DESIGN, FABRICATION AND TESTING OF SMALL SCALE TURBINE JET ENGINE

MOHAMAD RIDHWAN BIN RAJA MOHD

UNIVERSITI MALAYSIA PAHANG

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FACULTY OF MECHANICAL ENGINEERING

I certify that the project entitled “Design, Fabrication, and Testing of Small Scale Turbine Jet Engine” is written by Mohamad Ridhwan bin Raja Mohd. I have examined the final copy of this project and in my opinion; it is fully adequate in terms of scope and quality for the award of the degree of Bachelor of Engineering. I herewith recommend that it be accepted in partial fulfillment of the requirements for the degree of Bachelor of Mechanical Engineering with Automotive Engineering.

MR. MOHD FADZIL ABDUL RAHIM

Examiner Signature

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Report submitted in partial fulfillment of the requirements
for the award of Bachelor of Mechanical Engineering with Automotive Engineering

Faculty of Mechanical Engineering
UNIVERSITI MALAYSIA PAHANG

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SUPERVISOR'S DECLARATION

I hereby declare that I have checked this project and in my opinion, this project is adequate in terms of scope and quality for the award of the degree of Bachelor of Mechanical Engineering with Automotive Engineering.

Signature:
Name of Supervisor: MR. AMIR BIN AZIZ
Position: LECTURER
Date: 18 JUNE 2012

STUDENT'S DECLARATION

I hereby declare that the work in this project is my own except for quotations and summaries which have been duly acknowledged. The project has not been accepted for any degree and is not concurrently submitted for award of other degree.

Signature
Name: MOHAMAD RIDHWAN BIN RAJA MOHD
ID Number: MH08063
Date: 18 JUNE 2012

This work is dedicated to all those who have inspired me throughout my life, with special thanks to my family and friends

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ABSTRACT

A turbine jet engine comprises of four main parts, which are a compressor, a combustion chamber, a turbine and an exhaust nozzle. Turbine jet engine operates at an open cycle called a jet propulsion cycle. A small-scale turbine jet engine comprises of the same element as the gas-turbine engine but in a smaller scale. Both engines differ in utilization and purpose of its production. Turbine jet engines were constructed mainly for air transportation while the small-scale turbine jet engines are developed for a wider purpose, ranging for research activity to hobbyist enthusiastic. Hence, this thesis encompasses the design, fabrication, and testing a small-scale turbine jet engine. The engine was derived from an automobile turbocharger, which provided the turbine and compressor component. A combustion chamber was design and fabricated. Engine support system comprised of ignition, lubrication and fuel delivery system were installed at the engine. The engine assembly was mounted in a test setup. Thermocouples were installed at three different stations on the engine flow path to measure the temperature. Fuel regulators were utilized to measure the fuel flow. The engine was started using a specific procedure until it self-sustained. During testing, the engine was only able to self-sustain approximated for 10 seconds at 1.26×10^{-4} kg/s fuel mass flow rate. Troubleshooting and analysis regarding the failure of the engine was done. Analysis shows that there are four possible factors involves, namely, the uses of LPG fuel, large pressure drop at the exit of combustion chamber, low pressure pump and leaking at the turbocharger. Four recommendations were made for further studies, which are, utilize a brand-new turbocharger for the engine, use a pure propane gas as a source of fuel, avoid uses of pipe flange at the combustion chamber and utilize a higher pressure pump for lubrication system. Further modification was not made due to time and cost limitation.

ABSTRAK

Enjin jet turbin terdiri daripada empat bahagian utama, iaitu pemampat, kebuk pembakaran, turbin dan nozel ekzos. Enjin turbin jet beroperasi pada kitaran terbuka dipanggil kitaran pendorongan jet. Enjin jet turbin kecil terdiri daripada elemen yang sama seperti enjin jet turbin tetapi pada skala yang lebih kecil. Kedua-dua engine berbeza dari segi penggunaan dan tujuan pengeluaran. Enjin jet turbin telah dibina terutamanya untuk pengangkutan udara manakala enjin turbin jet turbin kecil telah dibina untuk tujuan yang lebih luas, dari aktiviti penyelidikan ke penggemar antusias. Oleh itu, tesis ini merangkumi reka bentuk, fabrikasi, dan pengujian enjin jet turbin kecil. Enjin dibina dari turbocharger kereta, yang menyediakan komponen turbin dan pemampat. Kebuk pembakaran direka bentuk dan difabrikasi. Sistem enjin sokongan yang terdiri daripada sistem penyampaian pencucuhan, pelinciran dan bahan api telah dipasang pada engine. Pengganding suhu telah dipasang di tiga stesen laluan aliran enjin untuk mengukur suhu. Pengatur bahan api untuk mengukur kadar aliran bahan api juga telah dipasang dalam persediaan ujian. Enjin dihidupkan degan menggunakan prosedur yang tertentu sehingga ia beroperasi sendiri. Semasa ujian, enjin hanya dapat beroperasi sendiri selama 10 saat pada 1.26×10^{-4} kg/s kadar aliran jisim api. Penyelesaian masalah dan analisis berkaitan kegagalan enjin telah dilakukan. Analisis menunjukkan bahawa terdapat empat faktor yang terlibat, iaitu, penggunaan bahan api LPG, kejatuhan tekanan yang besar pada kebuk pembakaran, tekanan pam yang rendah dan turbocharger bocor. Empat cadangan telah dibuat untuk pengajian seterusnya iaitu, menggunakan turbocharger baru untuk enjin, menggunakan gas propana tulen sebagai sumber bahan api, mengelakkan penggunaan paip bebibir di kebuk pembakaran dan menggunakan pam tekanan yang lebih tinggi untuk sistem pelinciran. Pengubahsuaian selanjutnya tidak dibuat kerana limitasi masa dan kos.

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LIST OF SYMBOLS

C_D	Discharge coefficient
c_p	Specific heat at constant pressure
F	Force
f_{st}	Stoichiometric
H	Height
h	Enthalpy
L	Length
P	Pressure
q	Heat transfer per unit mass
r	Radius
S'	Swirl number of primary air swirler
T	Temperature
U	Velocity component in direction of flow
V	Velocity
w	Work per unit mass
α	Bypass ratio
Δ	Finite change
ε	Combustion reaction progress variable
θ	Angle
τ	Temperature ratio
ϕ	Equivalent ratio
A	Annulus
c	Cooling

DZ	Dilution Zone
f	Fuel-to-air mass flow ratio
g	Gas
in	Inlet
MB	Main burner
max	Maximum
opt	Optimum
out	Outlet
PZ	Primary Zone
SZ	Secondary Zone
t	Total
1 - 4	Station location

LIST OF ABBREVIATIONS

CFD	Computational fluids dynamics
CITREX	Creation, innovation, technology and research exposition
COSMOFLOW	SolidWorks flow simulation
LPG	Liquefied petroleum gas
WWII	World war 2

CHAPTER 1

INTRODUCTION

1.1 PROJECT BRACKGROUND

A turbine jet engine is widely used in commercial aircraft and jet fighters. It comprises of four main parts, which are a compressor, a combustion chamber, a turbine and a nozzle. The compressor increases the pressure and temperature of air before entering the combustion chamber. The high pressure and high-temperature air then entered the combustion chamber where it is mixed with fuel and ignited, in a constant pressure process. The combustion gasses afterward flows through a turbine which connected to a compressor by a common shaft. The turbine extracts energy from the gasses resulting in a reduction of pressure and temperature of the gas. The remaining gasses, flows through a nozzle where it is accelerated to produce thrust. These processes of compression, combustion, extraction and exhaustion are continuous and self-sustaining.

Small scale turbine jet engine comprises of the same element as the commercial gas-turbine engine but in a smaller scale. Vast majorities of these jet engines are developed by amateurs and are design around an existing automotive turbocharger, since it consists of two components of a jet engine, a compressor and a turbine. The combustion chamber and an exhaust nozzle is then the only component to be constructed.

Hence, this project is subjected to the concept of developing a small-scale turbine jet engine. The engines are developing using an existing turbocharger. The design of combustion chamber and exhaust nozzle, and also construction of the engine support system will be explained. The instrument to monitor the engine performances is discussed. The testing procedure and the acquired data from experimental testing are discussed.

1.2 OBJECTIVE

The objective of this project is to design, fabricate and testing a small-scale turbine jet engine.

1.3 PROBLEM STATEMENT

The jet engine has made its name since the 60's and was developed by infamous aircraft industry with high expertise personnel and state of the art machinery. The concepts in designing the jet engines available today are subjected to large size. Hence, to design, nonetheless, to construct the jet engine in small scale is a challenge. Much method has been introduced in the designing of a small-scale turbine jet engine by amateur and even professional builder. However, none of them proposes a suitable engineering knowledge as an accurate guide.

The engine is design by proper calculation design to fulfill specific criteria. Simulation using CFD software is used based on parameters calculated from ideal condition. Once the results of the simulation satisfy the desired expectation, fabrication process took place.

Concern regarding fabrication of the engine is extensive. The hitch is not the construction but rather the material required for the engine. The availability of the suitable materials is limited. Since the engine as many other power devices are tied to a certain range, which would be failed if breached. Thus, the selection of proper material which can perform under these extend are a challenge. Still, the material is properly selected through extensive consideration of each parameter involved.

The fabricated components are installed to the turbocharger and certain set of a support system is introduced. The complete engine is test run, and its performance is measured.

1.4 SCOPE OF STUDY

The scope of this project is based on the following:

1. To design a combustion chamber.
2. To evaluate the design using COSMOFLOW simulation.
3. To fabricate the combustion chamber.
4. To use LPG as fuel.
5. To test run the turbine jet engine and measure its performances in terms of thrust produced, fuel flow rate and combustion and exhaust temperature.

1.5 LIMITATION

The limitations to carry out this project are as follows:

1. The availability of material.
2. The availabilities of components cost limitation.
3. Time limitation

CHAPTER 2

LITERATURE REVIEW

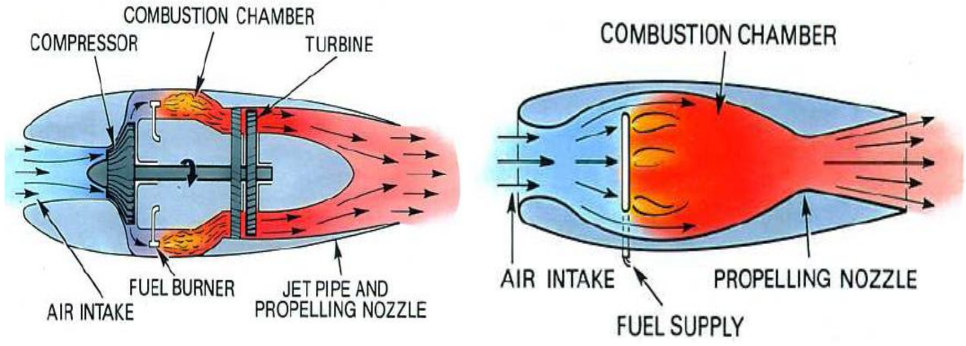
2.1 INTRODUCTION

This chapter presents the general concepts of turbine jet engine and its adaptation towards a small-scale jet engine. The jet engine operating cycle is explained and theory regarding the design of combustion chamber is described.

2.2 BACKGROUND

The idea of utilizing the principle of reaction in aircraft was started since early 17^{th} century and has been dominated by reciprocating engine. Not until the Second World War, the jet-engine technology has been developed to replace the unsuitability of the piston engine to produce high velocity of airflow necessary to carry a propulsive task. In 1913, Rene Lorin patterned the first jet engine. However, it is impossible to manufacture because the suitable heat resisting material was not yet available.

During the WWII, Sir Frank Whittle (Smith, 1995) made a major breakthrough in the jet propulsive technology. He patterns a gas-turbine engine that produces propulsive power, which is called the Whittle engine. The engine has formed the basis of gas-turbine engine, and the technology still widely used ten decades later. Figure 2.1 shows the comparison between Lorin’s and Whittle engine.



(a) Whittle engine **(b)** Lorin's engine

Figure 2.1: Comparison between Lorin's and Whittle engine

Source: Royce 1996

Small scale turbine jet engine is a smaller size turbine jet engine which exhibits similar operating principles as commercial or industrial gas turbine. Nevertheless, they differ in utilization and purpose of its production. While jet engines were constructed mainly for air transportation, the small-scale turbine jet engines are developed for a wider purpose, ranging for research activity (Benini, 2007) to hobbyist enthusiastic (Kamps, 2005).

2.3 JET ENGINE TYPES

There are three main types of jet engines, which are turbojets, turbofan and turboprop. Major components of these engines are similar as they consisted of a compressor, a combustion chamber and a turbine. The acceleration of air to develop thrust are divided into two concepts where the first; large amount of air accelerated at low speed and the second; small amount of air accelerated at high speed. Turbojet and turboprop engines provide propulsive force directly by the accelerated gas in the exhaust. Turbofan engines conversely differ from the other two where thrust is developed by the amount of air bypasses around the basis engine.

2.3.1 Turbojet

The turbojet is the earliest and the simplest form of jet engine. The major component of turbojet consists of a multistage compressor, a combustion chamber and a single or multistage turbine as shown figure 2.2. They produce high specific thrust, and it is the suitable for high subsonic and sonic flight speed.

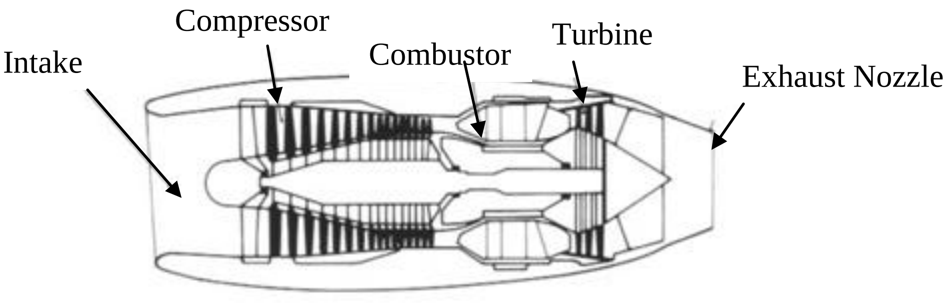


Figure 2.2: Schematic illustration of turbojet engine

Source: Hunecke 2003

2.3.2 Turbofan

Turbofan as shown in figure 2.3 has a core compressor, combustion chamber and a turbine. In addition, they have a fan in front of the core compressor and second power turbine behind the core turbine. Turbofan engine exhibits good efficiency at high subsonic flow and exhibit lower fuel consumption which is preferred in commercial aircraft transportation.

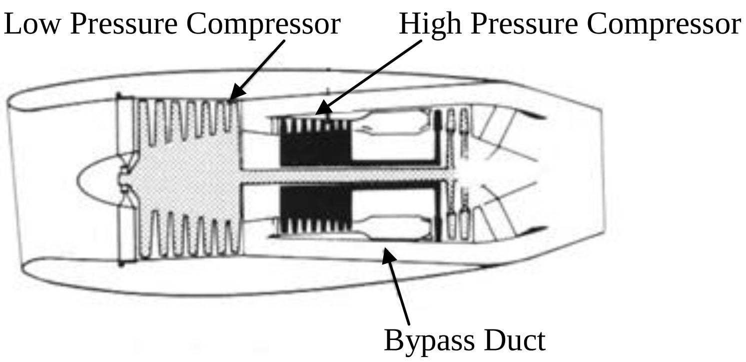


Figure 2.3: Schematic illustration of turbofan engine

Source: Hunecke 2003

2.3.3 Turboprop

Turboprop as shown in figure 2.4 is similar to turbofan, but they have a propeller in front of the compressor. Most of the energy extracted in the turbine is used to drive the propeller. The engine follows the second concept where they were design to accelerate a relatively small amount of air at high speed resulted in high propulsive efficiency. However, the fuel consumptions are relatively lower compared to turbojet at some thrust and flight speed.

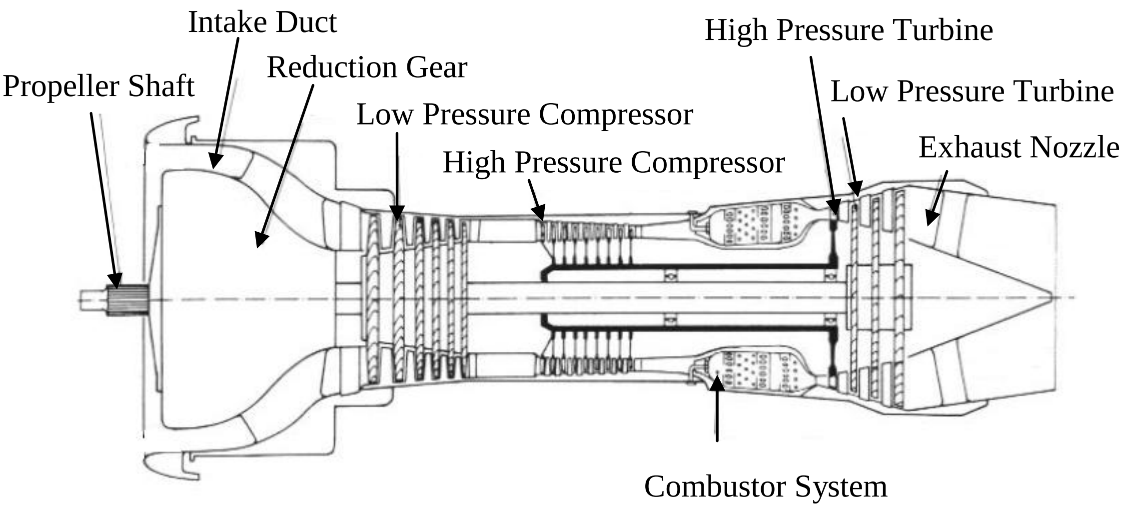


Figure 2.4: Schematic illustration of turboprop engine

Source: Hunecke 2003

2.4 PRINCIPLE OF JET PROPULSION ENGINE

2.4.1 Basic Operations

Jet engine operates at an open cycle called a jet propulsion cycle where it follows Brayton cycles but slightly differs. In an ideal Brayton cycle, the gas from the turbine exit expanded to atmospheric pressure. However, in a jet propulsion cycle, the gas is expanded to a pressure that the power produces by the turbine just sufficient enough to rotate the compressor rotor and other auxiliary equipments and gives power to the compressor and other auxiliary equipment within the aircraft.

