DESIGN AND FABRICATION OF TURBOJET ENGNE

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A Final Year Project Report

Presented to

SCHOOL OF MECHANICAL & MANUFACTURING ENGINEERING

Department of Mechanical Engineering

NUST

ISLAMABAD, PAKISTAN

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In Partial Fulfillment

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Bachelor of Mechanical Engineering

____

by

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ABSTRACT

The purpose of this project is to design, analyze and fabricate a turbojet engine. We went through the literature review of each component of the turbojet engine thoroughly. Based on that literature review, we designed our jet engine and then after thorough analysis of our design, we moved over to the manufacturing phase. Each component was fabricated, and final assembled prototype was developed.

ACKNOWLEDGEMENTS

The efforts of the project participants who discretized the project, researched each aspect of the project extensively, formulated practical design parameters and procured equipment based on those design parameters is to be acknowledged. The project supervisor, Dr. Sami-Ur-Rehman Shah is to be especially thanked for the supervision and advisory regarding the project who governed and helped with several of the design parameters. We also give credit to the whole sellers who gave us a description of the local marketplace in terms of components availability and their prices. NUST administration and staff members who facilitated our usage of workshops and laboratories in SMME for testing and set-up of our equipment are credited as well as the lab workers and supervisors who helped manufacture certain parts of the machine.

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ORIGINALITY REPORT

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ABBREVIATIONS

CAD	Computer-aided Design
MHI	Mitsubishi Heavy Industries
TIT	Turbine Inlet Temperature
CG	Centre of Gravity
RPM	Revolutions per minute
RC	Remotely controlled
NASA	National Aeronautics and Space Administration
TIG	Tungsten Inert Gas
CNC	Computer Numeric Control
CFD	Computational Fluid Dynamics
MRC	Manufacturing Resource Centre

NOMENCLATURE

Π, Pr	Pressure ratio
U	Blade tip speed
c	Absolute velocity
w	Relative velocity
r	radius
ω	angular velocity, bending critical angular velocity
A	area
b	Tip height
p	static pressure
p_0	stagnation pressure
ρ	density
R	General gas constant
T	Static Temperature

T_o	Stagnation Temperature
$\dot{m}$	mass flow rate
v	velocity
Ɣ, k	Heat capacity ratio
C_p	Specific heat at constant pressure
M, Ma	Mach number
α, β	Flow angles
f	fuel ratio
Q_R	Heating value of fuel
C	Distance from wheel's CG to first bearing
L	bearing space
D	shaft diameter
E	Modulus of elasticity
I	Moment of inertia
ξ	Blade stagger angle
ε	Deflection angle
ϕ	Flow co-efficient
ψ	Stage loading co-efficient

Chapter 1 Introduction

Motivation

A turbojet engine is one of the most commonly used engines in aircrafts. Aircraft is the fastest mode of transportation and it is one of the main components of a country's defense system. And keeping these necessities in mind, every country must be able to develop its own aircrafts. But despite that, Pakistan imports almost every component of an aircraft and just assemble them here. In 2018, Pakistani importers spent $217 million on turbojets and its parts. In the same year, Pakistan spent $228.7 million on different configuration of turbomachines. By 2024, aircraft engine manufacturer's market value is projected to be of $84 billion. This means that by then, importing those parts would be more expensive. Best solution for this issue is to manufacture, if not all, then some parts here in Pakistan. The main goal of this project is to be able to build our own jet engines here in Pakistan.

Problem Statement

Design, analysis and fabrication of an air breathing jet-engine for a model aircraft consisting of a micro gas turbine and a propelling nozzle. To fabricate this model jet engine, a centrifugal compressor from a turbocharger is acquired. For this compressor, an annulus combustion chamber, an axial turbine and auxiliaries will be modeled.

The deliverables of this project are as follows:

- ➢ Theoretical Design of:
- Centrifugal compressor
- Combustion chamber
- Turbine
- Nozzle
- Shaft

- A fully working prototype of model jet engine.
- Experimental analysis of prototype.

The problem statement of the project can be further written as:

"To design and fabricate a Turbojet Engine that can produce 39 newtons of thrust at maximum rpms by using a compressor rotor of a turbocharger."

Objectives

Some of the objectives of this project are:

- To be able to thoroughly design and fabricate a turbojet engine along with all its components.
- To be able to optimize and reproduce our design to manufacture jet engines for Pakistan.
- Reducing the imports' cost of Pakistan via covering most of its jet engine's needs.

Chapter 2 Literature Review

Gas Turbines

A gas turbine also known as combustion turbine is a kind of continuous, internal combustion engine. A basic gas turbine has a turbine, compressor and a combustor, as its three main elements. It is basically a machine that delivers mechanical power or thrust. The basic working principle of a gas turbine is that energy is added to the gas stream by a compressor, after which combustion takes place which results in the increase in the temperature, volume and velocity of the gas. Then the turbine rotates, powering back the compressor and then we can extract energy either in the form of shaft power, compressed air and/or thrust. A basic gas turbine is shown in the figure below:

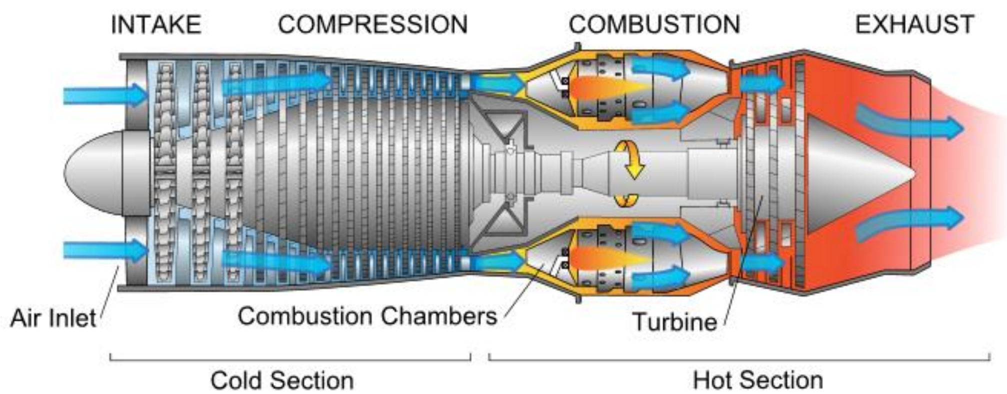


Figure 1: Basic Gas Turbine

Types of Gas Turbines

We can categorize the gas turbines as the following two types.

- Industrial Gas Turbines
- Jet Engine Gas Turbines

An industrial gas turbine is required to produce mechanical power while a jet engine gas turbine is required to produce thrust.

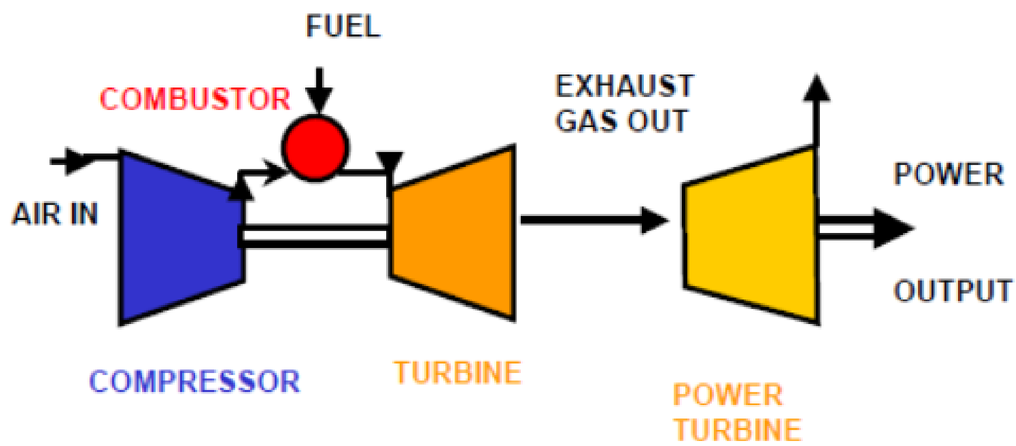


Figure 2: Industrial Gas Turbine Configuration

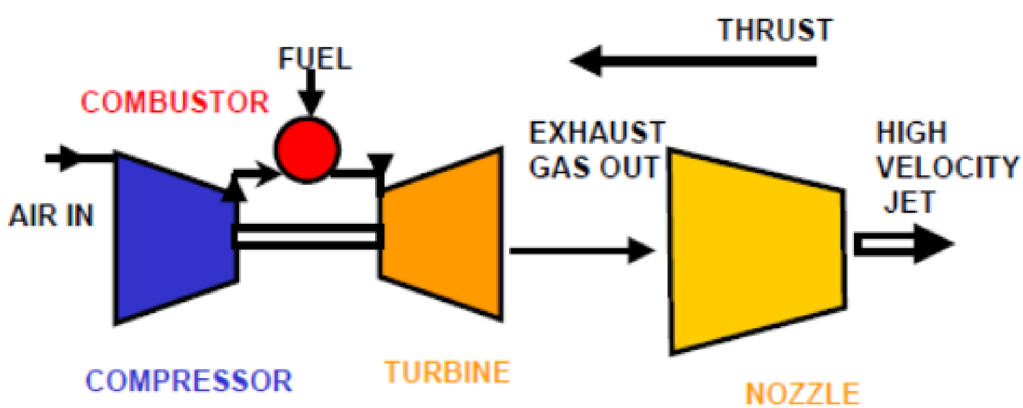


Figure 3: Jet Engine Gas Turbine Configuration

Jet Engine Gas Turbines

A jet engine is a type of reaction engine that releases a high-speed jet which in turn is used to produce thrust. A basic air breathing jet engine features an air compressor which is powered by a turbine and the leftover air containing high energy content us used to generate thrust using a propelling nozzle. The main configurations of a jet engine are:

- Turboprop Engine
- Turbofan Engine
- Turbojet Engine
- Turboshaft Engine

The basic working principle of Turbojet and turbofan engines are that the reactive force to the exhaust gases at high speeds generate thrust as a result. Turbofan has a turbine driven fan in the compressor upstream and is the most commonly used type in aerospace industries. In turbofan, the air adds cold thrust by bypassing the compressor and rejoining the downstream flow of turbine. On the other hand, in the turbojet engine, there is no air being bypassed with the compressor. The basic working principle of Turboshaft and turbo-prop engines is that they use exhaust gases to run a separate turbine that in turn is used to drive a propeller. In turboshaft, propeller is driven by all of the exhaust gases, whereas in turboprop, some of the exhaust gases is used to produce thrust [2]. The configurations are shown in the figure below:

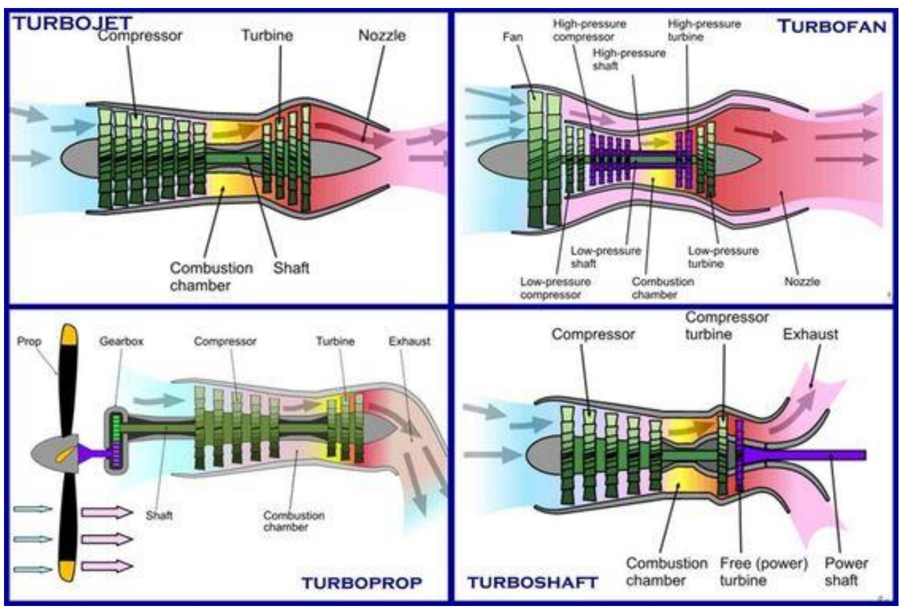


Figure 4: Four Types of Gas Turbine Engines

The configuration which we will be focusing on is turbojet engine.

Turbojet Engine

An airbreathing jet engine that has a gas turbine along with a propelling nozzle is called a turbojet engine. The main components of a gas turbine are inlet, compressor, turbine and a combustion chamber. The basic working principle of a turbojet engine is that the inlet air is compressed, which is then heated by burning fuel in the combustion chamber, then it is passed on to the turbine where it expands to provide back-work. Then, it is accelerated to high speeds through a propelling nozzle to produce thrust.

This thrust is generated based on Newton's second law of motion. From the conservation of momentum principle, the thrust equals the product of mass flow rate and the velocity of the air. In order to increase the thrust force, we can input more fuel assuming the efficiency remains the same. Besides that, afterburners can also be used to increase the thrust by once again consuming more fuel. But the afterburners are mainly used to achieve supersonic speeds in military aircraft and that's not the requirement of this model. The thermodynamics of a turbojet engine can be described by Brayton cycle [3].

