gases then are accelerated through special device called nozzle. The nozzle helps the gases to be imparted out of the engine with very high velocity and thus convey momentum to the engine. Thus, as a result of the momentum of high velocity gases the engine in reaction receives thrust which in turn helps to propel the engine forward.

Our liquid rocket engine consists of following parts:

- Liquid propellant: As already mentioned above liquid propellant consists of oxidizer and fuel. In our project the oxidizer is oxygen in gaseous form and the fuel is kerosene.
- Propulsion sustains system: Required pipes and other mechanical fittings is used for transport of liquid propellant into the combustion chamber through the injector.
- Combustion chamber: The combustion chamber is designed in which mixing and chemical reaction of the propellant occurs.
- Flow control valve: Various flow controllers is used at the required regions to regulate the required amount of fluid flow.
- Electric ignition coil: An electric coil from outside of the engine is set up and fed into the combustion chamber which helps to ignite the liquid propellant.
- Kerosene Stove: A locally available kerosene stove is used in our project which performs dual role, one is to store the fuel and the other is to pressurize the fuel.
- Fire extinguisher: In our project to avoid fire related hazards, a fire extinguisher is used.

## 1.6 Feasibility Analysis

### 1.6.1 Economic Feasibility

This project incorporates various materials having high melting point and high thermal conductivity which are expensive to purchase. At the beginning of our project, use of digital flowmeter was also considered by us but as it was very expensive in context of our project so that decision was canceled. Use of various valves for flow control, use of load cell for measurement of the thrust can be seen in the project which requires good sort of budget to accomplish. Similarly various high strength materials was also in our initial plan but due to the limited budget and availability it wasn’t considered later in the project. As the project seemed unfeasible at the beginning, various alternative materials were considered like use of stove for storage of fuel and pressurizing it, which proved to be a valuable alternative to reduce the cost of expensive pressurized nitrogen cylinder. Thus, such a use of alternative approach helped to make our project feasible to some extent.

### 1.6.2 Technical Feasibility

The design and manufacturing requirements for the completion of the project has not been totally met due to limited technology and resources in our country but still maximum use of the available resources was done to achieve the project target. The theory required by the project was already incorporated in the syllabus of Aerospace engineering which helped us on our project. The resources for welding , cutting, bending etc. was available in the department which helped to complete our project. Thus, our project was technically feasible.

### 1.6.3 Operational Feasibility

The testing of the liquid rocket engine requires an open space so that probable hazards can be minimized. The use of safety measures was greatly considered in our project. Use of control valves was incorporated to regulate the amount of fuel sent to the combustion chamber. Likewise, fire extinguisher was also kept intact to counter the fire related hazards. The liquid rocket engine was operated from a safe distance to consider the safety issues. The validation of the project was done before the test which eliminated any risk of the liquid rocket engine being damaged. The test of the liquid rocket engine was also done outside the incubation and innovation center which made sure that no harm would be done to the people inside. The nearby region around the testing zone was also cleared during the test so that no harm or injuries would occur to the people outside. Due to these reasons, our project was operationally feasible.

## 1.7 System requirements

### 1.7.1. Software requirements

- CAD software for design of various components (SOLIDWORKS AND CATIA)
- Software for simulation and system analysis of the liquid rocket engine (ANSYS)

### 1.7.2 Hardware Requirements

- Kerosene stove for storing and pressurizing of fuel
- Hollow tube made up of mild steel
- Bench setup for holding entire setup
- Combustion chamber holder
- Manually made fire stick for igniting the engine from a safe distance
- Pipes and fittings

# CHAPTER TWO: LITERATURE REVIEW

[1] A compact liquid rocket engine with self-pressurization, regenerative cooling with gaseous injection, and repeatable ignition was desired to be built for 25KN of thrust generation. For designing purposes, the right order of computations, such as propellant selection, oxidizer/fuel calculation, mass flow rate, nozzle size, combustion chamber, cooling, etc., was established. The pressurization of the tank is aided by the heat produced in the combustion chamber, which offers regenerative cooling. Basically, six essential components were taken into consideration when building the liquid engine system: the nozzle, throat chamber injector, combustion chamber, tanks, and feeding mechanism. The self-Pressurization was accomplished through regenerative cooling. The appropriate materials ought to be chosen. Due to the high temperature of the exhaust, some of the materials melt. The combustion chamber, where the combustion occurs, is an important component. The combustion chamber's temperature may increase more, which could cause it to melt. It is important to think about proper cooling. Regenerative cooling, film cooling, transpiration cooling, ablative cooling, and radiation cooling are the main methods for cooling the combustion chamber. Regenerative cooling is utilized in this design. Ammonia was the fuel chosen because it has an excellent cooling effect and can reach high pressure at low temperatures. The swirl injector was employed. Unnecessary instabilities are presented by the swirl injector. The chamber has a diameter of 30 mm, a pressure of 15 bars, a throat diameter of 3 mm, and is made of stainless steel. The mass flow rate is 1:4. The oxidizer was at a pressure of 40 bar and a temperature of 293 K. Fuel was at a pressure of 40 bar and a temperature of 300K, respectively. Since ethanol is safer and easier to handle than ammonia, it was initially chosen as the fuel for the experiment. Pressure, temperature, and pharynx measurements could all be made using the sensors. The initial test made use of external pressure. The test was then performed with ammonia. The estimated 25KN thrust was produced at a force of about 23N.

Using inexpensive and locally accessible components, a model of a liquid-fueled rocket engine was built [2]. Over solid fuel engines, liquid fuel rocket engines have many advantages. A liquid-fueled rocket has the advantages of being accurate, having greater control, and having better acceleration. Liquid propellants should have physical features like no freezing point, good heat transmission, high specific gravity, pumping characteristics, stability, etc. for greater efficiency. To reduce the likelihood of an explosion, the propellants should ignite as quickly as feasible. The time it takes to open valves, the amount of time it takes for propellant to reach the combustion chamber, and other parameters all have an impact on how the fuel burns in the combustion chamber. The mass and velocity of the propellant leaving the nozzle determine the thrust. The flow through the nozzle is compressible, unidimensional, and isentropic. At a low flow velocity in the chamber, the cross-sectional area of the combustion must be three times greater than the cross-sectional area of the nozzle throat. Fueled by gaseous oxygen and gasoline, the propellant was used. The engine's combustion chamber was under 300 psi of pressure, and a 20-pound throat was suggested. This design has an O/F ratio of 2.5 and a particular impulse of roughly 260 seconds. Copper is utilized to make the combustion chamber and nozzle. Using mathematical formulas, the combustion chamber's thickness was estimated. The brass fuel injector has a spray angle of 75 degrees. The average oxygen flow rate is 60.96 m/s. The fuel and oxygen should always be at a 45-degree angle. The throat chamber was constructed utilizing solid bronze for simple machining. The injector measures approximately 30mm in length, 12mm across, and 30mm deep. A 0.7mm drill bit was utilized to create the center fuel injection hole. It had a 1.9mm oxygen inlet hole. The injector lid had a diameter of 40mm and a length of 28.5mm. The cooling jacket has an exterior diameter of 42.27mm and a length of 73.82mm. The gasoline tank was 300mm length and 4.5 inches in diameter. Two 12mm-diameter grips were included on the lid. For the desired thrust, the gas tank's pressure should be 400 psi. Cotton wire was used to construct an ignition system. The engine was then put together using the assembling procedure. A 220N thrust was generated during the engine test.

The impacts of particle properties on flame propagation behavior during organic dust explosions in a half-closed chamber were investigated by [3]. A study was done to determine how particle properties affected the ways that flame spread during organic dust explosions. Clearly, three long chain monobasic alcohols were chosen to conduct experiments in a small, partially closed container because they are solids at room temperature and have comparable physical and chemical properties. The flame propagation process was captured on high-speed video, and images of direct light emission were also taken. A small thermocouple was used to measure the flame's temperature. Analysis of the flame propagation characteristics and temperature profiles of the organic particle cloud was done based on the experimental results. It was discovered as a result that the particle materials, particularly volatility, significantly influenced the behavior of the flame. The propagation of the combustion zone is greatly impacted by particle concentration as well. The maximum temperature of the combustion zone rises with increasing particle concentration at lower concentrations, reaches a maximum, and then falls at higher concentrations. The association between the maximum temperature and the propagation velocity of the combustion zone suggests that conductive heat transfer predominates in the propagation of flames for the three different volatile dusts.

A study on flame stabilization in high-pressure liquid oxygen/methane rocket engine combustion was conducted by [4]. Using optical diagnostics, the flame stabilization in the vicinity of the injector of a subscale liquid propellant rocket combustion chamber has been studied. Liquid oxygen (LOX) and methane injections have been performed under appropriate operating conditions using a variety of single shear coaxial injectors. Regarding the oxygen thermodynamic critical point, three primary operating points span the range of sub-, near-, and supercritical levels. While gaseous methane was injected at nearly ambient temperatures of approximately 270 K, liquid oxygen was injected at usual temperatures of about 120 K. In line with other studies using LOX=H2 combustion, spontaneous OH chemiluminescence has been found to identify the flame anchoring zone close to the LOX post tip. When a binary mixture of oxygen and methane is compared to an oxygen and hydrogen system, different behavior can be seen. It is thought that this will affect the mixing and spray evolution under supercritical conditions. Both the ignition transient and the steady-state operating points are covered by the experimental inquiry. It has been discovered that, under comparable working conditions, the LOX=CH4 and LOX=H2 flames exhibit remarkably similar properties. All hot runs have revealed critical flame stabilization at the start-up transient. The influence of the injection parameters on the flame shape is comparable to earlier LOX/hydrogen studies under steady-state conditions.

Due to its availability, high energy content, and low cost, kerosene has been utilized extensively as a liquid fuel in rocket engines. To create a high-energy, combustion-based propulsion system, it is frequently combined with liquid oxygen as the oxidizer.

The performance characteristics of kerosene-based rocket engines have been the subject of numerous investigations. For instance, [5] conducted a study comparing the performance of kerosene-based rocket engines with various injector designs and discovered that a coaxial injector performed best in terms of thrust and combustion efficiency. Another study [6] looked at the performance and combustion stability of a kerosene-based rocket engine and discovered that using a swirling injector increased the engine's combustion stability.