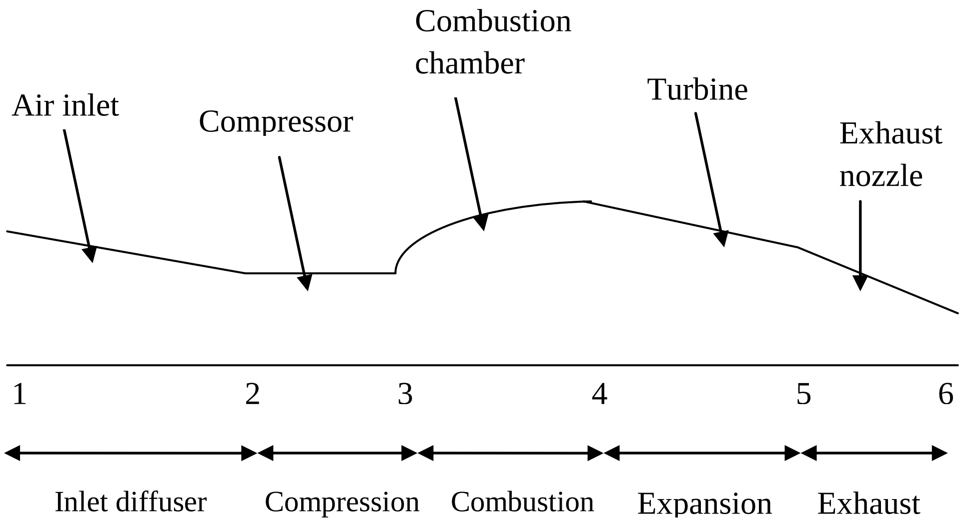


Figure 2.5: Schematic diagram of jet-propulsion cycle. The number represents the engine station locations and will be used throughout this report

By referring to figure 2.5, air enters the compressor through the inlet diffuser (station 1). As the air enters the diffuser, the air will decelerate and pressure is increased. The pressure will then further increase as it enters the compressor (station 2). The compressed air from the compressor subsequently mixes with fuel in the combustion chamber. At the combustion chamber (station 3), the mixture will burn at a constant pressure. The high pressure and high-temperature gasses will then expand in the turbine (station 4), producing enough power to drive the compressor and other equipments. The gasses with the ‘leftover’ pressure and temperature will leave the turbine and flow through the nozzle (station 5) where it accelerated to provide thrust (station 6).

2.4.2 Ideal Jet Propulsion Cycle

Jet propulsion cycle can be analyzed similar to Brayton cycle. The cycle is made up of four internally reversible processes

1-2 Isentropic compression of an ideal gas in a diffuser.

2-3 Isentropic compression of an ideal gas in a compressor.

3-4 Constant pressure heat addition.

4-5 Isentropic expansion of an ideal gas in a turbine.

5-6 Isentropic expansion of an ideal gas in a nozzle.

The T-s and P-V diagrams on an ideal propulsive cycle are shown in figure 2.6. These cycles are executed in a steady flow process thus analysis should be made using steady flow assumptions. When changes in potential and kinetic energy are neglected, the energy balanced equation can be expressed as

$$(q_{in} - q_{out}) + (w_{in} - w_{out}) = h_{exit} - h_{inlet} \tag{2.1}$$

Where;

q_{in}	=	Heat transfer entering the system, kJ/kg
q_{out}	=	Heat transfer leaving the system, kJ/kg
w_{in}	=	Work done on the system, kJ/kg
w_{out}	=	Work done by the system, kJ/kg
h_{in}	=	Specific enthalpy entering the system, kJ/kg
h_{out}	=	Specific enthalpy leaving the system, kJ/kg

Thus, heat transfer from and to the working fluid are

$$q_{in} = h_3 - h_2 = c_p (T_3 - T_2) \tag{2.2}$$

Where;

c_p	=	Constant pressure specific heat, kJ/kg K
T_2	=	Temperature at station 2, K
T_3	=	Temperature at station 3, K

And

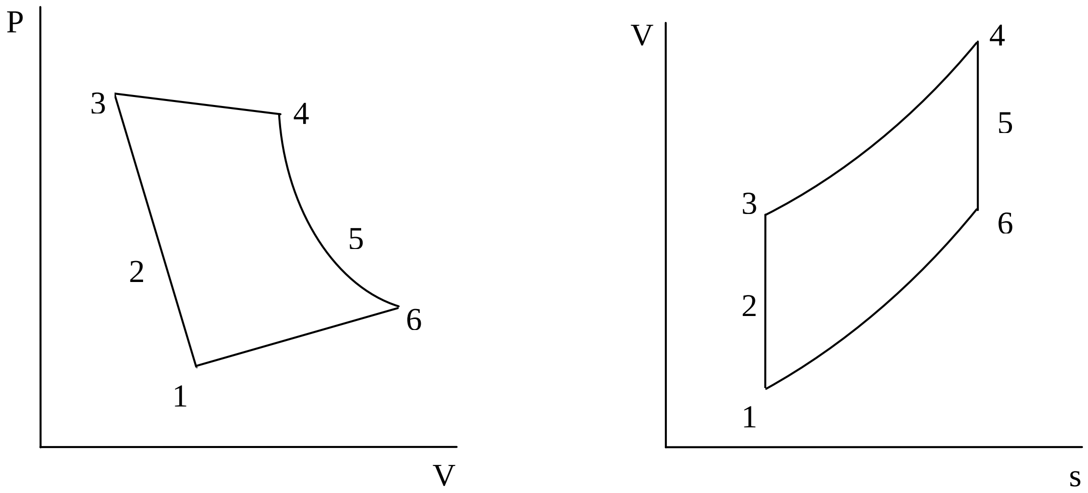
$$q_{out} = h_4 - h_1 = c_p\left(T_4 - T_1\right) \tag{2.3}$$

Where;

h_1	=	Specific enthalpy at station 1, kJ/kg
h_4	=	Specific enthalpy at station 4, kJ/kg
T_1	=	Temperature at station 1, K
T_4	=	Temperature at station 4, K

Process 2-3 and 4-5 are isentropic and $P_3 = P_4$ and $P_4 = P_1$. Thus,

$$T_3 = T_2\left(\frac{P_3}{P_2}\right)^{\frac{(k-1)}{k}} = P_4\left(\frac{T_5}{T_4}\right)^{\frac{k}{(k-1)}} \tag{2.4}$$



(a) P-V diagram (b) T-s diagram

Figure 2.6: The ideal jet propulsive cycle

Source: Cengel 2007

2.4.3 Thrust Development

Thrust develop by a jet engine is the unbalance force that is caused by the difference in momentum of low velocity air at the inlet and high-velocity exhaust at the nozzle leaving the engine. The pressure at the inlet and at the exit are identical thus, according to Newton's second law, the net thrust developed by the engine is

$$F = \left(\dot{m}V\right)_{exit} - \left(\dot{m}V\right)_{inlet} = \dot{m}\left(V_{exit} - V_{inlet}\right) \tag{2.5}$$

Where;

V_{exit} = Exit velocity of the exhaust gas, m/s

V_{in} = Inlet velocity of the exhaust gas, m/s

$\dot{m}$ = Mass flow rate of the air, kg/s

2.5 THE SMALL SCALE TURBINE JET ENGINE

Patnaik and Sanchdev (Patnaik, 2009) mentioned that there are several types of gas-turbine engines which are jet engines, amateur or home-built gas turbine, radial gas scale jet engine and micro turbines. Joshi (1997) stresses the importance use of a micro-jet engine for acquiring theoretical, computational and experimental test knowledge. He believed that by this approaches student will familiarize the complexity of the aerospace engineering system.

In recent years, manufacturers realize the demand for small-scale jet engine. The engine parts are manufactured with fined precision made with small tolerance and complete with a set of control systems that can regulate engine rotational speed, fuel flow and the exhaust temperature. It also included control unit software that includes a special sequence for the engine starting and can be controlled via remote control. These operations have made the engine operation easier and safer to run. Table 2.1 shows the characteristic of manufactured model jet engines.

Table 2.1: Characteristic of manufactured model jet engines

Characteristic	AMT Olympus	KH66	WREN MW54	JF-50 Bee
Engine Diameter, mm	130	112	87	80
Length, mm	267	230	150	173
Compressor diameter, mm	84	66	54	50
Turbine diameter, mm	84	66	55	50
Engine Weight, kg	2.475	0.930	0.8	0.8
Maximum rotation, rpm	108 000	115 000	160 000	180 000
Idle rotation, rpm	34 000	35 000	45 000	50 000
Thrust at maximum rotation, N	230	75	54	63
Pressure ratio,	4.0	2.2	2.3	2.3
Fuel Consumptions, mL/min	800	300	210	220
Mass Flow Rate, kg/s	0.45	0.23	0.18	0.2

Source: Kamps 2005

Due to lack of technical expertise, most amateur's builders use an automotive turbocharger as a compressor and turbine. The designs of combustion chamber are developed by guides from previous builder. Engine control system such as fuel delivery system and ignition system are introduced.

2.6 DESIGN CONSIDERATION OF SMALL SCALE TURBINE JET ENGINE

The designs of the small-scale turbine jet engine are a challenge due to its size and limitation. An abundance of information's regarding the construction of the jet engine to guides amateur is readily available for the net. However, these guides are basic in nature. Technical descriptions are extensive. However, it lacks of theoretical approach where knowledge of engineering none is applied. For example, the GR1 formula (Giandomenico, 2011) used for designing a combustion chamber are based on trial and error. Since these designs are not based on exact theoretical and engineering knowledge, large amounts of testing and modification are required in order for the jet engine to function and time consuming. Nevertheless, it can assist in giving a clear knowledge on building the small-scale turbine jet engine, especially for beginners.

Berg (2010) shows a successful attempt in building a small jet engine. He describes a home built jet engine design and test. His design began with an existing automotive turbocharger. A combustion chamber was fabricated, and propane was used as fuel. An oil pump was selected, and a vacuum blower used for engine starting. During testing, the combustion flame blows out and the engine could not sustain. He decided to start over the project with another turbocharger. The second attempt was more extensive, complete with a compressed air starter, a spark plug for ignition, and instrumentation to measure the rotor speed, combustion chamber pressure, exhaust gas temperature and thrust. The development was a success where the engine produces 29 N of thrust at a rotor speed of 140 000 rpm.

On the other hand, Mattingly (2002) proposes a suitable design theory for a turbine jet engine. However, his approach focuses on the development on the full-size turbine jet engine. The design theories are complex where each parameter plays a major role in the design and are dependent with each other. Nevertheless, by applying assumption and neglecting several parameters, the design of combustion chamber will be based on his approach.

2.6.1 Thermodynamic Cycle Analysis

Thermodynamic cycle of this design is similar to jet engine propulsion cycle but differ in their mechanics due to the limitation and scaling consideration. Analysis by ideal assumption at each state of the cycle will give a clear view (but not accurate) for the design. The results from the analysis will be used as a foundation for the design criteria.

2.6.2 Automotive Turbocharger

Turbocharger is famous for amateur builders to replace of turbine and compressor for their jet-engine model. Turbocharger consists of a turbine and a compressor and it is located in the exhaust manifold an automotive vehicle. It is design (Erjavec 2010) to increase engine power by compressing the air entering the engine without subsequently reduces in fuel consumptions. The high temperature and high pressure results from combustion process in the combustion chamber will spin the turbine blade, and in turn it rotates the compressor through rotating shaft. One might notice these applications are similar to jet engine cycles but there is a different in combustion section. Combustion process in spark ignition engine occurs only in one of the four cycle’s which is in power stroke whereas the combustion process of a jet engine cycles are continuous. A typical turbocharger has six components, which are a turbine, shaft, compressor, wastegate valve, actuator and center housing and rotating assembly. Therefore, given the lack of expertise and time constriction, the use of a turbocharger seems to suffice. Figure 2.7 shows a typical turbocharger system that is utilized in a vehicle.

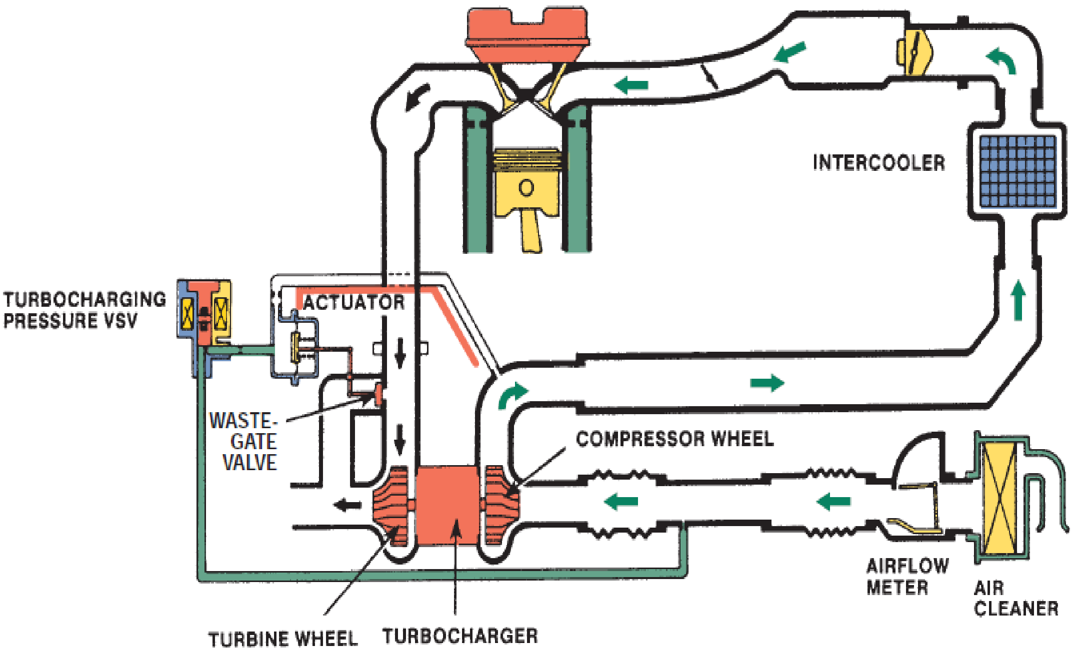


Figure 2.7: Exhaust gas and airflow of a typical turbocharger system

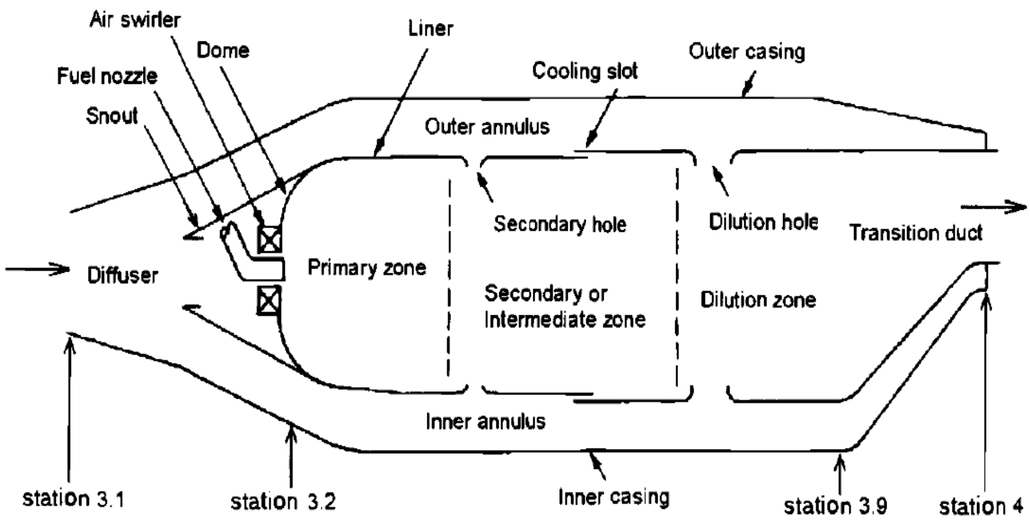
Source: Erjavec 2010

2.7 COMBUSTION CHAMBER DESIGN CHARACTERISTIC

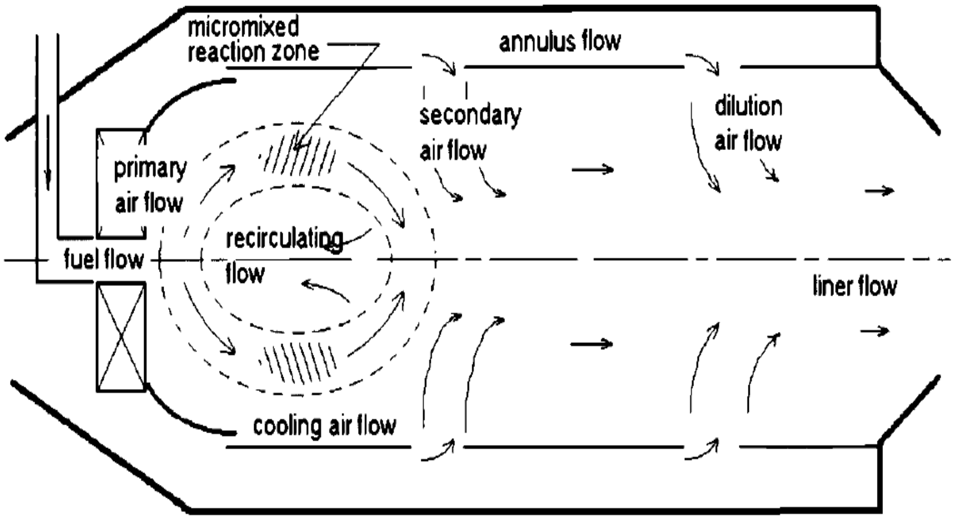
This characteristic is introduced to give a foundation in designing a combustion chamber. The final design will give a correct definitive in terms of dimension for the design. The characteristic as discussed earlier will be based on method introduced by Mattingly (2002). These methods, as mentioned before, obviously suits for design of a real size jet engine, however, due to size of the turbocharger some of the functions are made negligible. Thus, by integrated the parameter available, it has made the method possible to accommodate the turbocharger in smaller scale.

2.7.1 Principle Features

The purpose of a combustion chamber system in a jet engine is to increase thermal energy of incoming airstream from the compressor by combustion. The main requirements (Zucrow 1948) for the combustion chamber are; (a) low weight and small frontal area; (b) low pressure loss; (c) stable and efficient combustion; (d) reliability, serviceability and reasonable life, (d) uniform temperature distribution throughout the chamber. However, some of these requirements are in conflict with each other. To illustrate, consider the requirement of the combustion chamber is to have low pressure loss. To accommodate this feature the diffuser must be large, which results in a large frontal area. Thus, as with many complex engineering systems, the final design must be compromised to obtain satisfactory results. The design of jet engine combustion chamber is complex as it consists of chemical thermodynamic, gas-phase chemical kinetics and chemical reactor theory.



(a) Main features of the combustion chamber



(b) Flow pattern of the combustion chamber

Figure 2.8: General principle of a combustion chamber

Source: Mattingly 2002

The general principle of a combustion chamber consists of four main features, which are diffuser, primary zone, secondary zone and dilution zone as shown in figure 2.8. Incoming air from the compressor is undesirably at high velocity thus it must diffuse to a lower subsonic velocity. This is done by expanding the frontal area where the air enters. The diffuse airstream will be divided for distribution into the liner and annulus. The airstreams that entered the liner will flow through an air swirler into the primary zone where the flow will become turbulence. The remaining flow from the annulus will then enter the liner through holes at the walls of the liner.

In the primary zone, combustion process will occur where incoming fuel is mixed with incoming air. The turbulence airflow by the air swirler causes both the fuel stream and incoming air trapped in a recirculation bubble within the primary zone. This process is continuous and is called flameholding. From the primary zone, partially mixed, actively burning and incomplete burned gases flows downstream into the secondary zone where it will continue to burn down toward completion of the air coming from the secondary holes. Consequently, the combustion should be complete at the end of the secondary zone. The remaining airflow will enter the dilution zone before it enters the turbine where the temperature of the air flow is decreased to accommodate the turbine material limit.