The Brayton Cycle

Gas turbines are thermodynamically described by the Brayton cycle. In this cycle, both compression and expansion take place in a rotating machine. This is generally an open cycle as shown in the figure below:

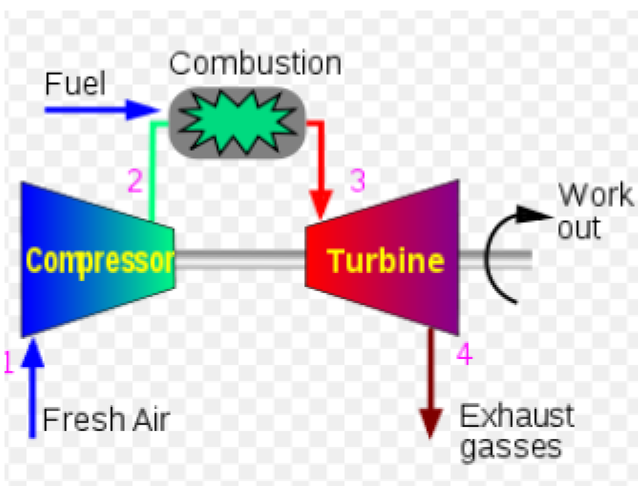


Figure 5: Example Geometry of Open Brayton Cycle

Since it's an open cycle, once the exhaust gases generate thrust, we cannot recirculate them for the next cycle of thrust generation. The compressor and turbine are connected through a single shaft. In an aircraft gas turbine, we can consider the net-work produced by the engine as the difference in kinetic energy between the input and exhaust gases. [4]

The T-s and P-v diagrams of an idealized Brayton cycle are shown in the figure below:

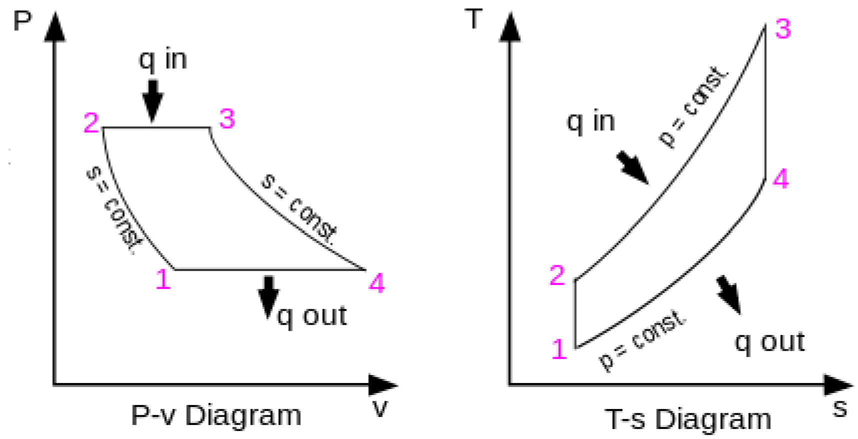


Figure 6: T-s and p-v diagram of Brayton cycle

Components

To have a better understanding of a turbo-jet engine, we must learn about each of its component, how they work and what's their role in a jet engine. The brief review of these components are as follows:

Air Intake

At the inlet of a turbojet, there is an intake tube which is required to direct the upstream air smoothly into the compressor so that there is less turbulence in the compressor itself. Stationary vanes also can do the job of smoothly regulating the intake air into compressor vanes. The upstream air is always subsonic, irrespective of the speed of aircraft. Another task of this air intake tube is that it should be able to provide air which is less turbulent, has small pressure variation and hasn't lost much energy along the way such that we have as much energy content available as possible. High pressure at the inlet means that we have higher pressure ratio across the engine and hence, better thermal efficiency [5].

Compressor

The intake air is supplied to the compressor where energy is added into the air through compression by raising its temperature and pressure. This temperature and pressure should be high enough so that the combustion takes place in the combustion chamber. There are two common types of compressors used in turbojet engines. These commonly used compressors are axial and centrifugal compressor. In an axial compressor, the incoming air is directed parallel to the rotational axis whereas in a centrifugal compressor, the incoming air is directed radially outward, perpendicular to the rotational axis. The centrifugal compressors are most commonly used in the small gas turbines, where the power requirement is less than 5 MW. One of the drawbacks of using a centrifugal compressor is that they have very low efficiency as compared to the axial compressor, but they can provide a very high-pressure ratio in a single stage. This pressure ratio directly effects the amount of thrust produced, fuel consumed and the efficiency of the engine.

Another task of a compressor is that is it required to provide air, known as secondary air, which is used for bearing cavity sealing, turbine cooling, anti-icing and making sure that the thrust bearing does not wear out prematurely. This is to be noted here that this secondary air, hence cannot be used to produce thrust.

A CAD model of a centrifugal compressor is shown in the figure below:

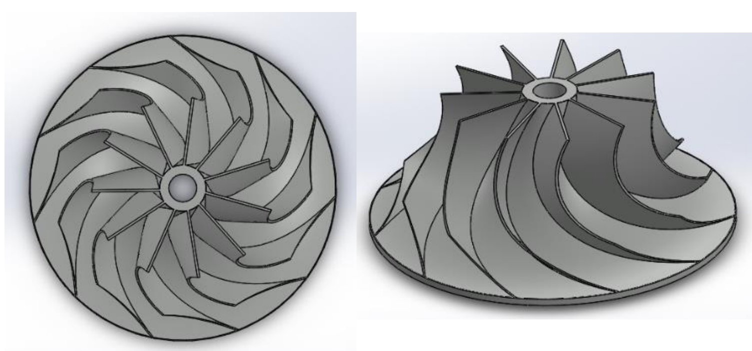


Figure 7: Centrifugal Compressor

The centrifugal compressor guides air radially along the axis of rotation. It compressors the air along the way and hence resulting in pressure and temperature rise. This rise in temperature results in the increase in enthalpy which means that more power must be provided by the turbine to the compressor to accommodate this pressure rise.

In order to figure out this pressure rise or the mass flow and rpms due to this pressure rise, a chart known as compressor map can be used. This is developed based on the computer simulations and empirical data. This chart provides us information about the mass flowrate, pressure ratio, rpm and efficiency of a centrifugal compressor [6]. One such compressor map is shown in the figure below:

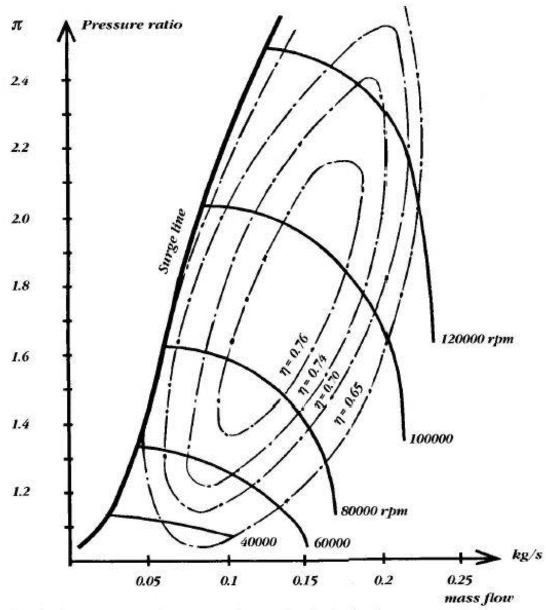


Figure 8: Compressor Map Example

Diffuser

The component which is right after a compressor is diffuser. Its purpose is to decelerate the incoming fluid, by converting that speed content into pressure. It does this by converting the kinetic energy into potential energy which serves as increase in pressure. Another reason why the air should be slowed down here in the diffuser is to facilitate the combustion chamber. The reduced speed of the air ensures that there a lesser flow loss in the combustion chamber. It also serves in stabilizing the combustion flame and the increased pressure improves the combustion efficiency. We can use several different diffusers for a small model of turbo-jet engine. The most popular one is the radial wedged diffuser. It was also used in the KJ66 jet engine. These diffusers can have different blade configurations. Some of them can have the curvature in the direction of rotation while others can have it opposite to the rotation of motion. There is also a configuration in which there is no curvature at all. Despite all these different configurations, each one of them can perform adequately [7].

Combustion Chamber

The decelerated, high pressure air from the diffuser is provided to the combustion chamber. The purpose of the combustion chamber is to increase the temperature of the air by adding heat. This heat is added by burning fuel. The most commonly used fuel is kerosene. The amount of fuel added in the combustion chamber is way less than the air flow through the engine. This air-fuel ratio usually ranges 45:1 to 130:1. But for the fuel to burn efficiently, this ratio should be around 15:1, which is also called stoichiometric air-fuel ratio.

Primarily, there are three common configurations of combustion chamber: can, can-annular and annular. For a smaller model of a turbo-jet engine, the annular configuration of the combustion chamber is commonly used.

Unlike compressor, the temperature in the combustion chamber is quite high. In order to bear this high of a temperature, the material used for the manufacturing of a combustion chamber should be selected carefully. Nickel-based alloys are most commonly used in the making of a combustion chamber.

An effective combustion chamber should fulfill following conditions:

- Continuous Combustion
- Uniform Combustion
- Low pressure loss
- Proper mixing of air and fuel
- Short length to cross-sectional area ratio

An annular combustion chamber is shown in the figure below:

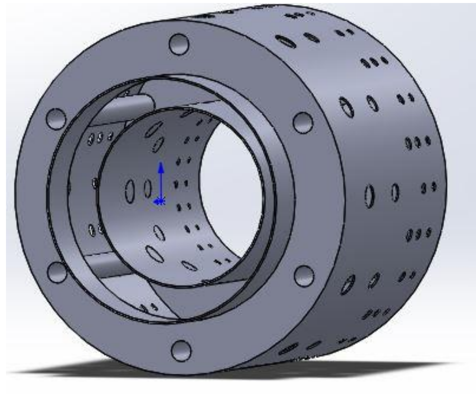


Figure 9: Annular Combustion Chamber

Combustion chamber is divided into three zones: primary, secondary and tertiary zones. This distinction is made on the base of hole sizes in those zones. The hole size decrease as we move from primary to the tertiary zone. Ignition takes place in the primary zone. In the secondary zone, more air is injected, and combustion completes. In the tertiary zone, leftover air is injected which is used to lower the temperature of the burnt mixture such that the combustion chamber can withstand the temperature and to make sure that the temperature is lowered down enough so that the turbine can withstand that.

As mentioned above, continuous combustion is necessary for a combustion chamber to be effective. For continuous combustion, we must make sure that the inlet temperature is high enough, fuel-air ratio is high, and pressure is high. If fuel-air ratio is not high enough, then instead of vaporizing the incoming fuel, the generated heat is used to increase the temperature of nitrogen and oxygen. Usually, to achieve this an alternative fuel is used initially to provide high enough temperature and then primary fuel is used to keep the combustion going.

Turbine Stage

Combustion chamber feeds this high temperature air to the turbine whose task is to expand that air and extract energy out of it while doing so. This energy is used to run the compressor and other auxiliary components of a turbojet engine. And the remaining energy content of this air is converted from high pressure to high speed fluid which is used to produce thrust.

There are two main types of turbines; axial and radial. In the axial-flow turbines, there are multiple stages to increase thrust and efficiency. They are used most commonly because they have high mass flow rates as compared to radial turbines. For a smaller model of jet engines, a single stage axial turbine is most commonly used. A single stage of a turbine comprises of a stator and a rotor. The stator works as guide vanes to direct the flow into the rotor without any swirl. They also help accelerating the flow. The stator plays a same role as the diffuser; they both guide the flow. But the difference is that the diffuser decelerates the flow whereas stator accelerates it. [8]

High temperature at the turbine inlets means higher thrust per unit mass but it also means that the turbine must bear higher thermal loads. Besides that, the angular velocities at the turbines are also very high. These working conditions means that we must use materials that can withstand high thermal loading and we also must try and keep the turbine somewhat cool using cooling techniques. Due to these issues, nickel-based alloys are used most commonly. Following figure shows a turbine wheel.

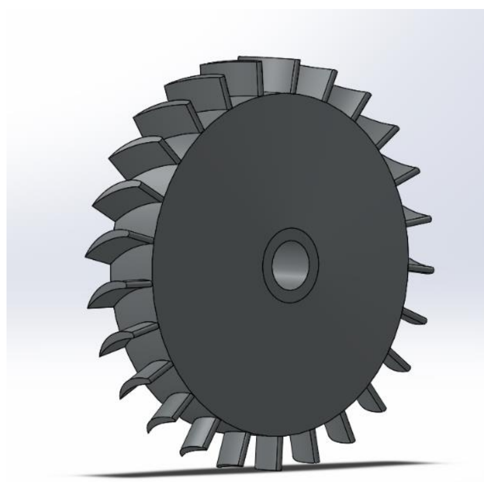


Figure 10: Turbine Rotor

While designing a turbojet engine, the turbine blade design is the most critical one. We must make sure that the blade produces a velocity vector which is identical everywhere at the exit and it has as little swirl as possible. Velocity triangles help us understand these stator and rotor much better. There are three commonly used variables when we talk about velocity triangles: absolute velocity of the blade ‘c’, flow speed ‘U’ and relative velocity ‘w’. The velocity vectors help us figure out states across the turbine which in turn help us design the blade itself.

Following figure shows a velocity triangle across a single stage of a turbine.

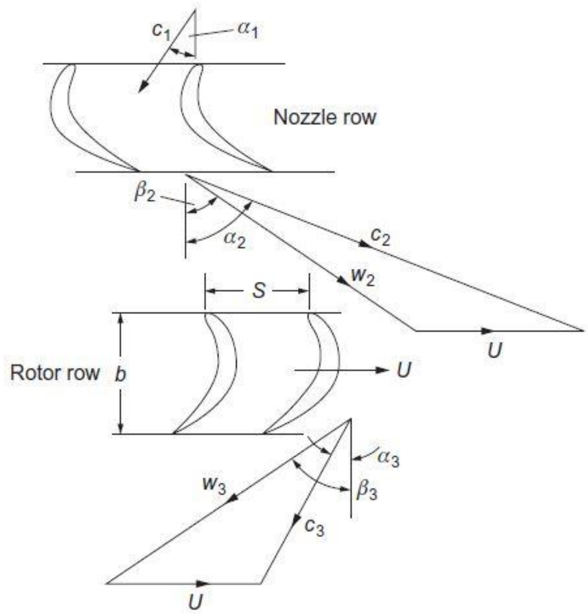


Figure 11: Velocity Diagram for Turbine Stage

Nozzle

A nozzle is a specially designed tube for the flow of hot gases in a turbojet. A nozzle converts potential energy of gases to kinetic energy, thus producing thrust. Nozzles decrease pressure by accelerating the hot gases whereas the mass flow rate remains unchanged. The length of nozzle is not an important factor; however, greater lengths can cause excessive frictional losses. Since we are dealing with a turbojet for a model aircraft, converging nozzle is used. Nozzle can also be converging diverging, but such nozzles are used for supersonic speeds. Nozzles also serve the purpose of avoiding shockwaves and noise reduction. Some assumptions are made for theoretical calculations of a converging nozzle. These include assuming a steady flow and steady state isentropic process, assuming the gas to be ideal, neglecting frictional losses and conservation of mass & energy. Based, on these assumptions a converging nozzle can be designed. Efficiency of more than 90%

is achievable if the nozzle is designed accurately. Choking can be avoided by reducing the change in cross-sectional are of a nozzle. [9]

Fuel System

Mixtures of liquid and gaseous fuels are usually used in gas turbine. Liquid fuels are commonly used in aircrafts as they can be more easily stored and do not impose the risk of gas-escape. Gaseous fuels have their own advantages as combustion can take place without vaporization. Since, the aim is to achieve high levels of energy using less amount of fuel, therefore, fuels with high specific heat are desirable for this purpose. Butane, diesel, propane and kerosene are usually used in gas turbines. These have specific heat values of about 40000 to 50000 kJ/kg.