Kerosene has also been investigated as a fuel for hybrid rocket engines, which use both a liquid oxidizer and a solid fuel. The performance of a hybrid rocket engine using kerosene as the fuel, for instance, was assessed in a study by [7] who discovered that it produced high levels of specific impulse and thrust.

The use of kerosene as a liquid propellant for rocket engines has thus far been demonstrated to be dependable and effective. Kerosene-based propulsion systems need to perform better and be safer, hence further research is required.

( [6] modelled the burning of kerosene in a fuel-rich environment. The primary power source for rockets and missiles is a liquid rocket engine. The liquid fuel that is fed into the gas generator powers the turbo pump and turbine. For the turbine to operate steadily, the combustion temperature must be kept below 1000 Kelvin. To achieve the desired thrust, the liquid propellant is burned in the combustion chamber of the engine at high temperatures and pressures. to improve the liquid rocket engine system's operational conditions and design. It's important to comprehend the kerosene oxidation thermodynamics in the gas generator. The 225 chemical species and 1800 reversible chemical reactions of Dagaut's kerosene oxidation process were utilized.

Kerosene oxygen combustion flow was simulated in a liquid rocket combustion chamber by ( [5]. Most launch vehicles and spacecraft employed liquid propellant rocket engines as their primary propulsion method throughout the early exploration of space. Here, kerosene is employed as the propellant and gaseous oxygen serves as the oxidizer. Kerosene and oxygen are both injected and mixed at high pressure using a swirl coaxial annular-shaped injector in the small recess. When using the convergence and divergence portion, De Laval nozzle type is used. This nozzle is used to create the acoustic conditions near the neck. The governing equations are approximated using the discretization approach. Methodology is used to convert the relevant calculus problem into an algebraic problem, which is subsequently solved using a computer. The outcome demonstrated that the CFD modeling allowed questions about the intricate interior dynamics of a rocket combustion chamber, which are difficult and expensive to see in a real laboratory experiment.

An investigation by [8] demonstrated that low Reynolds numbers result in flow separation, which lowers the nozzle's efficiency. Few research has particularly examined low Reynolds numbers in the past studies that have looked at the impact of nozzle shape on flow efficiency.

The authors examine much research that investigated how nozzle shape affected flow efficiency. They mention a 1995 study by Giguere et al. as an illustration, which revealed that the length of the nozzle significantly affects flow efficiency. They also mention a 2003 study by Li and Liou in which it was discovered that the nozzle's form significantly affects the flow impact.

The results of an experiment combining converging, diverging, and straight nozzle geometries in a wind tunnel demonstrated that the diverging nozzle had the maximum flow efficiency at low Reynolds numbers. Also, it was discovered that the nozzle's length significantly affected flow efficiency, with longer nozzles typically resulting in higher flow efficiency.

[9] examined methods for figuring out the essential measurements of simplified rocket engines. The thrust, specific impulse, combustion chamber pressure, and nozzle expansion ratio were among the important factors that the author listed as needing to be taken into account when designing a rocket engine. The method of characteristics, which entails solving a group of partial differential equations to ascertain the flow characteristics of the exhaust gases as they travel through the nozzle, is one of the analytical techniques examined by [9].With the use of this technique, the ideal nozzle geometry for a certain set of engine parameters can be found. The method of moments, which involves applying conservation principles to the flow of gases through the engine to derive the fundamental measurements of the combustion chamber and nozzle, is another analytical strategy examined by [9].With this technology, engineers can create engines with certain performance traits, such high thrust or high specific impulse.

In his investigation of rocket-engine transient regimes from 1977, [10] concentrated on the consequences of abrupt changes in the operating conditions on engine performance and stability. The rate of change of the engine operating conditions, the mechanical and thermal reaction of engine components, and the dynamics of the combustion process are a few of the crucial elements that the author highlighted as having an impact on the behavior of rocket engines during transient regimes.

Numerical simulation is one of the main techniques for investigating the transient regimes of rocket engines covered by [10].This method involves using computer models to solve the equations of motion and thermodynamics governing the engine's behavior during transient regimes. The prediction of engine performance and stability under a variety of operating situations, such as abrupt changes in engine thrust, pressure, and temperature, is possible through numerical modeling..

The use of test stands and thrust chambers to simulate engine operating conditions was covered in [10] assessment of experimental techniques for researching rocket-engine transient regimes. These tests can yield insightful information about how engine parts behave in transient regimes, including how the fuel injection system, combustion chamber, and nozzle react to alterations in operating circumstances.

The complexity of the combustion process, the impact of heat transport and thermal loads on engine components, and the necessity of precisely simulating engine dynamics are only a few of the major difficulties the author cited in his examination of rocket-engine transient regimes. [10] suggested the creation of more complex numerical models as well as the use of experimental data to evaluate and improve these models in order to overcome these issues.

The method for simulating transient combustion processes in rocket engines running on gaseous hydrogen and oxygen fuels was the subject of research by [11]. The dynamics of the combustion process, heat transmission and thermal loads on engine components, and the effects of turbulence on the fuel-air mixture are a few of the crucial elements that must be taken into account while simulating transient combustion processes.

Computational fluid dynamics (CFD) models are one of the main simulation methods examined by Zubanov et al. (2017). The combustion process and the movement of gases through the engine can be simulated using CFD models under a variety of operating situations. This method can give important insights into how engine parts behave during transient combustion processes, including how the fuel injection system, combustion chamber, and nozzle react to variations in operating circumstances.

The authors also discussed experimental techniques for examining transient combustion in rocket engines, including how to imitate engine operation using test stands and thrust chambers. The nozzle, combustion chamber, and fuel injection system responses to changes in operating circumstances are just a few of the engine components whose behavior these tests might shed light on during transient regimes.

The simulation of transient combustion processes in rocket engines is a difficult task that necessitates the fusion of several disciplines, including fluid dynamics, thermodynamics, and combustion science, according [11]. To effectively forecast engine performance and stability under transient regimes, they emphasized the significance of accurate modeling of the combustion process, heat transfer, and turbulence. To further expand our understanding of transient combustion processes in rocket engines, the scientists suggested that sophisticated CFD models and experimental techniques be developed in the future.

# CHAPTER THREE: THEORY

## 3.1 Design Equation

The main governing concept of the thrust obtained from our rocket engine is newton's third law and conservation of linear momentum of any system.

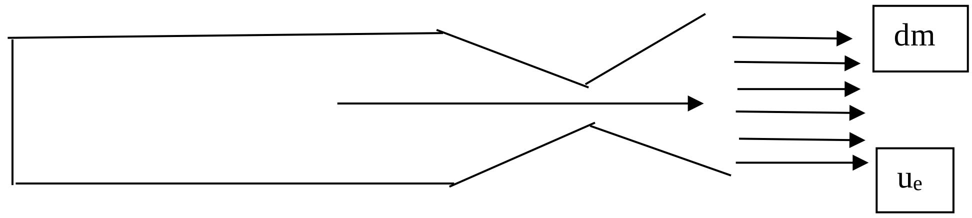



*Figure 7: Nozzle exit*

The mass of the system is fixed and there is no any flow into the system only there is outflow of exhaust gases which gave the thrust or the propelling velocity of the system in opposite direction. This happens due to the conservation of linear momentum. So, to get the maximum amount of thrust, increase dm*$u_e$ need to be increased. But there are some limitations for increasing both the parameters. The exhaust velocity is limited by the combustion chamber temperature. Mathematically we have, $u_e^2 = 2C_p * T_c$

Combustion temperature can't be increased due to combustion chamber material limitation. Similarly, the exhaust gases mass flow rate is limited by nozzle throat area. So, to design the engine calculate the parameters shown in the figure needs to be calculated.

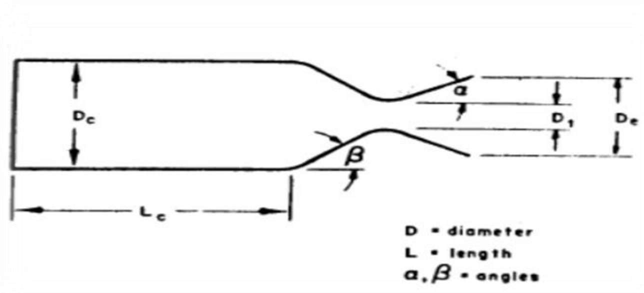



*Figure 8: Nozzle Parameters*

### 3.1.1 Nozzle Equation

For the required amount of thrust which is proportional to mass flow rate, we need to design specific throat area of our nozzle which is given by:

$$A_t = \frac{\pi}{4} d_t^2 = \frac{W_t}{P_t}\sqrt{\frac{RT_t}{\gamma g}}$$

From the above equation the required amount of throat diameter can be obtained.

