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Design and development of a liquid rocket engine closed loop control cooling system

Final year project (FYP 18-22)

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Declaration

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CAD Computer Aided Design

GOX Gaseous Oxygen

CFD Computational Fluid Dynamics

Abstract

Operation of rocket engines involve the generation of large amounts of heat during combustion. The thermal energy produced during propellant combustion is integral to rocket operation but can be detrimental to the structure of the rocket. There is therefore a need to cool the rocket body and protect it from heat damage. A number of techniques have been developed for cooling the chamber and the nozzle. These techniques have various advantages and disadvantages and typically involve test firing a liquid rocket to obtain results. This gives rise to the need to fabricate multiple chamber and nozzle assemblies. The project proposes development of a closed loop control cooling system test rig to be used in testing the cooling of a liquid rocket engine. This will eliminate the need for static firing to test the cooling of rocket engine thus promoting rapid prototyping. The proposed project will lead to the development of an optimized cooling system based on results obtained from various tests.

1 Introduction

1.1 Background

A rocket is any type of jet propulsion vehicle that carries propellants required for combustion and subsequent development of thrust. The propellants are usually either in solid or liquid state. Combustion occurs in a part of the rocket referred to as the rocket engine. The rocket engine generates thrust through this aforementioned combustion. Here a release of thermal energy is derived from the chemical reactions of the propellants. High temperature and high pressure gases result from the combustion of the propellants. These gases are then ejected at the rocket nozzle at high velocity.[1]

Liquid rocket engines are fed with liquid propellants stored under pressure from tanks and into a thrust chamber. The type of propellant used is either a bipropellant or monopropellant. A bipropellant consists of a liquid oxidizer and a liquid fuel. A monopropellant contains both oxidizing and fuel species. [2]

The combustion temperatures in a liquid rocket engine are very high and can be over 3000 °c. This is also accompanied by a high heat transfer rate from the gases to the chamber wall.

[1]

1.2 Problem statement

An important part of rocket development is iterative testing to find the most optimal parameters for each rocket flight.Testing of the cooling system of a liquid rocket engine or various coolants to be used should not necessarily involve fabrication of a new thrust chamber.

Currently testing of the cooling system of a rocket engine requires performing of a static firing test. This leads to having a damaged thrust chamber should the cooling system not work on the first time of asking.

Damaging a thrust chamber as a result of failure due to high temperatures leads to the need to fabricate a new thrust chamber which is resource consuming. As such there is need for a part of the test stand that is dedicated to cooling which would allow for dynamic changing of the cooling rate. To solve this a cooling system test rig with closed loop control is proposed to allow for testing without the need for static firing.

1.3 Objectives

1.3.1 Main Objectives

To design and develop as part of a test rig, a closed loop control cooling system for a liquid rocket engine.

1.3.2 Specific Objectives

1. To design and develop a mechanical structure for a thrust chamber with cooling facilities.
2. To design and develop an electrical system to power the sensors controller and actuator.
3. To develop and implement a control algorithm to achieve closed loop control of the cooling process.
4. To test the cooling system.

1.4 Justification of the study

Developing a closed loop control cooling system of a liquid rocket engine to be deployed on a test rig will eliminate the need to have static firing to test the efficiency of the onboard cooling system.This will reduce the strain on resources for iterative testing. The data points obtained from the testing will thereby be used to optimize the onboard cooling system.

1.5 Scope

This project will focus on developing a thrust chamber and a cooling system as part of a test rig specifically for the aforementioned thrust chamber.

2 Literature Review

2.1 Liquid rocket engine operation

A rocket engine produces thrust through combustion of propellants. These propellants include a fuel and an oxidizer. For a liquid rocket engine, these propellants are either in liquid or gaseous form. To determine the rocket engine performance before the actual flight several tests are carried out on a test stand as represented in Figure 2.1 below.