Along with the type of fuel, the quantity of fuel consumption relative to the air consumed is of importance too. A proper air-fuel mixture not only enhances the engine performance but also prevents the formation of deposits on various components of the turbojet due to unburnt fuel. Hence, an appropriate air-fuel mixture is inevitable for ensuring efficient combustion and smooth running of the engine. The minimum amount of fuel required can be calculated by the following formula:

$$\dot{Q} = \dot{m}\, c\Delta T$$

$$\dot{V}_f = \dot{Q} / (\dot{q}_f\, \rho)$$

In case of liquid fuel, vaporization is achieved with the aid of vaporization tubes. These are placed inside the combustion chamber and help in exchanging heat from hot gases. For this purpose, small gas turbines are initially started on gaseous fuels so that high temperature in combustion chamber is attained which can aid in vaporization of the liquid fuels.

Chapter 3 Methodology

This chapter describes extensively the approaches taken to design each component of the turbojet engine. The design approach of each component is described individually along with the problems faced during their design and the measures taken to resolve those issues.

Compressor

Compressor is the most complex component to design and fabricate. Keeping in mind the time constraint, we decided to design our turbojet engine for a compressor by Thomas Kamps. Initially, a compressor, named MHI TD04-09B, from a turbocharger was acquired and the turbojet engine was designed for that compressor, but the pressure rise across the cycle and the turbine inlet temperature were not suitable because of the material and cost limitations. Hence, a compressor from kamps was selected and the whole turbojet engine was designed for that compressor.

Most of the compressor's design parameters, such as shroud radii, hub radius, tip height, flow angle at the exit of rotor blades and the number of blades, were known along with the compressor flow map which is shown below. [10]

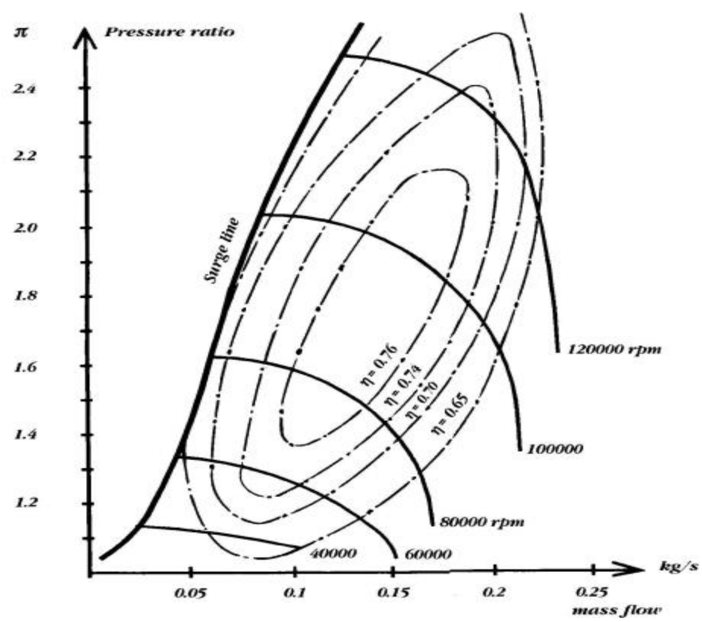


Figure 12: Compressor Flow Map

Complete thermodynamic analysis of the compressor, for a range of values of mass flow rate and pressure ratio, was carried out using Mathcad and excel. Results from Mathcad are shown in the Appendix.

The fundamental formulae used to calculate the thermodynamic states are as follows;

$$U = r\omega$$

$$A = 2\pi rb$$

$$p = \rho RT$$

$$\dot{m} = \rho Av$$

$$\frac{T_2}{T_1} = \pi^{\frac{k-1}{k}}$$

$$T_0 = T + \frac{v^2}{2c_P}$$

$$p_0 = p + \frac{1}{2}\rho v^2$$

$$M = \frac{v}{\sqrt{kRT}}$$

The remaining parameters such as both the velocities; absolute and relative, and flow angles were calculated using the velocity triangles at the rotor inlet and outlet as shown in the figures below:

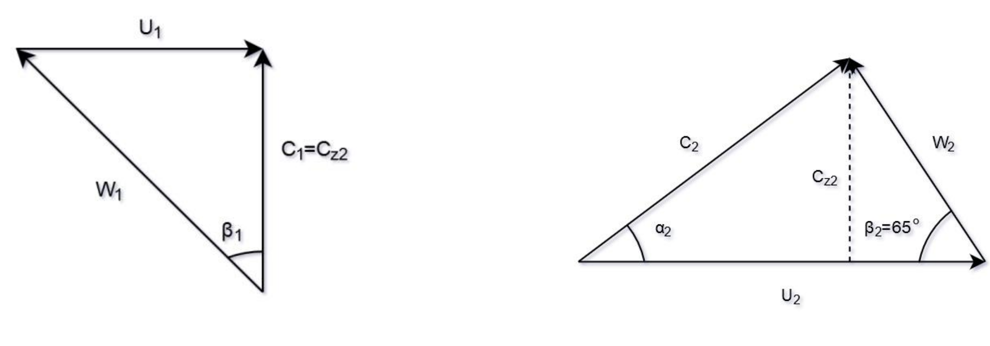


Figure 13: Velocity triangles at rotor inlet and exit (Compressor)

Vaneless Space between Compressor and Diffuser

The vaneless space between the compressor and diffuser is kept for the smooth flow of the fluid and to reduce the velocity to keep the Mach number low in a suitable range. Also, this area should not be increased that much so that the drag is increased beyond the acceptable limit. Keeping these things in mind, the radius of the rotor blades was increased by a factor of 1.15 to define the vaneless space i.e.,

$$r_{2d} = 1.15 r_{s_2}$$

Diffuser

The diffuser, located after the compressor and before the combustion chamber, is a divergent section of the turbojet engine. The main purpose of the diffuser is to slow down the speed of the incoming air from the compressor by increasing its pressure and guide that high-pressure, low-speed air into the combustion chamber. While designing the diffuser, the most important parameter to consider was the Mach number and hence we must decide the geometric parameters of the diffuser in way that the Mach number stays reasonable. Also, we had to consider the flow separation as the fluid flow into the diffuser vanes from the compressor vanes. The theory on turbojet design suggests that the number of vanes of the diffuser should be kept as many as the number of compressor vanes. By increasing the number of vanes past this limit increases the risk of undesirable flow behavior like flow separation and choking etc. Another factor which had to kept in mind while designing diffuser was to keep the frontal area of the jet engine as low as possible such that the drag is kept at a minimum. So, keeping the above-mentioned factors in mind, the number of diffuser vanes were decided to 13. The CAD model was developed and from that, the annulus areas at the diffuser inlet and exit were measured which were used in calculating the thermodynamic parameters across the diffuser.

$$\frac{dv}{v} = -\frac{dA}{A}\left(\frac{1}{1 - M^2}\right)$$

Combustion Chamber

While designing the combustion chamber, the most important factor to consider was the temperatures in the combustion chamber and its exit. Availability of materials along with the cost also played a role while designing the combustion chamber. All these factors limited us to keep our turbine inlet temperature to be at 700°C.

Thermodynamic states at the combustion chamber's inlet were known and now since, turbine inlet temperature was also decided based on our limitations, we can now calculate other thermodynamic states at the combustion chamber's inlet. Since combustion chamber has same area at the inlet and the exit so we can apply Rayleigh flow across the combustion chamber by ignoring the frictional losses since they are too small. The Rayleigh flow curves are given below:

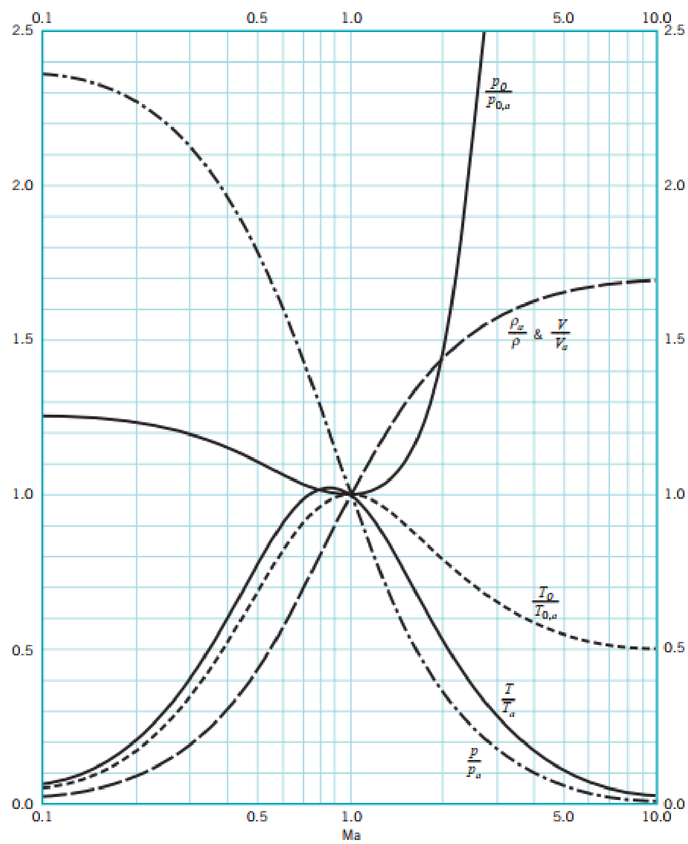


Figure 14: Rayleigh Flow Chart

Another important thing was to decide the fuel. After considering various fuels and going through their heating values, most feasible fuel which covered our heating requirements and was not too expensive as well was propane with a lower heating value of 46.4 MJ/kg. Now, after deciding the fuel, by applying the energy balance across the combustion chamber, the fuel ratio was calculated using the formula;

$$f = \frac{\frac{T_{04}}{T_{03}} - 1}{\frac{Q_f}{c_P T_{03}} - \frac{T_{04}}{T_{03}}}$$

Now, we must decide the holes on the combustion chamber. They were to be kept such that in the primary zone, the ratio of air-fuel mixture should be 14.85 and as we move along the combustion chamber, the size and number of dilution holes are increased as such that the required fuel ratio is achieved to justify the equations and achieve turbine inlet temperature equal to 700°C.

The first constraint to design a combustion chamber for a Turbojet is to decide which configuration of Combustion chamber will fit in our design and will be able to provide maximum heat input without minimum stagnation pressure loss.

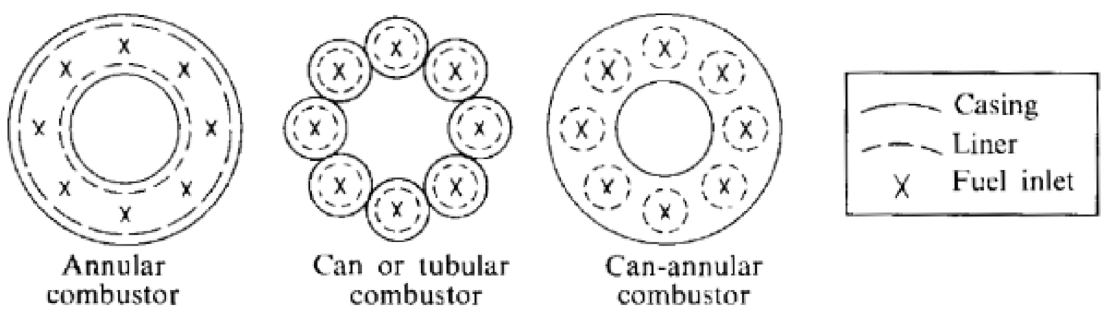


Figure 15: Types of Combustion Chambers

The calculations for fuel requirements at different temperatures for Rpm's is given in the excel file. Here, we will see a sample calculation for combustor outlet temperature of 923K. For this temperature the other conditions are:

$$m_a := 0.1575\ \frac{kg}{s}$$

$$T_{03} := 365\ K$$

Inlet Temperature of Combustor:

$$C_P := 1128\ \frac{J}{K \cdot kg}$$

$$\rho := 1.225\ \frac{kg}{m^3}$$

Specific Heat Constant:

$$Q_R := 46300000\ \frac{J}{kg}$$

Lower Heating Value:

$$T_{04} := 923\ K$$

Outlet Temperature of Combustor:

$$F := \frac{C_P}{Q_R}\left(T_{04} - T_{03}\right)$$

$$F = 0.01359$$

Fuel Air Ratio:

$$m_f := F \cdot m_a$$

Fuel Mass Flow Rate:

$$m_f = 0.00214\ \frac{kg}{s}$$

The combustion chamber is of annular type in which mass flow is distributed into three zones known as primary zone, secondary zone and dilution zone. These zones are demonstrated in picture below:

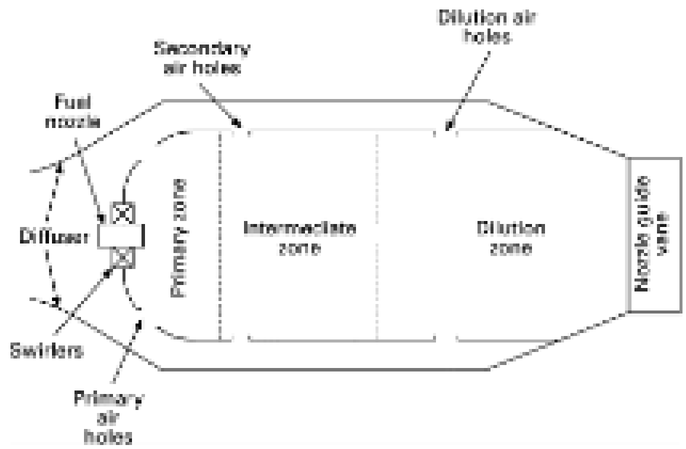


Figure 16: Zones of Combustion Chamber

In the zones shown in figure the flow rate is distributed as:

Primary Zone	21%
Secondary Zone	20%
Dilution Zone	59%

The approach for the design was to neglect the effect of change in density due to variation in temperature and divide the total area of liner openings into the percentages of primary, secondary and dilution zone.

$$m_{p_r} := 0.21 \cdot m_a$$

Air mass flow rate in Primary Zone:

$$m_{p_r} = 0.03308 \ \frac{kg}{s}$$

$$F_{P_r} := \frac{m_f}{m_{P_r}}$$

Fuel to air ratio in primary zone:

$$F_{P_r} = 0.06474$$

As shown above, the fuel to air ratio in Primary region is close to stoichiometric fuel air ratio of 0.067. This was our main goal to divide combustion chamber into zones.