From the assumptions of flow to be isentropic, compressible and one dimensional, relation between stagnation properties and properties of the gas can be obtained by following equations shown in figure below:

$$\frac{T_o}{T} = 1 + \left(\frac{\gamma - 1}{2}\right) M^2$$

$$\frac{p_o}{p} = \left[1 + \left(\frac{\gamma - 1}{2}\right) M^2\right]^{\frac{\gamma}{\gamma - 1}}$$

$$\frac{\rho_o}{\rho} = \left[1 + \left(\frac{\gamma - 1}{2}\right) M^2\right]^{\frac{1}{\gamma - 1}}$$

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

(Source: John D Anderson Jr., "Fundamentals of AERODYNAMICS" (Anderson, 2011))

$(T_0, p_0, \rho_0) = (T_c, p_c, \rho_c)$ Because the speed of gases can be assumed to be so small in combustion chamber and $(T, p, \rho)$ are the properties where the gases have Mach number $M$. $A^*$ is the area where mach number is 1 which we want our nozzle throat to be. A conical nozzle is used in our engine. Empirically, it is understood that the diverging angle should be no more than 15 degree and converging angle to be 60 degrees. Now for the exit diameter, experimental data shown in below figure will be used:

$\gamma = 1.2, P_{atm} = 14.7psi$ (Krzycki , Leroy J (1967))

*Table 1: Nozzle Parameters for various chamber pressures*

| $P_c$ | $M_e$ | $\frac{A_e}{A_t}$ | $\frac{T_e}{T_c}$ |
|---|---|---|---|
| 100 | 1.95 | 1.79 | 0.725 |
| 200 | 2.33 | 2.74 | 0.65 |
| 300 | 2.55 | 3.65 | 0.606 |
| 400 | 2.73 | 4.6 | 0.574 |

From this table we can calculate our exit diameter as:

$$A_e = \pi D_e^2/4 = 2.74A_t$$

$$D_e = \sqrt{\frac{4 * 2.74A_t}{\pi}}$$

### 3.1.2 Combustion Chamber Equations

Two of the parameters for combustion chamber needs to be calculated. They are combustion chamber length and combustion chamber diameter. For combustion chamber diameter, (Krzycki, L. J. (2018)) an empirical formula based on the minimization of losses of flow energy inside chamber which states that combustion chamber cross sectional area should be at least 3 times of throat area.

$$A_c \geq 3A_t$$

So, combustion chamber diameter can be obtained from the above expression. Now for combustion chamber length one term known as characteristic chamber length ($L^*$) needs to be defined which is defined as the required amount of combustion chamber volume for complete combustion and is given by:

$$L^* = \frac{V_c}{A_t}$$

Where $v_c$ is the chamber volume (including the converging section of the nozzle) and $A_t$ is the nozzle throat area. Therefore, we can find combustion chamber volume as:

$$v_c = A_c * L_c + convergent\ volume$$

Using the above equation, the required length can also be obatined. Now for combustion chamber thickness there is an equation given by S=PD/2t, Where S is maximum allowable stress, P is chamber pressure, D is diameter and t are thickness.

### 3.1.3 Injector

Injectors are one of the most important components of engine which function is to mix fuel and oxidizer thoroughly inside chamber and assure combustion stability. There are different types of injectors in use. some of commonly used are shown below:

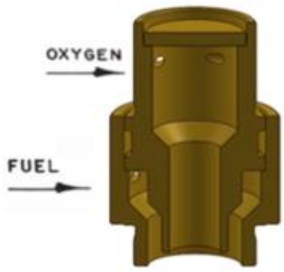



*Figure 9: Coaxial Swirl Cylinder*

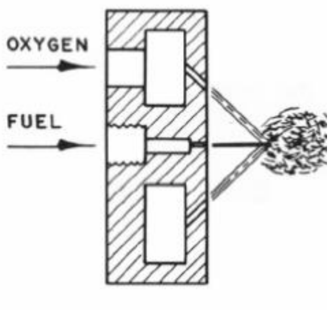



*Figure 10: Impinging Stream*

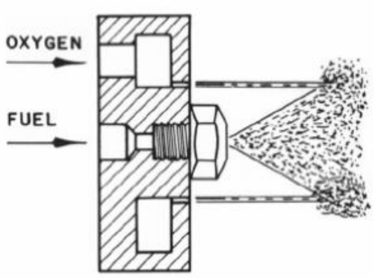



*Figure 11: Spray Nozzle*

The mass flow rate through the orifice of injector can be given by the expression shown below:

$$\dot{m} = C_d A\sqrt{2g\rho\Delta p}$$

Where $C_d$=coefficient of discharge, A=area of orifice, $\Delta p$=pressure drop across the orifice, $\rho$=density of fluid passing through orifice.

The injection velocity is given by the expression:

$$V=C_d\sqrt{2g\frac{\Delta p}{\rho}}$$

## 3.2 Propellent Choice

Liquid rocket engine has variety of propellent combinations. some of them will be tabulated below:

*Table 2: Performance of some propellent ( [12]*

| Propellant Combination Oxidizer/Fuel | Combustion Pressure, MPa | Mixture Ratio | Flame Temp (°C) | $I_{sp}$, sec |
|---|---|---|---|---|
| Liquid Oxygen & gasoline | 2.068 | 2.5 | 3020 | 242 |
| Gaseous Oxygen & gasoline | 2.068 | 2.5 | 3170 | 261 |
| Gaseous Oxygen & gasoline | 3.450 | 2.5 | 3240 | 279 |
| Liquid Oxygen & JP-4 (jet fuel) | 3.450 | 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 | 3.450 | 3.5 | 2480 | 363 |
| Red fuming nitric acid & JP-4 | 3.450 | 4.1 | 2840 | 238 |

In field of amateur rocketry, such propellent should be used which are easily available, safe, easy to handle and inexpensive. So, considering these points, use of gaseous oxygen and hydrocarbon as oxidizer and fuel respectively is recommended. The specific heat ratio of gaseous product of oxygen and hydrocarbon can be assumed to be

1.2. Different propellent have different flame temperature at different O/F ratio which can be seen in figure below:

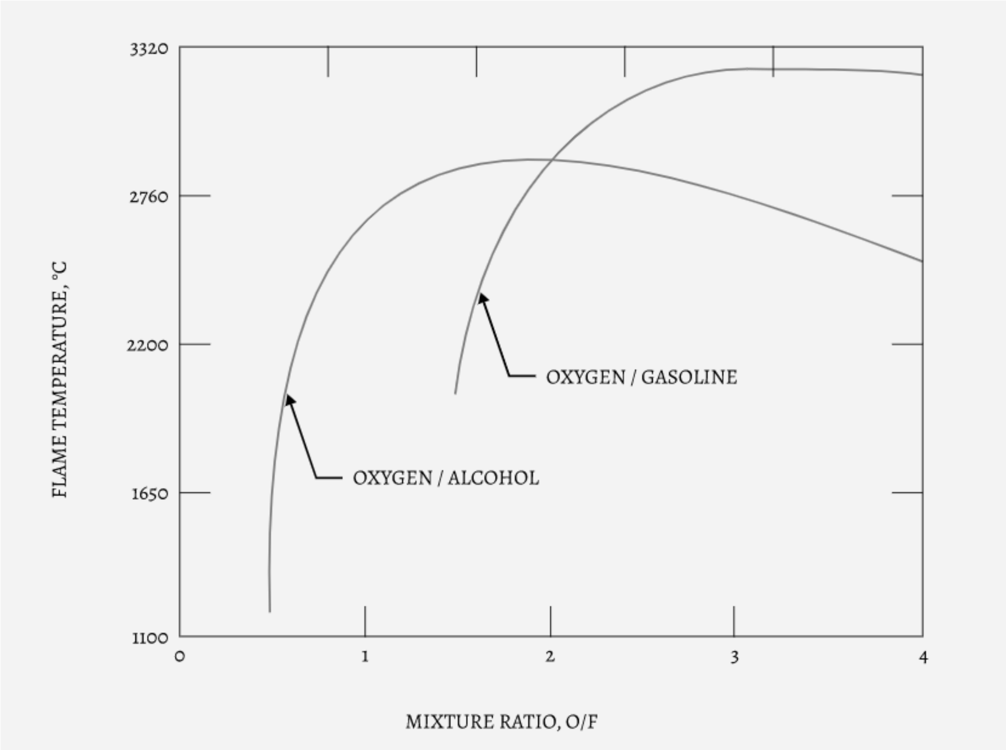



*Figure 12: Graph of O/F ratio vs Flame temperature ( [12]*

Flame temperature also varies with the combustion chamber pressure which can be seen by the figure below:

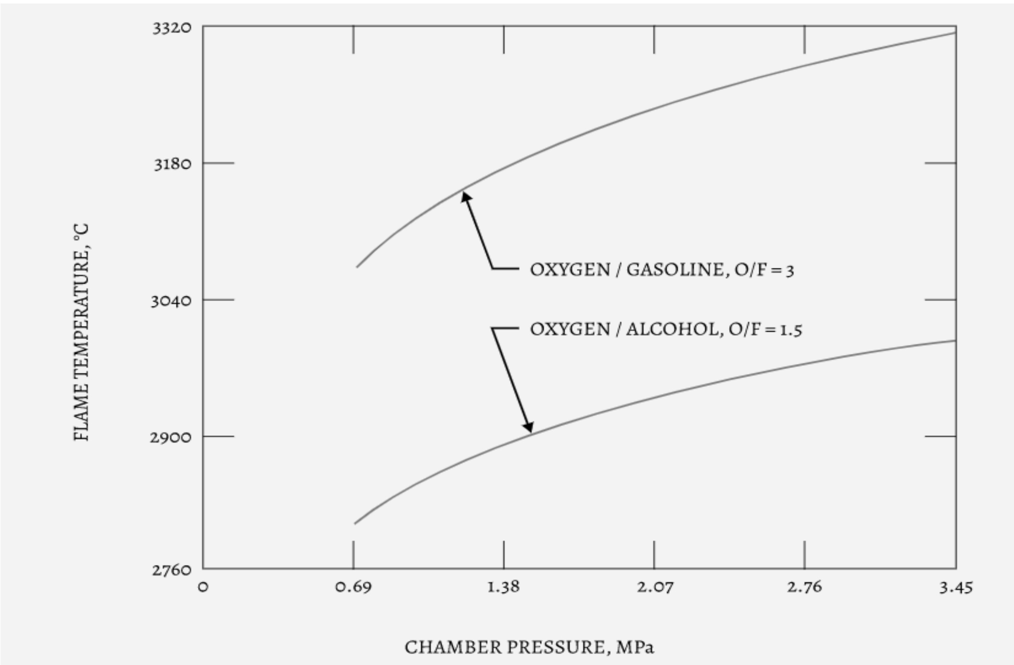



*Figure 13: Graph of chamber pressure vs Flame temperature [12]*

Different specific impulse is obtained at different combustion chamber pressure for a particular O/F ratio of a specific fuel. For gaseous oxygen and hydrocarbon as fuel, the below shown graph has been obtained:

