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Design and Manufacture of a mini-turbojet

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Resumo

O desenvolvimento e produção de pequenos motores a propulsão jato é relativamente recente, tendo em conta que, este tipo de turbina a gás começou a ser estudado e desenvolvido muito antes. No entanto, com a evolução dos tempos, as turbinas a gás foram-se tornando um desafio cada vez mais dificil de as desenvolver e melhorar. Este tipo de motor requere um estudo intenso das várias áreas relacionadas com o seu funcionamento, exigindo cada vez mais conhecimentos e perícia, para que um pequeno detalhe seja melhorado. Apesar de o detalhe poder ser pequeno, o efeito no desempenho geral é considerável.

Até tempos recentes, estes pequenos motores foram desenvolvidos sem um papel significativo na indústria aeronáutica, apenas sendo utilizados para aeromodelos. Contudo, à conta da evolução na ciência, estes motores começam a ser estudados e preparados para integrarem Veículos Aéreos Não Tripulados, UAV, como o seu sistema de propulsão [1].

Este projeto consiste no desenvolvimento de um turbojato, respeitando as dimensões de dois componentes previamente obtidos, o compressor e a turbina, do turbo modelo IHI RHB31 VZ21. Para perceber como se executa o design com o dimensionamento adequado, o estudo de todo componente presente num turbojato foi prosseguido, em paralelo com as áreas fundamentais relativas ao funcionamento do turbojato, por exemplo, ciclos termodinâmicos. No final de um estudo geral do turbojato, o autor prosseguiu para a fase de design, na qual o processo de dimensionamento começa com base na informação contida nas várias fontes de informação, encontradas na bibliografia. O dimensionamento foi realizado pelo o uso de um fator de escala. Este fator de escala foi obtido por uma razão de diâmetros de compressores. Sucintamente, no livro do Sr.Thomas Kamps, o autor aconselha o novato a dividir o tamanho do seu compressor pelo do compressor utilizado para o motor do Sr.Thomas Kamps. A razão de diâmetros, ou fator de escala, foi aplicada nos restantes componentes, produzidos pelo Sr.Thomas Kamps, permitindo chegar às medidas para esta turbina a gás, respeitando o recomendado. As dimensões da cobertura do compressor, flange de entrada, difusor, veio, túnel de acoplamento do veio, câmara de combustão, anel de distribuição do combustível, bocal anterior à turbina com pás guias para o escoamento, bocal dos gases de escape e, por último, o invólucro externo, foram obtidas. O próximo passo foi o processo de design dos componentes referidos, em relação aos designs observados na literatura estudada, utilizando o software de três dimensões CATIA V5R18. O design é um processo empírico, que se torna extremamente difícil de considerar um design como absoluto.

O processo de fabrico do turbojato foi realizado, aquando, o processo de design ter sido concluído. A próxima ação foi obter os materias necessários para a produção das peças, essencialmente, alumínio e aço inox. O alumínio usado foi alumínio fundido, sendo, depois, maquinado para adquirir as formas exigidas relativas ao desgin estabelecido. A maioria dos componentes foram produzidos com folhas de chapa de aço inox, na qual as peças foram cortadas, de acordo com as suas dimensões e forma, em geometria plana. O capítulo que descreve o processo de manufatura, assim como, o processo de design, é explicado para permitir uma futura reprodução do trabalho completado ou adaptação para um conjunto compressor/turbina diferente.

Infelizmente, a fabricação do difusor e cobertura do compressor não foi possível, sendo que

tinham dimensões extremamente pequenas para serem produzidas numa fresadora vertical de comando numérico de cinco eixos. Para além disso, a soldadura aplicada nas peças produzidas não foi executada com a qualidade exigida, mesmo tendo-se aumentado a espessura das peças para facilitar o processo, como foi explicado no capítulo 4.3. Portanto, um dos objetivos não foi atingido devido aos meios insuficientes que impediram a fabricação das partes do motor a jato.

Palavras-chave

Modelo de Turbojato, Mini-turbojet, Ciclos termodinâmicos, Design, Dimensionamento, CATIA V5R18, Processo de manufatura.

Abstract

The development and production of small engines with a jet propulsion system is, relatively recent, taking into account that this type of gas turbine started to be studied and developed many years before the first construction of these small turbojets. However, with the time evolving, the gas turbines turned out to be a greater challenge, becoming more and more difficult to develop and improve. The gas turbine requires an intense study of the several areas related to its functioning, demanding additional knowledge and skill to improve a small detail. Although the detail could be small, the effect on overall performance would be considerable.

Until recent times, these small engines were developed without a significant role in the aviation industry, being only used for model jet engines. Even though, in the account of the science evolution, these engines are being studied and prepared to integrate Unmanned Aerial Vehicles, UAV's, as their propulsion system [1].

This dissertation consists on the development of a turbojet, in a small-scale, respecting the dimensions of the two, previously obtained, components, the compressor, and turbine, from the model turbo IHI RHB31 VZ21. To understand how to execute a design with suitable dimensions, the study of every present component in a turbojet was carried out, in parallel with the fundamental areas, regarding the functioning of a turbojet, such as thermodynamic cycles. At the end of a general study of the turbojet, the author proceeded to the design phase, in which the dimensioning process starts based on the information contained in the various sources of information, found at the bibliography. The dimensioning was carried through by the use of a scale factor. This scale factor was obtained by the compressor's diameters ratio. In brief, in the Mr.Thomas Kamps' book, the author advises the novice to divide his compressor diameter by the compressor used for Mr.Kamps's engine. The diameters ratio, or the scale factor, was applied to the remaining components, produced by Mr.Thomas Kamps,in order to attain the measures for this gas turbine, respecting the recommended. The dimensions of the compressor shroud, inlet flange, diffuser, shaft, shaft housing, combustion chamber, fuel distribution ring, nozzle guide vanes, exhaust nozzle and, the last, outer casing were obtained. The next step was the design process of the referred components, in regard to the observed designs, found in the studied literature, using the three-dimensional design software CATIA V5R18. The design is an empiric process, which reveals itself as extremely difficult to consider one design as absolute.

The manufacturing process of the turbojet was executed, at the time, the design process had been concluded. The following action was to acquire the necessary material for the production of the pieces, essentially, aluminum and stainless steel. The aluminum used was cast aluminum, being, then, worked to acquire the requested shape concerning the established design. The majority of the components were manufactured with stainless steel sheets, in which, the pieces were cut, according to their dimensions and shape, in-plane geometry. The chapter describing the manufacture process, as well as, the design process, is explained to allow a future reproduction of the work completed or adaptation for a different compressor/turbine set.

Unfortunately, the fabrication of the diffuser and compressor shroud was not possible, since it had extremely small dimensions to be produced in the 5-axis vertical machining center. Moreover, the welding applied to the manufactured pieces was not executed with the required qual-

ity, even having increased the material thickness to facilitate the process, as it is explained in chapter 4.3. Therefore, one of the main objectives was not accomplished due to the insufficient means that disabled the manufacture of the jet engine's parts.

Keywords

Model Jet Engine, Mini-Turbojet, Thermodynamic cycles, Design, Dimensioning, CATIA V5R18, Manufacturing Process.

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Nomenclature

A	Cross-section area	m^2
a	Speed of Sound	m/s
c	Specific Heat	kJ/kgK
$\dot{E}$	Total Power	W
F	Thrust	N
$F/\dot{m}$	Specific Thrust	Ns/kg
FHV	Fuel Heating Value	J/kg
f	Fuel to Air Ratio	
h	Specific Enthalpy	J/kgK
ke	Kinetic Energy	J
M	Mach Number	
m	Mass	kg
$\dot{m}$	Mass Flow Rate	kg/s
pe	Potential Energy	J
P	Pressure	Pa
Q	Heat Rate	W
R	Real Gas Constant	kJ/kgK
S	Specific Fuel Consumption	mg/Ns
T	Temperature	K
V	Velocity	m/s
W	Work Rate	W
η_p	Propulsive Efficiency	
γ	Heat Capacity Ratio	
π	Pressure Ratio	
ρ	Density	kg/m^3
τ	Temperature Ratio	
τ_λ	Maximum Stagnation Enthalpy Ratio	
θ	Flowing Fluid Energy	J

Subscripts

a	Air
b	Burner
c	Compressor
cc	Combustion Chamber
f	Fuel
in	Input
m	Mechanical
o	Overall
out	Output
p	Constant Pressure
r	Free stream conditions
t	Turbine
th	Thermal
v	Constant volume
0	Total
1	Initial state
2	Final state

List of Acronyms

CAD	Computer Aided Design
CATIA	Computer Aided Three-Dimensional Interactive Application
CC	Combustion Chamber
CNC	Numerical Control
FABLAB	Fabrication Laboratory
NGV	Nozzle Guide Vanes
RC	Radio controlled
TSFC	Thrust Specific Fuel Consumption
UAV	Unmanned Aerial Vehicle
UBI	University of Beira Interior

Chapter 1

Motivation and Objectives

1.1 Motivation

The dream of a human to fly was one that intrigued us and, was in our thoughts, for a prolonged time. We started from the observation of birds flying, to the invention of a kite, the ornithopter of Leonardo da Vinci, the first hot air balloon, also known as the Montgolfier's Balloon, the glider, the aerodrome and, finally, the Wright Brothers Glider [2].

From the study to find sustainable wings, for the generation of lift, studies of propulsion systems were also in course, because, we understood to have a sustainable flight, we had to have a source of power. Steam engines were the first to be created, then the internal combustion engine followed by the gas turbine. Finally, we had our dream become true when the Wright Brothers made the first flight.