Let's find out Equivalence ratio,

$$F_{stoich} := 0.067$$

$$F = 0.01359$$

$$\phi_{Overall} := \frac{F}{F_{stoich}}$$

$$\phi_{Overall} = 0.2029$$

$$\phi_{Primary} := \frac{F_{P_r}}{F_{stoich}}$$

$$\phi_{Primary} = 0.9662$$

We know that when the equivalence ratio is 1 the combustion is stoichiometric. If it is >1 the combustion mixture is rich and if it is <1 the combustion mixture is lean with excess air. The difference between the overall equivalence ratio and the equivalence ratio in primary zone is shown. This allowed better combustion in primary zone and the flame temperature which is very high was reduced to the required temperature at the outlet of combustion chamber by introduction of air in dilution zone. [11]

Turbine

The preliminary design of turbine was estimated at mean radius. After applying the concepts of Rayleigh flow on combustion chamber we found out the inlet conditions for Turbine. Since, we know the static and stagnation pressure we found the pressure drop across the turbine stage. We assumed the process to be isentropic initially and found the temperature and isentropic enthalpy drop.

$$\frac{T_S}{T_0} = \left(\frac{p}{p_o}\right)^{\frac{k-1}{k}}$$

The total to total efficiency is written as:

$$\eta_{tt} = \frac{\Delta w}{h_{01} - h_{03s}}$$

Here, we used small but realistic approximation:

$$\frac{1}{2}c_3^2 = \frac{1}{2}c_{3s}^2 = \frac{1}{2}c_x^2$$

Since our turbine is axial, we can take α_1 to be zero degrees and for gas turbines we should have thrust vector in completely axial direction. We considered degree of reaction to be 0.5 as this helps in simpler model and manufacturing.

$$R = 1 - \frac{\phi}{2}(tan\,\alpha_2 - tan\,\alpha_1)$$

$$\Delta w = 2(1-R)U^2 = \eta_{tt}\left(\Delta h_{is} - \frac{1}{2}C_x^2\right)$$

$$C_x^2 = \frac{\eta_{tt}\Delta h_{is} - 2(1-R)U^2}{\frac{1}{2}\eta_{tt}}$$

After finding the angles from the equations shown above velocity triangles were drawn as shown in Fig.

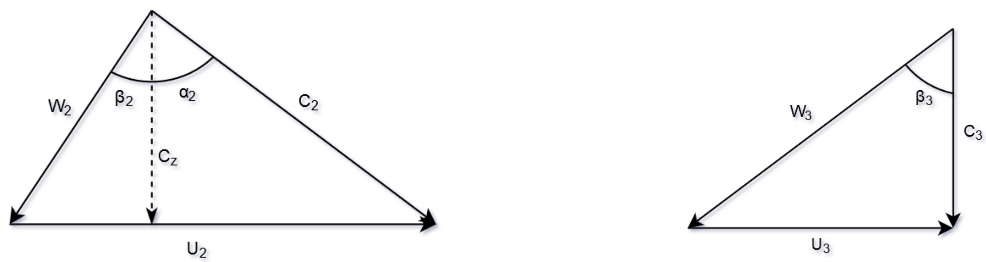


Figure 17: Velocity triangles at Rotor inlet and exit (Turbine)

Now, we found the nominal loss co-efficient using the simplest form of Soderbergh's loss co-relation and total to total efficiency using these formulas: -

$$\xi = 0.04\left(1 + 1 \cdot 5\left(\frac{\varepsilon}{100}\right)^2\right)$$

$$\eta_{tt} = \left(\frac{1 + \xi_R w_3^2 + \xi_N c_2^2}{2\Delta w}\right)^{-1}$$

Then, we found out the temperature drop across stator by using the velocity at exit of stator, which was found from velocity diagrams shown above. [12]

Now we must decide hub-to-tip ratio for turbine. The outer diameter of compressor will restrict the tip diameter of turbine, then we will find the annulus area required for turbine rotor which will decide the hub radius and height of blade is found using given eq:

$$r_h = r_t\sqrt{1 - \frac{A_2}{\pi r_t^2}}$$

After finding the flow parameters, we will choose a suitable geometry by estimating stagger angle and pitch-chord ratio using Zweifel's criterion as shown:

$$z = 2\left(\frac{s}{b}\right) cos^2 \beta_3 (tan\,\beta_3 + tan\,\beta_2)$$

$$tan\,\xi = \frac{cos\,\beta_2 - cos\,\beta_3}{sin\,\beta_2 + sin\,\beta_3}$$

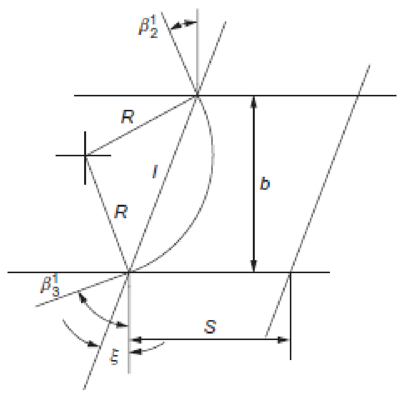


Figure 18: Flow through turbine

Material Selection for Turbine Rotor

As we are dealing with high rotation speeds and high temperatures, it is important to choose a material that can withstand such loads at high RPM's and high temperature. There were two materials available which could have provided such strength, but we had to choose the better. The tables showing the comparison of these two materials are shown below. Calculations were done for both materials, the stainless steel and Inconel (alloy of nickel majorly consisting of nickel, chromium and iron). The calculations for maximum deformation due to temperature and centrifugal stresses generated due to high RPM's were done and both materials were compared to analyze which gives better results.

Table 1: Calculations for Stainless Steel

Youngs Modulus (Pa)	Thermal expansion	Blade height (m)	Density (kg/m3)	Tip Radius(m)	Hub Radius (m)	Area (m2)	**Stainless Steel 304**		
2E+11	0.0000072	0.0055	8027	0.033	0.0273	1E-05			
Change in temperature	Change in length(m)	Elongation of Disc (m)	Total elongation (m)	Centrifugal Force (N)	RPM (rev/s)	RPM	Centrifugal Stress (Pa)	Yield Strength	Factor of Safety
200	7.776E-06	0.00004752	5.53E-05	267.750705	4186.7	40000	24340973.18	2.15E+08	8.832843
300	1.166E-05	0.00004752	5.918E-05	418.360477	5233.3	50000	38032770.6	2.1E+08	5.521554
400	1.555E-05	0.00004752	6.307E-05	602.439086	6280	60000	54767189.67	2.05E+08	3.743117
500	1.944E-05	0.00004752	6.696E-05	819.986534	7326.7	70000	74544230.38	2E+08	2.682971
600	2.333E-05	0.00004752	7.085E-05	1071.00282	8373.3	80000	97363892.74	1.8E+08	1.848735
680	2.644E-05	0.00004752	7.396E-05	1355.48794	9420	90000	123226176.7	1.6E+08	1.298425

Table 2: Calculations for Inconel 718

Youngs Modulus (Pa)	Thermal expansion	Blade height (m)	Density (kg/m3)	Tip Radius (m)	Hub Radius (m)	Area (m2)	**Inconel 718**		
1.724E+11	0.000013	0.0055	8193.25	0.033	0.0273	1E-05			
Change in temperature	Change in length (m)	Elongation of Disc (m)	Total elongation (m)	Centrifugal Force (N)	RPM (rev/s)	RPM	Centrifugal Stress (Pa)	Yield Strength	Factor of Safety
200	1.404E-05	0.0000858	9.984E-05	273.296183	4186.7	40000	24845107.58	9.8E+08	39.44439
300	2.106E-05	0.0000858	0.0001069	427.025287	5233.3	50000	38820480.59	9.8E+08	25.24441
400	2.808E-05	0.0000858	0.0001139	614.916413	6280	60000	55901492.05	9.8E+08	17.53084
500	0.0000351	0.0000858	0.0001209	836.969562	7326.7	70000	76088141.96	9.8E+08	12.8798
600	4.212E-05	0.0000858	0.0001279	1093.18473	8373.3	80000	99380430.32	9.8E+08	9.861096
680	4.774E-05	0.0000858	0.0001335	1383.56193	9420	90000	125778357.1	9.8E+08	7.791484

Material selection was pretty important step, but after selecting a material, we also had to do a complete analysis of that material so that we can find out the maximum deformation of the rotor and based on that, we can finalize the gap between the rotor and the stator so that both can function properly. The analysis of rotor is shown in the figure below:

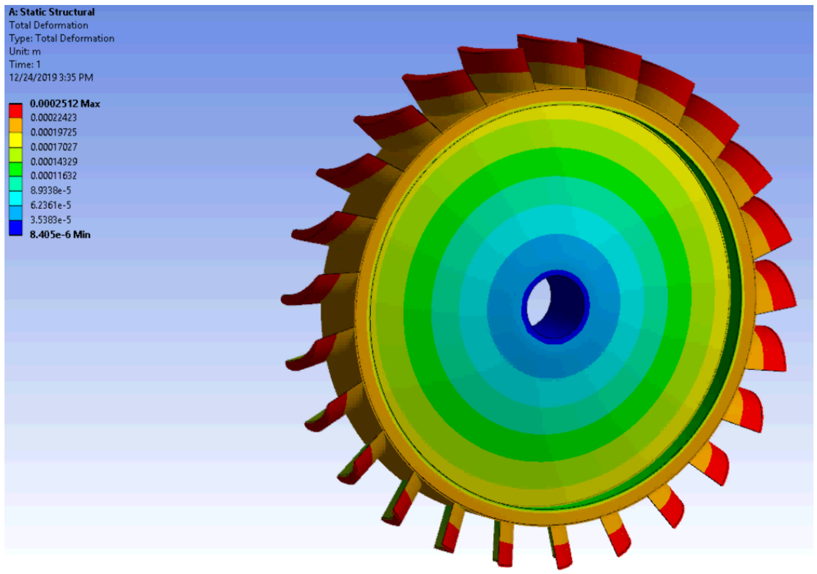


Figure 19: Total Deformation Analysis of Turbine Rotor

Nozzle

As diffuser is a divergent section of a jet engine, nozzle is the convergent one. The nozzle is a component after the turbine. The purpose of a nozzle is to produce thrust, throw exhaust gases back into the atmosphere and to set the mass flow rate through the turbojet engine. The nozzle increases the thrust of the turbojet engine by constricting the air flow. The aspect ratio of the nozzle is kept such that all the available energy is utilized to produce thrust. While designing the nozzle, we also must make sure that the Mach number remains less than 1 such that undesirable flow parameters like flow separation and choking don’t take place.

Shaft

Shaft is the component which transmits energy from the turbine to compressor. While designing the shaft, the most important factor to keep in mind is its balancing to minimize vibrations. For this purpose, the dynamic balancing of shaft is done, and the critical rotational speed of the shaft is calculated using Bohl's approximations which are as follows:

$$\omega = \sqrt{\frac{3EI}{m(l+c)c^2}}$$

$$\frac{1}{\omega^2} = \frac{1}{\omega_{compressor}{}^2} + \frac{1}{\omega_{turbine}{}^2} + \frac{1}{\omega_{shaft}{}^2}$$

The following figure shows the stress analysis of shaft:

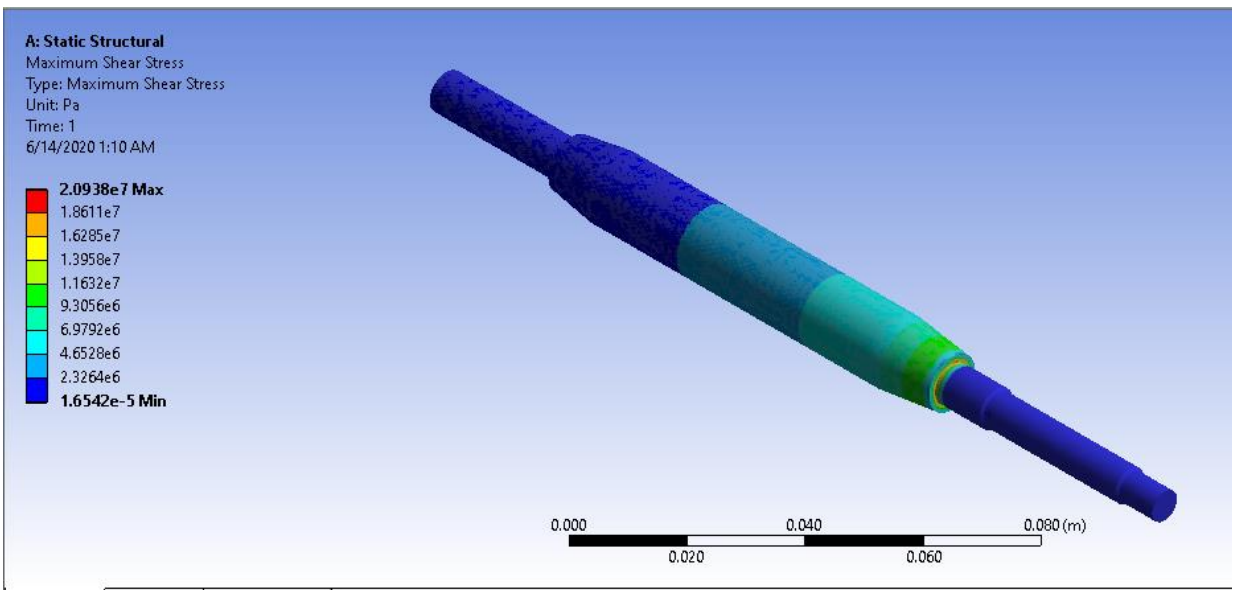


Figure 20: Stress Analysis of Shaft

Calculations

The complete calculations of turbojet engine at different rpm values of 40000, 60000, 80000, 100000 and 120000 were done using MATLAB. In order to explain the procedure adopted and formulae used to evaluate various parameters of turbojet, sample calculations at 80000 rpm were done using Mathcad. These calculations are given below.