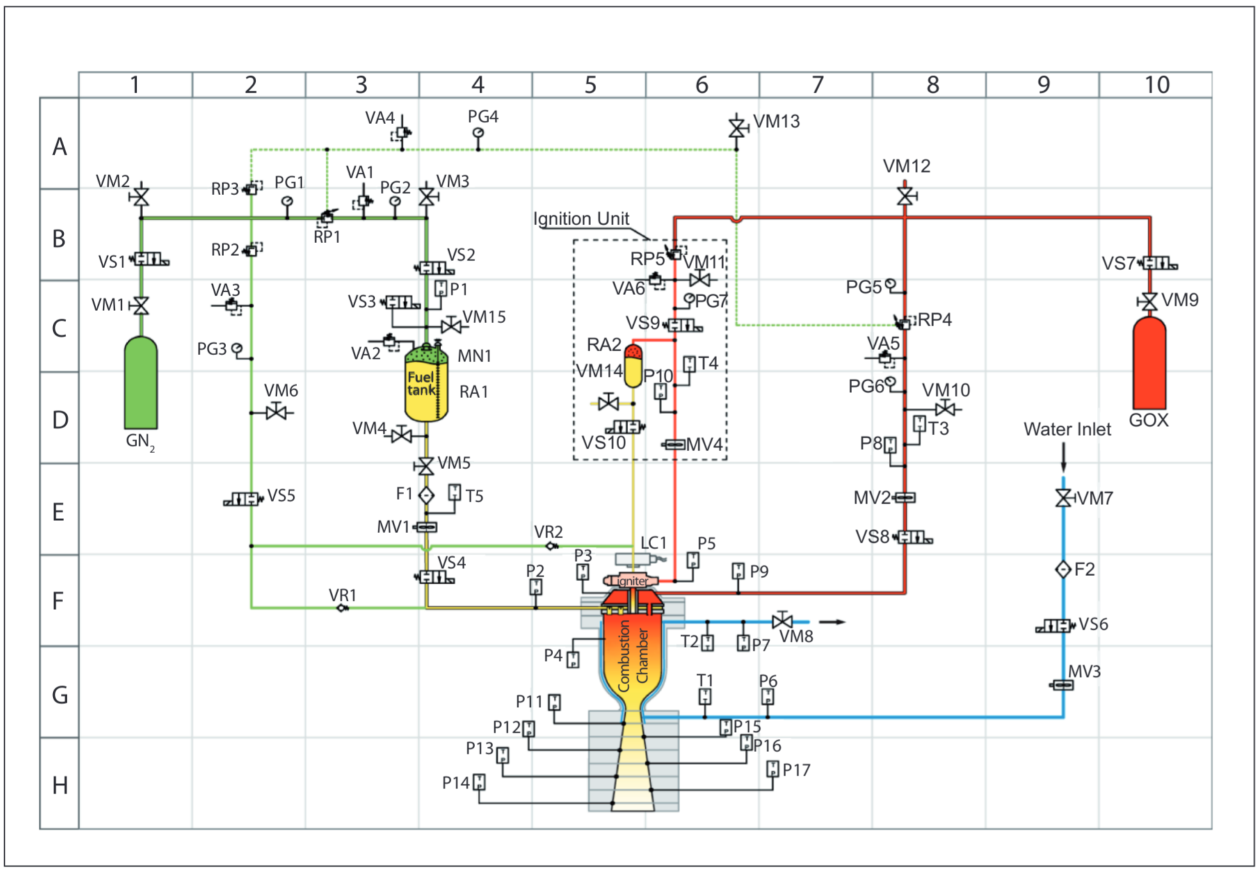

Figure 2.1: Liquid rocket engine test stand schematic[3]

From the above schematic, gaseous nitrogen in the green tank is used to pressurize the fuel towards the combustion chamber. The fuel used is ethyl alcohol and is stored in the fuel tank. Gaseous oxygen is the other propellant and is also in the ignition unit. The ignition unit also uses a bit of fuel and a small amount of this mixture is ignited when it reaches the igniter at the entrance of the combustion chamber. Water is pumped around the combustion chamber for cooling purposes. The entire schematic consists of manual, relief and solenoid valves that are used to control fluid flow. Flow meters are used to measure the rate of fluid flow throughout the system.[3]

2.2 Need for cooling

During rocket operation the propellants, oxidizer and the fuel, undergo combustion in the combustion chamber of the rocket. The reaction results in hot gases and is meant to liberate the chemical energy of the propellants, converting it into heat and pressure. If done stoichiometrically, combustion temperatures can range between 2500 to 3600 K for common propellants[2]. The combustion chamber and nozzle are designed to convert the heat and pressure generated into kinetic energy. The exhaust gases leave the nozzle at high velocity which generates a reaction force,thrust,which propels the rocket forward.

Not all the heat produced by the combustion is converted to kinetic energy.Some heat is absorbed by the combustion chamber and during nozzle operation a lot of the heat is dissipated to the nozzle. This is disadvantageous since most of the materials used to make rocket combustion chambers and nozzles lose strength as the temperature increases[4]. The chamber and nozzle also experience high pressure and if temperatures are allowed to rise this could lead to the material of the combustion chamber or nozzle failing or melting.

2.3 Types of cooling

Uncooled chamber walls can be used for a short duration up to a few seconds. For longer duration applications, one or a combination of the following cooling methods below is used.

2.3.1 Film cooling

Film cooling controls the chamber wall temperature by interposing a layer of coolant fluid between between the surface to be protected and the hot gas stream from combustion as shown in Figure 2.2 below.[5].