We started to develop new aircraft's wings, materials, structure and propulsion systems. They were all subjected to intense study and investigation until we reached the modern aircrafts, and, still, we do not stop investigating for further improvements. Developing the modern aircrafts, allowed us to cross continents and interact with different cultures. Moreover, to be possible to reach another continent in hours, the aircraft has to be extremely well designed, built and equipped. The aircraft's engines provide a serious amount of traction, enabling the aircraft to reach the necessary speed for the operation in quest. It is according to the purpose of the aircraft, for example, military long-range operations, commercial flights or combat situations, that they are developed and improved to the smallest detail. For instance, in combat situations, the aircrafts are built to fly at an astonishing speed that goes beyond the speed of sound, 343 m/s [3]. The gas turbine engines, with the technological and scientific progress, are no longer out of reach for someone that is interested and wants to build one. Thankfully, nowadays, there are mini-turbojet engines that one can build and improve.

The hand fitting jet engines appeared recently and are known as mini-turbojets or model jet engines. These small machines started to have their place in the industry, where they are being developed and produced by companies for a possible application in UAV's [1]. These gas turbines can be applied to a Radio Controlled (RC) model jet engine.

The challenge to build a mini-turbojet is itself appealing for someone interested in propulsion systems, as well as, knowing the possibility to contribute for further works, lead to embrace it. Therefore, through the study of gas turbine design and manufacture, a mini-turbojet prototype was designed and some parts were built and effortfully, very soon the remained will be fabricated and the final engine will be put through for experimental analysis.

1.2 Objectives

The primary objective of this thesis is concentrated on the design and construction of a small dimension turbojet. Initially, a three-dimensional design of the small jet engine was conducted in a Computer Assisted Design (CAD) software, in particular, CATIA. The design was carried out according to the dimensional values achieved throughout the study of small-scale jet engines design.

The dimensioning criteria found at the available literature was adopted by the author. This method relies on empirical data, specifically, developed designs by this time. This was a major aid, in order to obtain the adequate measures for the major components, giving the means to dimension the remaining pieces. The dimensioning approach is duly explained in the chapter 3.

The last and main goal of this work, is to describe the series of steps taken to produce a small model jet engine, for practical applications, in a transparent way, to help the reader to produce his small-scale jet engine.

1.3 Document Structure

The whole document is organized in five chapters, where each chapter focuses on several parts, in agreement with the custom format of the master's degree thesis.

The first, and current, chapter, manifests the author's motivation that preceded the development of this thesis. Additionally, the objectives and document organization are presented in a logical way, for easier comprehension of what is proposed for this thesis.

The second chapter presents a literature review, introducing the gas turbine, describing its types, in particular, the turbojet. Followed by a general description of the components that constitute the jet engine, alongside with the thermodynamic concepts that allow us to understand the jet engine overall functioning.

The third chapter, reports how the dimensioning process was effectuated, justifying the choices for the components' measures.

The fourth chapter sets up the procedures taken to design and manufacture the required components, completed by its assembly. Along with further modifications made throughout the process.

The fifth and last chapter, discloses the conclusions drawn from the practical procedure, the obstacles encountered along with the development of this thesis and future works suggestions.

Chapter 2

Bibliographic Review

2.1 Gas Turbines Historical Review

A first concept came up at the times of Roman-Egypt, created by Hero, or Hero of Alexandria. The aeolipile, the name of Hero's invention, is a radial steam turbine, which combines two nozzles, at opposites sides, where vapor water leaves due to the steam formed by boiling the water inside a sphere, causing the center of the sphere to spin, generating torque. An example of the steam engine is shown in figure 2.1.

Figure 2.1: *Aeolipile* [5]

The physical principle reaction was put in to practice in the thirteenth century by the Chinese people using fireworks [6]. After three centuries, 1687, Sir Isaac Newton made a crucial advance formulating the three laws of motion:

1. Inertia: An object, in a straight line, will remain in uniform motion unless an external force is applied to the object, changing its state [7].
2. $Force = m \times g$, the variation in velocity, g, depends on the mass, m, of the object, when an external force, F, is applied [7].
3. Action-Reaction Law: for example, a stone exerts a force on the earth as the earth applies an equal force to the stone [7].

These laws allowed us to take, within time, important steps towards the gas turbines. The first one was taken in 1791, by John Barber, an Englishman, that was granted a patent for the gas turbine thermodynamic cycle, known as the Brayton cycle, the same cycle of the actual gas turbines. Utilizing this cycle, Hans Holzwarth developed the electrical ignition of the mixture, in the combustion chamber, with controlled valves in 1908 [6].

In 1913, an engine using the jet propulsion system was patented by René Lorin. The first subsonic ramjet, although, it was not possible to concretize the project due to the quality of the material at that time. The materials could not resist the heat, as well as, the evolution of the jet propulsion system was on its first days, rebounding to the aircraft's efficiencies [8].

Jet propulsion engines were achieved in 1930 when Sir Frank Whittle patented the design of a centrifugal gas turbine for jet propulsion. Later on, in 1937, he made the first static test of the jet engine history. Despite Whittle made the first static test, it was Hans Joachin Pabst von Ohain, working for Heinkel aircraft company, that created a turbojet engine run by gaseous hydrogen, similar to Whittle's design, that was used as the propulsion engine for the aircraft He-178, realizing the first turbojet flight worldwide, in 1939. Three years later, Frank Whittle's engine was used for the first time as the propulsor of an aircraft [6].

The first axial-flow turbojet flight was in Germany, in the same year as Whittle's engine flight occurred. The axial turbojet, Jumo 004A, was the propulsion system of the aircraft Me-262. The leader of this project, chosen by Junkers company, was Anselm Franz [6]. Despite Frank Whittle's engine could not be built so quickly, he founded the basis of the modern gas turbine [8].

The jet propulsion system was and still is, studied extensively to search for improvements, with a future successful application in this engine type. Only years after, the turbojet engine was applied to an aircraft, the idea of reproducing the same engine in a small-scale started to appear. The miniature-turbojet history is difficult to date, however, it is assumed that it was started by Kurt Schreckling, German technician and amateur astronomer. Kurt was the first to replicate a turbojet in small-scale, opening doors for the small or miniature model jet engines [9].Gas Turbine Engines for Model Aircraft, the book from his authorship, explains how he built the engine, the FD 3/64, which created a starting point for miniature turbojets in the future. This enabled others to improve Kurt's turbojet, as well as, developing new small-scale gas turbines based on his engine, such as the KJ66 [10].

2.1.1 Types of Gas Turbines

Gas turbine history records show us the enormous and fast development of this engine. It is a product of a mixture of various areas like thermodynamics, mechanics, aerodynamics and other areas, which are still being studied to the fullest extent for improvements. Only after understanding these fields, that the utility of a gas turbine is thought, and, then, designed, dependent on whether it is used to, a space mission, aviation transport or for air combat situations. Teams of scientists, engineers and technicians created gas turbines with different ways of converting and supplying power, according to their purposes, such as jet propulsion engines: rocket, athodyd, also known as a ramjet, the pulse jet, and the turbo-jet, or, propeller jet engines: turboprop engine, turbofan, and turboshaft [8].

The ramjet, figure 2.2, is formed by a divergent inlet and a convergent or convergent-divergent exhaust. This engine requires forward motion to produce thrust. With no rotating parts, the air is forced to the divergent duct, followed by the combustion with fuel, where the gases will accelerate through the exhaust section to the atmosphere [8].

Another jet propulsion engine relative, the pulse jet, figure 2.3, uses a similar duct to the ramjet, but more robust because of the higher pressures involved. The air goes through open valves at the inlet, passes to the combustion chamber, where the combustion of fuel is realized, causing the gas to expand,thus, increasing the pressure. As a consequence of the rise, the valves close and the gas is ejected through the rear. Its high fuel consumption and unequal performance compared to the actual gas turbine make this engine inadequate for use in aircrafts [8].

The rocket engine, figure 2.4, is distinguished from the other engines by not utilizing the oxygen from atmospheric air for the combustion but, instead, using a specific fuel, decomposed chemically with oxygen [8].

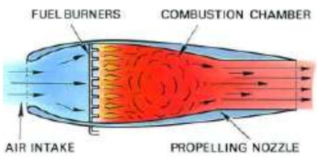

Figure 2.2: Ramjet [8]

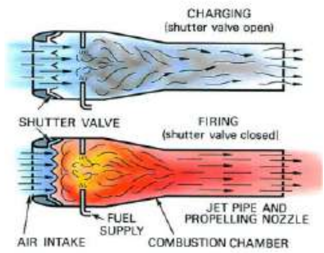

Figure 2.3: Pulsejet [8]

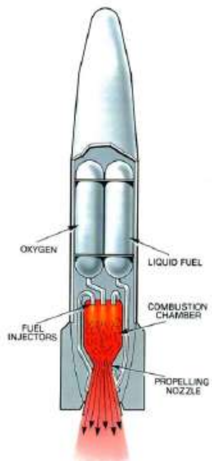

Figure 2.4: Rocket Engine [8]

The turbojet, shown in figure 2.5, is the junction of a compressor, combustion chamber, and turbine, called the gas generator, with an inlet and exhaust nozzle. The added exhaust nozzle will convert most of the energy of the airflow into velocity.

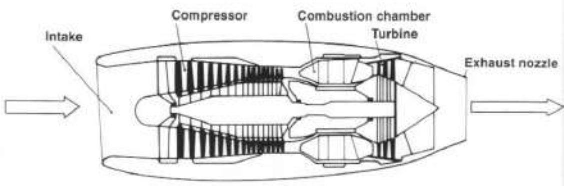

Figure 2.5: Turbojet [6]

With the propeller/turbine combination comes the ducted fan, prop fans and bypass engines. In the turboprop engine, shown in figure 2.6, the two turbines' functionalities are to sustain the compressor work demand and make the propeller run. In a similar engine, the turboshaft, shown in figure 2.7, the turbine drives the compressor and the second turbine will drive the shaft, which in turn, is connected to a transmission system that rotates the helicopter blades [6].