Sample Calculations at 80000 rpm

Shroud Inlet and Outlet Radius:

$r_{s1} := 0.021\ m \qquad r_{s2} := 0.033\ m$

Rpm value: $\omega_0 := 100000\ rpm \qquad \omega := 2 \cdot \pi \cdot \frac{\omega_0}{60} \qquad \omega_0 = \left(1.047 \cdot 10^4\right)\ \frac{1}{s}$

Tip Height: $b := 0.00508\ m$

Relative flow angle at Compressor Outlet: $\beta_2 := 65\ deg \qquad b_{2d} := 0.006\ m$

Density of air: $\rho := 1.166\ \frac{kg}{m^3}$

Number of compressor blades: $N := 12$

Gas Constant: $R := 287\ \frac{J}{kg \cdot K}$

Specific Heat Ratio of Air: $k := 1.4$

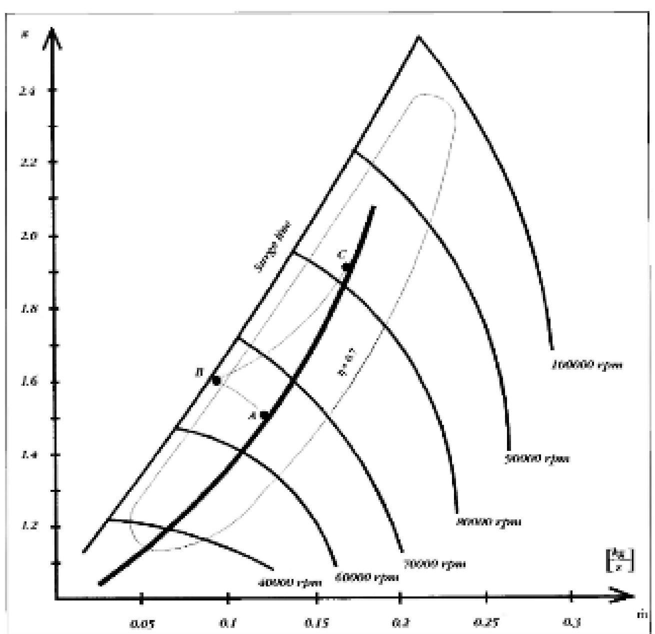
Specific Heat at Constant Pressure: $c_p := 1005\ \frac{J}{kg \cdot K}$

Efficiency of Compressor: $\eta := 0.7$

Standard Air Pressure: $P_1 := 98100\ Pa$

Standard Air Temperature: $T_1 := 293\ K$

Kamps has proved experimentally that if we can consider the working of compressor of turbocharger within the 70% efficiency range. The compressor surges due to simple reason which is when it tries to give more pressure ratio at restricted mass flow rate or the annulus area of turbine is small that whenever jet engine is accelerated mass flow rate is pushed at higher pressure ratio through small annulus area which cause compressor to surge. Whenever jet engine is accelerated it reaches towards point B but comes back to point C.



As we know that centrifugal compressor wheel which we used has back sweep angle of 65 degrees which means that the relative flow angle at outlet of compressor wheel is known and from compressor map we can find mass flow rate and pressure ratio which are:

Mass Flow Rate: $m_0 := 0.196\ \frac{kg}{s}$

Pressure Ratio: $P_{ra} := 1.72$

Kamps find out from his calculations and experimentations that half of the pressure ratio increases in compressor wheel. Also, it proved in its calculations and experimentations that we can take square root of pressure ratio shown in compressor map to do flow calculations at outlet of compressor wheel.

$$P_r := \sqrt{P_{ra}}$$

$$P_r = 1.311$$

Since in these preliminary calculations, we are considering the jet engine to be static. The reason for this is that we do not have flight speeds and conditions or details of the aircraft this is to be fitted in. Similarly, diffuser design is based on the fight conditions, weight and drag of aircraft in which it is to be fitted in. Since, we do not have such data we made our problem general. Also, the gas turbine produces minimum thrust at ground and in turbojets afterburner configuration is needed to increase the thrust and efficiency at ground level. Due to this reason the stagnation and static conditions at inlet are same:

$$T_{01} := T_1 \quad T_{01} = 293\ K \qquad P_{01} := P_1 \quad P_{01} = \left(9.81 \cdot 10^4\right)\ Pa$$

$$P_2 := P_r \cdot P_1$$

$$P_2 = \left(1.287 \cdot 10^5\right)\ Pa$$

Since, we know the angular velocity, inducer radius and exducer radius of compressor wheel we can find tip velocities at inlet and outlet of compressor wheel.

$$u_2 := r_{s2} \cdot \omega_0 \qquad u_2 = 345.575\ \frac{m}{s}$$

$$u_1 := r_{s1} \cdot \omega_0 \qquad u_1 = 219.911\ \frac{m}{s}$$

Static Temperature at Compressor Entry:

$$T_2 := T_1 \cdot P_r^{\frac{(k-1)}{k}} \qquad T_2 = 316.603\ K$$

Compressor Exit Area: $A := 2 \cdot \pi \cdot r_{s2} \cdot b \qquad A = 0.001053\ m^2$

Density of Air at Compressor Inlet $\rho_2 := \frac{P_2}{R \cdot T_2} \qquad \rho_2 = 1.416\ \frac{kg}{m^3}$

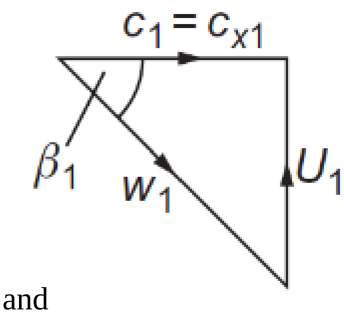
Absolute Axial Velocity at Compressor Inlet:

$$c_{x1} := \frac{m_0}{A \cdot \rho_2} \qquad c_{x1} = 131.42 \ \frac{m}{s}$$

A well-known assumption in design of turbomachinery was made which is that the axial velocity at inlet of compressor wheel is equal to radial component of velocity at outlet.

$$c_{r2} := c_{x1} \qquad c_{r2} = 131.42 \ \frac{m}{s}$$

By Using velocity diagrams, we can find relative flow angle and relative velocity at inlet.

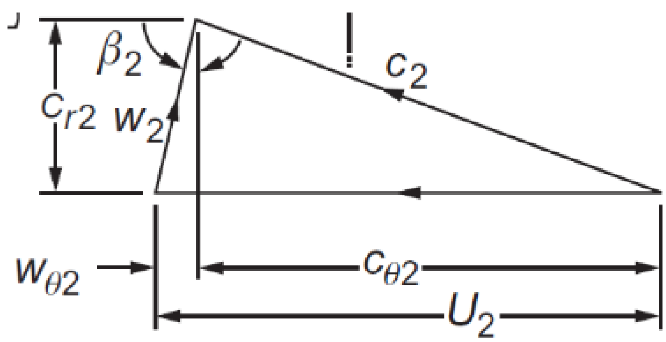


$$\beta_1 := \operatorname{atan}\left(\frac{u_1}{c_{x1}}\right) \qquad \beta_1 = 59.137 \ deg$$

Relative Velocity at Compressor Inlet: $= \sqrt{c_{x1}^{\ 2} + u_1^{\ 2}}$

$$w_1 = 256.188 \ \frac{m}{s}$$

Similarly, by using velocity triangles we can find relative and absolute flow conditions at outlet of compressor wheel.



Absolute Tangential Velocity at Compressor Exit: $c_{\theta 2} := u_2 - \left(\frac{c_{r2}}{\tan(\beta_2)}\right) \qquad c_{\theta 2} = 284.293 \ \frac{m}{s}$

Absolute Velocity at Compressor Exit: $c_2 := \sqrt{c_{\theta 2}^{\ 2} + c_{r2}^{\ 2}} \qquad c_2 = 313.199 \ \frac{m}{s}$

Absolute Flow Angle: $\alpha_2 := \operatorname{atan}\left(\frac{c_{r2}}{c_{\theta 2}}\right) \qquad \alpha_2 = 24.81 \ deg$

α_2 is the angle which was used to set the angle of diffuser guide vanes.

Relative Velocity at Compressor Exit: $w_2 := \sqrt{(u_2 - c_{\theta 2})^2 + c_{r2}^{\ 2}} \qquad w_2 = 145.006 \ \frac{m}{s}$

Stagnation Pressure at Compressor wheel outlet: $P_{02} := P_2 + \frac{1}{2}\left(\rho_2 \cdot c_2^{\ 2}\right)$

$$P_{02} = \left(1.981 \cdot 10^{5}\right)\ Pa$$

Stagnation Temperature at Compressor Entry: $T_{02} := T_2 + \dfrac{c_2^{\,2}}{2\ c_p}$ $\qquad T_{02} = 365.406\ K$

Vaneless Space Calculations

Vaneless space allows flow to be less turbulent by removing any disturbances in flow after compressor wheel. Experimentations have shown that vaneless space should be between 1.05 to 1.20 of exducer radius of compressor wheel.

$$r_{2d} := 1.15 \cdot r_{s2} \qquad r_{2d} = 0.03795\ m$$

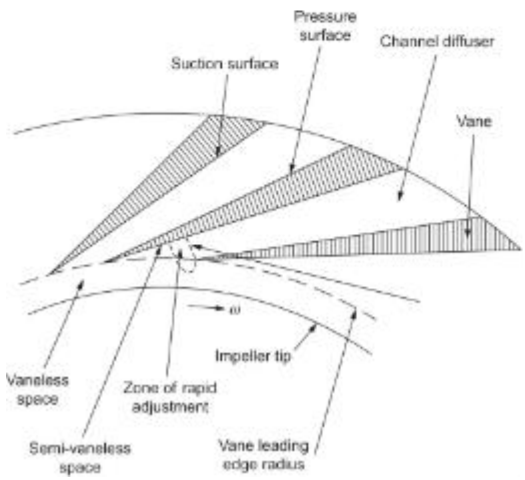
The approach used to solve the flow conditions in this region was that we know that there is no work interaction in vaneless space which means:

$P_{02d} := P_{02} \qquad T_{02d} := T_{02}$

Absolute and tangential velocity at the end of vaneless space was approximated as:

$$c_{\theta 2d} := \frac{r_{s2}}{r_{2d}} \cdot c_{\theta 2} \qquad c_{\theta 2d} = 247.211\ \frac{m}{s}$$

$$c_{r2d} := \frac{r_{s2}}{r_{2d}} \cdot c_{r2} \qquad c_{r2d} = 114.278\ \frac{m}{s}$$



By using the tangential and radial components of absolute velocities at outlet of vaneless space, absolute velocities were calculated. Then, absolute velocity was used to find out the flow conditions at the end of vaneless space and iterations were performed in MATLAB to find the results of flow conditions at outlet of vaneless space.

Absolute Flow Angle: $\alpha_{2d} := \operatorname{acos}\left(\dfrac{c_{r2d}}{c_{\theta 2d}}\right)$ $\qquad \alpha_{2d} = 62.466\ deg$

Absolute Velocity in Diffuser: $c_{2d} := \sqrt{c_{\theta 2d}^{\,2} + c_{r2d}^{\,2}}$ $\qquad c_{2d} = 272.347\ \dfrac{m}{s}$

Now as told before this c_{2d} will be used to find out static temperature, static pressure and other flow conditions at inlet of vanned diffuser.

$$T_{2d} := T_{02} - \frac{c_{2d}^{\,2}}{2 \cdot c_p} \qquad T_{2d} = 328.504\ K$$

Density of Air in Diffuser: $\rho_{2d} := \dfrac{P_{02d}}{R \cdot T_{02d}}$

Mach number at vaneless space exit $M_{2d} := \frac{c_{2d}}{\sqrt{k \cdot R \cdot T_{2d}}}$ $M_{2d} = 0.75$

$$c_{r2d} := \frac{m_0}{\rho_{2d} \cdot 2 \cdot \pi \cdot r_{2d} \cdot b_{2d}} \qquad c_{r2d} = 72.524 \ \frac{m}{s}$$

$$c_{2d} := \sqrt{c_{\theta 2d}^{\ 2} + c_{r2d}^{\ 2}} \qquad c_{2d} = 257.63 \ \frac{m}{s}$$

Static Air Pressure inside diffuser: $P_{2d} := P_{02} - \frac{1}{2} \left(\rho_{2d} \cdot c_{2d}^{\ 2}\right)$ $P_{2d} = \left(1.354 \cdot 10^5\right) \ Pa$

Here, we defined another parameter which is quite straight forward, the height of vanned diffuser which is going to equal to height of annulus space of compressor wheel.

$$b_{2d} := 0.006$$

Vanned Diffuser Calculations

The flow conditions at the exit of vanned diffuser were found by using compressible flow relations. The first step was to find out the Inlet and Outlet area of vanned diffuser, this was done by using solidworks.

$$A_{2d} := 6.36277 \cdot 10^{-5}$$

$$A_3 := 8.29918 \cdot 10^{-5}$$

As we know that there is no work interaction in vanned diffuser, the stagnation temperature would remain constant.

$$T_{03s} := T_{02d} \qquad T_{03s} = 365.406 \ K$$

Inlet Temperature of Combustor: $T_{03} := \frac{\left(T_{03s} - T_{2d}\right)}{0.97} + T_{2d}$ $T_{03} = 366.547 \ K$

where 0.97 is the efficiency of vanned diffuser which is basically due to the fact that there is a decrease in velocity due to friction also.

By Using Compressible flow relation,

Absolute Velocity at Combustion Inlet: $c_3 := c_{2d} + \frac{c_{2d} \cdot \left(A_{2d} - A_3\right)}{A_{2d} \cdot \left(1 - M_{2d}^{\ 2}\right)}$

$$c_3 = 78.643 \ \frac{m}{s}$$

By using absolute velocity at vanned diffuser outlet or combustion chamber inlet, we can find other flow conditions on vanned diffuser outlet.

Static Temperature at combustor inlet: $T_3 := T_{03} - \frac{c_3^{\ 2}}{2 \ c_p}$ $T_3 = 363.47 \ K$

$$P_{03} := P_{02d} \qquad P_{03} = \left(1.981 \cdot 10^5\right)\ Pa$$

Static Pressure at combustor inlet: $P_3 := \dfrac{P_{03}}{\left(1 + \dfrac{\left(c_3{}^2\right)}{2 \cdot R \cdot T_3}\right)} \qquad P_3 = \left(1.924 \cdot 10^5\right)\ Pa$

Density at Vanned diffuser exit $\rho_3 := \dfrac{P_3}{R \cdot T_3} \qquad \rho_3 = 1.844\ \dfrac{kg}{m^3}$

Mach Number at Compressor Exit: $M_3 := \dfrac{c_3}{\sqrt{k \cdot R \cdot T_3}}$

$$\boxed{M_3 = 0.20579}$$

This Mach number M_3 at the inlet of combustion chamber has much significance because its value determines the flow conditions at the end of combustion chamber. The value of Mach number M_3 determines whether the flow is going to be sonic or supersonic at the exit of combustion chamber. The reason is that as we move along the Rayleigh line, enthalpy of flow increases due to addition of heat, with that the velocity of flow also increases.