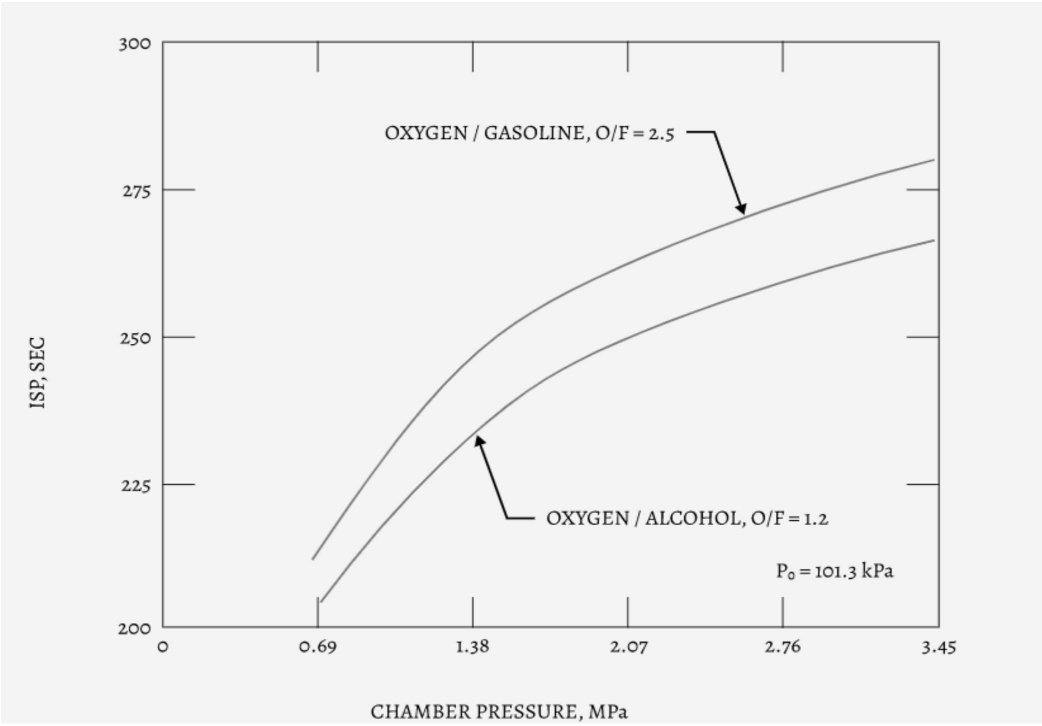



*Figure 14: Graph of specific impulse vs chamber pressure [12]*

## 3.3 Rocket Engine Calculations

### 3.3.1 Mass Flowrate Calculation

The rocket engine is designed to have chamber pressure 300 psi and capable of producing thrust of magnitude 200N. For this O/F ratio 2.5 is chosen. There is specific impulse corresponding to this O/F ratio as 261 s for kerosene. So, the mass flow rate of propellant will be given by:

$w_t = \frac{F}{I_{sp}} = \frac{200}{261} = 0.766N/s$=0.078kg/s

Now oxygen flow rate will be given by:

$$w_o = w_t * \frac{r}{r+1} = 0.0597kg/s$$

Where r =O/F ratio

Similarly, fuel mass flow rate will be given by:

$$w_f = w_t - w_o = 0.0183 kg/s$$

### 3.3.2 Nozzle Parameters

After knowing the propellant mass flow rate, calculation of nozzle throat area is done which is given by the equation shown below:

$$A_t = \frac{\pi}{4} d_t^2 = \frac{W_t}{P_t} \sqrt{\frac{RT_t}{\gamma g}}$$

So we have $w_t$,y,g=9.8m/$s^2$, R=350J/(kg.K) .Now the remaining two parameter needs to be calculated using the above equations :

$$\frac{T_o}{T} = 1 + \left(\frac{\gamma - 1}{2}\right) M^2$$

$$\frac{p_o}{p} = \left[1 + \left(\frac{\gamma - 1}{2}\right) M^2\right]^{\frac{\gamma}{\gamma-1}}$$

At throat, M=1 and$P_0 = 2.068 MPa$ , $T_0 = 3170^0 C$ then the value obtained from equations will be $T_t = T = 2881.81$℃ , $P_t = P = 1167332.087$ pascal.

So, by using all these data, the required values of throat diameter came as:

Area of throat=$A_t$=6.018 $cm^2$ and diameter at throat =$D_t$ =2.77$cm$.

For perfect expansion in nozzle, we get area ration as 3.65 from table 1. using this we can get area of exit=$A_e$=22.18 $cm^2$ and exit diameter =5.31 cm.

### 3.3.3 Combustion Chamber Parameter

Using the concept mentioned above, the required chamber diameter and length can be calculated as:

Combustion chamber area=3*area of throat=18.054$cm^2$

Combustion chamber diameter=4.8cm

Combustion chamber length= 62.44cm.

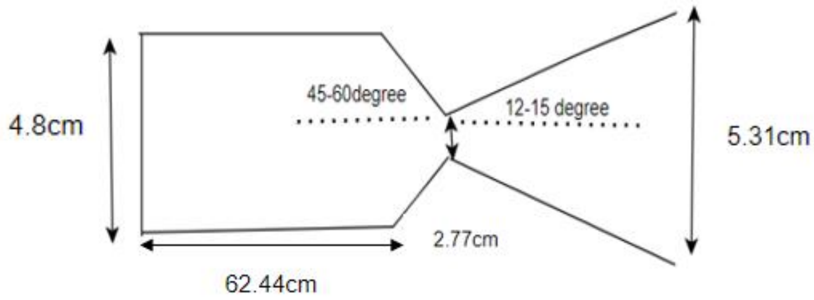



*Figure 15: Combustion Chamber Parameters*

# CHAPTER FOUR: METHODOLOGY

This chapter describes the methods, tools and the materials which are used to accomplish the project.

## 4.1 Algorithm and Flowcharts

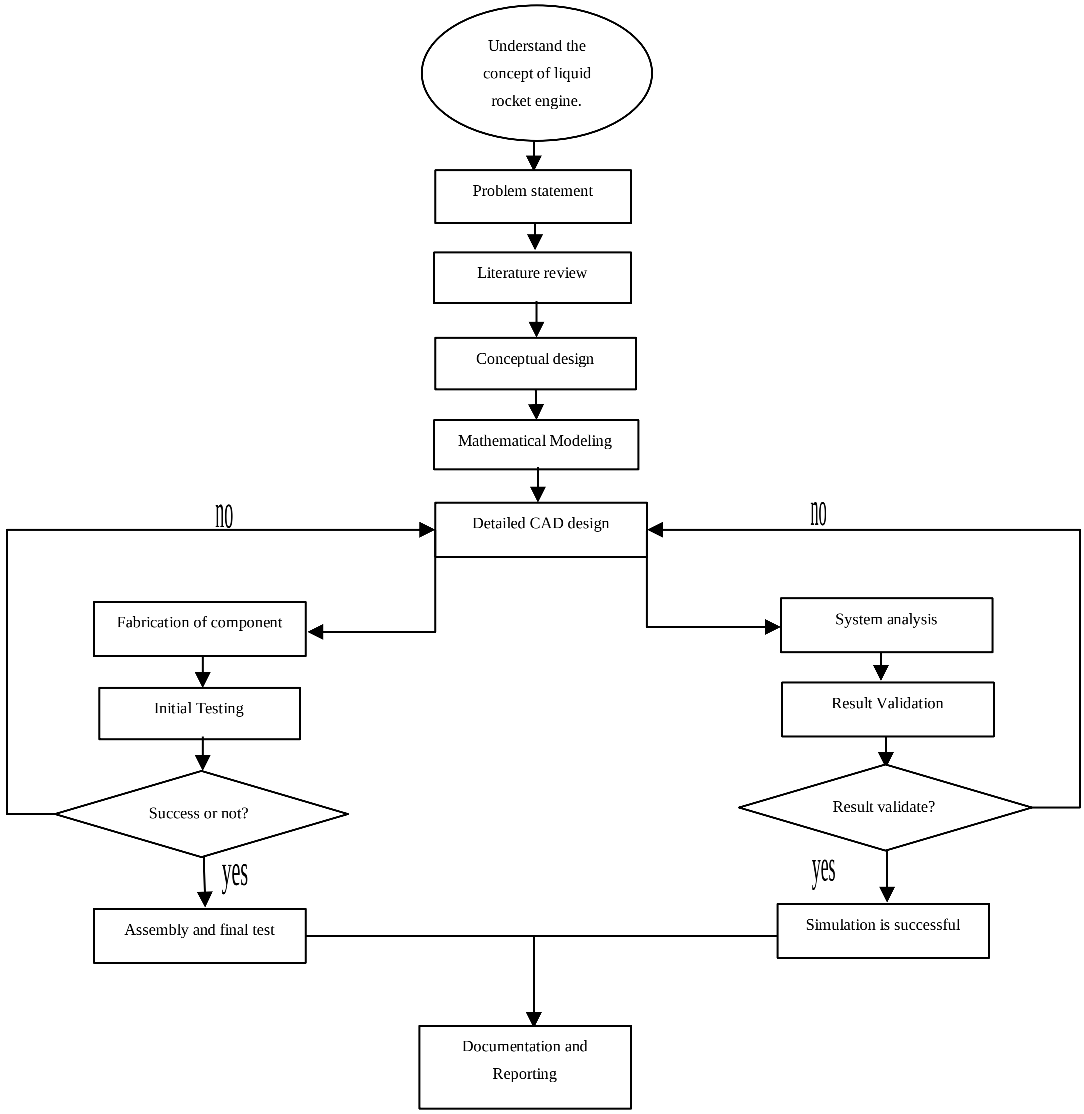



*Figure 16: Methodology Flowchart*

The algorithm of our project is as follows:

- First, the concept of liquid rocket engine is understood through the study of the related papers and literature review is done.
- After that the mathematical modeling of the engine parameters is done in excel

- Then SOLIDWORKS is used to create the detailed CAD design of our project
- The materials needed for our project is listed and market research is done to find out the availability of the required product
- Simulation of our project is carried in ANSYS, and results are analyzed
- The fabrication of the components of engine is done
- The components are tested to check their capability
- Finally, after the successful completion of the testing of components the final test of the liquid rocket engine is carried out and the obtained results are validated.

The process of creating a liquid rocket engine is intricate and normally entails the following steps:

Developing a liquid rocket engine is a complex process and it takes great attention to detail in design, manufacture, and testing, as well as a solid understanding of the physics involved, to construct a liquid rocket engine. In this project at first different journal articles, book sections and related papers were studied, and literature review was done and some basic ideas was gained from that. After that problem statement was defined and the design process was begun which is further explained below:

1. Conceptual design: The development of the engine's fundamental design begins at this point. The propellant type, engine cycle, thrust chamber, nozzle, injector, and other parts are all included in the design. The desired mission objectives and performance goals serve as the foundation for the design.