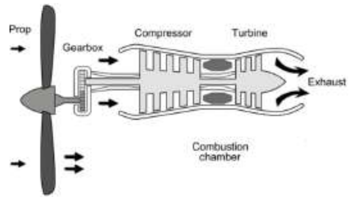

Figure 2.6: Turbopropeller [11]

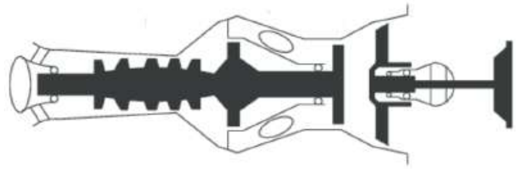

Figure 2.7: Turboshaft [12]

The turbofan, illustrated in figure 2.8, has also two turbines, in which, one, absorbs the energy from the hot airflow and, the other, utilizes the excess shaft work to drive a low-pressure compressor, a fan. It has a lower propulsive efficiency in comparison to the turboprop, when they are being operated at the same cruise speed and lower velocities. Nevertheless, at higher velocities the turbofan has advantage. Engines like the turbofan started to be and still are, widely used due to high propulsive efficiency values when compared to a turbojet. These values are explained due to the bypassed airflow [6].

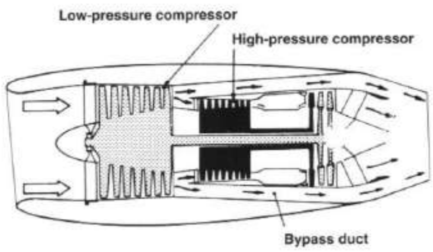

Figure 2.8: Turbofan [6]

Bypassed airflow's engines are distinguished in two types, low bypass-ratio and high-bypass ratio, in which bypass-ratio stands for the amount of air being bypassed in relation to the air going through the engine's core. The engine is constituted with a high and low-pressure compressor and the matching turbines, that are driven by two coaxial shafts. The air is sucked by the low-pressure compressor, the fan, that will divide the air into two flows. Most of the air is ducted through the sides of the engine's core and a small part goes for combustion, being, then, the two airflows joined at the exhaust section. This means a lesser fuel consumption than prior engines of similar thrust without this technology, allowing the engine to perform efficiently at high altitude flights. The dominant use of these technology, in particular, high-bypass ratio engines, in the propulsion systems of civil aviation and long-range military missions are justified for the low fuel consumption, considered the most important performance parameter [6]. An example of an high-bypassed engine is demonstrated in figure 2.9.

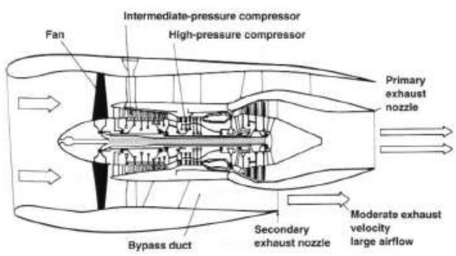

Figure 2.9: High-Bypassed Engine, Rolls Royce RB.211 [6]

The turbojet is a simpler turbofan, which means, does not have a bypassed airflow. Resulting in lower efficiency, although, compensates with speed. These engines can reach supersonic speeds being one of the reasons for their usage in military aircraft [6].

2.2 Cycle Review

The literature reviewed possesses dense, vast information about thermodynamics cycles along with its properties. Therefore only a summary will be presented to avoid an extensive and deep analysis. For more detailed information about thermodynamic cycles, and the corresponding gas turbines, the author recommends the reader to visit the following books, where more information can be found in the books *Thermodynamics: An Engineering Approach by Çengel, Yunus A. and Boles, Michael A.* and *Aerothermodynamics of Gas Turbines and Rocket Propulsion by Oates, George C.*, chapter 5.3, 7.4, 9.8 and chapter 2, respectively. Nonetheless, other literature was not disregarded.

In summary, it will be presented the overall functioning of the gas turbine, succeeded by the ideal thermodynamic cycle along with the ideal behavior analysis of the turbojet components. Afterward, jet performance equations are demonstrated to fulfill the analysis of a turbojet engine cycle.

2.2.1 Working Cycle

The working cycle subchapter aims to explain the functioning of an engine with a jet propulsive system, describing its thermodynamic cycle along with properties related to the latter.

The working cycle starts from the intake of air into the compressor, whose job is to increase the pressure via mechanical shaft power [6]. The pressure increase has a diminishable effect on the volume of the airflow, subsequently, raising the temperature of the air [8]. The pressurized air is discharged to the combustion chamber, where fuel is added and burnt, raising the temperature to extremely high values. While the gas is burned, the volume, as the temperature, increases due to the open structure of the combustion chamber, maintaining the pressure constant [8]. The combustion process raises the energy state of the molecules to high levels, enabling the necessary amount of energy to be effectively explored by the turbine [6]. As an effect of the work extracted from the gas, the turbine starts to rotate, converting the gas energy surplus to mechanical power by generating motion. This spinning motion will force the compressor wheel to rotate due to the work provided from the turbine, which is transferred by the rotation of the shaft to the compressor, on the other end of the spool[6]. At this stage, the gas variables, pressure, and temperature decrease, whereas, the volume increases. Finally, the gas flow reaches the final stage, at the exhaust nozzle, where the gas is ejected to the ambient at high velocities, producing thrust [8].

There are three main conditions in the engine cycle to retain:

- In the compression, there is an increase in pressure with a consequent decrease of volume and temperature rise [8].
- In the combustion, an increase in temperature occurs, while the pressure remains constant and the volume increases [8].

- In the expansion, the volume rises along with a decrease in pressure and temperature [8].

The working cycle of a gas turbine is, generally, compared to the working cycle of a four-stroke piston engine taking into account the four similar stages of each engine. Both engines initiate their cycle by the induction phase, followed by compression, combustion, and expansion. Nevertheless, all the stages of the piston engine are performed inside a cylinder, while in a jet engine, each component is assigned the corresponding function, resulting in a continuous action, instead of intermittent. Moreover, the combustion in a jet engine occurs at constant pressure, as opposed to the reciprocating engine, where the combustion process takes place in a closed space. Therefore, the jet engine is capable of operating larges masses of air with lightweight components. At the final stage, the exhaust phase, the gases, expanded by the turbine, leave the nozzle, generating a propulsive force essential to this engine. In contrast, the exhaust gases of the piston engine do not have the same significant effects [6]. Considering the differences stated, the gas turbine engine removes three idle strokes, consequently, more fuel is able to be burnt in a shorter period. Since the turbo-jet engine is a heat engine, with more fuel burnt, the higher the temperature of the combustion chamber is, hence, a substantial expansion of the gases occurs. Furthermore, a greater amount of power is produced for a given size [8]. In order for a piston engine to generate the same amount of power, it would have to be extremely large, heavy and the manufacture would be a serious challenge [6].

As the gas is discharged from the nozzle, it will be, progressively, dispersed to the ambient conditions, reverting to its original state. The gas throwback to its original condition implies that the state variables also revert to their original conditions, termed as a reversible process. A process to be considered reversible, has to combine the internal and external reversibility. If the gas states could be restored in a reverse sequence, while a system is submitted to a process in which the pressure and temperature gradients are minor, the process is defined as internally reversible. Meanwhile, for a process to be considered externally reversible, the atmospheric changes that go along with the process can be reversed in sequence. However, the reversible process is impossible to achieve due to the irreversible factors, for example, temperature, pressure and velocity gradients triggered by heat transfer, friction, chemical reaction and work applied to the system. Despite the irreversibility of the real processes, the reversible process is standardized to estimate the success of real processes taking into account the losses, as well as, to enable the thermodynamic relations to be derived to estimate reality [13].

2.2.2 Thermodynamic Cycle

2.2.2.1 Theoretical Notions

It is further presented notions to be acknowledged, in order to understand the conditions and properties behind the ideal thermodynamic cycle that comprehends all the gas turbines, in particular, the turbojet engine.

- Steady-Flow Process

  The gas turbines are built, generally, for continuous operation, in which there is an approximation of the conditions they operate on. Assuming it performs under the same conditions as the time passes, the process is termed as a steady-flow process. It means the fluid properties remain the same throughout the whole process. Flowing in a control volume, the fluid properties can alter from different fixed points but stay the same, from the start to the end. As a result, the mass, m, volume, V, and total energy rate or total power, E, are constant throughout this process [14].

The conservation of mass principle is applied, stating, considering a control volume, the total rate of mass entering equalizes the total rate of mass leaving it [14]. Since there is no increase or reduction of mass, the mass flow rate, $\dot{m}$, is equal from the beginning to the end of the process [14]. It is expressed in the form of,

$$\sum_{in} \dot{m} = \sum_{out} \dot{m} \quad [14] \quad (2.1)$$

Considering it for a uniform single stream, denoting the inlet and exit states, 1 and 2, respectively, the mass balance becomes,

$$\dot{m}_1 = \dot{m}_2 \rightarrow \rho_1 V_1 A_1 = \rho_2 V_2 A_2 \quad [14] \quad (2.2)$$

Where ρ, V and A, represent density, flow velocity, and cross-sectional area.

In the context of the total energy rate, the energy remains the same within a control volume, indicating no changes in the total power. This simplifies the energy balance to [14],

$$\dot{E}_{in} = \dot{E}_{out} \quad [14] \quad (2.3)$$

Remembering that the energy transfers occurs in the form of mass, $\dot{m}$, work, W and heat, Q, the energy balance is represented as,

$$Q_{in} + W_{in} + \sum_{in} \dot{m}\theta = Q_{out} + W_{out} + \sum_{out} \dot{m}\theta \quad [14] \quad (2.4)$$

Where the energy of a flowing fluid, θ, is described as,

$$\theta = h + ke + pe \quad [14] \quad (2.5)$$

Where h, ke and pe are defined as enthalpy or internal energy, kinetic energy and potential energy.