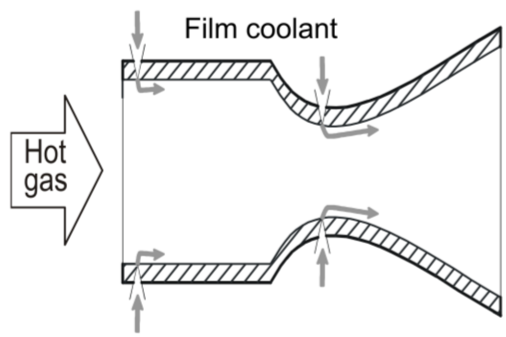

Figure 2.2: Film cooling [6]

2.3.2 Ablative cooling

Here select materials are used for sacrificial cooling by progressive endothermic disintegration of fiber-reinforced organic material and mass flow of pyrolysis gases away from the heated surface. This blocks heat transfer to the outer surface of the abrasive material[7].

2.3.3 Radiation cooling

Heat is transferred away from the surface of the outer thrust chamber wall[1].

2.3.4 Regenerative cooling

The fuel enters the cooling paths at the nozzle exit of the thrust chamber passes through the throat area and exits at the injector face as shown in Figure 2.3 below.[8].

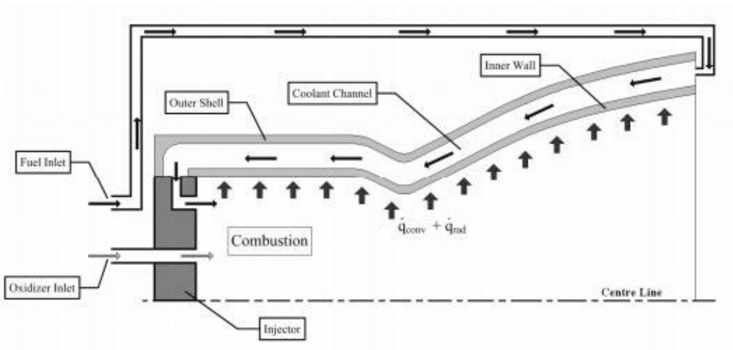

Figure 2.3: Regenerative cooling[9]

2.4 Heat transfer in the combustion chamber

A majority of the heat transferred from the hot gases produced during combustion is transferred to the wall of the combustion chamber is done by convection. Conduction makes u a very small percentage with radiation accounting for between 5 - 35% [10]. The basic relation of the heat transfer from the combustion gases to the wall of the combustion chamber can be expressed as:

$$q = h_g(T_{aw} - T_{wg})[10] \tag{2.1}$$

where:

q = Heat flux;heat transferred across the stagnant gas film per unit surface per unit time.

h_g = Gas side heat transfer co-efficient.

T_{aw} = Adiabatic wall temperature of the gas.

T_{wg} = Hot-gas-side local chamber-wall temperature

The adiabatic wall temperature of the combustion gass at any given location in the thrust chamber can be obtained from:

$$T_{aw} = (T_c)_{ns}\left[\frac{1 + r(\frac{\gamma-1}{2})M_x^2}{1 + (\frac{\gamma-1}{2})M_x^2}\right][10] \tag{2.2}$$

where:

$(T_c)_{ns}$ = Nozzle stagnation temperature

M_x^2= local Mach number

r =Local recovery factor

R = Effective recovery factor (from 0.90 to 0.98)

The local recovery factor represents the ratio of the frictional temperature increase to increase caused by adiabatic compression. This may be determined experimentally or estimated from the following equations:

$$r = (P_r 0.5(laminar flow)[10] \tag{2.3}$$

$$r = (P_r 0.33(Turbulent flow)[10] \tag{2.4}$$

Determination of gas side heat transfer co-efficient is a complex problem with data from analytical methods and experimental methods disagreeing.The disagreement is largely due to the initial assumptions of analytical methods than do not account for turbulent combustion pressure and the changing localised gas composition and temperature. The determination of heat flow is important in the analysis of the method of cooling to be chosen and the specific parameters.

In the case of regenerative cooling as shown in Figure 2.4. The general heat conduction will follow:

$$\frac{Q}{A} = -k\frac{dT}{dL} = -k\frac{\triangle T}{t_w}[2] \tag{2.5}$$

where:

Q = heat transferred to a surface area A

$\frac{dT}{dL}$ = temperature gradient

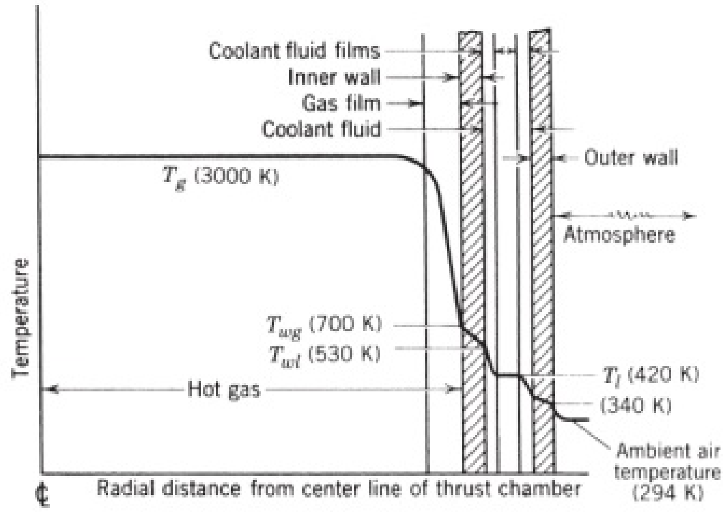

Figure 2.4: Temperature gradient of regeneratively cooled combustion chamber [2]