2.7.2 Stoichiometric Ratio

Stoichiometric ratio is an ideal air/fuel ratio of combustion of a fuel with air. The attention to the jet engine is restricted to hydrocarbon as it is in current use of an aircraft jet engine.

The ideal combustion process will occur when all the fuels are completely mixed with air to form reactant consists of water vapor H_2O, and CO_2. The ideal mixture of fuel is represented by a general atom balance equation,

$$C_xH_y + \left(x + \frac{y}{4}\right)\left[O_2 + \frac{79}{21}N_2\right] \rightarrow xCO_2 + \frac{y}{2}H_2O + \frac{79}{21}\left(x + \frac{y}{4}\right)N_2 \qquad (2.6)$$

The ideal stoichiometry air ratio can be determined readily from the ratio of a molar coefficient of the reactant on the left-hand side of equation 2.6. The stoichiometry serves as an ideal stoichiometry fuel/air ratio and is expressed as mass- basis given by

$$f_{st} = \frac{36x + 3y}{103(4x + y)} \qquad (2.7)$$

2.7.3 The Bragg Criterion

The possibilities of sustaining a self-propagating flame in steady flow combustion chamber can be achieved even though the velocity of reactant entering the greatly exceed the turbulence flame speed of a stiochiometric mixture of fuel and air at the entrance. The possibility is that the insertion of a flame holder who can separate the flow in the aerodynamic wake behind the flameholder which causes the flow to recirculate within the primary zone. Fractions of the reactant can reside for sufficiently long time to mix with hot product gases thus sustaining a continuous combustion reaction.

The mixing and chemical process that enables flameholding can be better understood by using chemical engineering concept called reaction engineering or chemical reactor theory. However, these concepts are rather complex but yet are important in understanding the combustion system, thus it will only be discussed briefly in this section.

A self stabilized, steady-state, and steady flow combustion system as in a gas-turbine engine requires two discrete sub region, which is a primary and secondary zone. An idealized combustion chamber with a primary zone and a secondary zone is known as Bragg combustion chamber (Bragg, 1956) as shown in figure 2.9.

The ideal primary zone is a region where intense recirculation in which reactant is mixed with previously burned gases that continue to recirculated within the primary zone. These reactions are necessarily occurring at a combustion efficiency of less than 80%. The secondary zone meanwhile, is to allow sufficient convective stay time for the primary zone effluent to burn out - at approaching 100% efficiency before exiting the chamber.

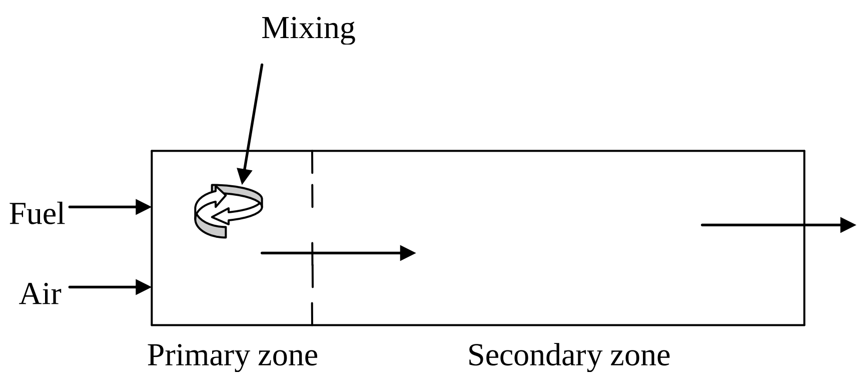


Figure 2.9: An idealized Bragg combustion chamber with a primary and secondary zone

2.7.4 Air Partioning

The term air partioning has been used since the beginning of the development of a gas-turbine jet engine. It is best described as the portion of airflow to be delivered for liner cooling, primary zone, secondary zone and dilution zone process. It is necessary to design a combustion system with near stiochiometric primary zone to ensure combustion stability. Since the gas temperature within the combustion chamber is extensively high due to combustion process, large amounts of cooling are needed to protect the liner. This can be achieved by dumping a certain amount of air into the hot gas to dilute hot gasses to a lower temperature.

Air partioning is used to control the temperature results from combustion product gases to fulfill certain set of requirements such as the integrity of liner material and to protection of the turbine. The liner cooling and dilution airflows are used only for thermal protection of the liner and the turbine respectively. Note that the liner cooling is not taken into consideration in this project.

Consider combustion is complete with the combustion chamber exit; the maximum exit temperature difference within the combustion chamber can be evaluated using the following equation.

$$\Delta T_{\max} = \frac{\left(T_{t4} - T_{3.1}\right)}{\phi_4} \tag{2.8}$$

Where;

T_{t4} = Maximum temperature at station 4

$T_{3.1}$ = Temperature at station 3.1

ϕ_4 = Equivalent ratio at station 4

The design principle for portioning of air at primary zone for the Bragg Criterion, the primary zone equivalence ratio can be solved using

$$\phi_{PZ} = \frac{\left(T_g - T_{3.1}\right)}{\varepsilon_{PZ}\Delta T_{\max}} \quad (2.9)$$

Where;

T_g = Liner gas temperature

ε_{PZ} = Combustion efficiency at primary zone

The primary zone air flow fractions are

$$\mu_{PZ} = \frac{\phi_4}{\phi_{PZ}} \quad (2.10)$$

Where the equivalent ratio at station 4 can be evaluated using

$$\phi_4 = \frac{\dot{m}_{fMB}}{f_{st} m_{3.1}} \quad (2.11)$$

It is desired to have combustion complete at secondary zone exit. The desired equivalence ratio at secondary zone exit is determined to be

$$\phi_{SZ} = \frac{\left(T_g - T_{3.1}\right)}{\Delta T_{\max}} \quad (2.12)$$

Thus, the secondary mass flow fraction can be determined to be

$$\mu_{SZ} = \frac{\phi_4}{\phi_{SZ}} - \frac{\phi_4}{\phi_{PZ}} \quad (2.13)$$

Lastly, the left over air where it is dump into dilution zone and can be expressed as;

$$\mu_{DZ} = 1 - (\mu_{PZ} + \mu_{SZ} + \mu_C) \quad (2.14)$$

2.7.5 Dome and Liner

The next step is to determine the optimal dome and liner height. The optimal liner-to-reference height ratio is

$$\alpha_{opt} = 1 - \left(\frac{m_A}{m_r}\right)^{\frac{2}{3}} \left(\frac{\Delta P_t}{q_r}\right)^{-\frac{1}{3}} \tag{2.15}$$

Where, the ratio of the liner to reference flow rate is defined by

$$\frac{m_A}{m_r} = \mu_{SZ} + \mu_{DZ} \tag{2.16}$$

And the total pressure loss coefficient is

$$\frac{\Delta P_t}{q_r} = \frac{\left(P_{t3.2} - P_{t4}\right)}{\left(P_{t3.2} - P_{3.2}\right)} \tag{2.17}$$

Thus, the outer liner height can be determined by

$$H_L = \alpha_{opt} H_r \tag{2.18}$$

Where;

H_r = Height of inner liner, mm

2.7.6 Primary Zone

With the references of velocity at the diffuser (based on ideal assumption) velocity of all jets can be determined.

$$Vj = U_r \sqrt{\frac{\Delta P_t}{q_r}} \tag{2.19}$$

Where;

U_r = Velocity at station 3.2

The swirl number for the design is obtained from

$$S' = \frac{2}{3}\tan\alpha \left[\frac{1-\left(\frac{r_h}{r_t}\right)^3}{1-\left(\frac{r_h}{r_t}\right)^2} \right] \tag{2.20}$$

Where;

r_h = Hub radius, mm

r_t = Swirler radius, mm

Also, it is known that

$$\alpha = \alpha_{opt} \tag{2.21}$$

The axial length measured from the can be estimated to be

$$L_{PZ} \sim 2S' r_t \tag{2.22}$$

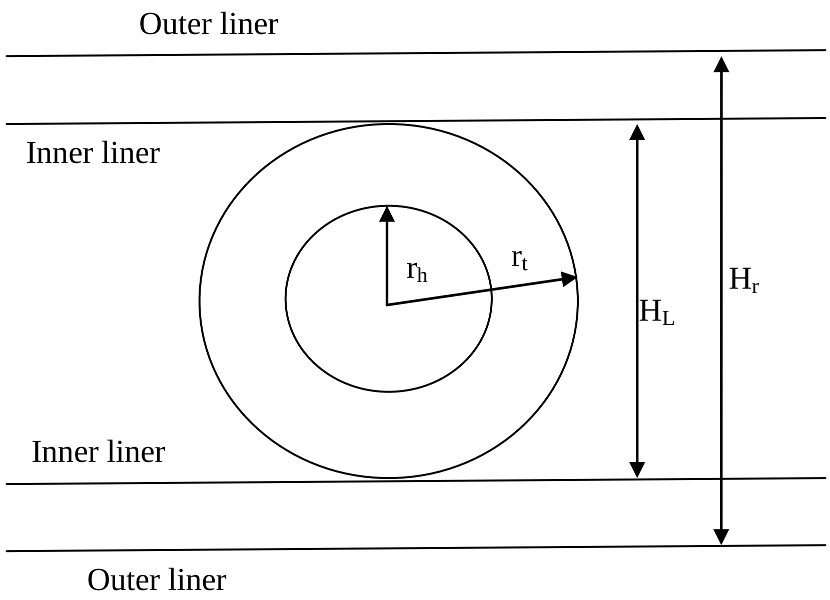


Figure 2.10: Layout distribution of the primary air swirler

2.7.7 Secondary Zone

It is essential to have combustion complete by the secondary zone exit. To design the number and location of the secondary zone air hole, it is necessary to calculate various dynamic pressure ratios.

$$\frac{q_j}{q_r} = \frac{\Delta P_t}{q_r} \tag{2.23}$$

$$\frac{q_A}{q_r} = \left(\frac{\mu_{SZ} + \mu_{DZ}}{1-\alpha}\right)^2 \tag{2.24}$$

and,

$$\frac{q_L}{q_r} = \tau_{PZ}\left(\frac{\mu_{PZ}}{\alpha_{opt}}\right)^2 \tag{2.25}$$

The maximum jet penetration of the jet centerline is obtained by

$$\frac{Y_{\max}}{d_j} = 1.15\sqrt{\frac{q_j}{q_r}\frac{q_r}{q_L}\left(1 - \frac{q_A}{q_r}\frac{q_r}{q_j}\right)} \tag{2.26}$$

The required jet area is obtain by

$$d_j = \frac{1}{4} H_L \left(\frac{Y_{\max}}{d_j}\right)^{-1} \tag{2.27}$$

The secondary jet enters the liner at a known angle and is obtain by

$$\sin\theta = \sqrt{1 - \frac{q_A}{q_r}\frac{q_r}{q_j}} \tag{2.28}$$

Thus, the diameter of the secondary hole is

$$d_h = \frac{d_j}{\sqrt{C_D}\,\sin\theta} \tag{2.29}$$

Lastly, the length of the secondary zone is approximated to be

$$L_{SZ} \simeq 2H_L \tag{2.30}$$

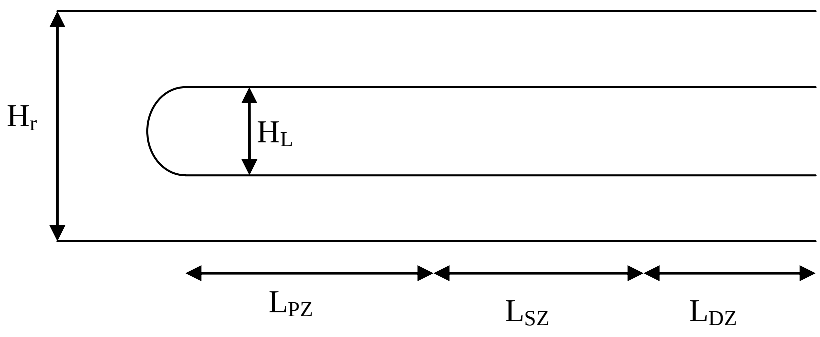


Figure 2.11: Layout distribution of the combustion chamber

2.7.8 Dilution Zone

The desired requirement for dilution zone is that any leftover air is dumped into this section. The same procedure is followed as of the secondary zone except the goal $Y_{max}=H_L/3$ rather than $Y_{max}=H_L/4$. Due to secondary jet have reduced the annulus airflow, only the dilution air now flows into the annulus, thus the annulus flow dynamic pressure is reduced to

$$\frac{q_A}{q_r}=\left(\frac{\mu_{DZ}}{1-\alpha}\right)^2 \tag{2.31}$$

Due to the increase by the airflow in the secondary air

$$\frac{q_L}{q_r}=\tau_{PZ}\left(\frac{\mu_{PZ}+\mu_{SZ}}{\alpha}\right)^2 \tag{2.32}$$

As previously stated, the maximum jet penetration of the jet centerline is obtain by

$$\frac{Y_{max}}{d_j}=1.15\sqrt{\frac{q_j}{q_r}\frac{q_r}{q_L}\left(1-\frac{q_A}{q_r}\frac{q_r}{q_j}\right)} \tag{2.33}$$

So, the required jet area is obtained

$$d_j=\frac{1}{3}H_L\left(\frac{Y_{max}}{d_j}\right)^{-1} \tag{2.34}$$

The secondary jet enters the liner at a known angle and is obtain by

$$\sin\theta=\sqrt{1-\frac{q_A}{q_r}\frac{q_r}{q_j}} \tag{2.35}$$

Thus, the diameter of the dilution hole is

$$d_h = \frac{d_j}{\sqrt{C_D}\sin\theta} \tag{2.36}$$

Where;

C_D = Discharge coefficient of jet hole (Dittrich 1956)

Lastly the length of the dilution zone is approximated to be

$$L_{SZ} \simeq 1.5 H_L \tag{2.37}$$

The layout distribution of the primary air swirler the combustion chamber is shown is figure 2.10 and 2.11 respectively.

CHAPTER 3

METHODOLOGY

3.1 INTRODUCTION

This chapter presents the experimental method used and is discussed clearly step by step. The systematic planning of methodology is crucial to keep the project running smoothly. Detail design, fabrication process and testing method are discussed in this chapter.

3.2 PROJECT FLOW CHART

A sequence of works has been planned carefully and is shown in figure 3.1. This project is divided into three sections, which are design, fabrication and testing.

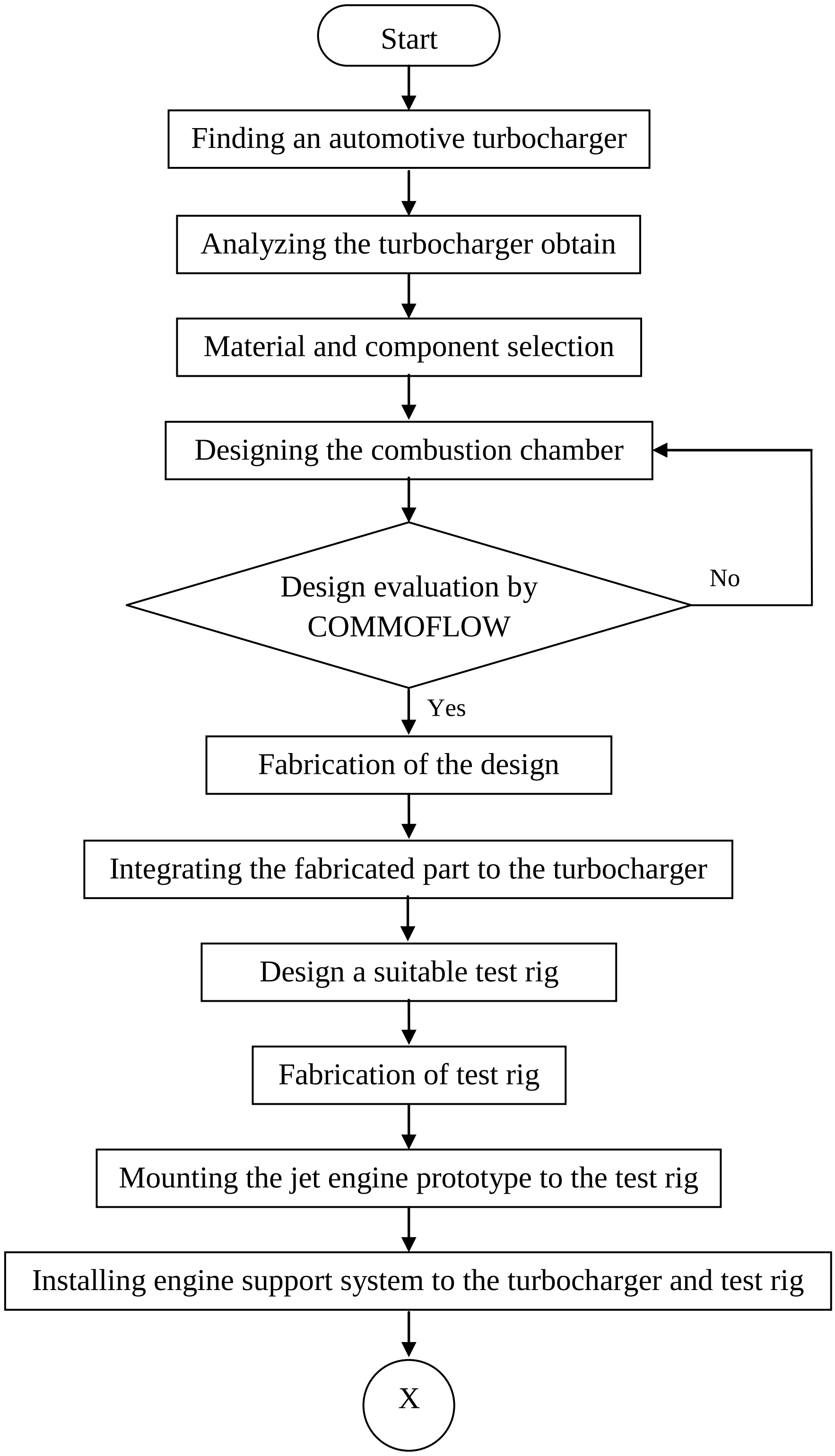


Figure 3.1: The methodology flow chart

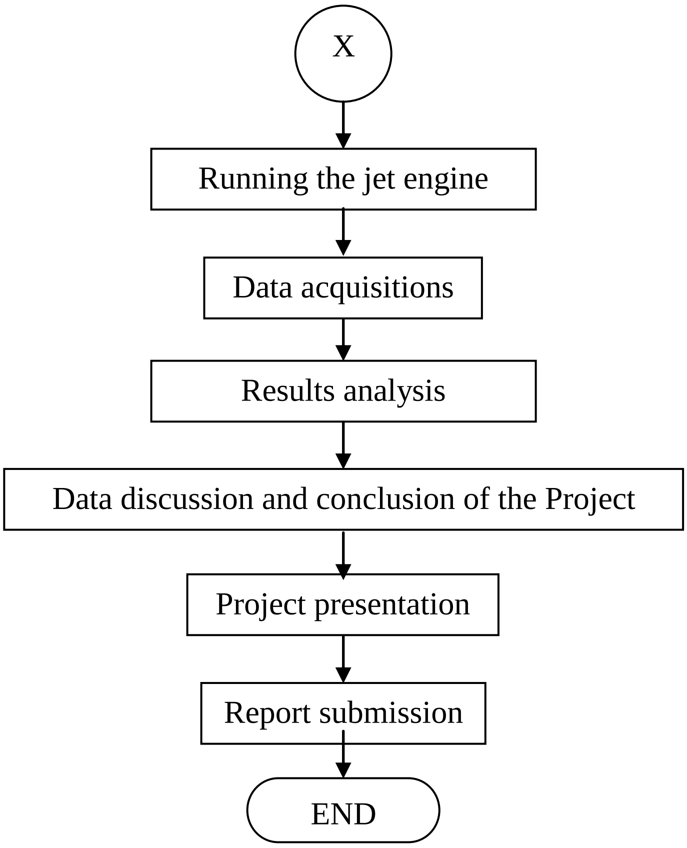


Figure 3.1: Continued

3.3 PRELIMINARY LAYOUT

Figure 3.2 shows the overall design of the jet engine. The combustion chamber is mounted at the bottom of the turbocharger where it will connect to the compressor outlet (station 3) and turbine inlet (station 4). The exhaust nozzle, on the other hand, will be connected to the turbine outlet (station 5).