Combustion Chamber Calculations

The calculations method in combustion chamber is explained earlier. Based on the Mach number at compressor exit, the intermediate state is determined using pressure, temperature and density ratios obtained from Rayleigh flow tables.

$$\rho_a := .191828 \cdot \rho_3 \qquad \rho_a = 0.354\ \frac{kg}{m^3}$$

$$T_{0a} := \frac{T_{03}}{0.34686} \qquad T_{0a} = \left(1.057 \cdot 10^3\right)\ K$$

$$T_a := \frac{T_3}{0.40887} \qquad T_a = 888.962\ K$$

$$P_{0a} := \frac{P_{03}}{1.1985} \qquad P_{0a} = \left(1.653 \cdot 10^5\right)\ Pa$$

$$P_a := \frac{P_3}{2.1314} \qquad P_a = \left(9.027 \cdot 10^4\right)\ Pa$$

The temperature required at turbine inlet is 900K for 80000rpm which means T04 is known which was used to find Mach number at combustion chamber exit and other properties were found using Rayleigh flow table.

$$T_{04} := 900\ K \qquad M_4 := 0.7$$

$$\rho_4 := 1.4337 \cdot \rho_a \qquad \rho_4 = 0.507\ \frac{kg}{m^3}$$

$$T_4 := 0.9929 \cdot T_a \qquad T_4 = 882.65\ K$$

$$P_{04} := 1.0431 \cdot P_{0a} \qquad P_{04} = \left(1.724 \cdot 10^5\right)\ Pa$$

$$P_4 := 1.4235 \cdot P_a \qquad P_4 = \left(1.285 \cdot 10^5\right)\ Pa$$

Turbine Calculations

Since, turbine exit, and initial entry are both atmospheric:

$$P_6 := P_1 \qquad u_5 := u_2 \qquad k := 1.354 \qquad c_p := 1158.4\ \frac{J}{kg \cdot K}$$

$$c_4 := M_4 \cdot \sqrt{k \cdot R \cdot T_4} \qquad c_4 = 409.961\ \frac{m}{s}$$

At first, we calculated the isentropic enthalpy drop across the turbine.

$$T_{6ss} := T_{04} \cdot \left(\frac{P_6}{P_{04}}\right)^{\frac{(k-1)}{k}} \qquad T_{6ss} = 776.625\ K$$

$$\Delta h_{is} := c_p \cdot T_{04} \cdot \left(1 - \frac{T_{6ss}}{T_{04}}\right) \qquad \Delta h_{is} = \left(1.429 \cdot 10^5\right)\ \frac{J}{kg}$$

Here, we finalized the value of reaction of turbine stage. The reaction in turbine stage most common in design of turbojet engines is 0.5.

Reaction of turbine stage: $R_d := 0.5$

Let's assume that the total to total efficiency of turbine stage is 90%. $\eta_{tt'} := 0.9$

Axial Velocity: $c_x := \sqrt{\dfrac{\eta_{tt'} \cdot \Delta h_{is} - 2\ (1 - R_d) \cdot {u_5}^2}{0.5 \cdot \eta_{tt'}}}$ $\qquad c_x = 143.012\ \dfrac{m}{s}$

$$\Delta W := 2\ (1 - R_d) \cdot {u_5}^2 \qquad \Delta W = \left(1.194 \cdot 10^5\right)\ \frac{J}{kg}$$

Stage Loading Co-efficient: $\psi := \dfrac{\Delta W}{{u_5}^2}$ $\qquad \psi = 1$

Flow Coefficient: $\phi := \dfrac{c_x}{u_5}$ $\qquad \phi = 0.414$

Absolute flow angle at Turbine Rotor inlet: $\alpha_5 := \operatorname{atan}\left(\dfrac{\Delta W}{c_x \cdot u_5}\right)$ $\qquad \alpha_5 = 67.518\ deg$

Relative Flow Angle at Turbine Rotor exit: $\beta_6 := \operatorname{atan}\left(\dfrac{1}{\phi}\right)$ $\qquad \beta_6 = 67.518\ deg$

Relative Flow Angle at Turbine Rotor $\beta_5 := \tan\left(\beta_6\right) - \dfrac{2 \cdot R_d}{\phi}$ $\qquad \beta_5 = -2.544 \cdot 10^{-14}\ deg$

Relative Velocity at Turbine Stage exit: $w_6 := \dfrac{c_x}{\cos(\beta_6)}$ $\qquad w_6 = 373.998\ \dfrac{m}{s}$

Absolute Velocity at Turbine Rotor inlet: $c_5 := \dfrac{c_x}{\cos(\alpha_5)}$ $\qquad c_5 = 373.998\ \dfrac{m}{s}$

Now, we used Soderberg's loss coefficients to find the total to total efficiency which was approximated earlier.

For rotor: $\varepsilon_r := \beta_5 + \beta_6$ $\qquad \varepsilon_r = 67.518\ deg$

$$\xi_r := 0.04 \cdot \left(1 + 1.5 \cdot \left(\frac{\varepsilon_r}{100}\right)^2\right) \qquad \xi_r = 0.04$$

For nozzle or stator: $\varepsilon_n := \alpha_5$ $\qquad \varepsilon_n = 67.518\ deg$

$$\xi_n := 0.04 \cdot \left(1 + 1.5 \cdot \left(\frac{\varepsilon_n}{100}\right)^2\right) \qquad \xi_n = 0.04$$

Total to total Efficiency: $\eta_{tt} := \left(1 + \dfrac{\xi_r \cdot {w_6}^2 + \xi_n \cdot {c_5}^2}{2 \cdot \Delta W}\right)^{-1}$ $\qquad \eta_{tt} = 0.955$

Total to Static Efficiency: $\eta_{ts} := .6232$

Since, we have found the value for absolute velocity at stator outlet, we can find the static temperature and Mach number at stator exit.

Temperature at Turbine Rotor inlet: $T_5 := T_{04} - \dfrac{{c_5}^2}{2\ c_p}$ $\qquad T_5 = 839.626\ K$

Mach Number at Turbine Rotor inlet: $M_5 := \dfrac{c_5}{\sqrt{k \cdot R \cdot T_5}}$ $\qquad M_5 = 0.655$

We know that the losses in stator are usually associated with friction and with good estimation of flow angles in stator we can assume the efficiency in stator to 97%.

$$\eta_n := 0.97$$

Since, we know the static temperature at the exit of stator, we can find other flow conditions.

Pressure drop across stator: $\pi' := 1 - \dfrac{1}{\eta_n}\left(1 - \dfrac{T_5}{T_{04}}\right)$

$$\pi' = 0.931$$

Static pressure at turbine rotor inlet: $P_5 := \pi'^{\left(\frac{k-1}{k}\right)} \cdot P_{04}$ $\qquad P_5 = \left(1.692 \cdot 10^5\right)\ Pa$

Density at turbine rotor inlet: $\rho_5 := \dfrac{P_5}{R \cdot T_5}$ $\qquad \rho_5 = 0.702\ \dfrac{kg}{m^3}$

Throat Area of Turbine Stage: $A_2 := 0.00108584\ m^2$

From A2, the throat area of turbine stage which is calculated using mass flow rate and density at stator. This area is used to calculate the hub and tip radius of turbine, which are further used to find the blade height.

Tip Radius: $r_t := \sqrt{\dfrac{A_2}{\pi' \cdot (1 - 0.75^2)}}$ $r_t = 0.05164\ m$

Hub radius: $r_h := 0.75 \cdot r_t$ $r_h = 0.039\ m$

Blade Height: $H = 0.013\ m$

By using the velocity triangles, we can find out the relative and absolute velocities at rotor inlet and exit.

Relative velocity at Rotor Inlet: $w_5 := \sqrt{{c_5}^2 - {u_5}^2}$ $w_5 = 143.012\ \dfrac{m}{s}$

Absolute Velocity at Turbine Stage $c_6 := \sqrt{{w_6}^2 - {u_5}^2}$ $c_6 = 143.012\ \dfrac{m}{s}$

Let's estimate pitch to chord ratio, for this Zweifel's criterion is used.

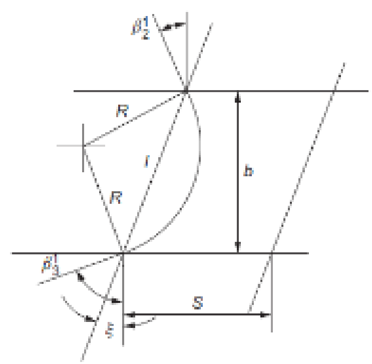
$z := 0.8$ $s := H$ $b := \dfrac{s}{0.8}$ $s = 0.013\ m$ $b = 0.016\ m$

Number of blades $N_b := 2 \cdot \pi \cdot \dfrac{r_t}{s}$ $N_b = 25.133$

Now that we have found out the thickness, spacing and number of blades, our goal is that how the airfoil should be placed to get most efficiency. For this we calculated stagger angle which tells us about the orientation of airfoil as shown in figure.

$$\varepsilon := \operatorname{atan}\left(\frac{\cos(\beta_5) - \cos(\beta_6)}{\sin(\beta_5) + \sin(\beta_6)}\right)$$

$\varepsilon = 33.759\ deg$



Chapter 4 Results and Discussion

At the end of the project tenure, many major components of the turbojet engine were either fabricated or acquired, whereas, some of the components were still in the manufacturing phase. The detailed specifications of all these components, their images and the results obtained have been discussed below:

Compressor

Since, the compressor is a complex part to design and fabricate, therefore, keeping the time constraint in mind we acquired the compressor wheel. After surveying the market, we found and acquired the required compressor model from a spare parts vendor Adnan Turbo in Rawalpindi. The specifications of this compressor formed the basis of our calculations and design of further components of turbojet. The Compressor wheel that we acquired had the following dimensions and specifications as shown in the table below:

Table 2: Compressor Specifications

Component	**Dimension**
Overall Height	31.5 mm
Tip Height	6 mm
Nose Diameter	13.5 mm
Blade Height	29.1 mm
Bore Size	6.5 mm
Inducer	52 mm
Tapered Tip Exducer	68 mm
Sweep Angle	65°
Tapered Tip Angle	30°
Number of Blades	12

Figure 21: Compressor

Turbine (Rotor)

Our design included an axial turbine which is commonly used in case of small turbojet engines. The turbine blade design is the most critical element while designing a turbojet. It is essential to keep the swirl as minimum as possible and the blade is designed such that the velocity is identical at the exit. Keeping these constraints in mind, high precision is required during fabrication. Such a facility was not accessible and easily available to us in Pakistan. Hence, it was decided to purchase the turbine rotor and after constant search, we finalized and ordered online from AliExpress from a Chinese vendor. It is shown in the figure below:

Figure 22: Turbine (Rotor)

Table 3: Turbine Specifications

Component	**Dimension**
Height	10 mm
Diameter (Hole)	7.59 mm
Weight	0.19 kg
Number of Blades	23

Shaft and Shaft Tunnel

One mild steel billet of 20cm in length and 1.8cm in diameter was bought from Nouman Steel, Sadar, Rawalpindi for the fabrication of shaft. Another aluminum billet of about 12cm in length and 6cm in diameter was bought from M Sardar, Sadar, Rawalpindi for the fabrication of shaft tunnel. Both billets were then lathed to size in MRC.

Stator

For stator, a piece of stainless steel was bought from Nouman Steel, Sadar, Rawalpindi. Then, that piece was plasma cut from Design Torch, Turnol, Rawalpindi as shown in the figure below:

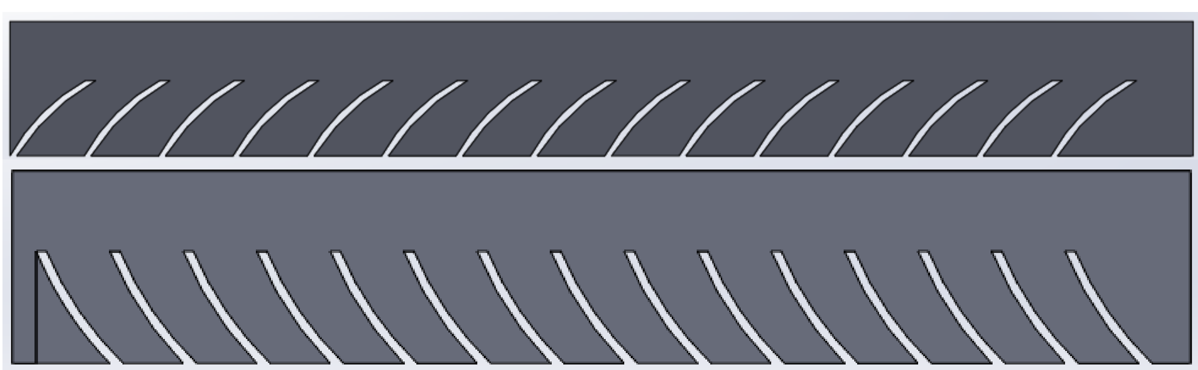


Figure 23: Plasma Cut Pieces for Turbine (Stator)

Then, those pieces were provided to Mr. Abdul Jabaar's workshop in Gawaalmandi, Rawalpindi where they were bent into shape through a process called sheet metal rolling and then connected through the stainless-steel pieces as shown in the figure below:

Figure 24: Fabricated Turbine (Stator)

Fuel line

The fuel line was manufactured by acquiring a 6mm stainless steel pipe from Mr. Abdul Jabbar's workshop in Gawaalmandi, Rawalpindi. That piece was bended to size in MRC and then the holes were also drilled in the MRC. Then, a piece of seamless pipe almost 30 centimeters in length with a diameter of 4mm was bought from Ismail Sons, Sadar, Rawalpindi. Then, from that seamless pipe, some smaller pieces of about 7-8 cm were cut and then were press-fitted in that steel pipe. After that, the steel pipe was permanently shut from one end so that there was no leakage. The fuel line is shown in the figure below:

Figure 25: Fuel Line

Combustion chamber

An 18-gage stainless steel sheet was bought from M Sardar, Sadar, Rawalpindi. From that sheet, two more rectangular pieces were cut for the outer and inner liner of the combustion chamber from the MRC. Then holes were drilled according to the zones of the combustion chamber in those two pieces. Two circular flanges were also fabricated in the MRC for the front and back covering of the combustion chamber. One of those flanges was solid while the other had holes particularly in place to fit the fuel line in. The sheet metal rolling for the liners and the welding of the flanges on those liners were done from Mr. Abdul Jabbar's workshop in Gawaalmandi, Rawalpindi. The final model of the combustion chamber is shown in the figure below:

Figure 26: Fabricated Combustion Chamber

Parts in Manufacturing Phase

The fabrication phase was halted due to unforeseen circumstances while a few parts were in the fabrication phase. Materials for these parts had been acquired and the manufacturing of these was underway. The details of these components, work completed and to be done along with their specifications is enlisted below:

Diffuser Guide Vanes

The billet of aluminum was acquired from Ismail Jee, Sadar, Rawalpindi. The dimensions of this billet are 150*125*19 millimeters. This billet was to be machined by CNC Milling Machine to convert it into diffuser guide vanes.