t_w =wall thickness and

k = thermal conductivity

The negative sign indicates that the temperature decreases as the thickness increases. The heat transfer to the walls can be treated as a problem of steady state heat transfer in series. It is a combination of convection at the boundaries of the flowing fluids and conduction through the chamber walls. This can be expressed by:

$$q = h(T_g - T_l) = \frac{Q}{A}[4] \tag{2.6}$$

where:

q = heat transferred per unit area per unit time

T_l =the absolute coolant liquid temperature

This relationship and the above bring a number of factors for cooling into the foreground:

- The specific temperature of the coolant
- The specific heat capacity of the coolant

- The thickness of the chamber wall
- The nature of the propellants
- The nature of combustion including temperature and pressure
- The properties of the material used to make the chamber wall

2.5 Factors affecting the choice of cooling system

The selection of the best cooling technique to use is specific to the design of the chamber, however there are some general rules that aid in this selection[10]:

1. **Propellants**

   The properties of the propellants after combustion such as temperature, specific heat, specific weight, viscosity have an effect on the heat-transfer rate and thus the choice of cooling to be used. The heat conductivity of the propellants as well as their flow rate is also determines whether they can be used for regenerative,film or transpiration cooling.

2. **Chamber Pressure**

   Higher pressures are associated with higher combustion- exhaust gases flow rate per unit area of the combustion chamber. This leads to increased heat transfer to the chamber. Thus for high chamber pressures a combination of cooling methods is often employed.

3. **Propellant feed system**

   The type of propellant feed system used determines how much pressure is available for cooling. For turbo pump fed systems there is a high pressure budget meaning that it is convenient to use regenerative cooling. For pressure fed systems which have a lower pressure budget which may not accommodate regenerative cooling other cooling methods such as ablative,film and radiation are preferred.

4. **Thrust chamber configuration**

   The shape of the combustion chamber determines the flow rate of the exhaust gases after combustion. Higher flow rates increase the rate of heat transfer. Spherical chambers offer the best cooling efficiency but are difficult to machine.

5. **Thrust chamber construction material**

   The choice of material for the combustion chamber will greatly affect the chosen cooling system. If the engine is small and the material chosen is of high conductivity then heat sinking can be used. Strength at elevated temperatures and the heat conductivity of the material affect the choice of regenerative cooling. For film cooling the material should have a high working temperature range. The success of ablative cooling wholly depends on the type of material chosen.

2.6 Design considerations for cooling

2.6.1 Material selection

The material to be used to develop the combustion chamber and nozzle as well as the cooling jacket has a considerable effect on the cooling[4]. The material for design of the combustion chamber should have the following characteristics:

- Good thermal conductivity
- High melting point
- High specific thermal strength at high temperatures
- Good wearing qualities against erosion by exhaust gases
- practical fabrication possibilities

The properties of some common materials used in rocketry are as shown in Table 2.1 below.[11]:

Metal	Density (g/cm^3)	Melting point	Cu 100 therm. conductivity	chamber hardness	tensile strength at 1000	nature of oxide at high temp
Aluminium	2.7	658	55	2.9	liquid	refractory
Duralumin	2.8	550	30	3.5	liquid	refractory
Copper	8.9	1083	100	3.5	low	powdery
Iron	7.9	1	30	4.5	low	powdery
Stainless steel	7.8	1250	5	5.5	42 MPa	partial refractory

Table 2.1: Properties of common rocket materials[11]

2.6.2 Selection of coolant

Depending on the type of cooling employed the type of coolant changes. For ablative cooling the material may be a solid ceramic. For dump cooling transpiration and film cooling the fuel is the coolant.For transpiration cooling For regenerative cooling the coolant may be either the oxidizer or the fuel. Depending on the mass flow rate and the specific heat carrying capacity, one of the propellants is chosen. Despite the large temperature difference offered by liquid oxygen it is offers compatibility issues[4].