Heat and work interaction is defined by, a heat transfer into the system, heat input, and the work produced by the system, work output [14]. Considering there are no changes in kinetic and potential energy, the energy balance is expressed in way,

$$Q - W = h_2 - h_1 \quad [14] \quad (2.6)$$

- Stagnation Properties

In control volumes analysis, it is usual to put together the internal energy and the fluid energy to form one variable already referred, specific enthalpy, h. For most of the cases, the kinetic and potential energy are disregarded, defining the enthalpy as the total energy of the fluid. However, when the kinetic energy is not neglected, generally, it is appropriate to convert the kinetic energy into enthalpy of the fluid, combining them into one term defined as stagnation or total specific enthalpy, shown in equation 2.7 [14].

$$h_o = h + \frac{V^2}{2} \quad [14] \quad (2.7)$$

In equation 2.7, the enthalpy is distinguished by two types, the static and stagnation enthalpy, h and h_0, respectively.

As a result of the kinetic energy conversion to enthalpy, the temperature and pressure increase. These fluid properties are recognized as stagnation properties or isentropic stagnation properties. Enthalpy and the stagnation temperature of an isentropic stagnation state, and actual, are the same, given the fluid is an ideal gas. The actual stagnation pressure differs from the isentropic stagnation pressure because the entropy raises due to fluid friction [14]. Assuming the fluid as an ideal gas, the enthalpy can be substituted by the constant specific heat times the temperature, shown in equation 2.9.

$$c_p T_o = c_p T + \frac{V^2}{2} \quad [14] \quad (2.8)$$

Becoming,

$$T_o = T + \frac{V^2}{2c_p} \quad [14] \quad (2.9)$$

Where T_0 indicates the stagnation or total temperature, in other words, the temperature that the ideal gas reaches when it is brought to rest in an adiabatic process, and, the term $\frac{V^2}{2c_p}$, represent the temperature increase throughout the process named as dynamic temperature [14].

The relation between the temperature and pressure is demonstrated at equation 2.10.

$$\frac{P_0}{P} = \frac{T_0}{T}^{\frac{\gamma}{\gamma-1}} \quad [14] \quad (2.10)$$

Where, P_0 and γ, are termed as stagnation pressure and specific heat ratio.

2.2.2.2 Brayton Cycle

It is a thermodynamic cycle, idealized, present in all gas turbines equipped with the fundamental components, such as the compressor, combustion chamber and turbine [15]. The cycle is divided into two types: open and closed cycle. The former cycle, consists of air, at atmospheric conditions, pulled to the compressor that raises the temperature and pressure of the air. The pressurized air will follow to the combustion chamber, where it is mixed with fuel, succeeded by combustion, at constant pressure. When the combustion process has been finalized, the gas exits to the turbine, at extreme temperatures, where the expansion of the gas occurs [14]. At this phase, the interaction of the gas with the turbine is used to drive the compressor. The remaining work of the gas is taken to accelerate the fluid ducted by the exhaust nozzle to the exterior [16]. For the reason of the gas being expanded to the exterior, the cycle is classified as an open cycle. On the contrary, if the gas had been recirculated, the cycle would be considered to be closed. The figure 2.10 and 2.11, describe an open and closed cycle [14].

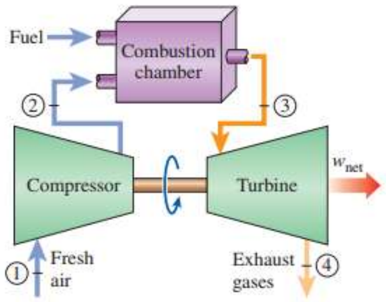

Figure 2.10: Open cycle [14]

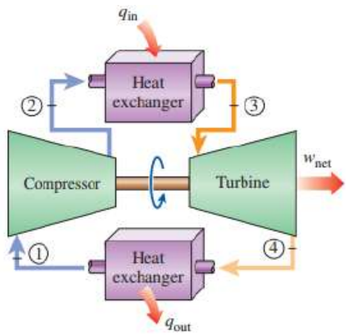

Figure 2.11: Closed cycle [14]

Noticing in the closed cycle, the combustion process was substituted by a constant pressure additional heat, accompanied by the replacement of the exhaust process for a heat rejection process, at constant pressure, to the exterior [14].

The turbojet engine working cycle corresponds to an open Brayton cycle, which is the usual type of cycle for the gas turbines [14]. Figure 2.12 illustrates the components of a jet propulsion device with the corresponding Brayton cycle.

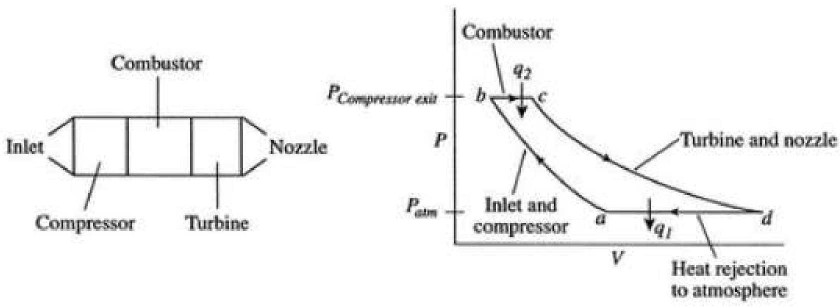

Figure 2.12: Jet Engine Components and analogous thermodynamic states [16]

The thermodynamic states observed in figure 2.12, are distinguished in four processes. The legend of this cycle is formulated in the following manner:

- a-b: Compression at the inlet and compressor, in an isentropic process [16].
- b-c: Combustion of fuel at constant pressure [16].
- c-d: In the expansion, the volume rises along with a decrease in pressure and temperature [16].
- d-a: Air cooling at constant pressure [16].

2.2.2.3 Cycle Behavior Analysis

Cycle analysis is a process to obtain estimates for performance parameters as thrust or specific fuel consumption, calculated after assuming some conditions and design specifications, presented below [17].

Conditions

- Working fluid is considered an ideal gas with constant heat capacity and specific heat ratio [17].
- Isentropic Compression/ Expansion [17].
- The external source of heat for combustion and fuel mass is disregarded [17].

Design

- Atmospheric pressure and temperature values [17].
- Compression ratio [17].
- Inlet Mach number [17].

In this subsection, the ideal and actual behavior of the components are presented, indicating the temperature and pressure for each station. The stations will be distinguished by a number for easier referencing, as demonstrated in figure 2.13. The actual turbojet cycle analysis is presented in Appendix D.

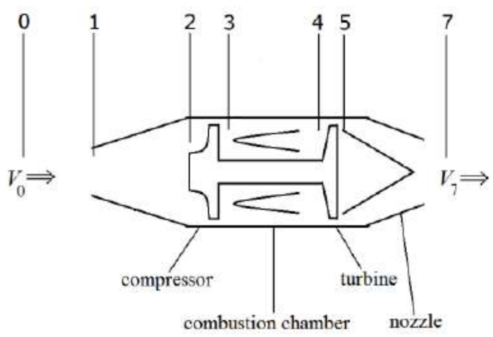

Figure 2.13: Turbojet station numbering [16]

Stations descripitions

- 0 : Free stream
- 1 : Inlet Entry
- 2 : Compressor Entry
- 3 : Compressor Exit
- 4 : Turbine Entry
- 5 : Turbine Exit
- 6 : Nozzle Entry
- 7 : Nozzle Exit

Ideal Behavior Component Analysis

- Inlet(i)

For the ideal case, when the flow goes across the inlet, it is considered to be isentropic. This will make the induction, a process with the enthalpy constant. The ratios resuming the inlet ideal behavior are presented in equation 2.11 and equation 2.12, [19].

$$\pi_i = \frac{\text{Pressure at Inlet Entry}}{\text{Free stream Air Pressure}} = \frac{P_{02}}{P_{0r}} = 1 \quad [19] \quad (2.11)$$

$$\tau_i = \frac{\text{Temperature at Inlet Entry}}{\text{Free Stream Air Temperature}} = \frac{T_{02}}{T_{0r}} = 1 \quad [19] \quad (2.12)$$

- Compressor (c)

At this phase, the compressor will add energy to the flow by the form of work and, consequently, increase the temperature and pressure of it. The equation 2.13 expressing the compressor work is demonstrated below [20].

$$W_c = \dot{m}_a c_p (T_{03} - T_{02}) \quad [20] \quad (2.13)$$

The temperature at the compressor exit can be obtained using the temperature and pressure relation found in equation 2.14.

$$\tau_c = \pi_c^{\frac{\gamma_c - 1}{\gamma_c}} \quad [20] \quad (2.14)$$

Being the pressure ratio, equation 2.15 and temperature ratio, equation 2.16

$$\pi_c = \frac{\text{Pressure at Compressor Exit}}{\text{Pressure at Compressor Entry}} = \frac{P_{03}}{P_{02}} \quad [20] \quad (2.15)$$

$$\tau_c = \frac{\text{Temperature at Compressor Exit}}{\text{Temperature at Compressor Entry}} = \frac{T_{03}}{T_{02}} \quad [20] \quad (2.16)$$

- Burner/Combustor (b)

Previously explained in the working cycle, the combustion process is executed at constant pressure, which in the ideal case, the pressure ratio, equation 2.17, and temperature ratio, equation 2.18 are,

$$\pi_b = \frac{\text{Pressure at Turbine Entry}}{\text{Pressure at Compressor Exit}} = \frac{P_{04}}{P_{03}} \quad [21] \quad (2.17)$$

$$\tau_b = \frac{\text{Temperature at Turbine Entry}}{\text{Temperature at Compressor Exit}} = \frac{T_{04}}{T_{03}} \quad [21] \quad (2.18)$$