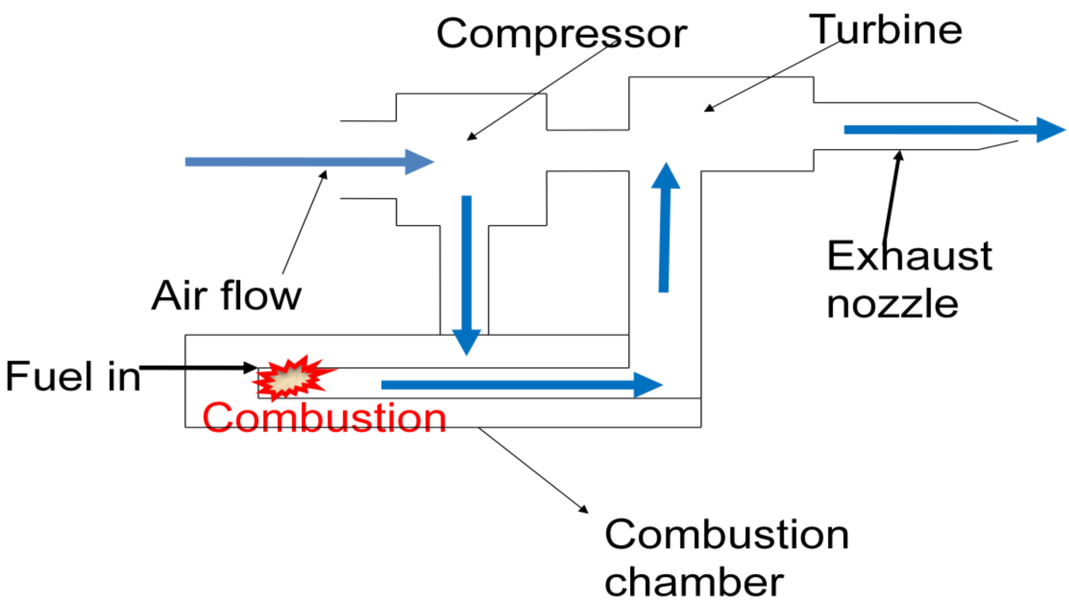
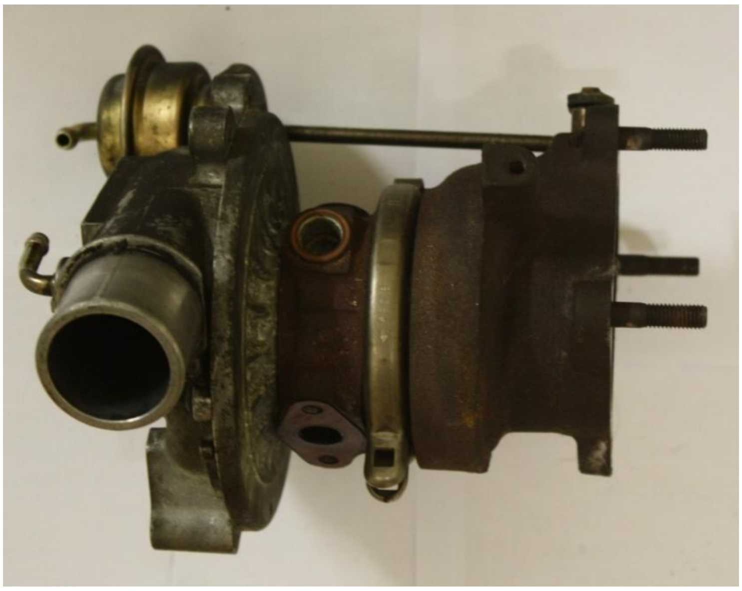


Figure 3.2: Overall design of the jet engine

3.4 TURBOCHARGER SELECTION

Due to limitation of cost, second hand turbocharger (figure 3.3) is chosen. The characteristic of the turbocharger is shown in table 3.1.



(a) Front view

(b) Isometric view

Figure 3.3: The second hand turbocharger used

Table 3.1: Characteristic of turbocharger used

Characteristic	**Value**
Volume flow rate range	0.023 - 0.18 m^3/s
Maximum pressure ratio	2.8
Maximum speed	180 000 rpm
Maximum allowable temperature (Turbine)	950 K
Weight (without wastegate valve)	3.2 kg
Compressor inlet diameter	4.1 cm
Compressor outlet diameter	3.6 cm
Turbine inlet diameter	4.5 cm
Turbine outlet diameter	4.0 cm

Source: IHI Turbo 2005

3.5 THERMODYNAMIC PARAMETER ANALYSIS

As discuss in previous chapter, the designs of combustion chamber are based on the method discussed by Mattingly (2002). Several parameters must be first be determined by applying simple thermodynamic analyses based on ideal condition assumption, using equation 2.1 to 2.5 and variable from table 3.1. Detail calculations are shown below, and the result is summarized in table 3.2.

3.5.1 Detail Calculation of Thermodynamic Parameter Analysis

The assumptions for the calculation are as follows:

1. Steady operating conditions exist.
2. Kinetic and potential energy are negligible except at the exhaust nozzle exit.
3. The turbine and the compressor are rotates at 100% efficiency.
4. The turbine work output is equal to the compressor work input.

Standard T-s diagram from figure 2.3 (b) is used for clarity.

Process 2-3 (isentropic compression of an ideal gas in a compressor): There is no diffuser at the compressor inlet, thus, standard atmospheric assumptions are applicable where the pressure is 101.3 kPa, and the temperature is 300 K. The velocity at the inlet can be determined by knowing that the area of the compressor inlet. The compressor inlet area is

$$A = \frac{\pi d^2{}_{comp,in}}{4} = \frac{\pi (0.041)^2}{4} = 0.00132 \text{ m}^2$$

Thus the velocity is

$$V = \frac{\dot{V}}{A} = \frac{0.18}{0.00132} = 136.36 \text{ m/s}$$

The pressure and temperature at station 3 can be determined by

$$P_3 = P_2 r_p = (2.8)(101.3) = 283.64 \text{ kPa}$$

$$T_3 = T_2\left(\frac{P_3}{P_2}\right)^{\frac{(k-1)}{k}} = (300)\left(\frac{283.64}{101.3}\right)^{\frac{(1.4-1)}{1.4}} = 402.606\,\mathrm{K}$$

Process 4-5 (Isentropic expansion of an ideal gas in a turbine): Neglecting the kinetic energy changes across the compressor and the turbine, by using equation 2.2 and equation 2.3, the temperature and pressure at the turbine exit are

$$w_{comp,in} = w_{turb,out}$$

$$h_3 - h_2 = h_4 - h_5$$

$$c_p\left(T_3 - T_2\right) = c_p\left(T_4 - T_5\right)$$

$$T_5 = T_4 - T_3 + T_2 = 1173 - 402.606 + 300 = 1070.394\,\mathrm{K}$$

Since combustion occurs at a constant pressure thus, $P_3 = P_4$, the pressure at station 5 is

$$P_5 = P_4\left(\frac{T_5}{T_4}\right)^{\frac{k}{(k-1)}} = \left(283.64\right)\left(\frac{1070.394}{1173.000}\right)^{\frac{1.4}{(1.4-1)}} = 205.886\,\mathrm{K}$$

Process 5-6 (isentropic expansion of an ideal gas in a nozzle); to find the air velocity, it is essential to determine the nozzle exit temperature.

$$T_6 = T_5\left(\frac{P_6}{P_5}\right)^{\frac{(k-1)}{k}} = 1070.394\left(\frac{101.300}{205.886}\right)^{\frac{(1.4-1)}{1.4}} = 874.055\,\mathrm{K}$$

By applying steady state equation

$$h_6 + \frac{V_6^{\,2}}{2} = h_5 + \frac{V_5^{\,2}}{2}$$

$$0 = c_p\left(T_6 - T_5\right)\left(\frac{{V_6}^2}{2}\right)$$

$$V_6 = \sqrt{2c_p\left(T_5 - T_6\right)} = \sqrt{2\left(1005\right)\left(1070.394 - 874.055\right)} = 627 \text{ K}$$

Additionally, the mass flow rate entering the compressor is equal to the mass flow rate through the engine, thus,

$$V_6 = \sqrt{2c_p\left(T_5 - T_6\right)} = \sqrt{2\left(1005\right)\left(1070.394 - 874.055\right)} = 627 \text{ m/s}$$

$$\dot{m} = \rho VA = \left(1.2\right)\left(136.46\right)\left(0.00132\right) = 0.216 \text{ kg/s}$$

Hence, by using equation 2.5, the thrust produce is

$$F = \dot{m}\left(Ve - Vi\right) = \left(0.216\right)\left(627 - 136.46\right) = 103.957 \text{ N}$$

Table 3.2: Relevance calculated parameter at different stations

Station	Parameter	Value
1	Inlet velocity	136.46 m/s
1	Inlet temperature	300 K
1	Inlet pressure	101.3 kPa
1	Compressor ratio	2.8
2	Compressor outlet-temperature	402.606 K
2	Compressor outlet-pressure	283.64 kPa
4	Turbine inlet pressure	283.64 kPa
4	Turbine inlet-temperature	1173 K
5	Turbine outlet-pressure	205.886 kPa
5	Turbine outlet-temperature	1070.394 K
6	Nozzle exit-pressure	101.3 kPa
6	Nozzle exit-velocity	627 m/s
6	Maximum thrust produce	107.68 N

3.6 COMBUSTION CHAMBER DESIGN

The detail design is calculated using equation 2.7 until equation 2.37 is divided into four parts. The parameter for the calculation is based on table 3.2. The assumption was made by trial and error to obtain satisfactory results. The calculated and assumed parameters are shown in figure 3.2. The completed design is shown in figure 3.6 and 3.7

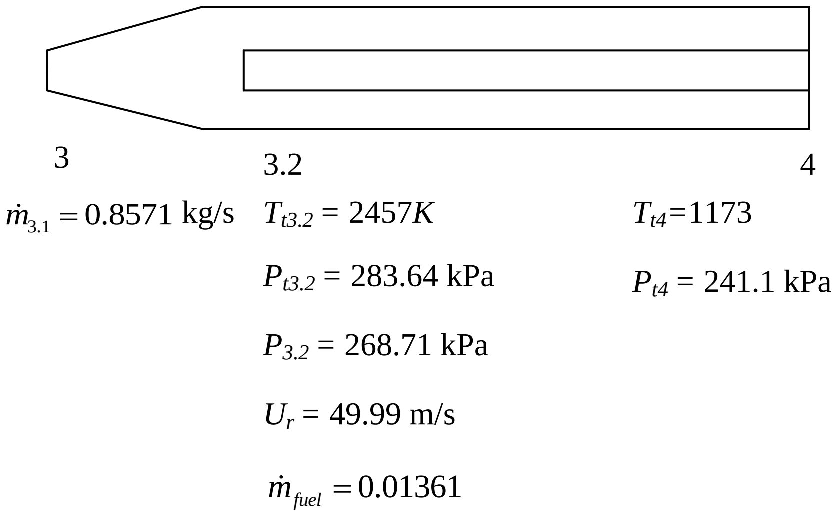


Figure 3.4: Summary of calculated and assumed parameter at different station in the combustion chamber

3.6.1 Air Partioning

The first step is to determine the ideal stoichiometric fuel/air for the combustion process. Since the fuel used is a LPG (Liquefied Petroleum Gas) which mainly consists of propane C_3H_8, thus by using equation 2.7, the ideal stoichiometric fuel/air ratio is

$$fst = \frac{36x + 3y}{103(4x + y)} = \frac{36(3) + 3(8)}{103[4(3) + (8)]} = 0.064$$

Then, the equivalent ratio at station 4 can be evaluated using equation 2.11. The mass flow rate of fuel is assumed to be 0.013 kg/s and based from ideal assumptions, the mass flow rate of air at station 3.1 is 0.085 kg/s, substitute the following parameter into equation 2.1,

$$\phi 4 = \frac{\dot{m}_{fMB}}{f_{st} m_{3.1}} = \frac{0.013}{(0.064)(0.85)} = 0.24$$

Considering that the combustion is complete at station 4, the maximum exit temperature difference within the combustion chamber can be evaluated using equation 2.8,

$$\Delta T_{\max} = \frac{(T_{t4} - T_{3.1})}{\phi_4} = \frac{(1173 - 402.61)}{0.24} = 3029.96\,\mathrm{K}$$

The combustion temperature is assumed to be equal to the temperature of the inner liner. The design target for liner gas temperature can be calculated by rearranging equation 2.9, by assuming that ϕ_{PZ} is 0.7 and ε_{PZ} is 0.8,

$$T_g = T_{3.1} + \phi_{PZ}\varepsilon_{PZ}\Delta T_{\max} = 402.61 + (0.7 \times 0.8 \times 3029.96) = 2099.39\,\mathrm{K}$$

With, $T_g = 2143.6093\,\mathrm{K}$ the primary zone air flow fractions is

$$\mu_{PZ} = \frac{\phi_4}{\phi_{PZ}} = \frac{0.24}{0.7} = 0.34$$

The desired equivalence ratio at secondary zone exit is

$$\phi_{SZ} = \frac{(T_g - T_{3.1})}{\Delta T_{\max}} = \frac{2099.39 - 402.61}{3029.96} = 0.56$$

From equation 2.13;

$$\mu_{SZ} = \frac{\phi_4}{\phi_{SZ}} - \frac{\phi_4}{\phi_{PZ}} = \left(\frac{0.24}{0.56}\right) - \left(\frac{0.24}{0.7}\right) = 0.086$$

The remaining mass flow can be calculated using equation 2.14. Since the design does not consist of liner cooling due to hard to manufacture, the value is assumed,

$$\mu_{DZ} = 1 - (\mu_{PZ} + \mu_{SZ} + \mu_C) = 1 - (0.34 + 0.086 + 0.09) = 0.49$$

3.6.2 Dome and Liner

With the air partioning is determined; it is now possible to determine the inner liner height. The outer liner height, however, for the simplicity of design, a good choice is to assume it to be 68.6 mm. The next step is to determine the optimal dome. By equation 2.16 and 2.17, the total pressure loss coefficient and ratio of annulus reference flow can be calculated respectively. From ideal assumption calculation, the combustion occurs at a constant pressure. However, that is impossible, since there always pressure loss at throughout the combustion chamber, both value of pressure at station 3.2 and station 4 is assumed slightly lesser. Hence,

$$\frac{m_A}{m_r} = \mu_{SZ} + \mu_{DZ} = 0.086 + 0.49 = 0.58$$

and

$$\frac{\Delta P_t}{q_r} = \frac{(P_{t3.2} - P_{t4})}{(P_{t3.2} - P_{3.2})} = \frac{(283.64 - 241.1)}{(283.64 - 268.71)} = 2.85$$

The optimal liner-to-reference height ratio is

$$\alpha_{opt} = 1 - \left(\frac{m_A}{m_r}\right)^{\frac{2}{3}} \left(\frac{\Delta P_t}{q_r}\right)^{-\frac{1}{3}} = 1 - (0.58)^{\frac{2}{3}} (2.85)^{-\frac{1}{3}} = 0.59$$

Thus, the inner liner height is

$$H_L = \alpha_{opt} H_r = 0.59 \times 68.6 = 40.5\,\text{mm}$$

3.6.3 Primary Zone

It is desirable to assume the reference velocity at the diffuser to be 164. Thus the velocity of all jets from equation 2.19,

$$Vj = Ur\sqrt{\frac{\Delta P_t}{q_r}} = 49.99\left(\sqrt{2.85}\right) = 84.39\ \text{m/s}$$

The hub radius and swirler radius is assumed. Since the r_h and r_t located within the inner liner, it must be smaller than Hr. It is desirable to assume r_t half of H_L and r_t accommodate the fuel inlet. From equation 2.20 and 2.21, the swirl number for the design is,

$$S' = \frac{2}{3}\tan\alpha\left[\frac{1-\left(\frac{r_h}{r_t}\right)^3}{1-\left(\frac{r_h}{r_t}\right)^2}\right] = \frac{2}{3}\tan\left(45\right)\left[\frac{1-\left(\frac{19}{20.25}\right)^3}{1-\left(\frac{19}{20.25}\right)^2}\right] = 0.97$$

and the axial length measured from the can be estimated to be

$$L_{PZ} \sim 2S' r_t \cong 2\left(0.97\right)\left(20.25\right) \cong 39.5mm$$

3.6.4 Secondary Zone

To design the number and location of the secondary zone air hole, it is necessary to calculate various dynamic pressure ratios.

$$\frac{q_j}{q_r} = \frac{\Delta P_t}{q_r} = 2.85$$

$$\frac{q_A}{q_r} = \left(\frac{\mu_{SZ} + \mu_{DZ}}{1-\alpha_{opt}}\right)^2 = \left(\frac{0.086+0.49}{1-0.59}\right)^2 = 1.98$$

and

$$\frac{q_L}{q_r} = \tau_{PZ}\left(\frac{\mu_{PZ}}{\alpha_{opt}}\right)^2 = 0.5\left(\frac{0.34}{0.59}\right)^2 = 0.17$$

The maximum jet penetration of the jet centerline is obtained by

$$\frac{Y_{\max}}{d_j} = 1.15\sqrt{\frac{q_j}{q_r}\frac{q_r}{q_L}\left(1-\frac{q_A}{q_r}\frac{q_r}{q_j}\right)} = 1.15\sqrt{\frac{2.85}{0.17}\left(1-\frac{1.98}{2.85}\right)} = 2.60$$

The required jet area is obtained

$$d_j = \frac{1}{4}H_L\left(\frac{Y_{\max}}{d_j}\right)^{-1} = \frac{1}{4}(68.6)(2.60)^{-1} = 6.60 \text{ m/s}$$

The secondary jet enters the liner at a known angle and is obtain by

$$\sin\theta = \sqrt{1-\frac{q_A}{q_r}\frac{q_r}{q_j}} = \sqrt{1-\frac{1.98}{2.85}} = 0.56$$