2.6.3 Selection of cooling passages geometry

- **Regenerative cooling**

  There are two common cooling passage designs. the first comprises of cooling tubes which are brazed together to an outer shell that forms the contour of thrust chamber. In this technique the cross-sectional area of the tubes are changed according to the region of thrust chamber. For the high heat flux regions, tubes are elongated and

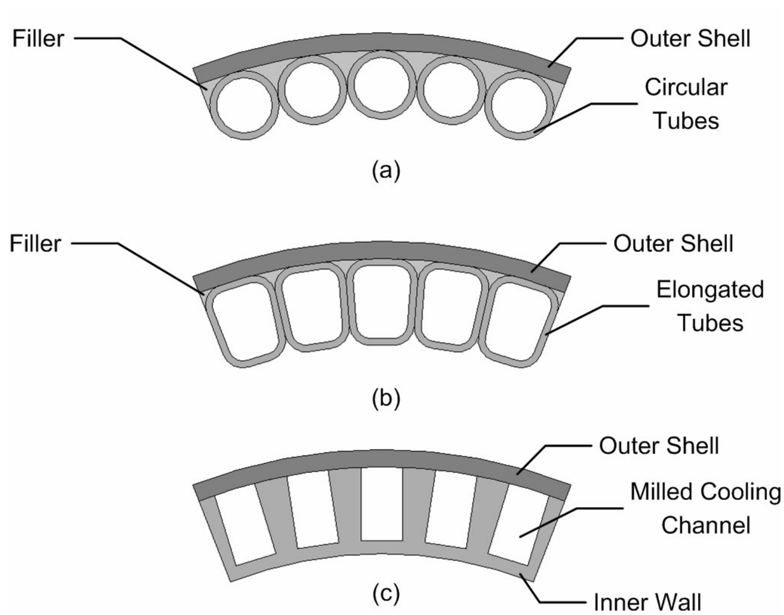

Figure 2.5: Regenerative cooling geometry[2]

squeezed to increase the velocity of the coolant and to increase the heat transfer area.This is as shown in the image. The second technique involves the milling of rectangular cooling channels along the contour of a thick thrust chamber. as shown in Figure 2.5.

- **Film cooling**

  This type of cooling can be achieved using two techniques[12] The first involves the use of the injector. The injector configuration can be used to spray propellant onto the walls of the combustion chamber. The second method involve drilling holes or slots onto the wall of the combustion chamber to introduce the coolant. This is as shown in Figure 2.6.

- **Dump cooling**

  The geometry of dump cooling is similar to that of regenerative cooling.

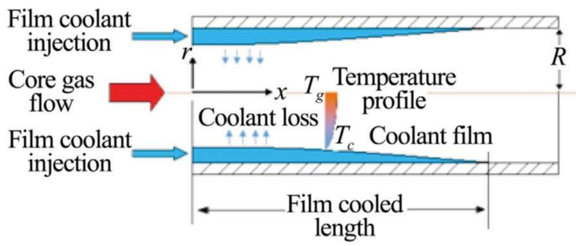

Figure 2.6: Film cooling geometry[12]

2.7 Water Cooling on Test Stands

Here the temperature of the chamber wall is kept at a controlled temperature by running water at a constant rate through the cooling jacket. This is the same cooling jacket that would be used with a coolant during the flight of the rocket[3]. This process uses a constant mass flow rate which is determined using the following equation:

$$m_w = \frac{1.1\pi q_{cw}(D_c + 2t_{cw})L_c}{c_w \triangle T_w}[13] \tag{2.7}$$

where:

q_{cw} = Average heat transfer rate of chamber material.

D_c = Combustion chamber diameter.

t_{cw} = Thrust Chamber wall thickness.

L_c = Combustion chamber length.

c_w = Coolant water specific heat capacity.

$\triangle T_w$ = Desired water temperature value.

2.8 Gap Analysis

The above testing methods all use a constant coolant flow rate while the rocket engine is tested on a test stand. This means there is no way to gauge the performance of a coolant without running the risk of a damaged thrust chamber.

3 Methodology

3.1 Introduction

T The proposal seeks to develop a dynamic cooling system for a liquid rocket engine test stand. The system will feature the synergistic combination of mechanical electrical and control modules as illustrated in the figure below:

3.2 Mechanical module

3.2.1 Design Requirements

1. Functional Requirements

    (a) The designed thrust chamber and nozzle should be able to handle the high temperature and pressure of combustion.

    (b) The designed cooling system should be able to lower the temperature of the combustion chamber and nozzle.

    (c) The designed test rig should be strong enough to hold the engine during testing.

    (d) The piping system and pump should be able to deliver cooling to the engine with minimal leakage.

2. Non-functional requirements

    (a) Safety.

    (b) Low cost

    (c) The assembly should be aesthetically pleasing.

    (d) Aesthetically pleasing connections.

3.2.2 Conceptual Design

The conceptual design of the mechanical module comprises of:

1. **Thrust chamber and nozzle assembly**

   Some of the design requirements for Thrust chamber and nozzle assembly include:

   (a) The chamber should be able to withstand the high temperatures and high pressure of the combustion chamber.