Nozzle

The stainless-steel sheet of 18-Guage was acquired to make combustion chamber, nacelle and nozzle. As shown in cad model of nozzle, it basically consists of two parts, a nozzle shaped outer part and inner cone. These two parts are to be made by same process, stainless-steel sheet is to be rolled into their particular shapes and then TIG welding would be used make the permanent joint on rolled sheet. Then, these parts would be joined by drilling 3 adjacent holes in both and bolt and nut would be used to join these parts into required result.

Shroud

First approach to manufacture this part was to acquire the intake port (to engine) of turbocharger and convert it according to our desired shape of shroud which would be compatible to our assembly. This approach was cost effective, but the acquired part of turbocharger did not have compatibility of compressor rotor and this approach was rules out. Now, a better approach was finalized in which nylon billet would be acquired and CNC machined into the required shape of shroud.

Nacelle

The stainless-steel sheet was cut into desired dimensions and a flange of 115.6 mm diameter was machined on lathe. The stainless-steel sheet would be rolled into desired diameter, then it would be TIG welded. After this both the flange and rolled stainless-steel sheet and flange would be welded using TIG welding.

Velocity Triangles

The velocity triangles at 80000rpm at compressor and turbines are shown in the figures below:

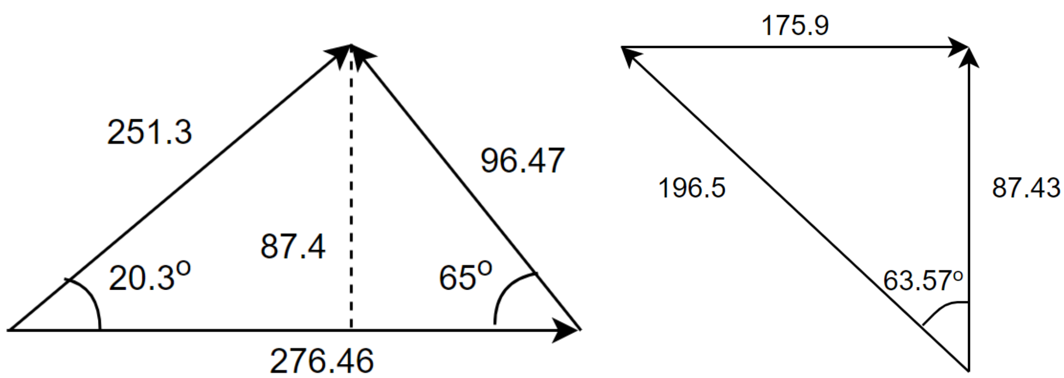


Figure 27: Velocity Triangles across Compressor Rotor

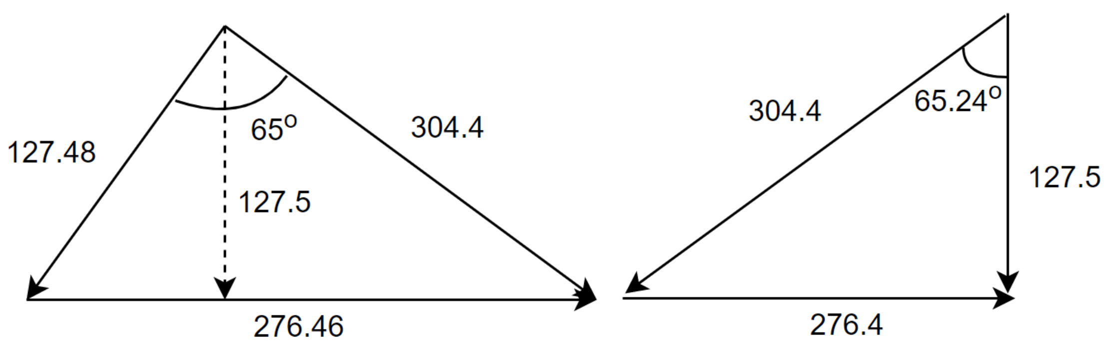


Figure 28: Velocity Triangles across Turbine Rotor

Values at Different Stages

The following table shows various states of our turbojet engine while running at 80000 rpms having produced a static thrust of 20.57 newtons.

Table 4: Values at various state of turbojet engine at 80000rpm

Stages	**P_o(kPa)**	**T_o(K)**	**M**	**c(m/s)**
Compressor Inlet	98.1	293	0.25	87.48
Compressor Outlet/Diffuser Inlet	164.13	342.72	0.71	251.38
Diffuser Outlet/Combustion Chamber Inlet	164.13	343.43	0.31	112.98
Combustion Chamber Outlet/Turbine Inlet	142.85	900	0.7	391.43
Turbine Outlet	102.73	819.7	0.53	127.22

T-s and P-V Diagrams

The following figure shows T-s and p-v diagrams of our turbojet engine at 80000 rpms:

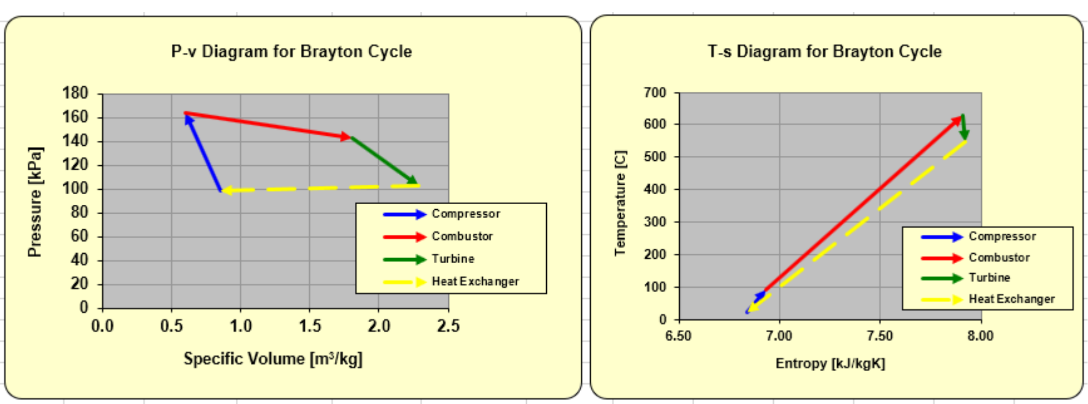


Figure 29: T-s and P-V Diagrams at 80000 rpm

Chapter 5 Conclusion and Recommendation

Conclusions

The design, analysis and manufacturing of a turbojet engine is a very tough but at the same time a very leaning experience. The field of jet engines is evolving by each passing day. We are having breakthroughs in this field every day. But this also means that we have yet to go very far in this field. But despite that, the present knowledge on the jet engines is a very deep one. Each single component of a jet engine is a different field of study. Despite all the theory work, while manufacturing a turbojet engine, empirical data takes precedent. Before designing a jet engine, we had to get answers to a lot of our questions like How does the air behave in a jet engine? What basic thermodynamic principles are followed? How does the fluid behave across the engine? Does a phase change occur? What materials are commonly used while manufacturing a jet engine? What is most feasible fuel? To get answers to these questions, we went through the theory of jet engine and its components as much as we could in the given time frame. Based off that learning, the engine was designed. But what we learnt is that no matter how good your design is, you must go through a lot of difficulties in the manufacturing phase. Is this component available in the market? Do we have the resources available in our country to manufacture our component according to our design? Can we avail those resources? The materials which we are looking for, are they available in the market? Does our budget allow us to use those materials in our product? And many more questions like these arise when you try to manufacture a jet engine.

Besides that, you also must make sure that once you build a jet engine, you have a testing bench available to measure the thrust produced. You have the tools available to read rpm at all time. You can measure the temperatures of your engine at every possible stage. But these foreseen issues remained just foreseen as the project never came to an end because of the unforeseen circumstances. But to conclude, we were able to grace ourselves with the theory behind a jet engine. We learnt about its various configurations. We went into about the fluid and thermodynamics of a jet engine. We were able to manufacture all the components of a jet engine and faced a lot of issue while manufacturing these components and we learnt that how to deal with those issues while keeping in mind the resources available.

Recommendations

The design of a turbojet engine is a time taking and iterative process. In this project, we completed the preliminary design calculations for a scaled down turbojet engine. Our design was based on the available data and we completed our objectives of design.

Whenever a turbojet engine or any other configuration of jet engine is designed, the process has multiple steps which include preliminary design, optimization through simulations, experimentation and testing in wind tunnel and other mediums. Moreover, there are integration programs in which the Gas turbine engine is modified to be integrated efficiently according to the needs of aircraft. The future recommendations for this project are in these phases for the improvement of jet engine.

The first recommendation is that our preliminary design can be taken as the basis for a future project in which this design is to be optimized using simulation techniques. The use of Computational Fluid Dynamics can help to optimize the design of Diffuser Guide Vanes, Turbine Stage and other components. Also, CFD optimization can be used on combustion chamber to improve the fuel vaporization, Flame propagation and flame stabilization.

The second recommendation is that the components design in this project should be tested in wind tunnel and other mediums to find out their actual performance. The preliminary design is based on theoretical mathematical modelling which has several assumptions based on theory. So, by doing experimental analysis and re-evaluating our design parameters and conditions by comparative analysis, performance of jet engine can be increased.

The last and the most important phase in any gas turbine engine program is its integration with a specific aerial vehicle. Mostly, the tycoons of gas turbine engine manufactures design engine for a particular aerial vehicle and all the data regarding that is available. But there are several examples in which a gas turbine engine designed for a specific aircraft was modified and optimized to be integrated in another aircraft. A very common example of this is our pride of nation JF-17 Thunder fighter aircraft. JF-17 Thunder operates Klimov RD-93 Turbofan engine which is a modified form of primary engine Klimov RD-33 of a Russian fighter jet MiG-29.

Our design for scaled model of turbojet engine can be modified according to a specific aerial vehicle in which it is to be integrated. This requires a lot of data regarding that aerial vehicle like how much drag it will offer, what would be cruise speed, what would be max speed and max acceleration, how much restriction it will offer in maneuverability, how it’s motion characteristics like pitch, roll and yaw are defined. These are some of the key parameters that effects the design modification and integration with a particular aerial vehicle. If these phases are introduced further to our preliminary design of turbojet engine it will become a market competitor product in gas turbine engine industry.

Another integration related recommendation or improvement would be that kerosene should be used as the primary fuel. The main advantage of kerosene is that it's Energy Density is much higher than propane. In initial phase of design, we selected kerosene as the primary fuel, but we switched to propane because of the fact that for using kerosene we would have needed a fuel system which would contain a pump. We were not able to get such a pump that could provide low flow rates with non-variable pressure. That's why we used a simpler fuel which had simple fuel delivery system. But, during integration phase of Gas turbine engine with an aerial vehicle, the space or available volume of aerial vehicle is very important, in that scenario kerosene is the more appropriate solution because it will take less space.

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Appendix 1: Bill of Materials

Final Year Project Title		Design and Fabrication of Jet Engine					
Name of Student 1		Umar Ali					
Name of Student 2		Bilal Majeed					
Name of Student 3		Mohsin Manzoor					
Name of Final Year Project Supervisor		Dr. Sami Ur Rehman					
Sr. No.	Brief Description of Part	Vendor's Name	Quantity (numbers)	Dimensions in cm (L x W x H)	Weight (KG)	Specification (if any)	Price (PKR)
1	SS Sheet for Nozzle Guide Vanes	M Sardar	1	7.6 x 26 (L x W)	0.35	14 guage	280
2	Mild Steel Billet for Shaft	Nouman Steel	1	20 (L), φ1.8	1.25	N/A	400
3	Aluminum Billet for Compressor	Ismail Sons	1	10(L), φ2.1	0.85	N/A	150
4	SS Sheet 18 guage	M Sardar	1	8.2 x 22 (L x W)	0.3	18 guage	240
5	Aluminum Billet for Shaft Tunnel	M Sardar	1	12 (L), φ6	3.4	N/A	850
6	Bearings	Khurram Bearing	2	2.2*.8*.7	N/A	ISO 608	300
7	Seamless pipe	Ismail Sons	2	70 (L)	0.1	3mm and 5mm	800
8	Turbine	Ali Express	1	φ6.6	0.19	Height 10mm,Hole 7.59mm	33630
9	Bolts and Nuts	Manzoor Sons	11	N/A	N/A	M4/M6/M8	290
10	Drill Bits	Vaseem Entreprise	5	N/A	N/A	5.9mm/4.9mm/3mm/2mm	490
11	Compressor	Adnan Turbo	1	φ6.8	0.15	t 32mm,Hole 6mm,Inducer 42mm,Exducer	1500
12	S Sheet for flanges and outer coveri	Ismail Sons	1	8 x 28 (L x W)	0.25	N/A	250