2. Mathematical Modeling: The development and study of liquid rocket engines rely heavily on mathematical modeling. Mathematical modeling is mostly used to simulate engine behavior under various operating conditions and to forecast its performance characteristics. The following significant elements are modeled in a liquid rocket engine:

- Combustion: The combustion process that takes place in the thrust chamber is simulated using mathematical models. This involves heat transfer, the creation of combustion products, and the chemical processes that take place between the propellants.

- Fluid dynamics: To simulate the flow of propellants via engine parts like pumps, valves, and injectors, mathematical models are used. Modeling the pressure drop, heat transfer, and other flow properties of the propellant flow is part of this.

- Heat transfer: The heat transmission between the combustion chamber and the engine parts is simulated using mathematical models. This covers the simulation of heat transport in the nozzle, the walls of the combustion chamber, and other engine parts.

- Structural Analysis: The structural behavior of the engine components under various operating situations is simulated using mathematical models. This includes simulating the stresses and strains on the engine's parts, including the nozzle and combustion chamber.

3. Detailed CAD design: A thorough CAD design of a liquid rocket engine might incorporate 3D models of the engine parts, such as the combustion chamber, nozzle, and turbopump, together with details on their measurements, material characteristics, and tolerances. Annotations and notes about the design, such as guidelines for assembly or production, may also be included in the CAD model.

4. System analysis: Instead of studying individual components, the system analysis phase of a liquid rocket engine concentrates on analyzing the behavior and performance of the entire system. Examining the relationships between various parts and the overall system reaction to various operating situations are both necessary for this. The finite element analysis and computational fluid dynamics is carried out in simulation software like ANSYS, and the performance is analyzed.

4. Fabrication of components: The fabrication of the engine's components comes after the design has been completed. The use of additive manufacturing processes, welding, or machining techniques are used to fabricate the desired components like nozzle, combustion chamber, bench setup etc.

5. Assembly and testing: After fabricating the engine parts, the next step is to assemble the engine and put it through a series of tests to check its functionality and make sure it complies with design specifications. This involves testing that involve firing an engine while it is stationary in order to gauge its stability and performance.

6. Documentation and Reporting: After carrying out all the tests and simulations successfully the results, process, methods, materials and analysis of all tasks performed to complete the project are reported in a systematic way and proper documentation is done so that it can be looked up to in the future if anyone has interest in doing the similar projects.

## 4.2 Systems Designs and Block Diagrams

The CAD model is designed using the SOLIDWORKS software. This helped us to visualize the actual realistic model of our engine. The CAD design of initially proposed model is shown below:

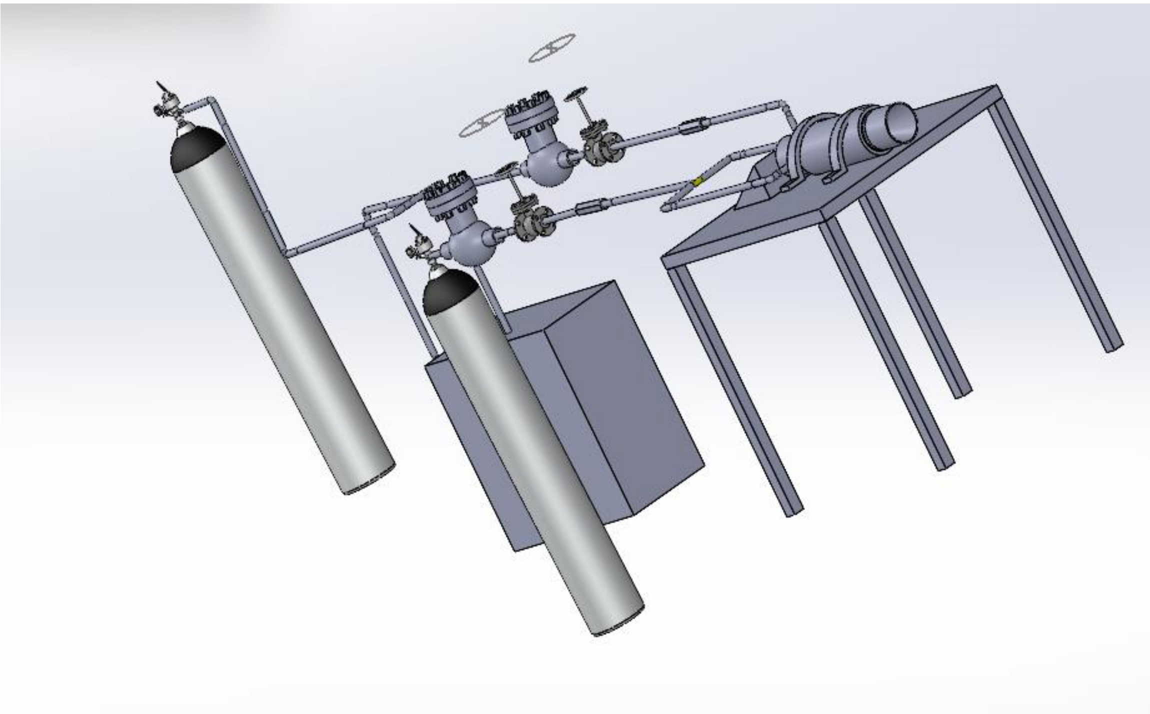


*Figure 17: Detailed CAD design of proposed model*

The model consists of fuel and oxidizer tank, all fittings, bench setup, combustion chamber and nozzle. This model can be more understand by the schematic diagram shown below:

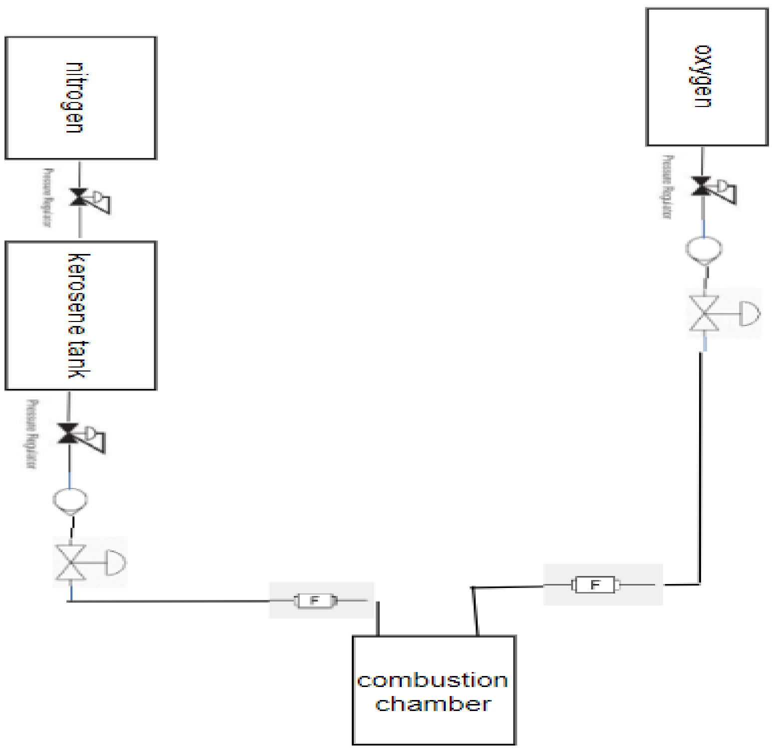



*Figure 18: Block diagram for the project*

Due to safety concern and the resource constraint, the proposed model shown in figure 17 couldn't be built up and according to the advice from the supervisor one small prototype was built whose CAD design is shown below:

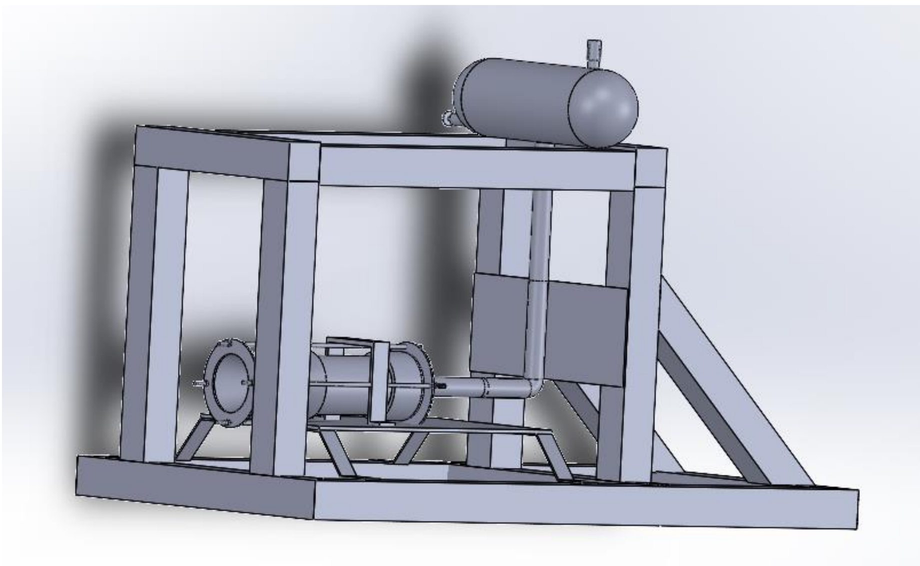


*Figure 19: CAD design of the prototype design*

## 4.3 Fabrication

Different components of the liquid rocket engine such as the combustion chamber, nozzle and injector were fabricated which is further explained below:

### 4.3.1 Combustion Chamber

A mild steel hollow pipe of required diameter and thickness was used for the combustion chamber which was available in the market. Firstly, three materials for the combustion chamber were shortlisted which were aluminum, copper, and mild steel. The combustion chamber made up of copper would have been best suited for our project because of its high melting point and high thermal conductivity. But due to its availability in the Nepalese market it couldn't be possible, so mild steel was preferred.

*Figure 20: Mild Steel tube*

### 4.3.2 Nozzle

The nozzle for our project was brought from the market which was a hollow nozzle of convergent-divergent nature, and the nozzle holder was manufactured in the workshop lab of campus by using the rod made up of mild steel.