   (b) The chamber and nozzle should accelerate the exhaust gases thus reducing their pressure to match that of the exit plane.

   (c) The chamber and nozzle should dissipate heat quickly.

   Design of the assembly begins with find the coefficient of thrust,C_f which indicates the quality of exhaust gas expansion by the . It is given by:

   $$C_f = \frac{F}{A_t(P_c)_{ns}}[4] \tag{3.1}$$

   The size and shape of the combustion chamber must offer sufficient volume to adequately atomize, mix, evaporate and thoroughly combust. Different propellant combinations and states yield different mixing,vaporization and reaction timed therefore require different chamber volumes.One method of designing chamber length is by first determining the throat size as it is a relatively trivial characteristic to determine. Characteristic chamber length,L^* is dependant on the propellant combination and is generally given by the equation:

   $$L^* = \frac{V_c}{A_t}[4] \tag{3.2}$$

   A cylindrical chamber is easiest to construct. The forward section is approximated as a cylinder. Since we opt for a converging diverging nozzle the converging section of the chamber is approximated as a truncated cone. The length & volume of the convergent section is dependent on the convergent half angle α. This angle ranges from 20 °to 45 °. The nozzle should be designed with the cross-sectional area sufficiently large to avoid combustion instability. For this condition the convergent half angle,α , can be assumed to be equal to the divergent half angle of the equivalent minimum length nozzle which is equal to half the Prandtl-Meyer angle,v which is given by:

$$v(M_e) = \frac{\gamma_e + 1}{\gamma_e - 1} tan^{-1} \sqrt{\frac{\gamma_e - 1}{\gamma_e + 1}(M_e^2 - 1)} - tan^{-1}\sqrt{(M_e^2 - 1)}[4] \tag{3.3}$$

where:

M_e = Mach number at exit

γ_e = specific heat ratio of exhaust products

v = Prandtl-Meyer angle

With the above we can determine the chamber diameter,D_c

$$D_c = \sqrt{\frac{D_t^2 + \frac{24}{\pi} tan(\alpha) V_c}{D_c + 6tan(\alpha) L_c}}[4] \tag{3.4}$$

where:

D_c= the chamber diameter

D_t = the throat diameter

The expansion ratio,ϵ, must be determined for the fully expanded nozzle. The Mach number of the exhaust gases at the exit plane of the nozzle can be determined by a function of atmospheric pressure.

$$\frac{(P_c)_{ns}}{P_e} = \left[1 + \frac{\gamma - 1}{2} M_e^2\right]^{\frac{\gamma_e}{\gamma_e - 1}} [4] \tag{3.5}$$

To reach the exit Mach number desired the nozzle exit area,A_e can be defined using the isentrolpic Area Mach relation:

$$\frac{A_e}{A_t} = \frac{1}{M_e}\left[\frac{2}{\gamma + 1}(1 + \frac{\gamma - 1}{2} M_e^2)\right]^{\frac{\gamma + 1}{2(\gamma - 1)}} [4] \tag{3.6}$$

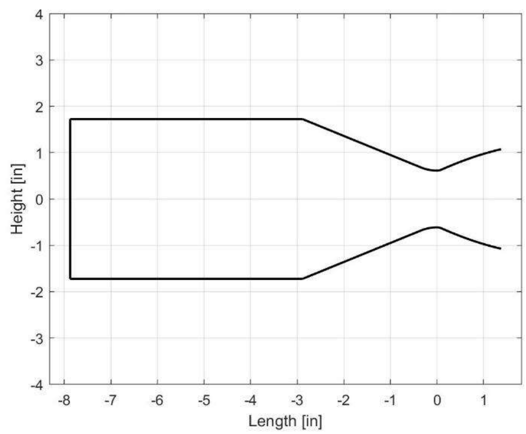

Figure 3.1: Basic nozzle geometry[4]

With this information the basic nozzle geometry is obtained as displayed in Figure 3.1. The specific nozzle design can be chosen between the simple conical nozzle,the ideal bell nozzle and the non-ideal bell nozzle.

2. **Cooling Jacket**

   This will handle the flow of the coolant close to the combustion chamber to enable cooling. The design considerations for this module include:

   (a) The cooling jacket should be able to withstand the pressure from the coolant feed system and minimize pressure drop across the jacket.

   (b) The cooling jacket should be corrosion resistant.

   (c) The cooling jacket should offer a large surface for cooling.

3. **Coolant feed piping**

   This will include pipes, valves and fittings that will be used to deliver the coolant to the cooling jacket of the engine. The design considerations for this include:

   (a) The pipes should be able to deliver high pressure coolant with minimal pressure losses.

   (b) The system should be dynamic to be able to change the amount of coolant being delivered.
   (c) The coolant pipes should have few leakages.