Thus, the diameter of the secondary hole is

$$d_h = \frac{d_j}{\sqrt{C_D}\sin\theta} = \frac{6.60}{\left(\sqrt{0.64}\right)(0.56)} = 14.73 \text{ mm}$$

The diameter of the secondary hole is too large for the design. A good choice is to divide it by half thus,

$$d_h \approx 8 \text{ mm}$$

The hole should be space evenly in a single lateral line within the wall of the inner liner. Lastly, the length of the secondary zone is approximated to be

$$L_{SZ} \simeq 2H_L \simeq 2(40.5) = 81 \text{ mm}$$

3.6.5 Dilution Zone

The annulus flow dynamic pressure is reduced to

$$\frac{q_A}{q_r} = \left(\frac{\mu_{DZ}}{1-\alpha_{opt}}\right)^2 = \left(\frac{0.49}{1-0.59}\right)^2 = 1.43$$

Due to the increase by the airflow in the secondary air

$$\frac{q_L}{q_r} = \tau_{PZ}\left(\frac{\mu_{PZ}+\mu_{SZ}}{\alpha_{opt}}\right)^2 = 0.5\left(\frac{0.34+0.086}{0.59}\right)^2 = 0.12$$

As previously stated, the maximum jet penetration of the jet centerline is obtained by

$$\frac{Y_{\max}}{d_j} = 1.15\sqrt{\frac{q_j}{q_r}\frac{q_r}{q_L}\left(1-\frac{q_A}{q_r}\frac{q_r}{q_j}\right)} = 1.15\sqrt{\frac{2.85}{1.43}\left(1-\frac{1.43}{2.85}\right)} = 1.15$$

so that the required jet area is obtained by

$$d_j = \frac{1}{3}H_L\left(\frac{Y_{\max}}{d_j}\right)^{-1} = \frac{1}{3}(68.6)(1.15)^{-1} = 19.88$$

and the secondary jet enters the liner at a known angle and is obtain by

$$\sin\theta = \sqrt{1-\frac{q_A}{q_r}\frac{q_r}{q_j}} = \sqrt{1-\frac{1.43}{2.85}} = 0.71$$

Thus, the diameter of the dilution hole is

$$d_h = \frac{d_j}{\sqrt{C_D}\sin\theta} = \frac{19.88}{\left(\sqrt{0.64}\right)(0.71)} = 35 \text{ mm}$$

The diameter of the secondary hole is too large for the design. A good choice is to divide it by five thus,

$$d_h \approx 9 \text{ mm}$$

The hole should be space evenly in a single lateral line within the wall of the inner liner. Lastly, the length of the dilution zone is approximated to be

$$L_{SZ} \simeq 1.5H_L \simeq 1.5(40.5) \simeq 60.8 \text{mm}$$

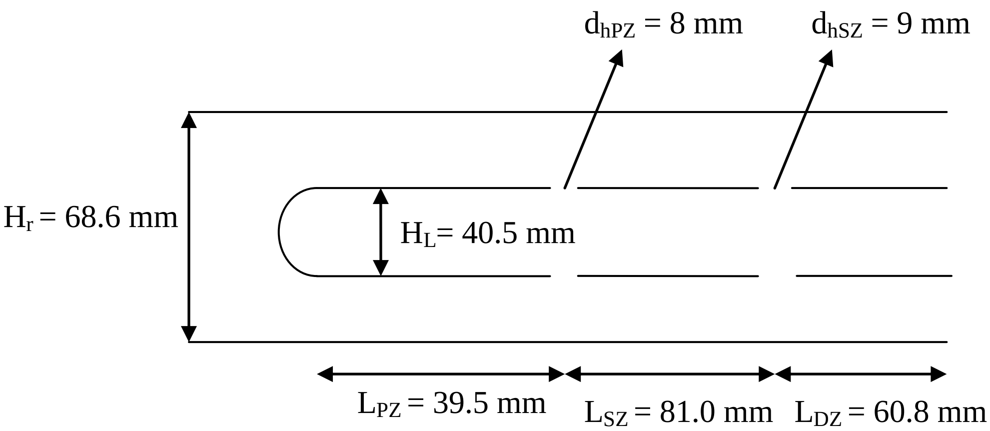


Figure 3.5: Dimension of the combustion chamber

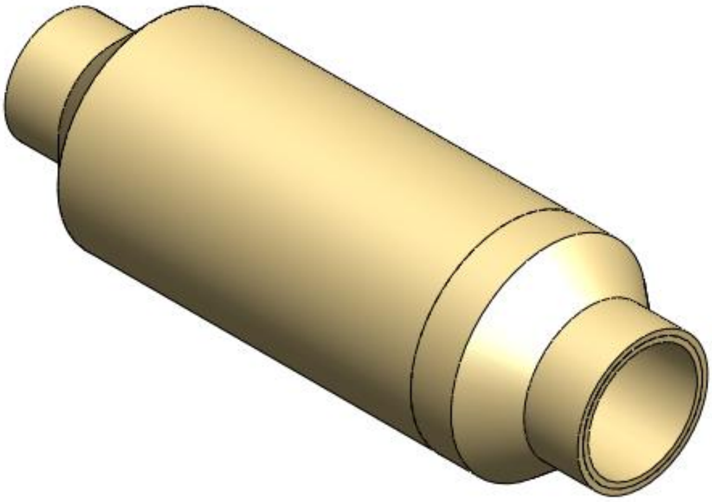


Figure 3.6: Isometric view of combustion chamber

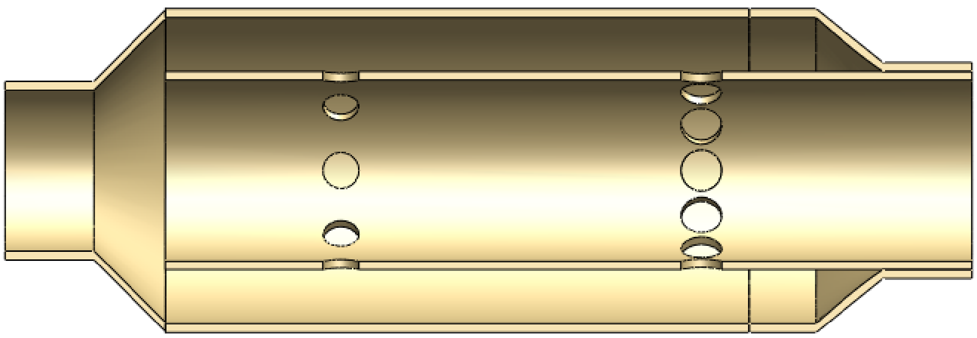


Figure 3.7: Sectional view of combustion chamber

3.7 EXHAUST NOZZLE DESIGN

A simple exhaust nozzle was design. The overall dimension of the design was based on the turbine diameter, and a simple straight exhaust with 7 degree nozzle was made (Kamps 2005).

Figure 3.8: Isometric view of exhaust nozzle design

3.8 MATERIAL AND COMPONENT SELECTION

Stainless steel is the best material as it high melting temperature, however, due to limitation on the availability of the material, as suggested by Kamps (2003) the second best material best material is mild steel. Thus, cylinder mild steel will be use to fabricate the combustion chamber.

Operation of these jet engines is not independent and thus, it cannot run without a set of a support system. Such support systems required are lubricating system, ignition system, and fuel delivery system. To satisfy these requirements, a large number of components are essential. All of these systems are installed on a test rig and connected to the turbocharger accordingly. Complete list of all components used is shown in APPENDIX A.

3.9 FABRICATION

Fabrication of the combustion chamber and the exhaust nozzle was made, and the material used is mild steel. For the combustion chamber, additional part was made to assemble it to the turbocharger which is a holder at attached at the end of the combustion chamber and a diffuser attached at the front.

(a) Top View **(b)** Side View

Figure 3.9: Fabricated outer liner

Figure 3.10: Side view of fabricated inlet diffuser

Figure 3.11: Side view of fabricated inner liner

Figure 3.12: Fabricated inner liner with spark plug and holder attached

3.10 ENGINE SUPPORT SYSTEM

3.10.1 Lubrication System

Lubrication system consists of an aquarium pump connected to the turbocharger oil inlet. An oil reservoir was made to supply additional lubricant oil. The system is a closed system where the lubricant oil recirculates back to the reservoir from the turbocharger oil outlet.

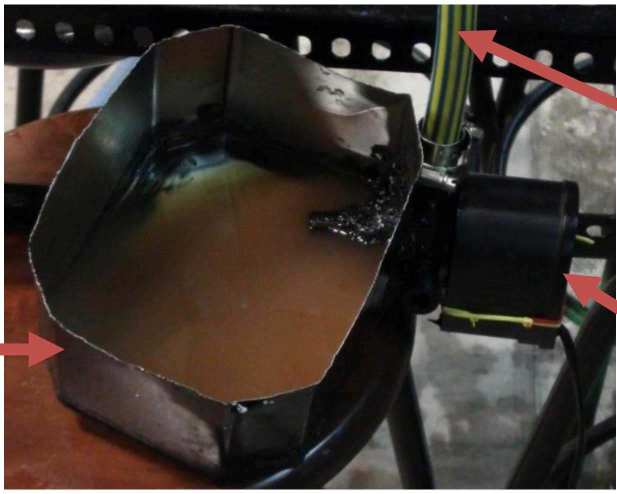


Figure 3.13: Oil reservoir

Figure 3.14: Water pump

3.10.2 Fuel Delivery System

A connection was made from the LPG tank to the combustion chamber. Pressure regulator was used to control the fuel mass flow rate that enters the combustion chamber.

Figure 3.15: Fuel used

3.10.3 Ignition System

The ignition system consists of an automotive spark plus park plug, an ignition coil, a condenser, a relay and powered by a 12 volts battery.

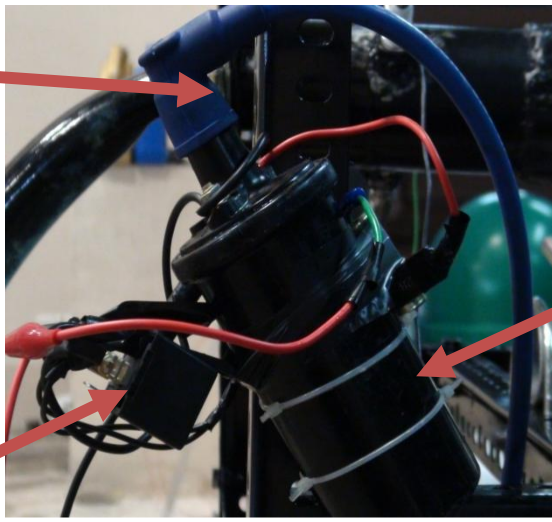


Figure 3.16: Major components of ignition system

3.11 EXPERIMENTAL TEST BENCH

3.11.1 Test Rig

The completed assembled jet engine is mounted on a test rig. Figure 3.17 shows the engine installed test rig.

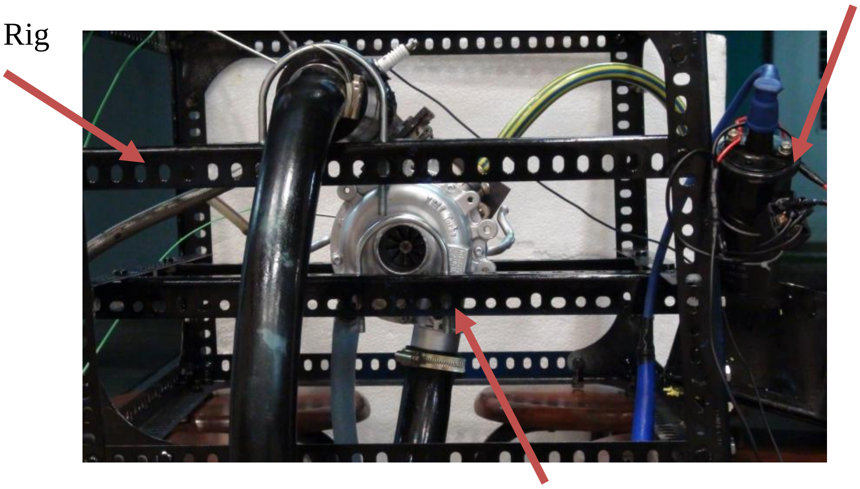


Figure 3.17: Jet engine mounted on a test rig

3.11.2 Instrumentation

The instrumentation use is thermocouples and was connected to three points as suggested by Santeler (2005) which at station 3.2, station 3.9 and station 6 to measure temperature.

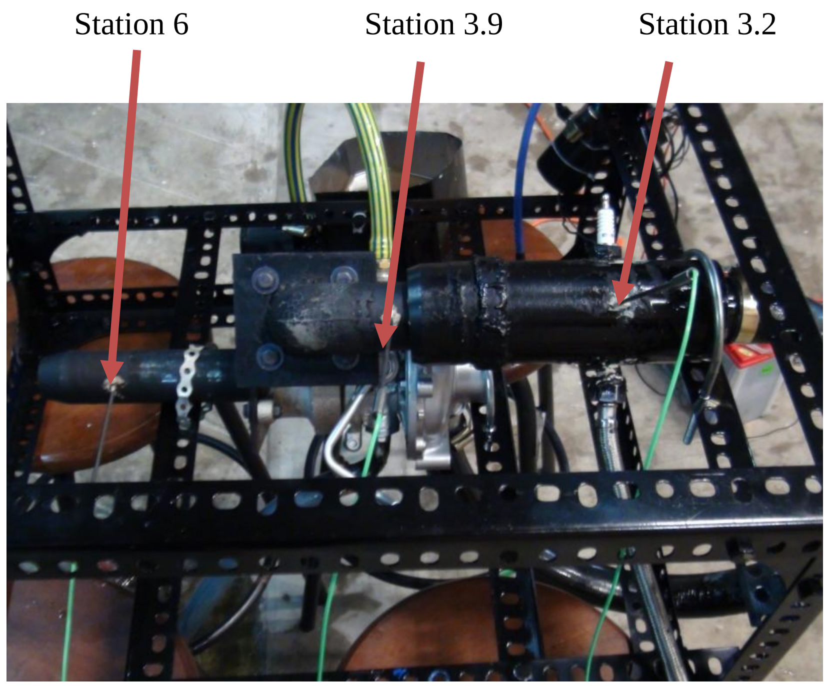


Figure 3.18: Thermocouple attached at different stations

3.11.3 Engine Starting Procedure

The engine requires a specific procedure (Santeler, 2005) in order for the engine to start. The starting procedure is as describe below:

1. Lubrication system is turned on.
2. Compresses air at the compressor inlet is introduced to rotate the rotor.
3. Then, ignition is turned on.
4. Fuel is introduced at the lowest mass flow rate.
5. A large bang will occur indicated the fuel is burned in the combustion chamber.
6. The mass flow rate increased slowly as the compressed air remains constant.
7. Once the turbine rotates faster than the compress air, the compressed air is turned off.

It is uncommon to determine the point where the engine will start to self-sustained, which varies with different engines. Thus, these points can be determined by trial and error.

CHAPTER 4

RESULTS AND DISCUSSION

4.1 INTRODUCTION

This chapter presents the result obtained from the experiment and based on the data recorded, discussion and analysis were made.

4.2 COSMOFLOW ANALYSIS

As mentioned in chapter 2, the design of the combustion chamber is analyzed by COSMOFLOW software. The purpose of the simulation is to evaluate the design, whether it follows the combustion chamber criteria and observes the flow pattern within the combustion chamber. Since this analysis was meant to evaluate the design based on flow pattern, combustion analysis was not analyzed. The result of the analysis is shown in figure 4.1 and 4.2.

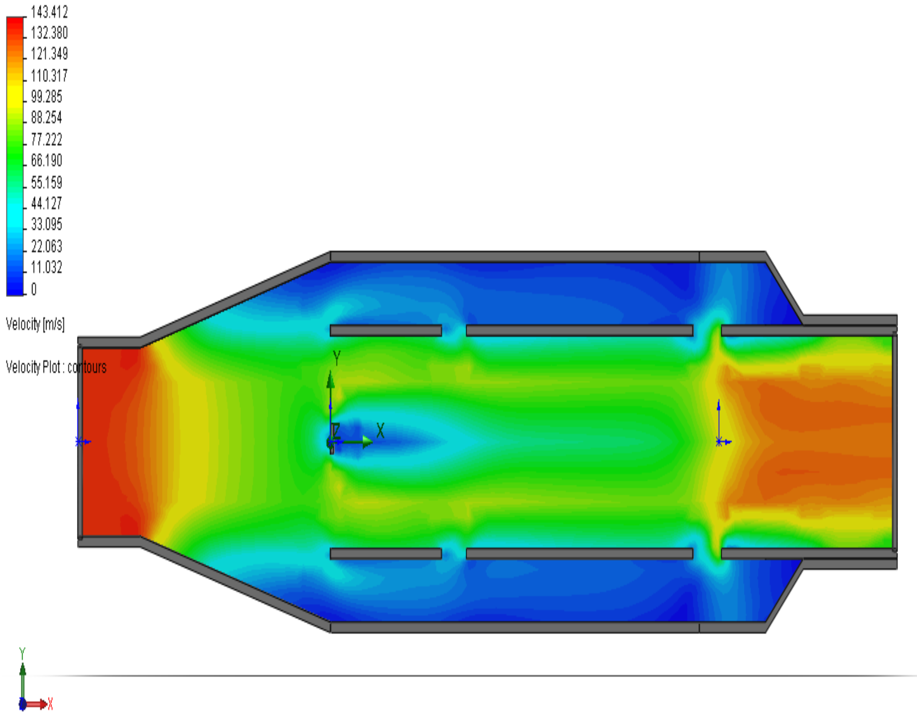


Figure 4.1: Velocity contour plot within the combustion chamber

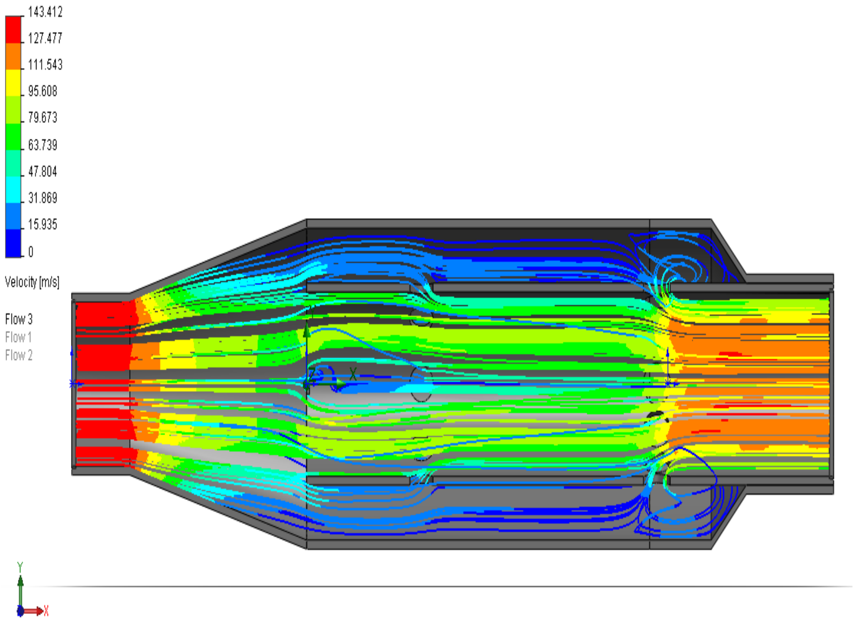


Figure 4.2: Velocity flow path of the combustion chamber

As mentioned in chapter 2, the principal feature of a jet-engine combustion chamber consists of four factors. Results from the analysis show that the flow decreases within the diffuser (station 3.1) thus, enabling the incoming air to mix with the fuel for combustion process. At station 3.2, the incoming air is divided into the inner and outer liner. Within the primary zone, small recirculation bubble occurs at low velocity. The recirculation process permits the burned gasses (due to combustion) to continuously self-ignite. However, the recirculation bubbles are small and not filling up the primary zone because of the swirler was not introduced in the design. As mentioned in chapter 3, the swirler is replaced with blunt bodies thus causes the small recirculation. Downstream the primary zone, the air from the outer liner enters the secondary and dilution zone through the primary and secondary holes respectively.