Reminding this is an adiabatic and constant pressure process, with complete combustion, hence, the combustion efficiency, η_b, is equal to one. In addition, the enthalpy is the product of the specific heat coefficient at constant pressure and temperature [21]. Therefore, the energy equation 2.19 is,

$$(1+f)c_{pt}T_{04} = c_{pc}T_{03} + fFHV \quad [21] \quad (2.19)$$

Where FHV and f refer to the fuel heating value and fuel to air ratio. Utilizing equation 2.19, it can be related to the temperature ratio, equation 2.20.

$$\tau_b = \frac{T_{04}}{T_{03}} = \frac{1 + fFHV/c_pT_{03}}{1+f} \quad [21] \quad (2.20)$$

Where,

$$f = \frac{\dot{m}_f}{\dot{m}_a} \quad [22] \quad (2.21)$$

- Turbine (t)

The heated flow goes through the turbine is subjected to pressure and temperature decrease. The turbine as it was explained before, exploits the energized flow realizing work. In equation 2.22 , it is illustrated the turbine work per mass of the airflow.

$$\Delta W_m = c_{pt}T_{04}(1-\tau_t) \quad [17] \quad (2.22)$$

In the account of the cycle be ideal, the expansion stage is considered to be an isentropic process . Thus, the turbine does not have any losses, making complete use of the energy surplus due to the combustion stage, resulting in an efficiency value equal to one [17]. The relation between the temperature and pressure ratio of the turbine can be described as shown in equation 2.23.

$$\tau_t = \pi_t^{\frac{\gamma_t}{\gamma_t - 1}} \quad [23] \quad (2.23)$$

Where,

$$\pi_t = \frac{\text{Pressure at Turbine Exit}}{\text{Pressure at Turbine Entry}} = \frac{P_{05}}{P_{04}} \quad [23] \quad (2.24)$$

And,

$$\tau_t = \frac{\text{Temperature at Turbine Exit}}{\text{Temperature at Turbine Entry}} = \frac{T_{05}}{T_{04}} \quad [21] \quad (2.25)$$

- Nozzle (n)

The behavior of the nozzle in comparison to the inlet behavior is equal, that is, the flow is isentropic, while going through the specified stage [17]. Granted these conditions, the total pressure and temperature, equation 2.26, of the nozzle are,

$$\pi_n = \frac{P_{07}}{P_{05}} = \tau_n = \left(\frac{T_{07}}{T_{05}}\right)^{\frac{\gamma}{\gamma-1}} = 1 \quad [24] \quad (2.26)$$

Jet Engine Performance

This part comprises the principal measures to evaluate the turbojet performance in an ideal scenario. Assuming the gas is calorically perfect along the cycle, the pressure at the turbojet's exit is equivalent to the ambient pressure, as well as, the fuel-to-air ratio is much less than unity [17].

$$\tau_r = 1 + \frac{\gamma-1}{2}M_0 = \frac{T_{0r}}{T_r} \quad [17] \quad (2.27)$$

$$\pi_r = (1 + \frac{\gamma-1}{2}M_0)^{\frac{\gamma}{\gamma-1}} = \frac{P_{0r}}{P_r} \quad [17] \quad (2.28)$$

However, it is frequent to limit the design in regard to the maximum permitted turbine inlet stagnation temperature, hence the term established in equation 2.29 [17].

$$\tau_\lambda \equiv \frac{c_{pt}T_{04}}{c_{pc}T_{0r}} \quad [17] \quad (2.29)$$

The thrust (F), equation 2.30, and specfic thrust $(\frac{F}{\dot{m}})$, equation 2.31, can now be obtained.

$$F = \dot{m}(V_7 - V_0)[17] \quad (2.30)$$

$$\frac{F}{\dot{m}} = a_0\left(\left[\frac{2\tau_r}{\gamma-1}(\frac{\tau_\lambda}{\tau_r\tau_c} - 1)(\tau_c - 1) + \frac{\tau_\lambda}{\tau_r\tau_c}M_0^2\right]^{\frac{1}{2}} - M_0\right)[17] \quad (2.31)$$

Where the speed of sound is found in equation 2.32.

$$a = \sqrt{\gamma RT} \quad [17] \quad (2.32)$$

Finally, the specific fuel consumption, S can be obtained by the equation 2.33.

$$S = \frac{f}{\frac{F}{\dot{m}}}[17] \quad (2.33)$$

Actual Behavior Component Analysis

The actual, or, nonideal cycle analysis presents the equations for analysis of the components and the engine's performance, ignoring the gas velocities throughout the gas generator. The only velocities considered are at the inlet, intake, and outlet nozzle, exhaust.Besides, the components are considered irreversible, but adiabatic. Therefore, isentropic efficiencies are assumed for the inlet, compressor, turbine and nozzle [11].

- Inlet

At the inlet, when the air is being ducted to the compressor, there is a reduction of the total pressure from the free stream pressure. This occurs due to friction during the intake. Consequently, the temperature increases being higher than the ideal case, which is conditioned by the inlet efficiency, η_i. The equations 2.34 and 2.35 presented below, obtain the pressure and temperature at the inlet, in which the outlet temperature is calculated as in the ideal cycle [11].

$$P_{02} = P_{0r}\left(1 + \eta_i \frac{\gamma_c - 1}{2} M_r^2\right)^{\frac{\gamma_c}{\gamma_c - 1}} \quad [11] \quad (2.34)$$

$$T_{02} = T_{0r}\left(1 + \frac{\gamma_c - 1}{2} M_r^2\right) \quad [11] \quad (2.35)$$

The pressure ratio, equation 2.36 is,

$$\pi_i = \frac{P_{02}}{P_{0r}} \quad [11] \quad (2.36)$$

- Compressor

At this stage, the compression occurs in an irreversible adiabatic process. Thus, the association with the compressor's isentropic efficiency, η_c [11]. Since this is an actual cycle, the compression of the air suffer losses along the way due to friction, turbulence and many other unfavourable factors, which will cause the temperature to rise. This rise is related to the compressor efficiency,η_c [11]. So, the exit conditions, pressure, equation 2.37, and temperature, equation 2.38, at the compressor exit are,

$$P_{03} = P_{02}\pi_c \quad [11] \quad (2.37)$$

$$T_{03} = T_{02}\left[1 + \frac{\pi_c^{\frac{\gamma_c - 1}{\gamma_c}} - 1}{\eta_c}\right] \quad [11] \quad (2.38)$$

- Burner

During the combustion, there are losses derived from defective combustion, for example, conduction, radiation, which are accounted by introducing the burner efficiency, η_b [11]. As a result, the pressure at the burner exit, equation 2.39, is,

$$P_{04} = P_{03}(1 - \Delta P_{cc}\%) \quad [11] \quad (2.39)$$

The temperature of the turbine outlet is pre-determined to respect the material limitations. Therefore, the fuel-to-air ratio, equation 2.40 can be calculated by,

$$f = \frac{c_{pt}T_{04} - c_{pc}T_{03}}{\eta_b FHV - c_{pt}T_{04}} \quad [11] \quad (2.40)$$

- Turbine

For this process, the turbine available power is less than in the ideal process. Thus, the turbine efficiency is linked to the expansion, in which the turbine pressure ratio expression, equation 2.41 turns into,

$$\pi_t = \left(1 - \frac{(c_{pc}/c_{pt})T_{02}}{\lambda(1+f)\eta_c \eta t T_{04}} \left[\left(\frac{P_{03}}{P_{02}}\right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1\right]\right)^{\frac{\gamma_t}{\gamma_t - 1}} \quad [11] \quad (2.41)$$

Where, λ stands for the ratio between the power required to drive the compressor, and the available power generated by the turbine. The values of this parameter vary from 75$\%$ to 85$\%$ [11].
The latter equation can be associated to the equation 2.42, to obtain the exit temperature [11].

$$\frac{P_{05}}{P_{04}} = \left[1 - \frac{1}{\eta_t}\left(1 - \frac{T_{05}}{T_{04}}\right)\right]^{\frac{\gamma_t}{\gamma_t - 1}} \quad [11] \quad (2.42)$$

- Nozzle

At this stage, a critical pressure is introduced to verify if the nozzle is choked or not. Therefore, the critical pressure is obtained by the equation 2.43 [11].

$$\frac{P_{06}}{P_{crit.}} = \frac{1}{\left[1 - \frac{1}{\eta_n}\left(\frac{\gamma_t - 1}{\gamma_t + 1}\right)\right]^{\frac{\gamma_t}{\gamma_t - 1}}} \quad [11] \quad (2.43)$$

If $\frac{P_{06}}{P_{crit.}} > \frac{P_{06}}{P_{0r}}$, the nozzle is unchoked. The exhaust velocity is calculated from equation 2.44.

$$V_7 = \sqrt{\frac{2\gamma_t \eta_n R T_{06}}{(\gamma_t - 1)}\left[1 - (P_{0r}/P_{06})^{\frac{(\gamma_t - 1)}{\gamma_t}}\right]} \quad [11] \quad (2.44)$$

From the equation above, the exhaust temperature is obtained from the equation 2.45.

$$T_{07} = T_{06} - \frac{V_{07}^2}{2c_{pt}} \quad [11] \quad (2.45)$$

If $\frac{P_{06}}{P_{crit.}} < \frac{P_{06}}{P_{0r}}$, the nozzle is choked, altering the exhaust temperature equation, 2.46, to,

$$\frac{T_{06}}{T_{07}} = \frac{\gamma_t + 1}{2} \quad [11] \quad (2.46)$$

Turning the exhaust velocity equation, 2.47, to,

$$V_{07} = \sqrt{\gamma_t R T_{07}} \quad [11] \quad (2.47)$$

Jet Engine Performance Parameters

Specific Thrust, equation 2.48, is now expressed as,

$$\frac{F}{\dot{m}_a} = [(1+f)V_{07} - V] + \frac{A_7}{\dot{m}_a}(P_{07} - P_{0r}) \quad [11] \quad (2.48)$$

And, the thrust specific fuel consumption equation, TSFC equation 2.49, can be demonstrated as,

$$TSFC = \frac{\dot{m}_f}{F} \quad [11] \quad (2.49)$$

2.3 Components

In this section, a background information of each component is given, as well as, the different types and their functionalities.