4. **Frame**

   This will be used to hold the liquid engine during testing. The design considerations for the frame include:

   (a) Should be able to support the liquid engine during testing.
   (b) Should offer provisions for the piping system.
   (c) The frame should have safety or fallback parameters in case of emergency.

3.2.3 Data collection

Data from the mechanical module will be obtained from:

1. **Modeling and simulation**

   The mechanical module will be modelled in CAD software such as Autodesk Inventor. The software will also be used for Finite Element Analysis of the designed chamber. Computational Fluid Dynamics (CFD) software will be used to model and simulate flow in the nozzle.

2. **Experimental Testing**

   Testing will be carried out on the fabricated mechanism to determine the appropriate pressure and flow rate for effective cooling.

3.2.4 Data analysis

The CAD software will be used for both the collection and analysis of data. Data from the simulation will include stress, strain and flow parameters. The data obtained from the analysis will be used to:

1. Determine the appropriate dimensions for the frame to prevent failure.
2. The nozzle parameters to prevent combustion instability.
3. To determine the most suitable method machining method.

3.3 Electrical Module

3.3.1 Design Requirements

1. Functional Requirements
   (a) The pump should have enough power to pump the fluid through the plumbing system effectively.
   (b) The module should facilitate variation of power delivered to the pump.
   (c) The electrical module should allow for an interface to receive user inputs and display outputs to the user.
   (d) The electrical module should allow for delivering of appropriate power to the microcontroller and sensor suite.
2. Non-functional requirements
   (a) Power efficiency.
   (b) Safety.
   (c) Minimum heat dissipated.
   (d) Aesthetically pleasing connections.

3.3.2 Conceptual Design

The conceptual design comprises of:

1. Power Distribution Circuit

   The design considerations include:

   (a) Should avail appropriate power in terms of voltage and current to each of the components from the main power supply.

   (b) Incorporate safety features to prevent damage to the components.

   (c) Design should protect from electrical noise to ensure the signal integrity is maintained throughout the circuit.

   (d) Allow for manual overrides to the system in the event of a power emergency.

2. Pump

   The design requirements include:

   (a) Should provide enough power to pump the desired fluid at the desired variable rates.

   (b) Should fit well into the mechanical structure.

3. Sensors

   These will include thermocouples and mass flow rate meters. The design considerations will be:

   (a) The sensors should provide accurate and consistent data.

   (b) The sensors should be compatible with the chosen microcontroller.

   (c) The sensors should have the capability to achieve the required range.

4. Display panel.

   The display panel should appropriately and promptly relay communication from the system to the user.

5. Input panel

The input panel should allow the user input to the system to allow for a change in operation parameters.

3.3.3 Data collection

This will be achieved primarily through modelling and simulation. The electrical system will be modelled using electrical modelling software. This will be to primarily check whether the circuit achieves the primary objective of providing sufficient power safely. The model will then be optimized for efficiency.

The data collected from the simulation software will include:

1. Current consumption.
2. Voltage drop at each component.

3.3.4 Data analysis

This is analysis of the data collected using from carrying out simulations and preliminary tests. This data will be used to determine:

1. Control parameters of power delivered to the pump.
2. Amount of power consumed by the proposed solution.
3. Processing required on the data acquired from the sensors.

3.4 Control module

3.4.1 Design requirements

1. Functional requirements

(a) The module should take sensor data as an input.

(b) The control module should determine the actuation required from the sensor data depicting the current state of the system.

(c) The control module should provide appropriate signals to allow for the required actuation.

2. Non-functional requirements.

   The algorithm should be straightforward and allow for direct implementation.

3.4.2 Control Algorithm

The control algorithm is as shown in figure 3.2 below.