Analysis shows that the design agrees with all four principles. Even though the recirculation bubble is small, but it is acceptable due to time and cost limitation.

4.3 EXPERIMENTAL TESTING

The complete assembled jet engine was prepared for testing and method used for starting the engine mentioned in chapter 3 is used. The test run was done numerous times because various problems occur during the testing period. Consequently, several modifications and troubleshooting of the engine are done. For preliminary testing, the first test run was not as expected. Combustion didn’t occur even though an ignition system was started, and fuel is introduced. Based on visual observation, it gas was seen coming out at the exhaust. Thus, it was thought perhaps that the oxygen within the combustion chamber was not enough to start combustion. By following several trial and errors; the combustion still does not occur.

For the engine troubleshooting, another testing method is introduced whereas the ignition using a spark plug is replaced with a lighter at the exhaust. Once the flame ignites at the exhaust, compressed air is quickly introduced at the inlet to spin the rotor. Flame was seen coming out at the exhaust; however, the flame would blow out, and the rotor would stop spinning once the compressed air is removed. Temperature data shows that the combustion chamber temperature did not increase above ambient proved the hypothesis. There is not enough oxygen inside the combustion chamber as a result the flame would burn at the exhaust.

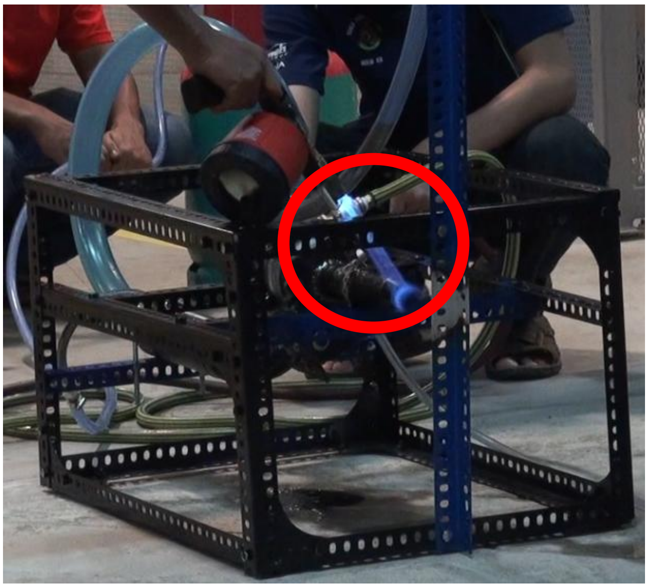


Figure 4.3: Method of ignition by lighter at the exhaust

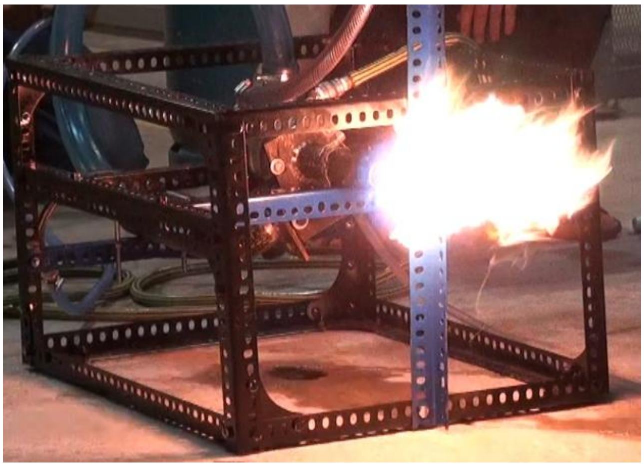


Figure 4.4: Unburned fuel burning at the exhaust

It was decided to remove the engine for the assembly and run the combustion chamber with compressed air as the source of airflow. Understand what happening inside the combustion chamber is the first step of troubleshooting the engine. The combustion chamber was tested run at fuel flow rate of 5.86×10^{-6} kg/s and 2.35×10^{-5} kg/s and the method of ignition with the means of lighter. At both mass flow rates, flames were seen coming out at the exit of the combustion chamber. It seems that air was not enough air inside the combustion chamber causing a fuel rich condition, thus any unburned fuel was burning outside the combustion chamber exit. The decision was to modify the combustion chamber and enlarge the primary and secondary holes, enabling more air to mix with fuel. The test was repeated at fuel flow rate of 5.86×10^{-6} kg/s and 2.35×10^{-5} kg/s was a success. At both flow rates, it could be seen that the flame contained inside the combustion chamber. However, the flame won't ignite by the spark plug. Then, it was suspected that the distance between the spark plug, and the fuel injector was too far for combustion to occur thus to resolve the problem; the spark plug hole is increased and reduce its distance from the fuel injector. Another test was done at fuel flow rate of 5.86×10^{-6} kg/s and 2.35×10^{-5} kg/s was a success. At both flow rate, the combustion is easily ignited and self-sustained when the spark is turned off.

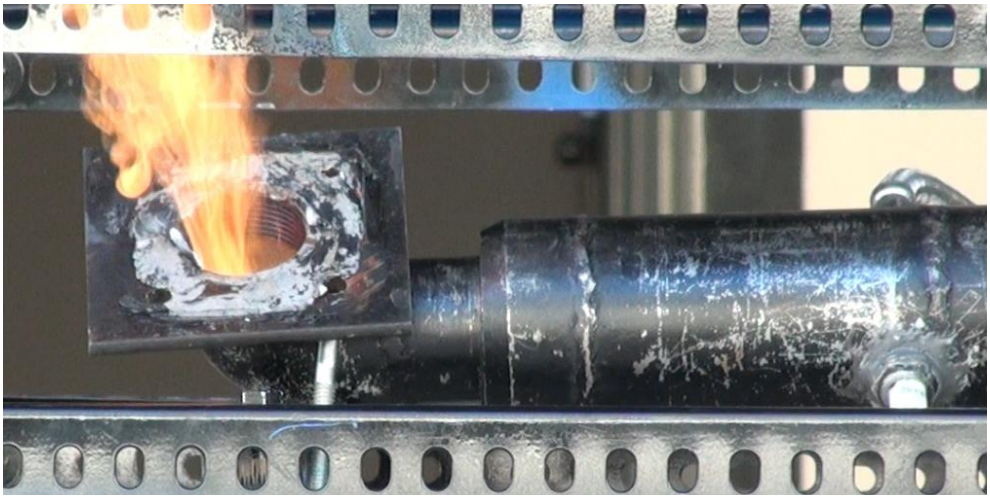


Figure 4.5: Flame coming out from the combustion chamber

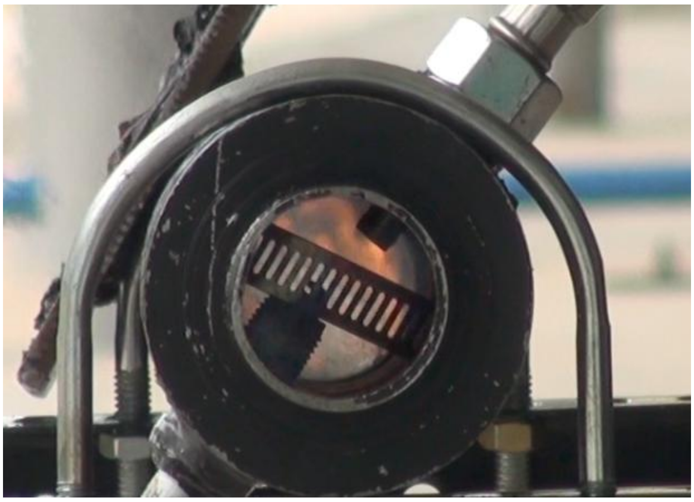


Figure 4.6: Flame contained inside the combustion chamber

The turbocharger was then reinstalled to the combustion chamber and the engine was prepared for another round of testing. Once preparation was done, the engine was ready to run and ignition occurs easily at lowest fuel flow rate of 5.86×10^{-6} kg/s. Large 'bang' were heard confirming that combustion occur within the combustion chamber. Fuel mass flow rate then steadily increased as the compressed air introduced at the intake remained constant. At fuel flow rate at 6.174×10^{-6} kg/s, the turbine started glowing red, and flame were coming out at the exhaust.

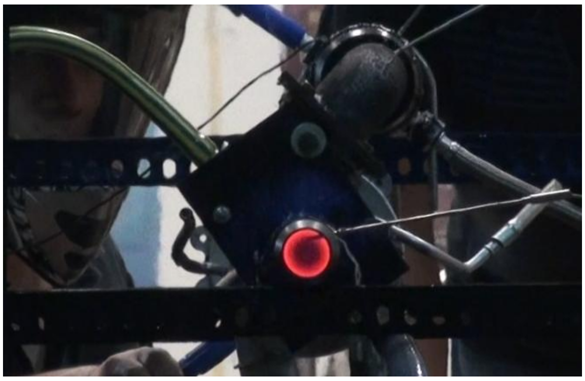


Figure 4.7: Turbine blade glowing red

Based on the testing results, the modified combustion chamber improved mixing and causes more complete combustion downstream of the combustion chamber. At higher fuel flow rate of 1.26×10^{-4} kg/s, combustion becomes inconsistent and large 'bang' occur frequently. This phenomenon can be described as a compressor stall. The turbine spins so fast that the compressor cannot spin at the same revolution causing it to stop for a brief of moment. This is not desirable, thus it was decided to limit the fuel mass flow rate up to 1.26×10^{-4} kg/s. The subsequent test was begun with 5.86×10^{-6} kg/s and the fuel was ignited successfully. During the test run, the compressed air at the intake was turned off at different fuel flow rate to determine the point where the engine would sustain itself. However, the engine could sustain approximated only at 10 seconds at maximum fuel flow rate. The results during the testing period are shown in graph 4.7 to graph 4.11. In addition, the modification and troubleshooting of the engine during the testing from the first experiments up to this point are summarized in Appendix B.

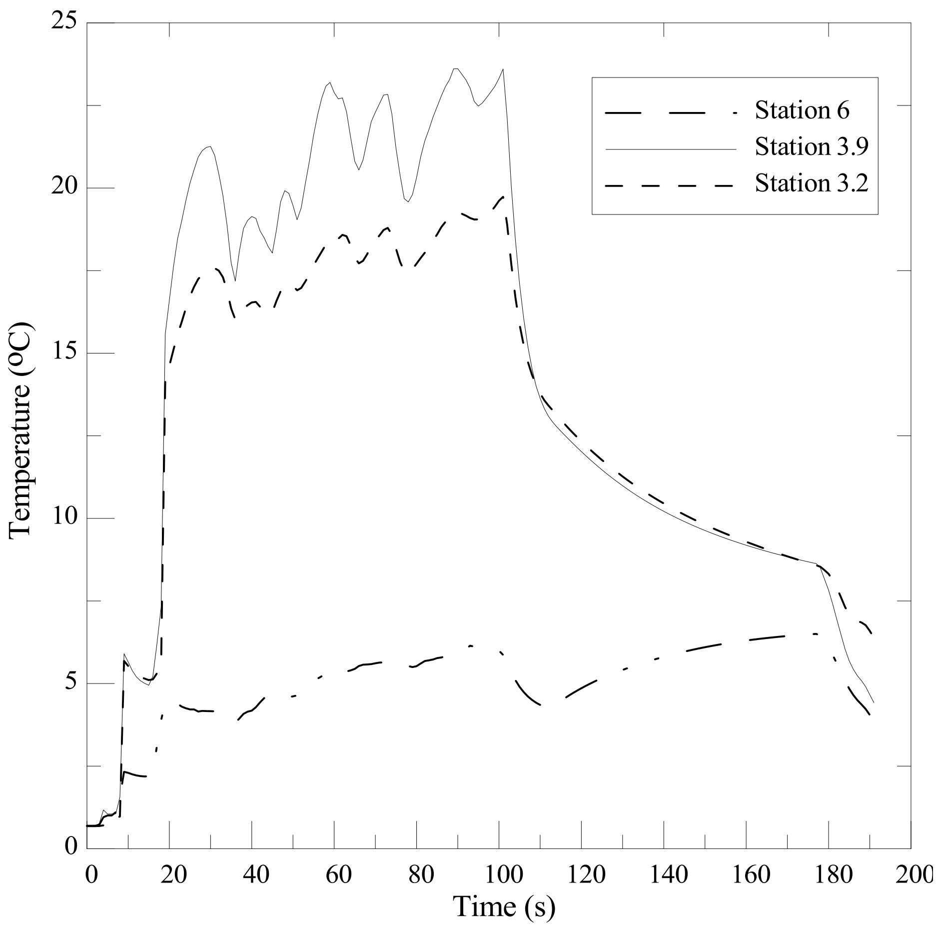


Figure 4.8: Graph of temperature distribution of the jet engine at 3 different stations during testing period

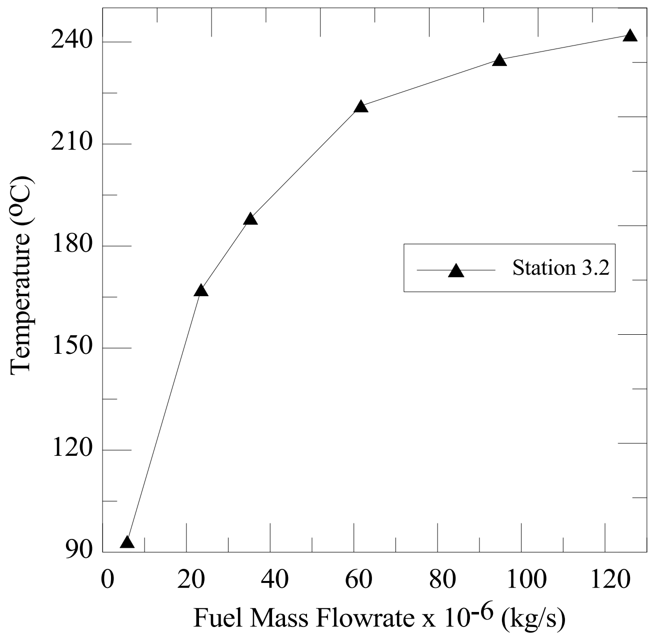


Figure 4.9: Graph of temperature mass flow rate at station 3.2

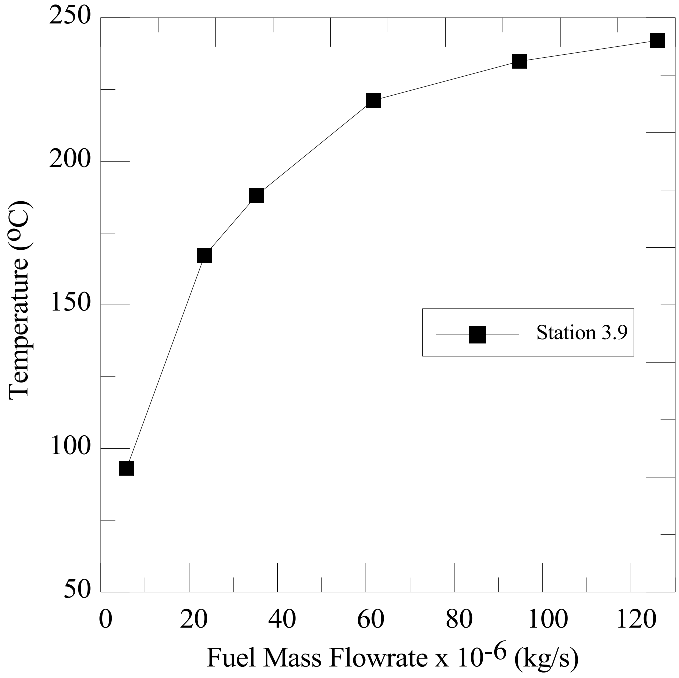


Figure 4.10: Graph of temperature mass flow rate at station 3.9

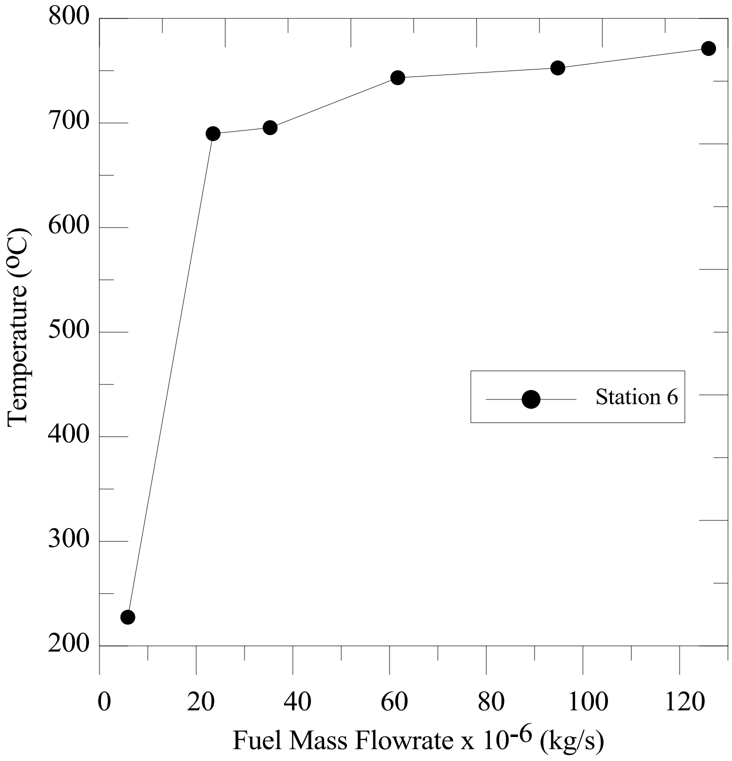


Figure 4.11: Graph of temperature versus fuel mass flow rate at station 6

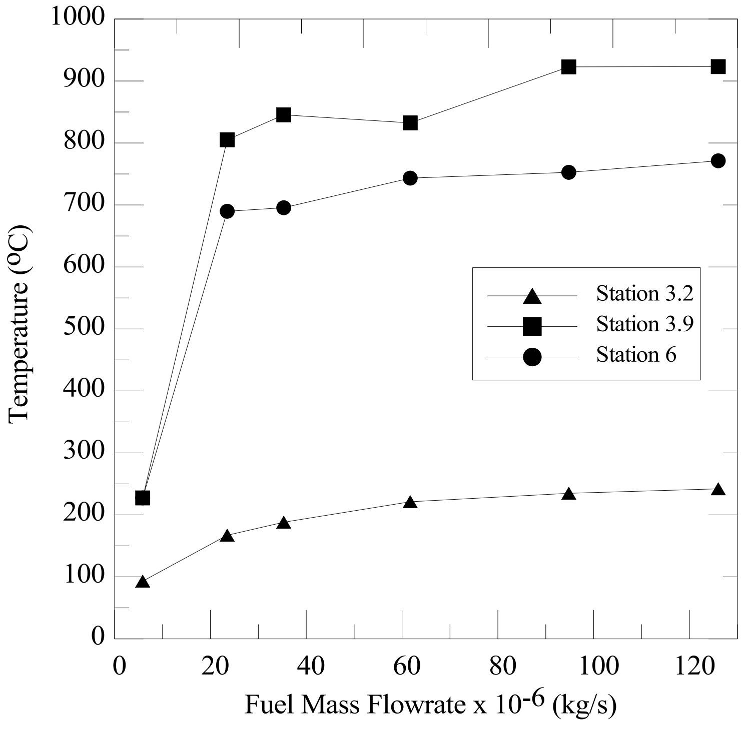


Figure 4.12: Graph of temperature mass flow rate at all three stations

Figure 4.7 show the temperature distribution during the testing period at all three stations. The temperature during the first 10-second remains at ambient and not until the fuel ignited in the combustion chamber the temperature increases. The temperature slowly increases over time as the $\dot{m}_f$ increases. At maximum $\dot{m}_f$, the temperature is the highest and at this point the engine self-sustained over a brief period which at seconds 90 to 100. The relationship between temperature and fuel mass flow rate is shown in Figure 4.8 to Figure 4.10. The temperature increases as the $\dot{m}_f$ increases for all stations. As more fuel is burned, more energy released increasing it temperature. However, the temperature at station 3.9 is higher than station 3.2 and station 6. CFD analysis shows that there are more mixing of air and fuel at the end of the combustion chamber thus more combustion occurs, increasing its temperature. Temperature drops were seen at station 6 due to energy loss to rotate the turbine and heating the turbine blade.