2.3.1 Inlet Nozzle

The inlet nozzle is like an air filter for the compressor. In other words, when the air is being ducted to the compressor, the inlet nozzle is responsible for allocating and controlling the airflow, that is, providing the required amount of airflow to the engine. Before the airflow enters the compressor smoothly, the inlet nozzle performed his task by providing an uniform, stable and high-quality airflow to the compressor. The inlet is, indirectly, responsible for generating thrust, being considered one of the major components belonging to a turbojet [6].

The development of the inlet nozzle is an extensive study and application of dynamics fluid laws in order to control the airflow at subsonic or supersonic flight speeds. The distinction of the flight regime is made according to the Mach number, in which the intake is operating. For a Mach number lower than 1.0, it operates in a subsonic condition and, if the Mach number is higher than 1.0, the flight speed is supersonic [25]. The number of Mach is shown in equation 2.50.

$$M = \frac{V}{a} \quad (2.50)$$

Considering the flow of air enters the compressor uniformly is an idealized presumption because, in reality, components like engine pylon, wing, and fuselage stray from the ideal, which affects the airflow and undermines the performance of the inlet nozzle. Cross-wind and vortex are examples of undermining the intake performance. In a cross-wind scenario, the stronger it is, the riskier it becomes, for the reason that, the velocity at the lip might exceed the speed of sound in that zone, jeopardizing the blades due to the consequent increase of the flow speed added by the windward side of the air intake. An example of a vortex is the ground vortex.

It can develop as a result of the engine placement under the wings being next imbibed by the intake, which could have a negative impact on the intake performance [6].

Intake design for a subsonic airflow has a well-rounded lip, found in figure 2.14, to prevent the flow separation, resulting in a thicker lip as opposed to a sharp lip for a supersonic airflow. As for the inlet cross-section, for subsonic speed, they have a round or an elliptical shape, whereas, for the supersonic speed inlets, they have a central cone to drastically reduce the flow to subsonic speeds or a rectangular shape intake, shown in figure 2.15 and 2.16 [26].

Figure 2.14: Subsonic Inlet [26]

Figure 2.15: Axisymmetric Supersonic Inlet [26]

Figure 2.16: Rectangular Supersonic Inlet [26]

2.3.2 Compressor

The compression of the air is vital for the gas turbines and the responsibility for assuring it is the compressor. Its principle is to convert mechanical energy, which is supplied by the turbine in the form of work, into pressure energy. So, the better the conversion, the better the overall functioning of the gas turbine. To improve it, there are three important parameters that characterize a compressor performance, the first, compressor efficiency, indicates the energy loss during the conversion, in other words, it shows the energy that the compressor needs to increase the pressure energy. The second, compressor pressure ratio, is the ratio of the total pressure at compressor release and compressor entrance. The third, airflow rate, means the volume of airflow that the compressor is able to process within a unit time. These three parameters are interrelated playing an important role in the performance of the compressor. For example, the compressor pressure ratio is connected directly to the thrust, fuel consumption and engine efficiency [6].

2.3.2.1 Centrifugal

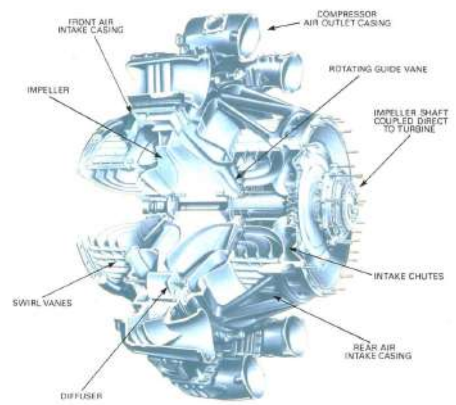

Figure 2.17: Centrifugal Compressor [8]

This compressor, figure 2.17, has a rotating impeller to accelerate the air and a fixed diffuser, that produces the required pressure rise. The impeller rotates at high speed, inducing, continuously, the air into its center generating a radial airflow orientated by vanes to the impeller tip that changes the airflow to perpendicular, in relation to the rotation axis. The action of the impeller will accelerate the airflow causing the pressure to rise [8]. At the exit of the impeller, the air goes through a vaneless space followed by a vaned diffuser constituted by vanes tangent to the impeller that turns the kinetic energy into pressure energy [27].When the air leaves the impeller, it is dependent on the configuration of the impeller that will dictate the airflow direction. The different types of impellers are illustrated in figure 2.18.

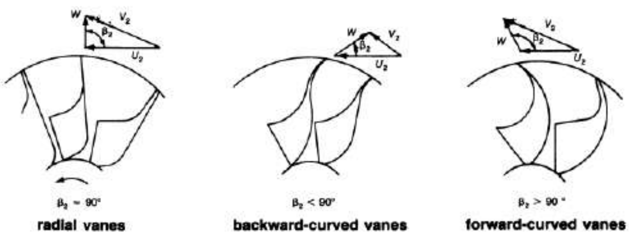

Figure 2.18: Types of impeller [27]

When the air is going through the impeller, there are changes in pressure and velocity. These alterations can be seen in the figure 2.19, where a graphic of pressure and velocity is traced, describing the airflow passage through the elements of the compressor.

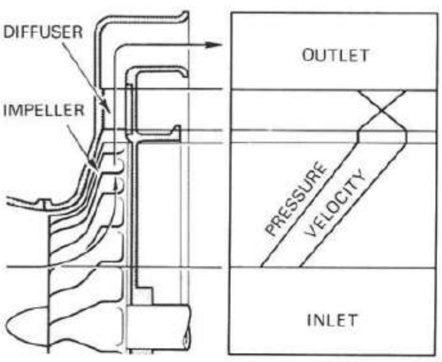

Figure 2.19: Pressure and Velocity evolution throughout the impeller-diffuser system [27]

This compressor is a choice for smaller gas turbines because it is simpler, has low-cost manufacture, high stability, which means a greater operating range. Radial compressors obtain a high-pressure ratio, like 13:1 on experimental modes and, on average 5:1 [17]. An overall balance of the positive and negative points about the three impellers can be seen in table 2.1.

Table 2.1: Advantages and disadvantages of the different impellers [27]

Types of Impellers	Advantages	Disadvantages
Radial vanes	1. Reasonable compromise between low energy transfer and high absolute outlet velocity 2. No complex bending stress 3. Easy manufacturing	1. Surge margin is relatively narrow
Backward-curved vanes	1. Low-outlet kinetic energy = low-diffuser inlet mach number 2. Surge margin is wide	1. Low-energy transfer 2. Complex bending stress 3. Hard manufacturing
Forward-curved vanes	1. High-energy transfer	1. High-outlet kinetic energy = High-diffuser inlet mach number. 2. Surge margin is less than radial vanes 3. Complex bending stress 4. Hard manufacturing

2.3.2.2 Axial

The axial compressor, figure 2.20, is the most used in engine designs for the actual aircrafts because it can deliver high-pressure ratios and high mass flow rates at the same time, allowing high-thrust engines to be made. This compressor is formed by several rotors, connected to the central shaft, which increases the kinetic energy and static pressure. Each rotor goes along with stators, which are fixed rotors that reduce kinetic energy, hence, increase the static pressure and prevent the flow from spiraling [28]. Acting as air straighteners that remove the swirl, stator vanes do this due to their varying angle that corrects the flow from the rotor and directs the flow in the correct direction for the next rotor [6]. Lowering the angle of attack, the variable stators reduce the tendency to stall [17]. The pressure and velocity throughout the axial compressor are demonstrated in figure 2.21.

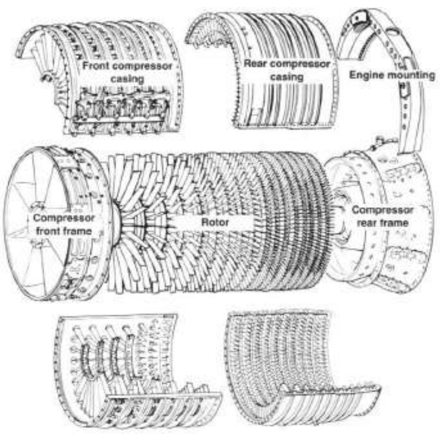

Figure 2.20: Axial Compressor [6]

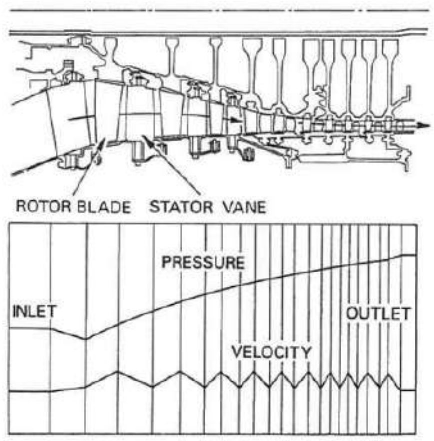

Figure 2.21: Pressure and Velocity changes in an axial compressor [8]

A row of a rotor, with a row of stator vanes is called a stage. Each stage has a low pressure increase. The increase is small because of the deflection angle of the blades, and, the rate of diffusion has to be limited to prevent air breakaway at the blades that could be followed by blade stall. To suppress this effect, “bleed valves” were created to release part of the air from the blades in the intermediate rows and variable stator. Controlling as much as is possible to prevent these losses, the body of the compressor can be built to have multiple stages, that, when comparing to the centrifugal compressor with the same frontal area, it obtains higher-pressure ratios, coming up with much more thrust. That is why the axial compressor is chosen for most aircraft engines [6]. Other advantages are the reduced aerodynamic drag due to a smaller cross-section and there is no need for turning the flow because of the airflow streams in an uniform direction to the turbine [8].