### 4.3.3 Injector

Injector is one of the most important components of rocket engine which better performance is necessary for combustion stability. As already mentioned in theoretical design, swirl injector was used for our 200N thrust engine and for our prototype the burner of stove was used as an injector.

## 4.4 Ignition System

Our rocket engine will be ignited using ignition coil from outside. The diagram of ignition system is shown here:

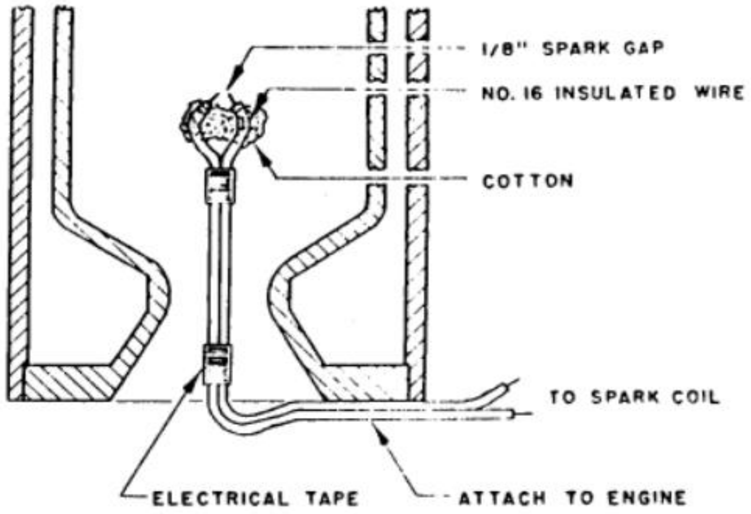



*Figure 21: Injection System*

## 4.5 Fuel Storage

For the storage of the fuel which is kerosene the storage space of the stove was used for the prototype design. Any tank which is good enough to handle the desirable pressure of fuel and gaseous oxygen could have been used to store the fuel and oxidizer. But as already mentioned the inbuilt storage tank of kerosene stove was used for the storage of the fuel and oxygen was supplied through the compressor.

## 4.6 Feeding System

There are two types of feeding system. They are pressure feed and pump feed. For simplicity pressure feed system was used. Oxygen cylinder available in market are already available in range of 1500-2000 psi pressure so an extra feeding system isn’t needed since its pressure is greater than combustion chamber pressure. But for fuel extra pressure feeding system is needed. For that purpose, an extra nitrogen tank was used which is be highly pressurized and connected to fuel tank for pressure supply. For small prototype engine, the fuel was pressurized manually, and oxygen was sent from the tank which was already pressurized.

## 4.7 Ansys Simulation (CFD)

### 4.7.1 Geometry:

The fluid domain is like the actual test setup geometry. The length of fluid domain is 9inch which is length of the combustion chamber, and the width is 3inch as the diameter of tube. There

Was two setup one where no nozzle used and one where nozzle was used. The domain of nozzle used setup can also be seen in figure. Nozzle was just of converging section with converging diameter of 1 inch. There was two oxidizer inlet point each of 0.15 inch and one fuel inlet point 0.2 inch which can be seen in geometry. The geometry shown below is half portion of domain which is symmetric about axis.

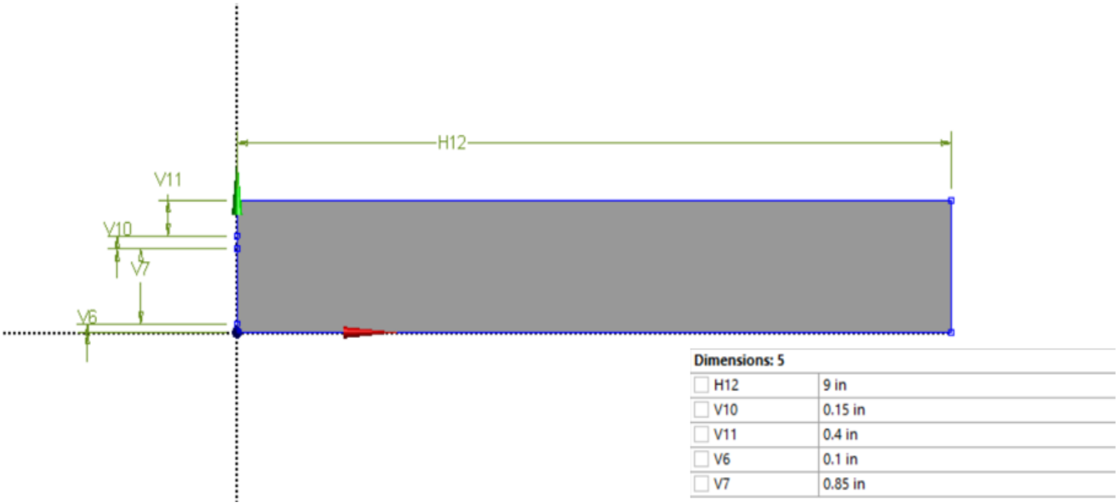



*Figure 22: Geometry without nozzle*

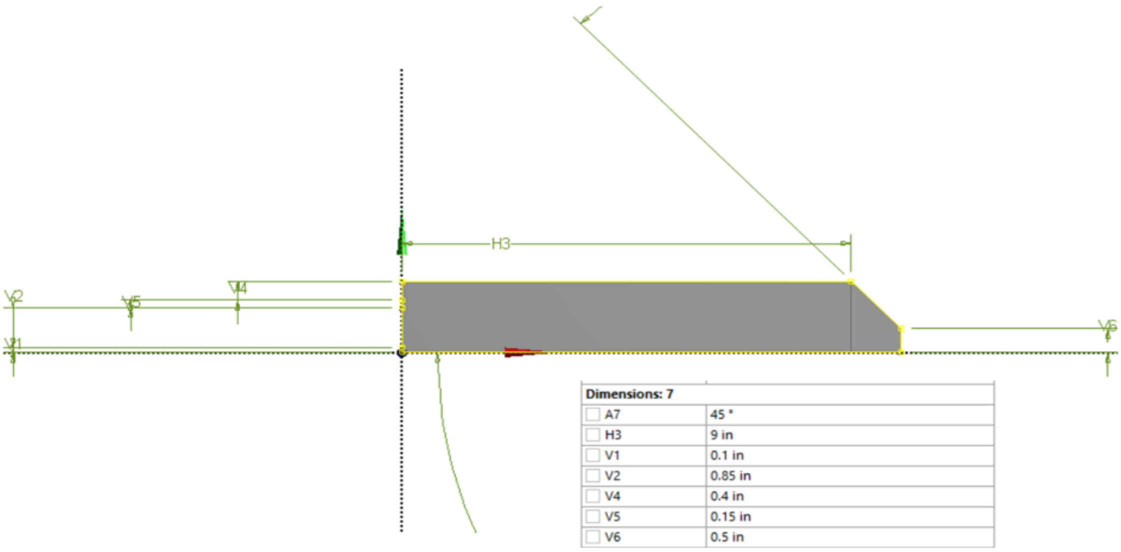



*Figure 23: Geometry with nozzle*

### 4.7.2 Meshing

Face meshing is used to generate structured quadrilateral mesh. Element sizes were decreased, and mesh independence test was carried out. To conduct mesh independence test, a set of parameters (element size, number of nodes, number of elements) is given, according to which 10 different meshes are generated and their corresponding value of outlet temperature is studied.

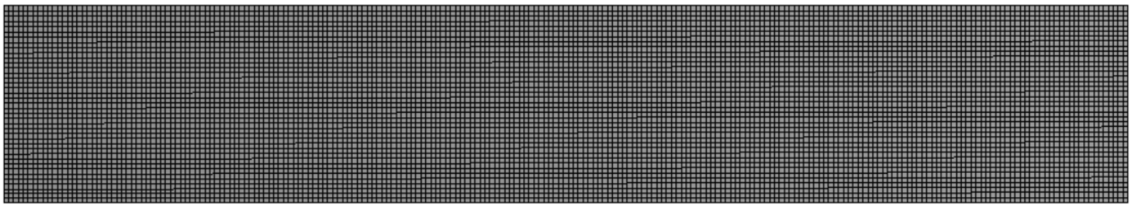


*Figure 24:Mesh with 9200 nodes and 8931 elements*

### 4.7.3 Mesh independence test:

The results were carried out for different number of elements and nodes for both cases (nozzle and without nozzle) and were plotted in excel.

# The parameters defined in the project are:
# P1 - Mesh Element Size P2 - Mesh Nodes P3 - Mesh Elements P4 - temperature outlet [K]
#
# The following header line defines the name of the columns by reference to the parameters.

| Name | P1 | P2 | P3 | P4 |
|---|---|---|---|---|
| DP 0 | 0.005 | 423 | 368 | 1561.105 |
| DP 1 | 0.004 | 638 | 570 | 1650.142 |
| DP 2 | 0.003 | 423 | 988 | 1357.961 |
| DP 3 | 0.002 | 2300 | 2166 | 1463.021 |
| DP 4 | 0.001 | 9200 | 8931 | 1548.741 |
| DP 5 | 0.0005 | 35266 | 34732 | 1612.575 |
| DP 6 | 0.0004 | 55008 | 54340 | 1548.973 |