No thrust data were obtained during the test, due, mainly to the failure of the engine to self sustained. Due to time and cost limitation, further modification and improvement cannot be done. Nevertheless, the factors regarding the failure of the engine to self-sustain were analyzed and are discussed in the next section.

4.4 ENGINE FAILURE ANALYSIS

The engines were then dissembled for another troubleshooting, and analysis was made on why the turbine won't self sustained for a longer period. During the troubleshooting, it was found that there are four factors that affected the engine failure.

4.4.1 Leaking at the Turbocharger

During troubleshooting, it was found that there is lubricant oil coming out from the turbine blade as shown in figure 4.12. In addition, there was white smoke coming from the turbine housing (figure 4.13) during the test run the jet engine, and it was thought that lubricant oil was burned together with the fuel.

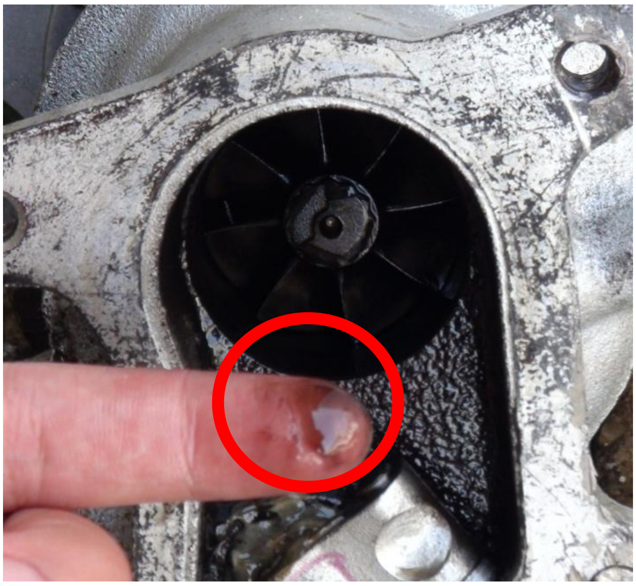


Figure 4.13: Lubricant oil leaks at the turbine blades

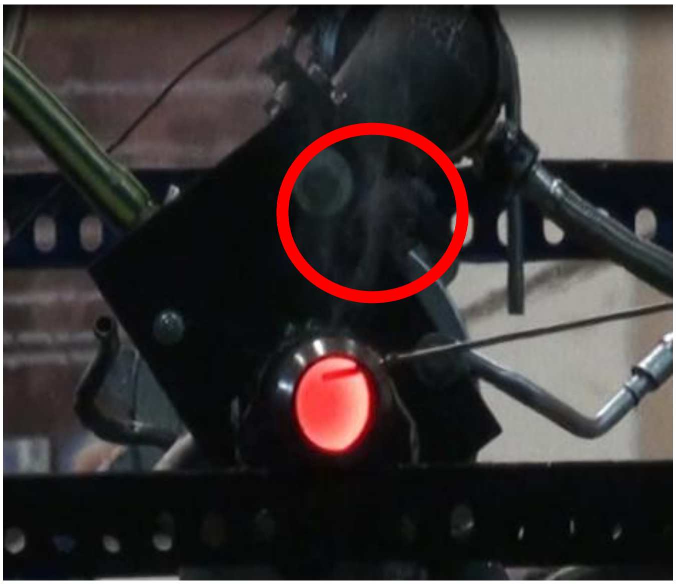


Figure 4.14: White smoke appear at the turbine housing during testing

4.4.2 Low Lubricant Pressure Pump

As mentioned in chapter 3, aquarium pump was used to drive lubricant oil to the turbocharger. Though it can supply oil to the turbocharger, it can’t supply enough pressure to accommodate the turbocharger specification. The maximum pressure the pump can supply is approximated 13 kPa, which are lower than average pressure required for a turbocharger which is between 310 kPa to 480 kPa (Erjavec, 2009).

4.4.3 LPG (Propane and butane mixture) not suitable to be used as fuel

LPG typically used as cooking gas and was used as fuel as it is cheap and readily accessible. Commercial LPG in the market normally consists of propane and butane with 30 percent and 70 percent (Hashim, 2008) in composition respectively. However, its composition will vary accordingly and subject to the application, country and surrounding temperatures. On the other hand, the use of LPG of mixture of gas as fuel for the homemade turbine jet engine is a novelty, where Kamps (2001) and Santeler (2005) mentioned the use of propane as fuel. Propane has higher-energy content than and more reactive LPG, which makes it more suitable to be used as fuel. Table 4.1 shows the comparison of commercial propane, commercial butane and LPG.

Table 4.1: Comparison between LPG, propane and butane

Properties	**LPG**	**Propane**	**Butane**
Composition	70% Propane, 30 Butane	100% propane	100% Butane
Specific gravity	0.5 @ 20ºC	0.508	0.584
Upper flammable limit	9.5%	10.0	9.0
Lower flammable limit	1.8% -1.9%	2.2	1.8
Calorific Value, gross	49.5 MJ/kg	50.0	49.3
Calorific Value, net	45.7 MJ/kg	46.3	45.8

Source: Ishak 2008

4.4.4 Large Pressure Drop

The large pressure drop was mainly due to the use of $90°$ pipe flange at the end of the combustion chamber as a connection to the turbine housing. This concept was similar to the early Whittle combustion chamber (figure 4.15). Similarly, Whittle type combustion chamber has a reverse flow thus created a considerable amount of pressure loss (Royce, 1986). The utilization of the $90°$ pipe constructs the engine neatly. However, pressure loss was thought to be insignificant and was unexpected.

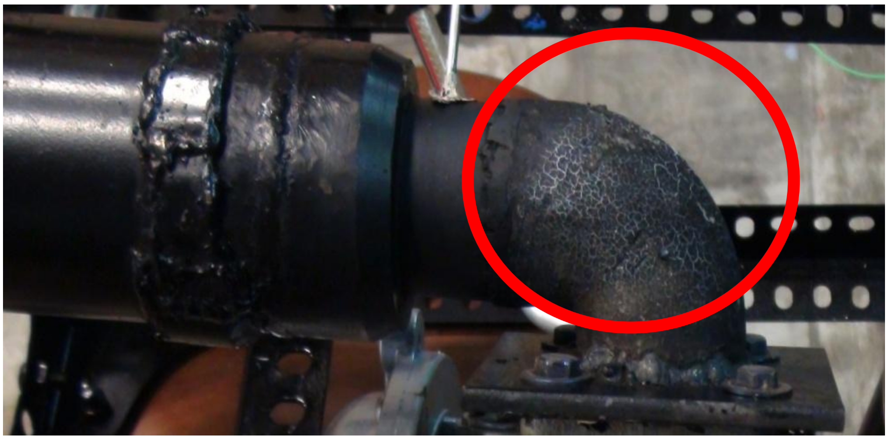


Figure 4.15: 90 degree pipe flange attached to the exit of the combustion chamber. The design is similar to the Whittle combustion chamber (Figure 4.15) where the exit of the combustion chamber is tuned more than 90°

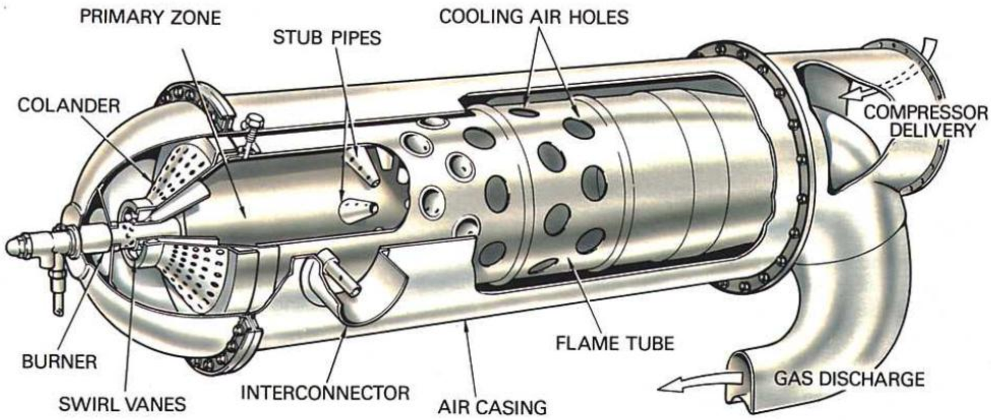


Figure 4.16: The Whittle combustion chamber

Source: Royce 1986

CHAPTER 5

CONCLUSIONS AND RECOMENDATIONS

5.1 CONCLUSIONS

A small-scale turbine jet engine was designed, fabricated, and tested. A second-hand turbocharger was used, and a combustion chamber was designed and fabricated. The engine support system was introduced to the engine which comprised of a lubricating, ignition and fuel delivery system. During testing, the jet engine could start however it was only able to self-sustain approximated for 10 seconds.

Various problems were encountered with starting the engine, early in the testing phase. The modification of the combustion chamber and the modification of the ignition system facilitated the successful starting of the engine. Temperature and fuel mass flow rate data were taken during the testing period. There are no thrusts measurements due to the engine won't self-sustain for a longer period. The engine could self-sustain for a brief period at maximum fuel mass flow rate.

Troubleshooting and analysis of the failure were done, and it was found that, there were four main factors involve, which are; LPG unsuitable to be used as fuel, leaking at the turbocharger, large pressure loss in the combustion chamber and pressure pump not large enough. Due to time and cost limitation; further modification cannot be done.

5.2 RECOMMENDATIONS

For further studies of this project, the recommendations are as follows:

1. Brand new turbocharger should be use.
2. Pure propane or pure butane should be used as fuel.
3. Avoid uses of pipe flange in the combustion chamber to reduce pressure loss
4. Use a higher pressure pump for lubricant system and install a pressure regulator.

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APPENDIX A

LIST OF COMPONENTS AND MATERIAL REQUIRED

Table 6.1: List of components and material required for the development of the small scale turbine jet engine

System	Component	Material	Description
Jet engine system	Turbocharger	-	Consists of turbine and compressor
	Combustion Chamber	Mild steel	Fabricated
	Exhaust Nozzle	Mild steel	Fabricated
Ignition and starting system	Spark Plug	Automotive spark plug	Required for ignition of combustion process.
	Battery 12V	Car battery	Power supply
	Wires	-	Connection between spark plug and battery
	Switches	-	Required to ignite the spark plug when desirable
	Leaf Blower	-	Required for starting procedure
Lubrication system	Lubricant		To supply to the turbocharger to avoid wear and excessive heat
	Electric pump	Aquarium pump	To deliver the lubricant to turbocharger
	Valve	-	To control the flow of lubricant entering the turbocharger
	Oil tank	-	To store the lubricant
Fuel delivery system	LPG tank	LPG gas	Fuel for combustion
	Hoses	-	To deliver fuel to the combustion chamber
	Fuel regulator	-	Controlling the mass flow rate of fuel entering the combustion chamber
	Safety valve	-	Connected to the LPG for safety purpose
	Fuel Injector	-	To inject the fuel into combustion chamber

Table 6.1: Continued

System	Component	Material	Description
Test rig system	Test stand	Mild steel	To mount the engine and other component appropriately.
	Thermocouple	-	To measure the temperature of the combustion chamber and exhaust nozzle

APPENDIX B

JET ENGINE TROUBLESHOOTING SUMMARY

Table 6.2: Summary of the jet engine troubleshooting

No	Description	Date	Problem Encountered/ Observations	Remarks/ Modifications
1	Test Run the engine	26/04/2012	1. Unburned fuel burned at the exhaust nozzle 2. Fuel did not ignite within the combustion chamber.	Increase the diameter of primary and secondary holes
2	Test run the combustion chamber	28/04/2012	Fuel did ignite with ignition system	Decreased the distance between the injector and spark plug
3	Test Run the engine	1/05/ 2012	1. The engine wont self sustained 2. Compressor rotor does not spin fast enough	Modifies the lubrication system
4	Test run the lubrication system	5/05/2012	-	The compressor rotor spins faster
5	Test Run the engine	11/05/2012	1. Ignition system failure. 2. Lubrication system failure	1. Change spark plug 2. Repair the Lubrication system
6	Test Run the engine	15/05/2012	1. The engine only sustained for approximated 10 seconds 2. White smoke appear at turbine during testing	1. Dissembled the engine 2. Analyzed failure of the jet engine