These engines can be composed of more than one spool. The spool is the shaft that connects the compressor and turbine, in which the latter rotates on. If there is only one set of compressor and turbine, the propulsion device is classified as a single-spool, as shown in figure 2.22. A multi-spool consists of two or more rotor assemblies with each rotor being driven by their own turbine [8]. An example of a multi-spool is demonstrated in figure 2.23.

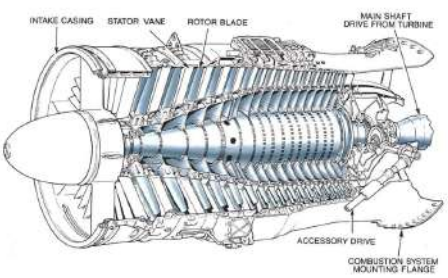

Figure 2.22: Single-spool axial compressor [8]

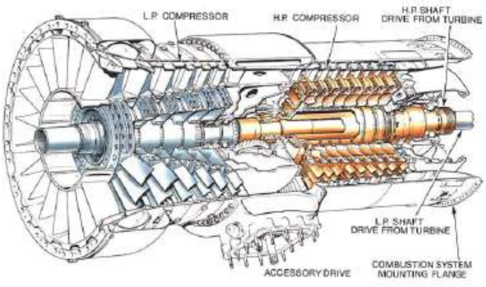

Figure 2.23: Twin-spool axial compressor [8]

If the compressor rotates at an optimum speed, it reaches higher-pressure ratios. With high-pressure ratio values, fuel efficiency is improved but it has limitations. The casing tends to expand and distort, making necessary a heavier compressor casing to support vibration stresses [8].

2.3.3 Diffuser

The diffuser or stator is a subcomponent of the compressor system, which redirects the high-speed radial airflow that comes from the impeller to the combustion chamber, converting the residual speed energy into pressure energy [29].

The diffuser system can be distinguished between two types, bladed or non-bladed. The non-bladed or bladeless annular space is easily produced and could be efficient. Since it has no blades, the correct blade angle does not become a worry with the consequence of breaking away the flow. However, the bladeless diffuser would not be the appropriate choice because the flow duct would widen, hence, the possibility of flow breakage. Considering the Bernoulli law, the total energy of the flow remains constant, in other words, if the speed increases, the pressure decreases. Furthermore, since the relation between the speed and the diffuser diameter, based on the vortex law, is constant, concluding that widening the flow would not have the desired impact on the pressure conversion, hence, the flow could break down [29].

The bladed diffuser, can have the guide vanes curved in the direction of the compressor rotation or curved in the opposite direction or even blades that widen, forming thick wedges that can be drilled. The latter allows the fixation of bolts with no interference in the gas flow. These types, shown in figure 2.24, are identified as, straight diffuser blades, forward curved blades and wedge-shaped blade diffuser, respectively. Moreover, this diffuser types possesses blades that counter the twisting motion of the gases, avoiding a fall of the gas pressure, by eliminating the residual spiral motion [29].

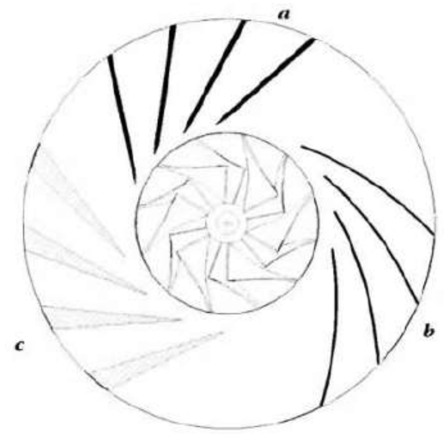

Figure 2.24: Diffuser Types [29]

2.3.4 Combustion Chamber

It is in the combustion chamber, see figure 2.25, that almost of internal energy increase is made and is one of the most important components for the thrust generated. It is conceived to provide an efficient combustion limiting pressure losses at the minimum due to the relevant role it has in the operating and range costs [17]. It is where the fuel is burnt after mixing with the air induced by the compressor, followed by a release of thermal energy, after the combustion, where the air will be expanded and will accelerate through a stream [6]. For the air to be uniformly

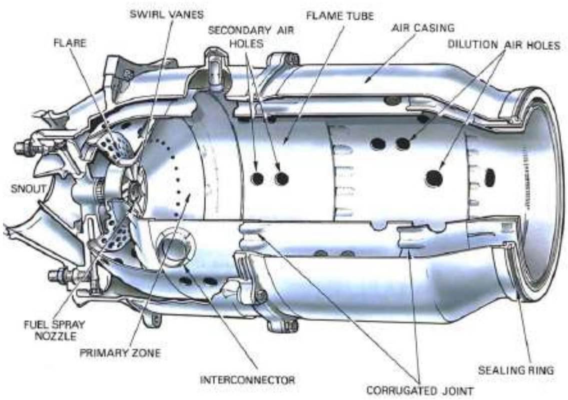

Figure 2.25: Combustion Chamber [8]

heated, there has to be a temperature control at the time of the combustion and adequate material to resist extreme temperatures throughout the combustion process. The combustion chamber supports heat increase differences from 650 to 1150 degrees Celsius of pre-heated air, which enters the chamber at 250 to 550 degrees Celsius. However, only 20 percent of the air mass flow passes to the primary zone (where the fuel burning takes place) of the combustion chamber, through swirl vanes to stimulate recirculation of the hot gas and small orifices of disk supporting the swirl generator. This recirculation is created by the swirl vanes, which belong to a whole part designated as snout, that lowers the airflow velocity to be in the desired values to keep the flame lit, constantly, throughout the range of the ongoing operation. In addition, there is also, another 20 percent that goes through the secondary air holes, joining the hot gas to lower the temperature before it reaches the turbine [8].

In the secondary air holes, the air goes through the external and internal walls of the flame tube, getting inside, after the adjacent flow passes the primary zone. Uniting these two streams, create low-velocity recirculation, originating a toroidal vortex to achieve a stabilized flame [8].

With so high temperatures, it would be extremely difficult for a material to support and to keep having the same performance independently of how high the temperature is. Therefore, to enable the proper functioning of the material, the combustion chamber was designed to dictate the course of the airflow. This enables the cooling of the material and the hot gas. To prevent an undesired performance of the chamber, spacing in the flame tube was developed, so the airflow cools the flame tube walls preventing it from overheating. On average, for cooling purposes is used 40 percent from a total of 60 percent of air is not used for combustion [6]. Then another 20 percent of air passes through the secondary air holes, into the dilution zone, reducing the hot gas temperatures of 1800 to 2000 degrees Celsius before it reaches the turbine [8].The air distribuition inside the burner is demonstrated, below, in figure 2.26.

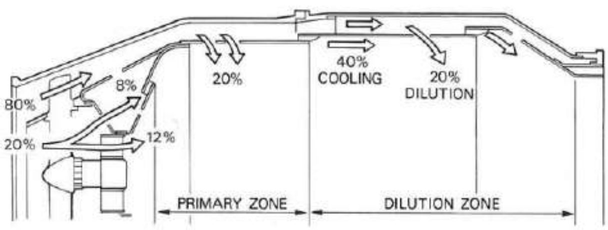

Figure 2.26: Distribution of air inside the burner [8]

Types of combustion chambers

- Multiple combustion chambers or can-type combustion chamber

It has multiple single chambers in parallel forming a circle around the central shaft. The combustion is easily achievable since it is only necessary ignition at one or two combustors. Creating a flame that spreads flame tube to flame tube by a connection link named interconnector, allows the burners to operate at equalized pressure [6]. An example of this chamber is shown in figure 2.27.

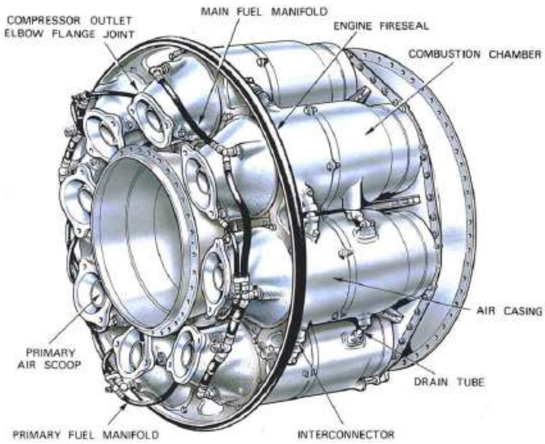

Figure 2.27: Multiple combustion chamber [8]

- Tubo-annular or can-annular combustion chamber

  Came as a middle term of the can-type and annular chamber improving the flaws of each one, combining their strongest characteristics. This combustion chamber, shown in figure 2.28 differs from the multiple combustion chambers with an outer casing surrounding the several chambers for secondary air supply, becoming more mechanically stable than before. This improvement made a more efficient usage of the available space [6].

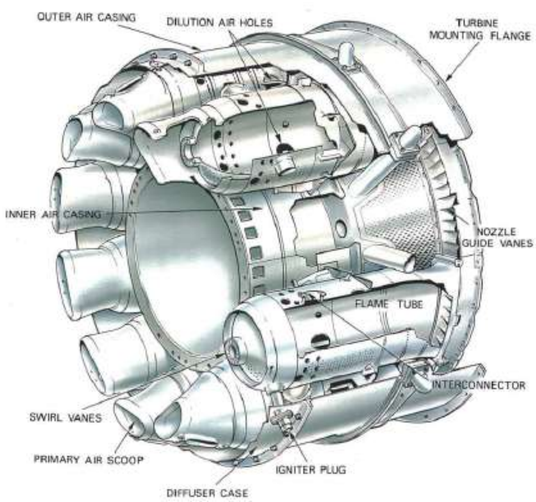

Figure 2.28: Tubo-annular combustion chamber [8]

- Annular combustion chamber
  It is a single flame tube, demonstrated in figure 2.29, in an annular form, that uses more efficiently the volumetric space, allowing an even combustion process. Its simpler design has a beneficial reduction of the overall weight, since, the energy expanded through the burners is the same, with 25 percent less of length. Subsequently, impacts the production cost by decreasing it [8].