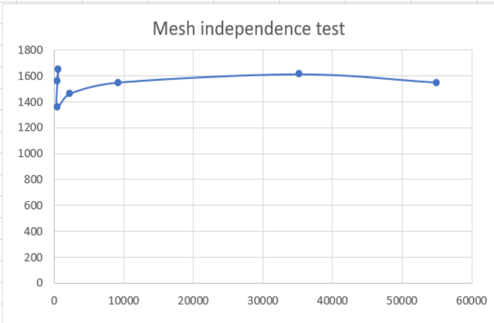
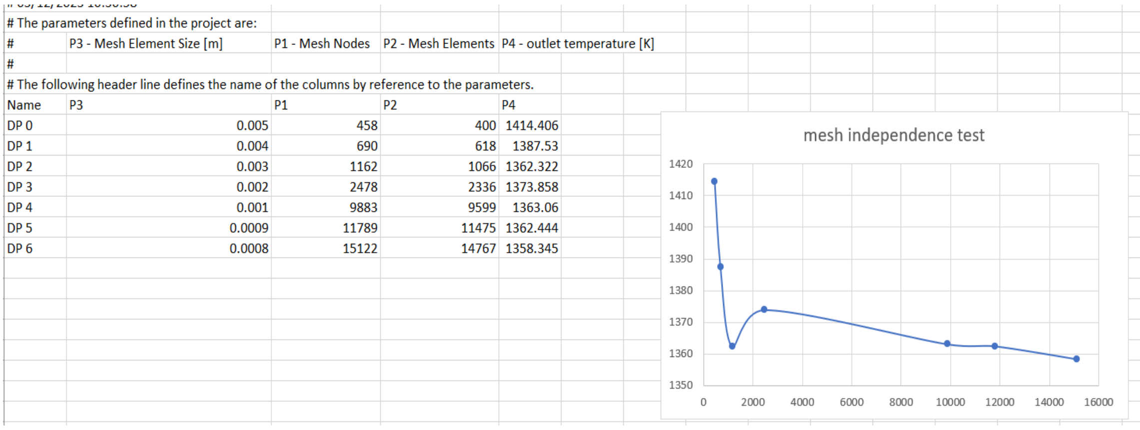



*Figure 25: Case I (Without nozzle)*



# The parameters defined in the project are:

| # | P3 - Mesh Element Size [m] | P1 - Mesh Nodes | P2 - Mesh Elements | P4 - outlet temperature [K] |
|---|---|---|---|---|
| # | | | | |

# The following header line defines the name of the columns by reference to the parameters.

| Name | P3 | P1 | P2 | P4 |
|---|---|---|---|---|
| DP 0 | 0.005 | 458 | 400 | 1414.406 |
| DP 1 | 0.004 | 690 | 618 | 1387.53 |
| DP 2 | 0.003 | 1162 | 1066 | 1362.322 |
| DP 3 | 0.002 | 2478 | 2336 | 1373.858 |
| DP 4 | 0.001 | 9883 | 9599 | 1363.06 |
| DP 5 | 0.0009 | 11789 | 11475 | 1362.444 |
| DP 6 | 0.0008 | 15122 | 14767 | 1358.345 |



*Figure 26: Case II (With nozzle)*

From grid independence test for case one when no nozzle was used and when nozzle was used, element size of 0.001 are set to give best result with least number of elements and nodes.

### 4.7.4 Physics setup and solver:

Physics setups were similar for both cases. To account combustion, series transport module was used.

### 4.7.5 Setup

*Table 3:Setup parameters*

| | |
|---|---|
| Solver | Type: Pressure Based<br>Velocity Formulation: Absolute<br>Time: Steady<br>2D Space: axis-symmetric |
| Models | Energy on<br>SST k-omega<br>Series transport, reactions |
| Materials | Fluid Property:<br>Kerosene-air mixture imported from chemkin library |

| | |
|---|---|
| Boundary Conditions | Air inlet -0.02kg/s<br>Fuel inlet-0.005466kg/s<br>Outlet: Pressure = 0 Pa.<br>Wall Motion: Stationary<br>Shear Condition: No Slip |

### 4.7.6 Solution

*Table 4: Solution Parameters*

| | |
|---|---|
| Initialization | Hybrid Initialization |
| Calculation | No. of iterations: 1000 |

# CHAPTER FIVE: RESULT AND DISCUSSION

## 5.1 Setup

Liquid rocket engine is a non-air breathing engine which carry fuel and oxidizer in liquid form within itself and utilize the energy released during combustion of fuel and oxidizer for propulsion. Any prototype of engine must contain storage for fuel and oxidizer, combustion chamber for combustion and nozzle for expansion of flow from chamber. The

Schematic of our prototype can be shown in figure 1. Due to safety concern associated with the use of oxygen cylinders, Compressor were used as a source of oxidizer. Kerosene stove tank were used as fuel tank and was compressed manually. GI tube of 3 inch were used as a chamber and GI fittings were used to connect fuel tank to chamber which can be shown in figure 2. Gate valves are used to control the flow of oxidizer and fuel.

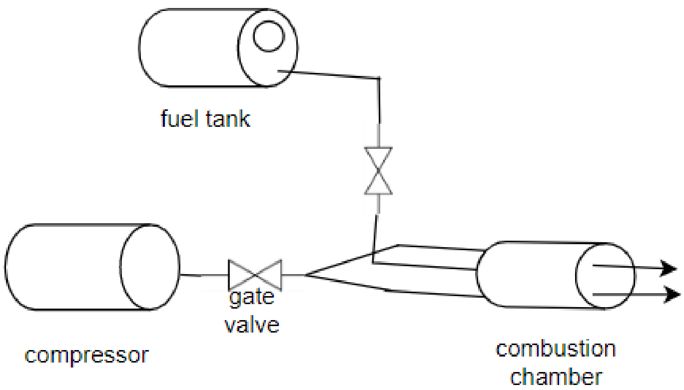



*Figure 27: Schematic diagram of Prototype design*

## 5.2 Leakage test

After setting up the prototype, leakage test were carried by flowing pressurized fuel and oxidizer through the fittings into the chamber. Leakage test are one of the serious and important tests carried in aerospace industries where leakage hold extreme danger. At first the fittings were cleaned and then fitted using water tape in joints. After fitting all joints, m-seal were used on joints to completely eradicate the leakage chance. Then sometimes were left for m-seal to get set. After which fuel was put in fuel tank and manual pressure was given for it to flow and then fuel coming out from any point was noticed. At first some joint had little leakage which was resolved by reusing m-seal.

## 5.3 Test and Observation

After setting up the prototype and carrying the leakage test, it was time for the final test. For conducting test, the following step were taken. The first step is to fill the fuel tank and compressor tank. After filling them, all joints and linkage were tested and were fixed in order to prevent them for moving randomly due to any pressure generated during combustion. After following these steps, our test setup get ready for ignition. Due to safety concern, our test was started with low mass flow rate of fuel and oxidizer and gradually the flow rate was increased. Many tests were carried out with different stoichiometric ratio of fuel and oxidizer.

### 5.3.1 Test 1

This test was carried out with low mass flow rate of air and fuel. This test was successfully conducted with the burn time of 30sec. The flame can be seen oozing from the top of chamber only. This was due to oxidizer being feed from the top side only which can be seen in background where there is a pipe on the top side of chamber. There was single point of oxidizer feeding. So, the second test was carried out with two points of oxidizer feeding.

*Figure 28: Test 1*

### 5.3.2 Test 2

This test was different from first one in regard to oxidizer feeding point. First test was carried with the oxidizer feeding from one point where flame was observed to be coming from upper surface only. In this test flame can be seen to be coming from whole region uniformly. After setting up the setup, test was carried out again from small mass feed of oxidizer and gradually increased to higher. When air flow was increased bluish flame was seen which shows that it is fuel lean combustion and can be seen in figure.

When the air mass flow rate was reduced by closing the gate valve between compressor and combustion chamber, yellowish flame was obtained which shows us that fuel rich combustion is taking place. This condition can be seen in figure below:

*Figure 29: Test 2 (Fuel Lean Combustion )*

*Figure 30: Test 2 (Fuel Rich Combustion)*

### 5.3.3 Test 3

The Third and the final test was carried out in similar manner to second test but in second test nozzle wasn't used whereas in third test nozzle was used. Combustion was carried out in chamber during second test in different stiochimetric condition.For nozzle, we used readily available reducer of 3inch-1inch.After fitting with combustion chamber , test were carried out for 15 sec.The result can be seen in figure.

*Figure 31: Test 3 (Final Test)*

## 5.4 Analytical Approach for Thrust Approximation

Analytical Approach is the exact approach that comprises use of theoretical equations of different discipline used to model the system. It is an ideal approach guided by a set of assumptions and gives a theoretical hypothetical and ideal system design. The function of a liquid rocket engine is to generate thrust through chemical combustion reaction; release of the thermal energy derived from the chemical reaction undergoes between propellants. The generated force imparts a momentum to the elements of combustion products. In accordance with Newton's 3rd law, the momentum in opposite direction is imparted to rocket. In practice, high temperature and high-pressure gases are produced in combustion chambers through chemical reactions of liquid propellants. These gases are ejected through a nozzle at high velocity to the surrounding environment. Analytical approach is done via two methods which are based on some additional assumption parameter discussed later in this chapter.

### 5.4.1 Predefined performance of propellent combinations:

Fuel and oxidizer : kerosene and atmospheric oxygen

Feed pressure [$P_f$] : 50 psi oxygen from centrifugal compressor

Feed temperature [$T_f$] : 15°C = 288 K

### 5.4.2 Stoichiometry of Propellant Combustion Process:

For a complete combustion process, kerosene burns in the presence of oxygen and gives out carbon-dioxide and water vapor as by-product. The chemical equation can be given as (Eq. 1)

$$C_{12}H_{26} + 18.5O_2 \quad \rightarrow 12CO_2 + 13H_2O \tag{1}$$

Hence 18.5 mole of oxygen is required of complete combustion of one mole of kerosene.

The heat released during combustion is what makes kerosene a useful fuel for powering various engines.

Mixture Ratio may be defined as the amount of oxidizer (oxygen in this case) required to be mixed with the fuel for a complete combustion process. Mixture ratio for Eq. 1

chemical reaction can be given by Eq. 2

$$r = \frac{weight\ of\ oxidizer}{weight\ of\ fuel} = \frac{n_{o_2} * m_{o_2} * g}{n_{c_{12}h_{26}} * m_{c_{12}h_{26}} * g} = \frac{18.5*32}{170*1} = 3.48 \tag{2}$$

The temperature after the combustion so formed can be estimated by the temperature of flames at the combustion process. This temperature is maintained throughout the combustion process. Hence, the temperature inside the combustion chamber can be givenby Eq. 3

$$T_c \;=\; 2093\ K^{[1]} \tag{3}$$

The pressure inside the combustion chamber after the combustion process varies according as the temperature varies after combustion and is dependent of feed pressure and feed temperature. So, the pressure inside a combustion chamber can be given by

Eq $$P_c = P_f \frac{T_c}{T_f} = 50 * \frac{2093}{288} = 363.36psi \tag{4}$$

The pressure at the exit of the nozzle can be defined as the pressure exerted by the burnt gases leaving out the nozzle and can be given as $P_e = P_{atm} = 14.7$ psi, for a perfectly expanded nozzle. However, in the case of under-expanded or over-expanded nozzle, the exit pressurewill not be equal to atmospheric pressure.