APPENDIX C1

DRAWING OF INLET DIFFUSER

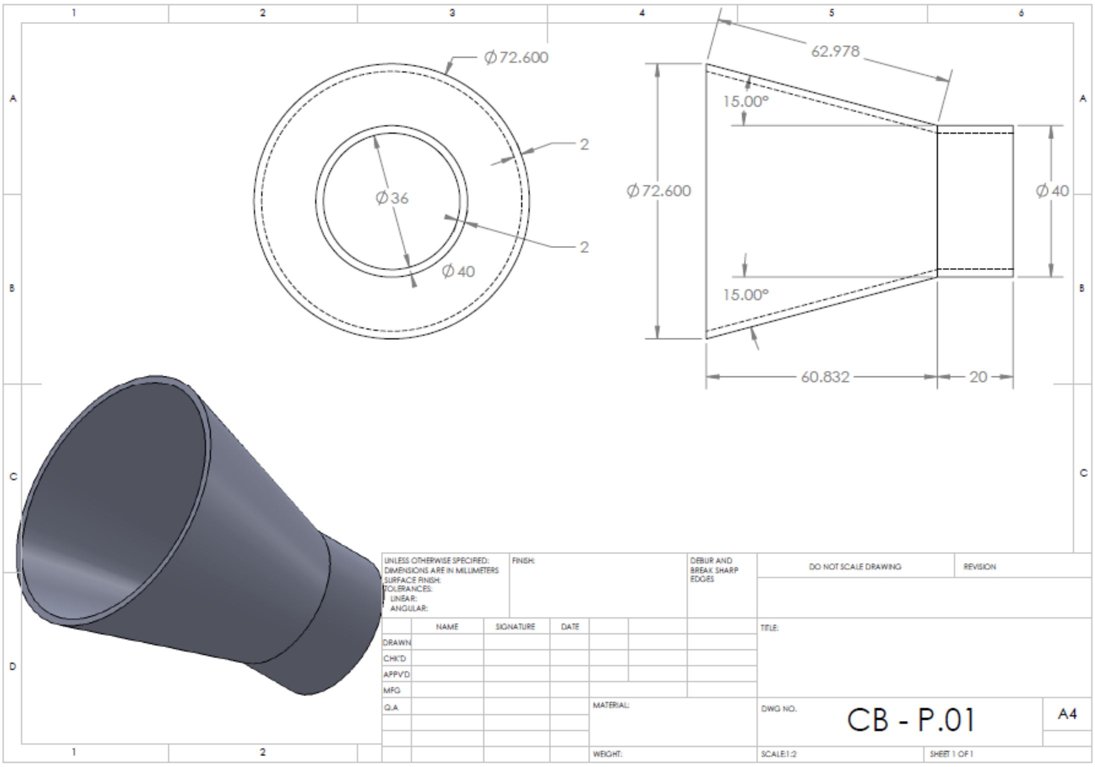

Figure 6.1: Two-dimentional drawing of inlet diffuser

APPENDIX C2

DRAWING OF INNER LINER

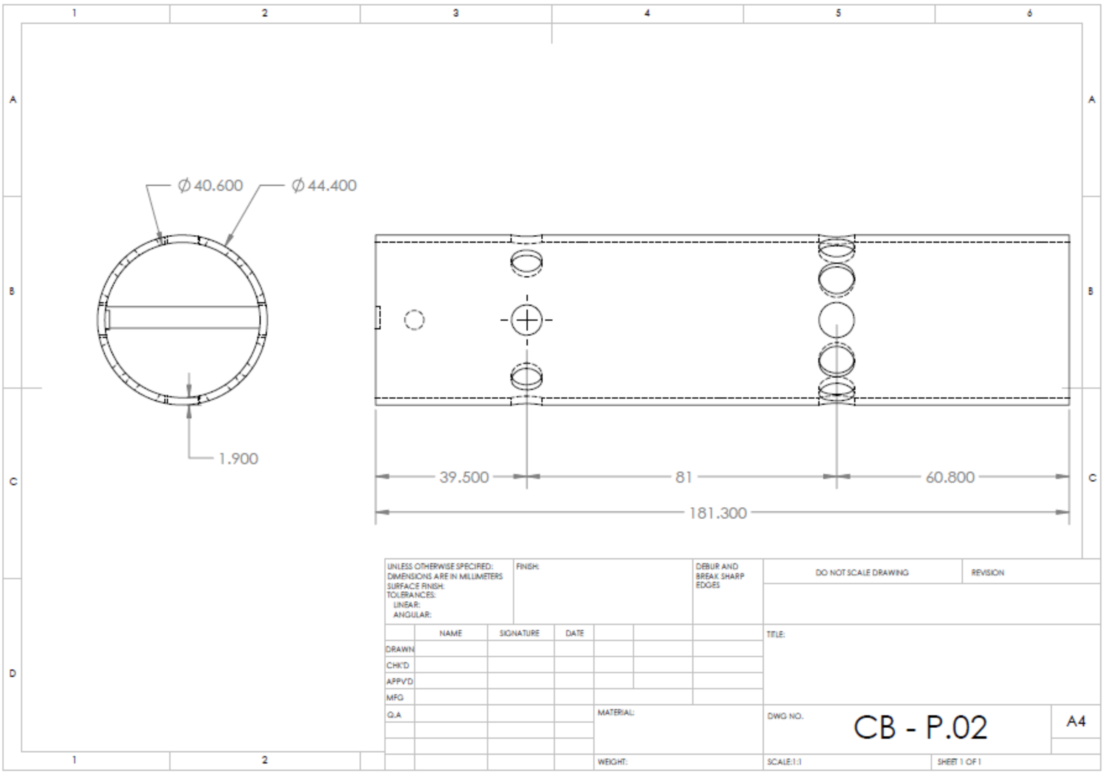

Figure 6.2: Two-dimentional drawing of inner liner

APPENDIX C3
DRAWING OF LINER HOLDER

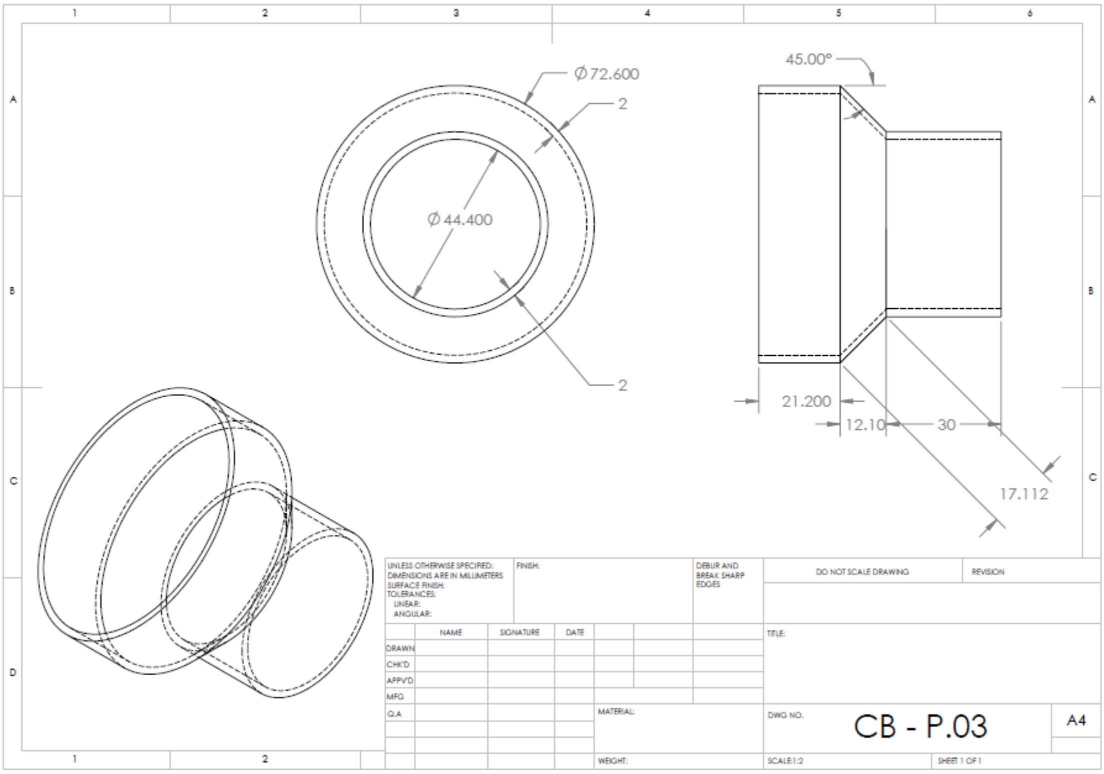

Figure 6.3: Two-dimentional drawing of liner holder

APPENDIX C4

DRAWING OF OUTER LINER

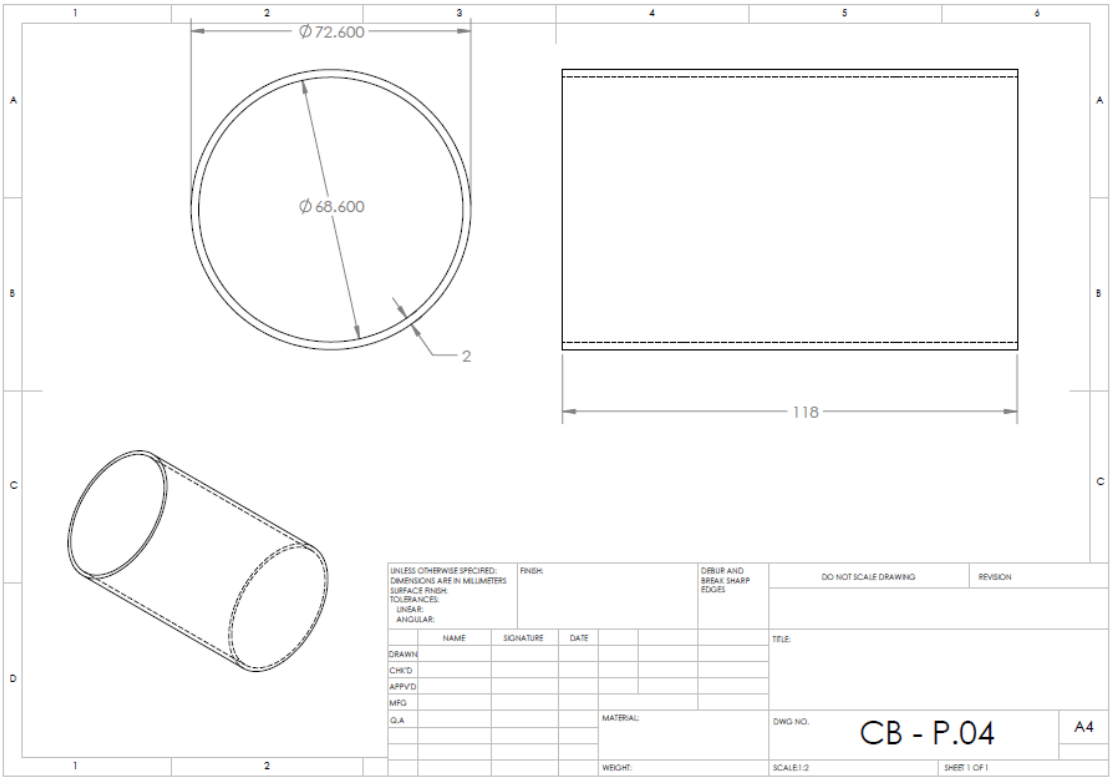

Figure 6.4: Two-dimentional drawing of outer liner

APPENDIX C5

DRAWING OF COMPLETED ASSEMBLED COMBUSTION CHAMBER

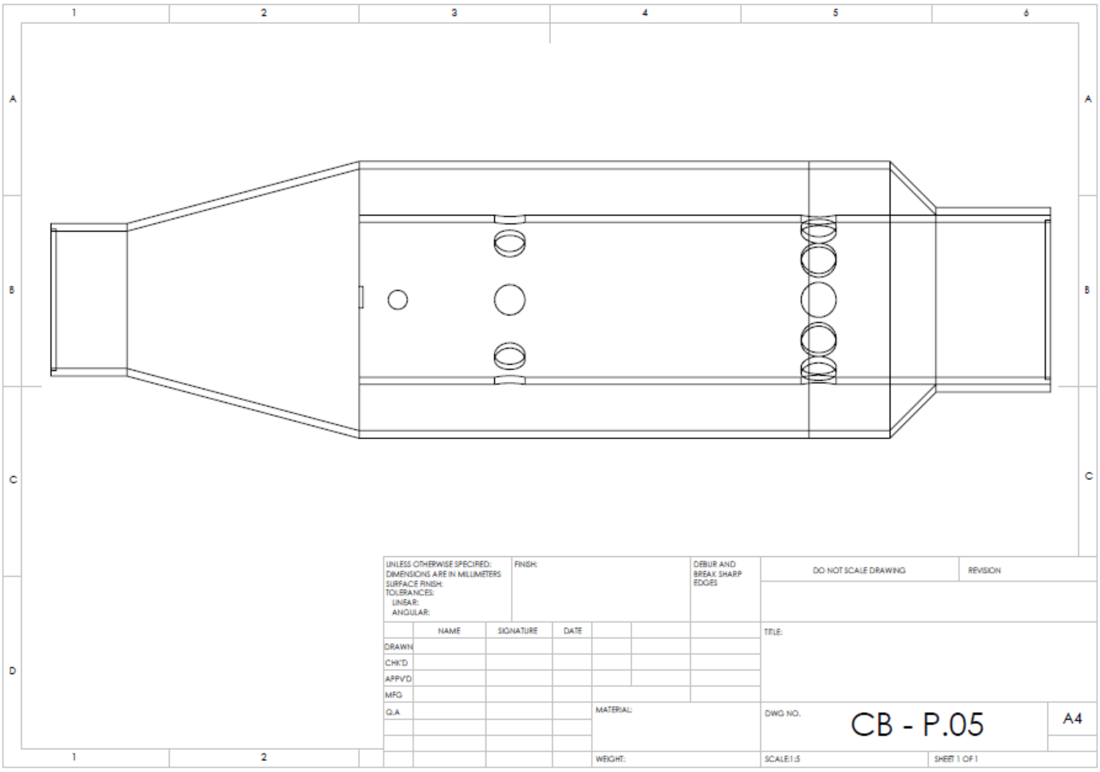

Figure 6.5: Two-dimentional drawing of completed assembled combustion chamber

APPENDIX D1

DRAWINGS OF NOZZLE

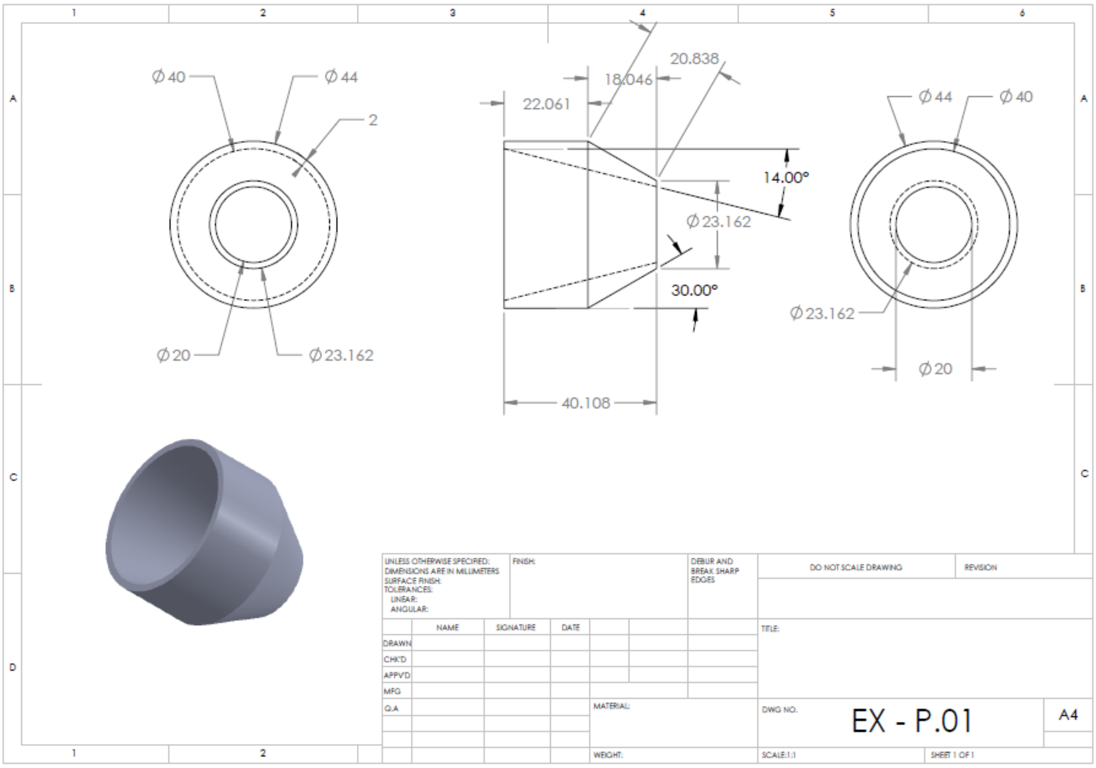

Figure 6.6: Two-dimentional drawings of nozzle

APPENDIX D2
EXHAUST LINER

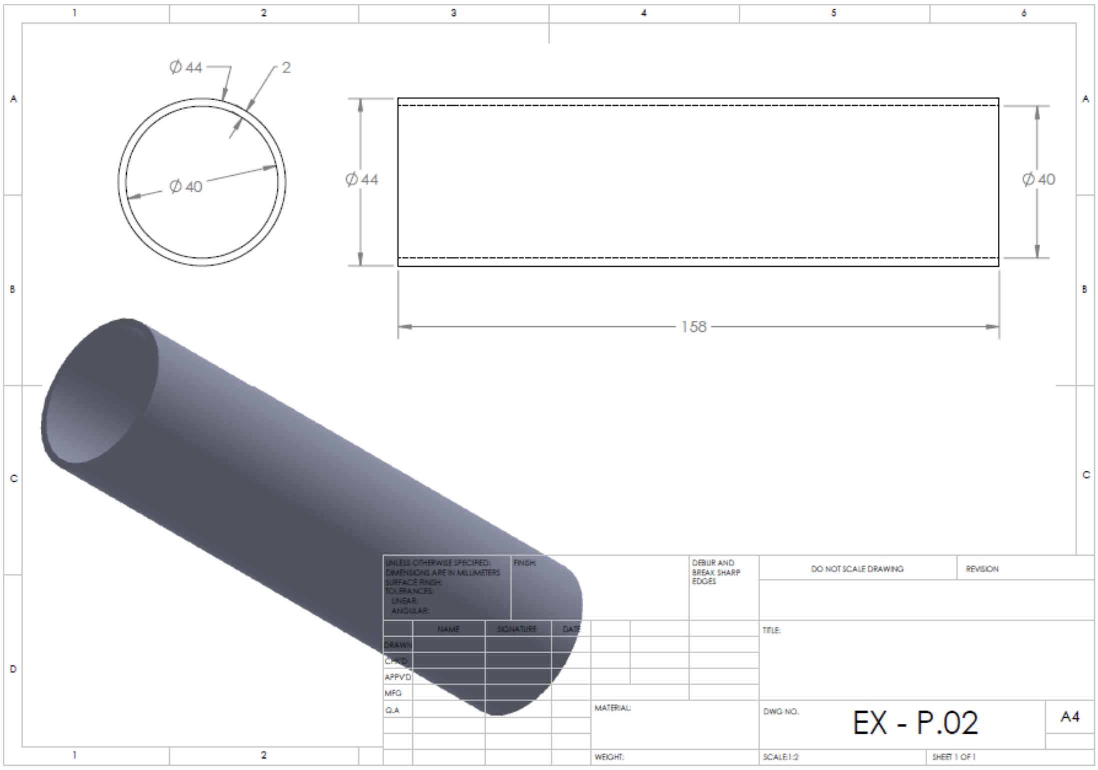

Figure 6.7: Two-dimentional drawings of exhaust liner

APPENDIX D3

DRAWINGS OF ASSEMBLED EXHAUST NOZZLE

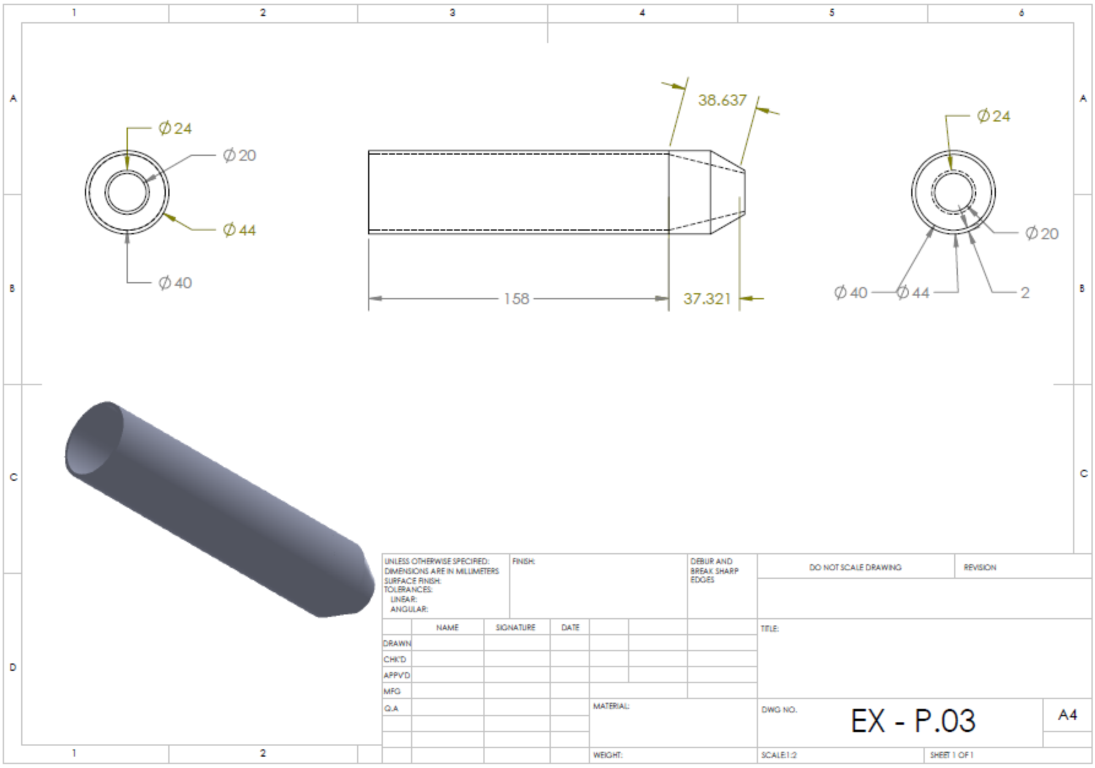

Figure 6.8: Two-dimentional drawings of assembled exhaust nozzle

APPENDIX E
CITREX CERTIFICATE OF AWARD


Universiti Malaysia PAHANG
Engineering • Technology • Creativity
CITReX
Certificate of Award
BRONZE MEDAL
This Certificate of Award is presented to
Mohamad Ridwan bin Raja Mohd, Dr. Rizalman Mamat, Amir bin Aziz
For the invention / innovation of
Compact Turbine Jet Engine
CREATION, INNOVATION, TECHNOLOGY & RESEARCH EXPOSITION
26 – 28 March 2012, Universiti Malaysia Pahang
PROFESSOR DATO' DR. ROSLI BIN MOHD YUNUS
Deputy Vice-Chancellor (Research & Innovation)
Universiti Malaysia Pahang
UNIVERSITI MALAYSIA PAHANG
UMP

Figure 6.9: CITREX certificate of award