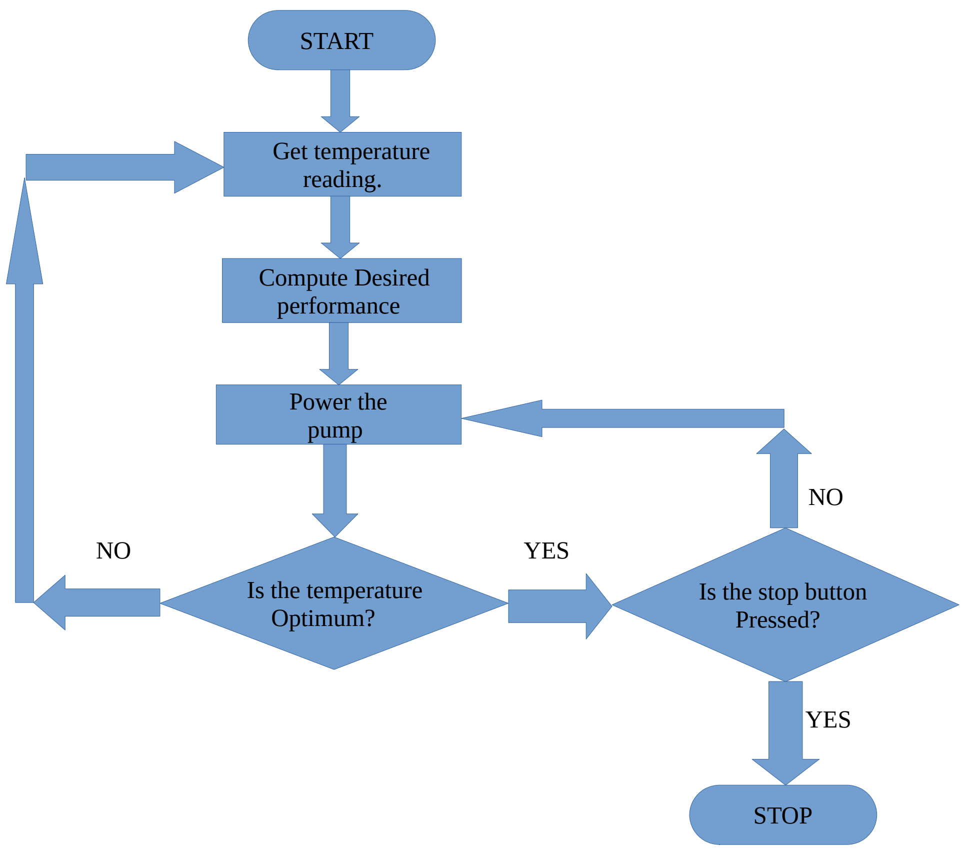

Figure 3.2: Program Flow Chart

3.4.3 Data collection

1. Modelling and simulation

   The control algorithm will be modelled using computer software and different algorithms simulated and responses observed. The data collected will be visualized to compare input signals and the responses that consequently occur.

2. Experimental testing

   The fabricated mechanism will undergo iterative testing to acquire the optimum tuning parameters of the control algorithm.

3.4.4 Data analysis

The data from the simulation will allow for the determination of the most appropriate control algorithm to be used in implementing the dynamic cooling rate. This will then be actualized through firmware to be run on the system's microcontroller.

4 Expected Outcomes

The expected outcomes are:

1. Mechanical structure with consisting of a combustion chamber with a cooling jacket and a plumbing to achieve the cooling system.

2. Electrical system to power the sensors, microcontroller, valves and pumps.

3. A control algorithm to achieve closed loop cooling control.

5 Time plan

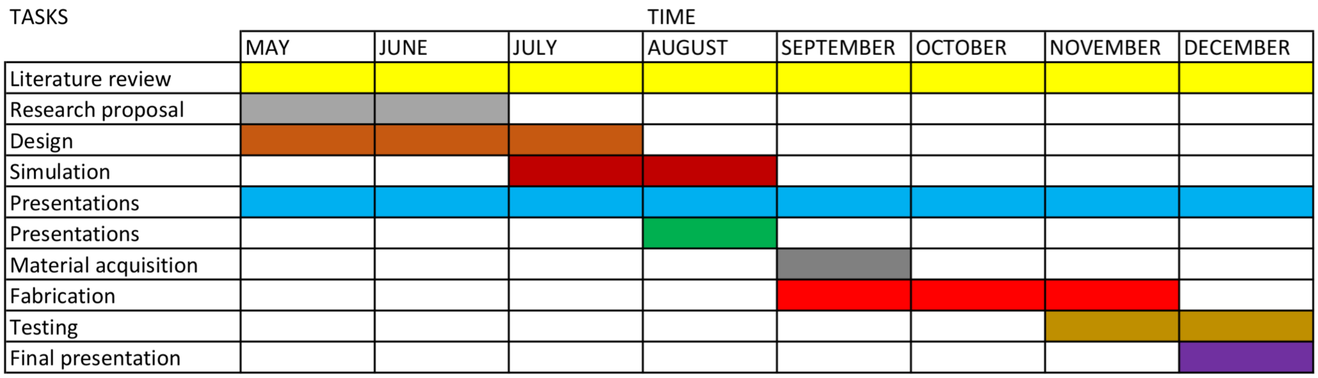

Table 5.1: Time plan

6 Budget

	Item	Description	Cost per unit(Kshs.)	Total(Kshs.)
1	Chamber material	Machining purposes	5000	5000
2	Pump	Fluid flow	5000	5000
3	Microcontroller	Function control	2000	2000
4	Valves	Flow control	2000	2000
5	Assorted sensors	System state	2000	2000
6	Assorted electronics	Electronic wiring	1000	1000
7	Plumbing fittings	Fluid flow	2000	2000
8	Coolant tank	Stores the coolant	3000	3000
				22000

Table 6.1: Budget

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