Appendix 2: Computer Aided Designs of Turbojet Engine

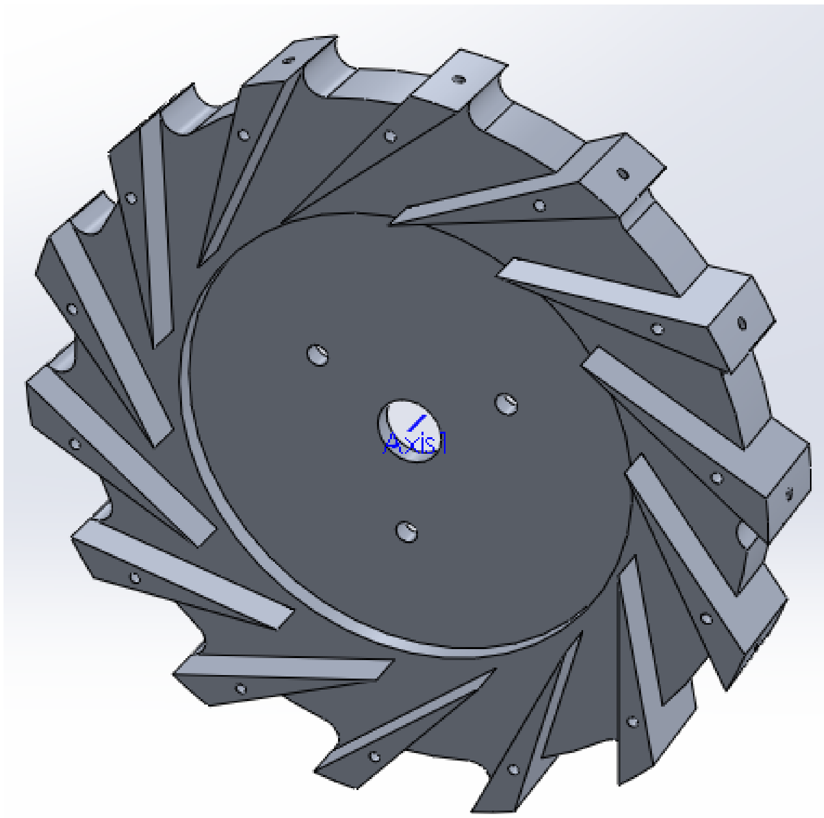

Figure 30: Diffuser Guide Vanes

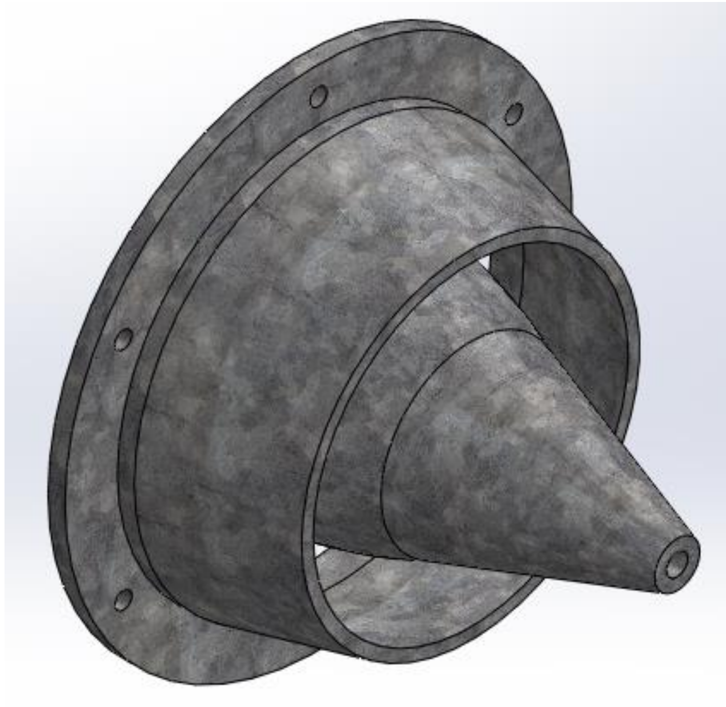

Figure 31: Nozzle

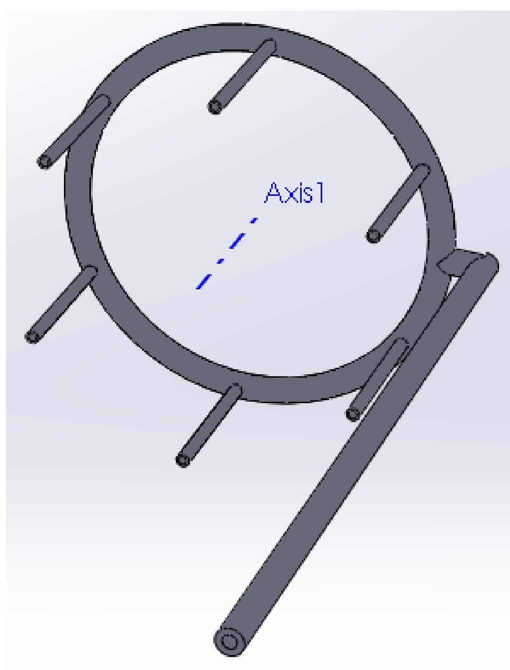

Figure 32: Fuel Line

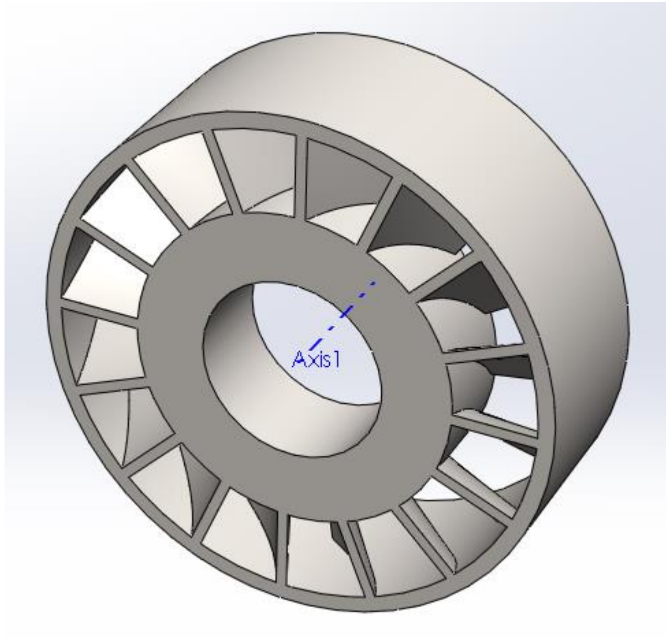

Figure 33: Turbine Nozzle Guide Vanes

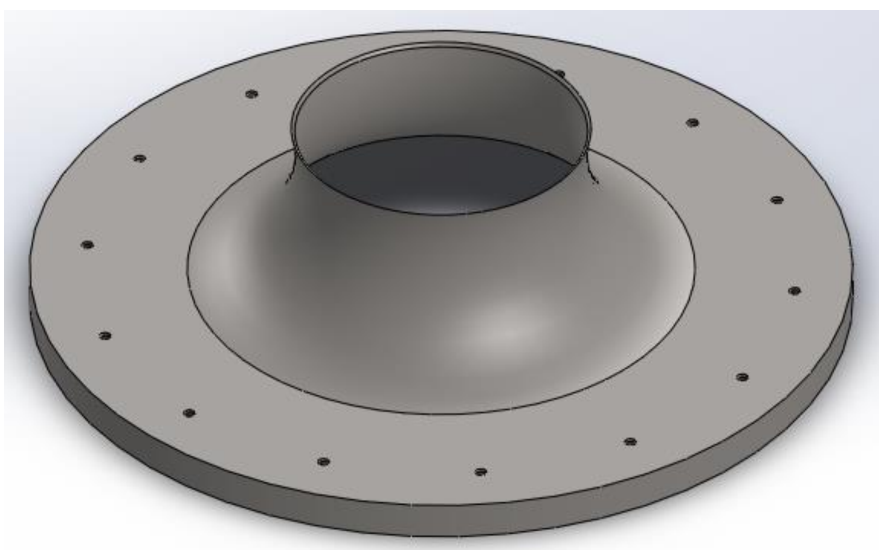

Figure 34: Shroud

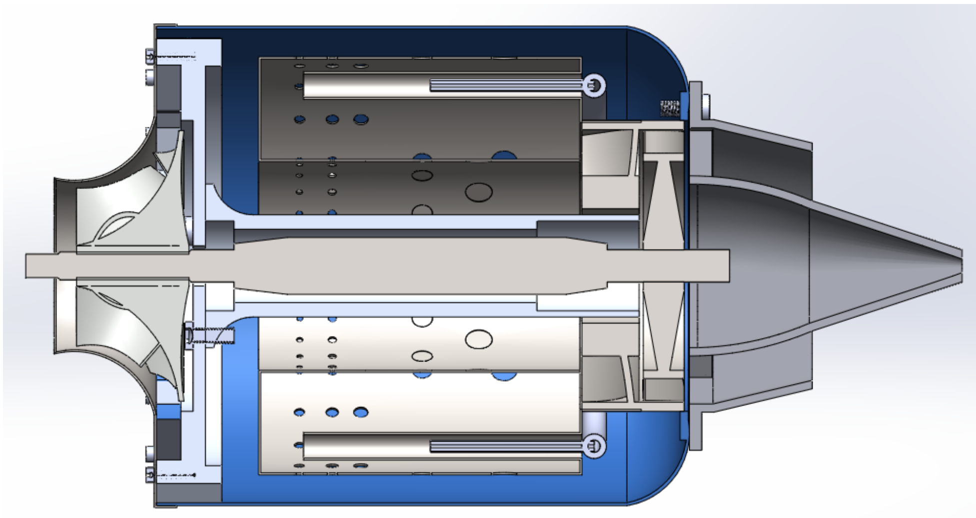

Figure 35: Cross-sectional View of Turbojet Engine

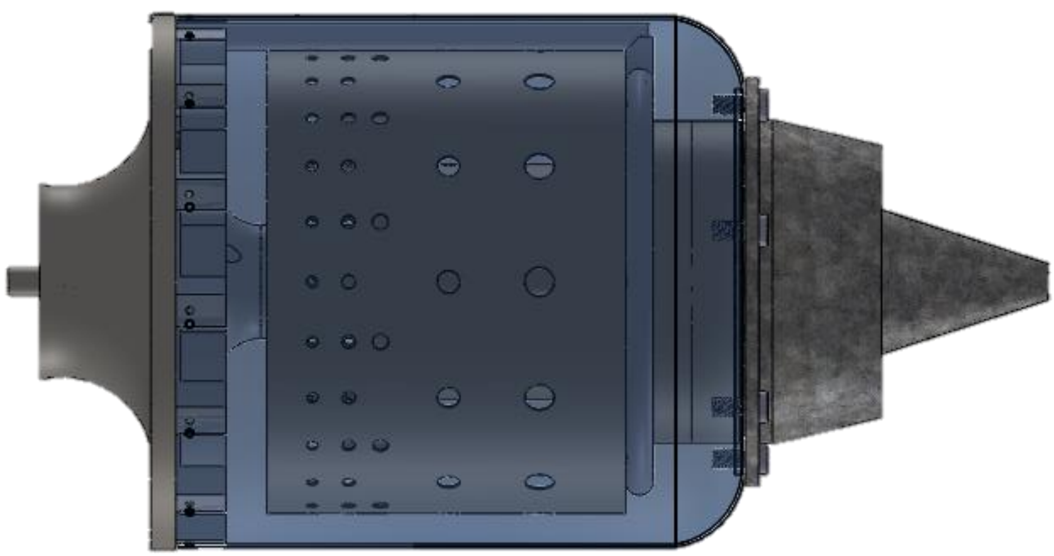

Figure 36: Right View of Turbojet Engine

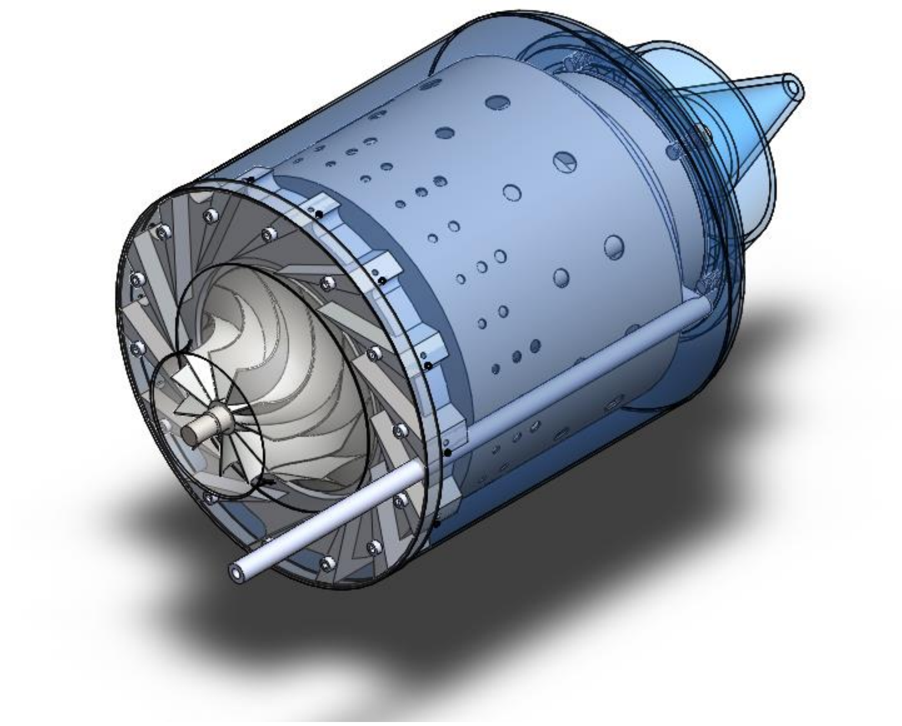

Figure 37: Isometric view of Turbojet Engine

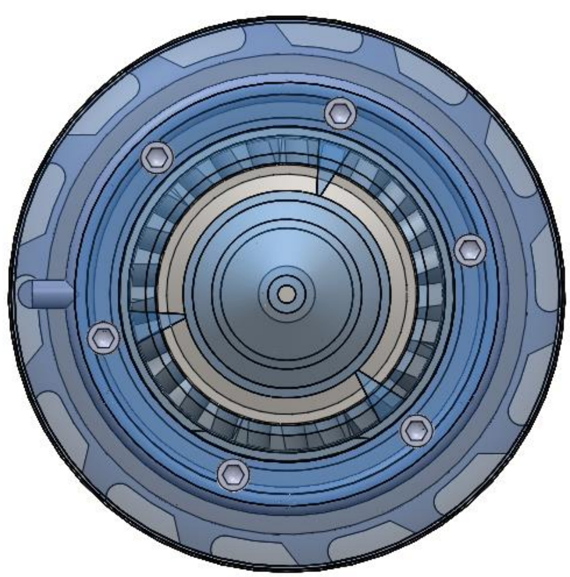

Figure 38: Back View of Turbojet Engine

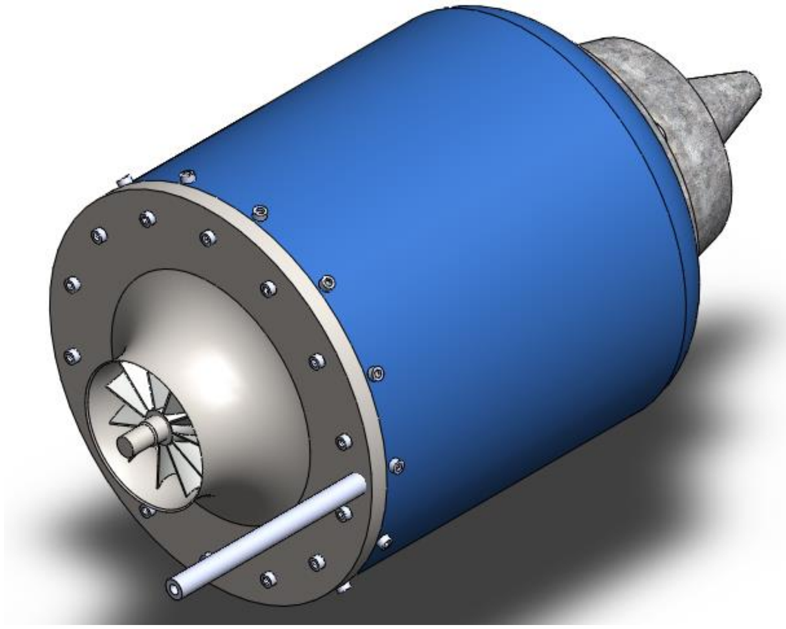

Figure 39: Final Assembly of Turbojet Engine