### 5.4.3 Effective Average Molecular Mass [ £ ]

It is the average effective mass of the molecules of the gas taking part in the chemical reaction which can be given by Eq. 5

$$£ = \frac{\sum_{i=1}^{m} n_i M_i}{\sum_{i=1}^{m} n_j} = \frac{1*170+18.5*32}{12+13} = 30.48 \qquad (5)$$

Where, $M_i$ is molecular mass of reactants; $n_i$ and $n_j$ are number of moles of reactants and products.

### 5.4.4 Exhaust Velocity [U]

It is the average velocity of the burnt gases leaving out the nozzle. It is responsible for thegeneration of the thrust and specific impulse and can be given by Eq. 6

$$U = \sqrt{[1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}] \frac{2\gamma*R*T_c}{(\gamma-1)*£}} = 1684 m/s \qquad (6)$$

Where, $\gamma$ = 1.2 Specific Heat Ratio; R = 8314.3 J $g^{-1}$ $K^{-1}$ Gas Constant;$P_e = 14.7 psi$ assuming nozzle to be perfectly expanded.

### 5.4.5 Specific Impulse [$I_{sp}$]

It is a measure of how efficiently a reaction mass engine creates thrust. It is exactly proportional to exhaust gas velocity. A propulsion system with a higher specific impulse uses the mass of the propellant more efficiently i.e., The higher the specific impulse, the less propellant is needed to produce a given thrust for a given time and the more efficient the propellant is. Specific Impulse for this system is given by Eq. 7

$$I_{sp} = \frac{U}{g} = \frac{1634}{9.8} = 171.85\ s \qquad (7)$$

### 5.4.6 Thrust calculation:

For perfectly expanded nozzle, thrust can be calculated by eq. 8

$$T = \dot{m} * I_{sp} \qquad (8)$$

By varying the mass flow rate of propellent, the required amount of thrust can be generated with the relation as shown in eq.8.

### 5.4.7 Chamber thickness calculation:

Combustion chamber pressure : 400 psi=2.758Mpa

Ultimate stress of GI pipe ( $\sigma_{ult}$ ) : 320MPa=46.1kpsi

Allowable working stress for factor of safety 6($\sigma_{all}$) : 7.68kpsi

Diameter of chamber : 3 inch

The thickness of chamber can be calculated using the eq.9,

$$t = \frac{P * D}{\sigma_{all}} = \frac{400 * 3}{7680} = 0.15625 inch$$

$$= 4mm$$

## 5.5 Result Obtained From CFD

The CFD solution was carried to obtain maximum temperature of combustion between kerosene and air in combustion chamber. The solution was carried for 1000 iteration. The value of maximum flame temperature in normal combustion in our setup came out to be 2243 K where in nozzle case it came out to be 2004 K which is quite closer to theoretical value of adiabatic flame temperature of 2366 K.

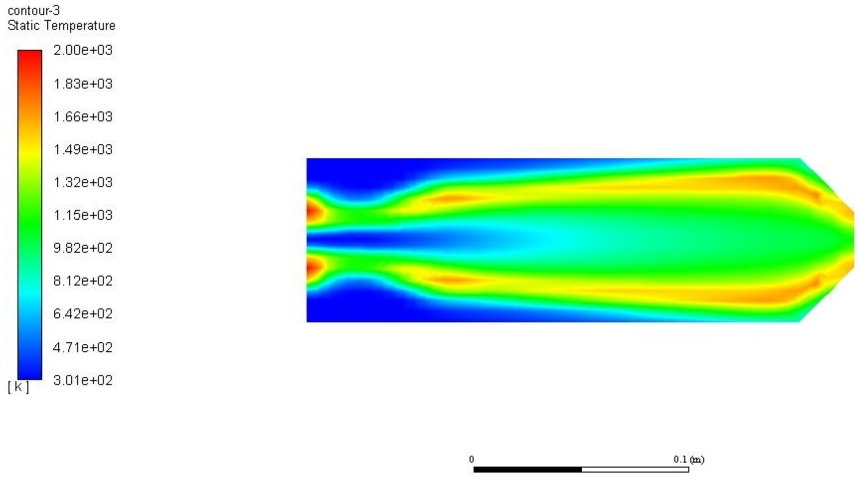



*Figure 32: CFD solution for nozzle case*

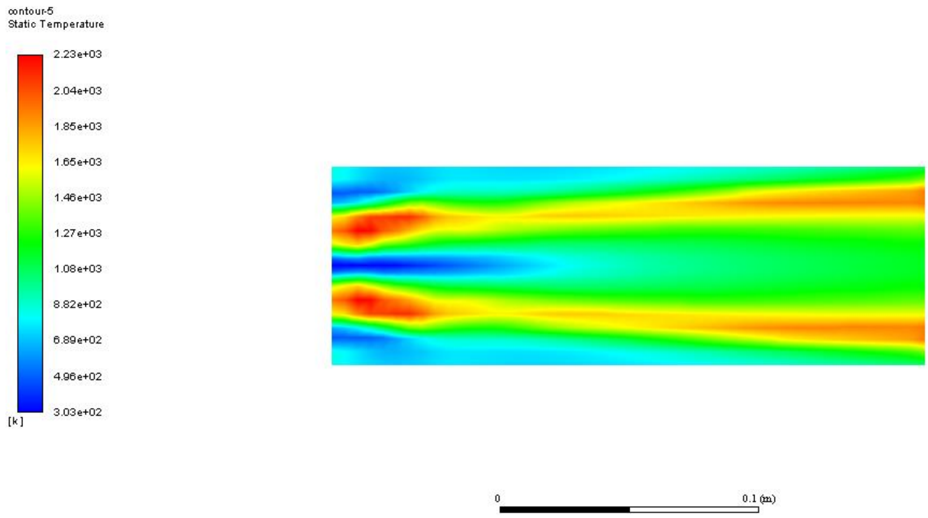



*Figure 33: CFD solution for without nozzle case*

## 5.6 Limitations

The project was constrained by some limitations. These are:

The project is constrained by its limits. These are:

1) Real gas's properties are extremely difficult to predict. While ideal gas is the foundation of all theoretical analysis, real gas always deviates from ideal instances.

b) The neglect of higher order differential and integral equation leads to less precise design geometry in theoretical analysis.

c) The residual objective is 10-4 because there are no high-performance computing tools available.

Some assumptions must be made to validate the project in order to overcome the limitation. These are:

a) The gas is assumed to be an ideal gas, which means that PV = nRT governs its state.

b) Here, it is assumed that the fluid flow is both adiabatic and reversible, meaning that neither heat nor energy is introduced to the flow.

c) An air-standard study analysis is utilized to examine the open gas turbine engine under the presumptions that air is the operating ideal gas and a heat transfer source is employed to transmit the energy produced by combustion.

## 5.7 Problems Encountered

1) Connecting the fittings with our fuel tank was a challenging task to accomplished due to different dimensioning of thread and holes. This problem was solved by connecting the joints with gas welding.

2) Ignition of engine was also a difficult task due to burner taking some time to vaporize the fuel as burner vaporize the fuel only after getting heated itself first.

## 5.8 Budget Analysis

*Table 5: Expenses*

| S. N | Name of the component | Quantity | Price | Remarks |
|---|---|---|---|---|
| 1. | Stove (Bhatti Chulo) | 1 | 1000 | |
| 2. | Pipe and Fittings | - | 2250 | |
| 3. | Bench setup | - | - | Available |
| 4. | Load cell | 1 | - | Available |
| 5. | Air compressor | 1 | - | Available |
| 6. | Injector | 1 | - | Stove burner |
| 7. | Fire extinguisher | 1 | - | Available |
| 8. | Fuel (Kerosene) | - | 860 | |
| 9. | Ignition coil | 1 | 2000 | |
| 10. | Injector | 1 | 1000 | |
| 11. | Fuel Storage Cylinder | 1 | 500 | |
| 12. | Control Valve | 2 | 1500 | |
| 13. | Check Valve | 2 | 2500 | |
| 14. | Metallic Nozzle | 1 | 600 | |
| 15. | Miscellaneous | - | 2500 | |
| | Total | | 14710 | |

From the table shown above the budget used for our project can be analyzed. Initially the expenditure for our project was expected to be around 35-40K but as we move on into our project some changes were made into our project due to resource constraint. Many materials required in our project was found available in our campus which helped a lot to reduce the cost of the project. Therefore, at the final stage of our project, the total cost of the project is found to be 14,710 only.

# CHAPTER SIX: CONCLUSION AND RECOMMENDATION

## 6.1 Conclusion

The design of liquid rocket engine consists of huge challenges and tasks. Developing engine need a lot of resources, machining tools and equipment. Designing of engine need most extensive mathematical equations for the better efficiency. So, we built the engine with the most local resources available. We have used kerosene stove parts for our fuel supply which need to get pressurized manually and in initial phase we used compressed air from our compressor for an oxidizer purpose. Initially we have used the hit and trial method for the dimensioning and other purpose. We came across few problems like leakage at the beginning which we have mitigated up to some extent and pressurizing fuel tank to sufficient pressure was also a big hurdle for us which mitigated with manual pressurizing technique. Finally, we built the liquid rocket engine and tested it successfully.

## 6.2 Recommendation

The model of our rocket engine can used by the upcoming students to study and analyze how it functions. The setup built by us can be further modified and analyzed for different results. The characteristic length of the combustion chamber can be changed and thrust variation can be studied. With the regular improvement in the technology and facilities available for rocket engine, advanced materials can be used to improve the efficiency of the liquid rocket engine by the upcoming generation.

Due to resource constraint, we couldn’t build up the originally designed liquid rocket engine which included advanced components. The upcoming students who are interested in doing similar project can build up our original model and can look upto our project to get insights about the designing, building and performance analysis of the liquid rocket engine.

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# Appendix

*Figure 34: Assembly of the whole system*

*Figure 35: Ignition test*

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