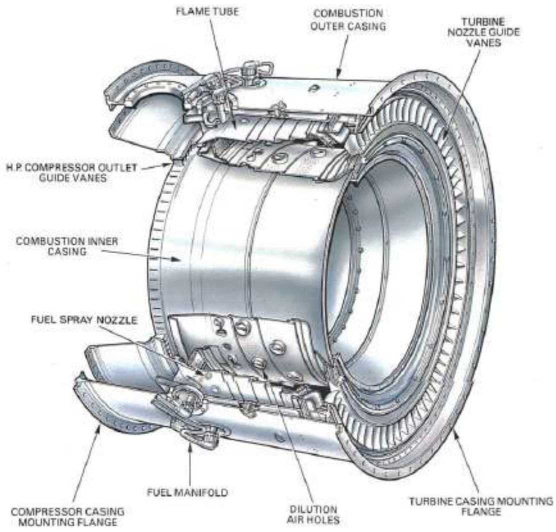

Figure 2.29: Annular combustion chamber [8]

2.3.5 Turbine

The turbine is a primary component for the gas generator, that drives the compressor. It is the priority of its functionality to withdraw the energy contained in the hot gas, that comes from the combustion, by absorbing the energy on the impact of the gas flow with the blades. Extracts the energy when the hot gas, at high temperatures, reaches the limit of space available in the turbine. This cycle, repeatedly, will promote the rotation, at a high speed, of the turbine. Fast enough to drive the compressor in the form of mechanical shaft power, which is achievable by converting kinetic energy to pressure energy and work [8].

2.3.5.1 Axial

Most of the engines in aircrafts have axial-flow turbines because of the higher mass flow intake. This type of turbine is characterized by a set of one stator and one rotor, in the respective order. A stator is a stationary nozzle with guide vanes of an airfoil section that adjusts the flow from the combustion chamber to the turbine by straightening the flow path, accelerating the gas, creating a nozzle effect hence the reason of its name. Followed by a rotating wheel that uses the kinetic energy to create motion [8].

For the extraction of energy to be successful, the turbine blades induce a swirl in the gas flow in order to obtain a uniform work across the length of the blades. The blades design will have an effect on the flow expelled from the turbine, turning to more of axial flow, before getting into the exhaust system [6].

To be an efficient turbine, its design and development have to be according to the specifications of the engine. An important specification is the power demand which will affect the number of stages in a turbine. Although, the number of stages that have to be added, not only depends on the power demand but also on, rotational speed, maximum permissible turbine diameter and the number of compressor spools. For example, engines with high compression ratio use to have two shafts to drive the low and high-pressure compressor [6].

The rotational movement causes stress in the turbine disc that builds up along with the square of the speed [8].

New materials and cooling techniques were developed to support the high temperatures and pressures. One material, generally used, in blade speeds, is the nickel-based superalloys due to a high resistance of creep and high-temperature strength [30]. Even so, efforts are being made to prevent and manage losses in any part and functioning of the turbine. However, it is extremely difficult to prevent losses like friction, turning the flow or tip clearance [6].

The way a turbine converts energy, designates the type of it, by distinguishing it in three types: impulse, reaction, and impulse-reaction.

The impulse turbine, shown in figure 2.30, is similar to a water wheel that extracts energy from the gas flow by the impact of the gas on the turbine blades. The gas flows by the nozzle guide vanes exiting with a higher velocity, decreasing pressure and temperature. In the rotor stage, the accelerated flow will impact on the rotor blades, reducing the velocity due to the energy conveyed from the gas to the blades. This momentum exchange will result in wheel rotation [6].

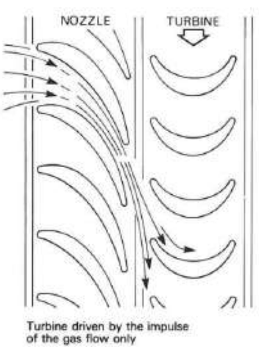

Figure 2.30: Impulse turbine [8]

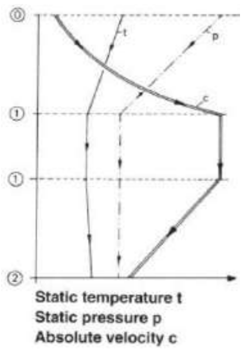

Figure 2.31: Properties variation throughout [8]

Observing the figure 2.31, there are three distinctive phases. When the gas enters the nozzle guide vanes, correspondent to the phase zero to one suffers a pressure and temperature decrease with a velocity increase. Afterward, throughout phase one, as the gas leaves the nozzle guide vanes and reaches the rotor blades, temperature, pressure, and velocity are constant. It then goes to the rotor blades path, phase one to two, where energy is extracted in the form of work, reducing the gas flow speed. It can be observed a slight temperature rise as a consequence of friction [8].

The reaction turbine does not differ much from the impulse turbine. The difference is in the rotor blades. In other words, the rotor blades path, due to the design of the blade, creates a nozzle effect, accelerating more the gas flow. Because of its design, an aerodynamic force is also generated, besides the generated momentum from the impact of the gas on the blades, causing the rotor to spin [8].

The constriction in the cross-sections of the flow path, causes the nozzle effect explained previously. In addition, there is a slightly less reduction of temperature and pressure than in the impact turbine.

2.3.5.2 Radial

From the first use in a jet engine flight, at the end of 1930, to turbochargers utility, the radial-inflow turbine is used for various purposes, for example, gas liquefaction expanders in the petrochemical industry, small gas turbines that provides power to helicopters or as generating units [27].

A radial turbine presents a great benefit for work production. This type of turbine provides more power than two or more stages of an axial-flow turbine. In addition, the cost for its fabrication its lower than a single or multistage axial turbine. However, in terms of efficiencies, this type of turbine loses to the axial-type, which is the major reason why the axial type is mostly applied to aircraft engines [27].

The radial-inflow turbine is divided into two types:

- Cantilever radial-inflow turbine

  This turbine, observed in figure 2.32, is similar to a low-reaction or impulse turbine due to no acceleration of the flow across the turbine. It is characterized by not using radial inlet angles and having two-dimensional cantilever blades. Low-efficiency values and manufacturing complications are the reason for their rare usage [27].

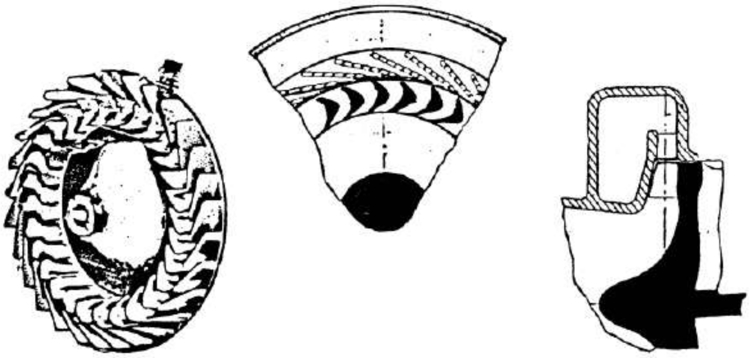

Figure 2.32: Cantilever radial-inflow turbine [27]

- Mixed-flow radial-inflow turbine

  This turbine, shown in figure 2.33, is composed, firstly, by a scroll that collects, from a single duct, the flow, to nozzle blades that are often used as vaneless nozzles, like in turbochargers where the efficiency is not of importance due to the excess energy in the exhaust gases. With a vane design, the flow is directed by the blades, accelerating it. These blades are merged in the hub or the disc, provoking a force normal to the flow guideline. When the flow reaches the curved end section of the blades or exducer, part of the tangential velocity force is removed. Finally, the flow makes to the outlet diffuser, where high absolute velocity from the exducer is transformed into static pressure [27].

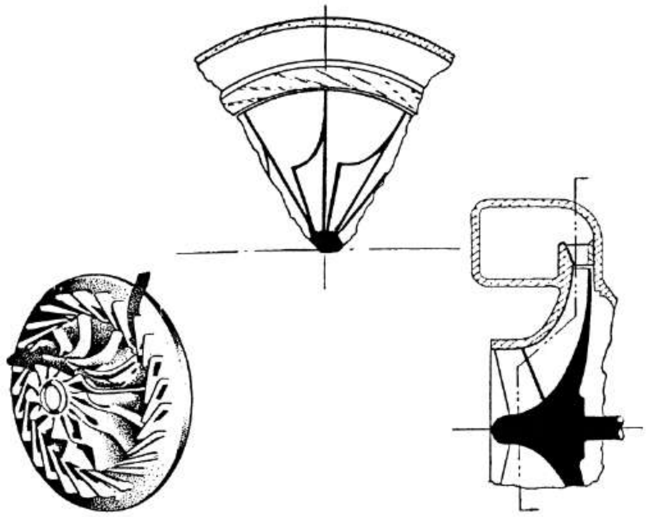

Figure 2.33: Mixed-radial inflow turbine [27]

2.3.6 Exhaust Nozzle

The last component of a turbojet is responsible for the generation of thrust, separating the turbojet from being a gas generator, as referred previously. Its method of producing thrust consists of converting the remaining enthalpy into exhaust speed [29].

The nozzle is developed according to the required performance and takes various shapes. Concerning the design, three types are distinguished:

- Convergent Nozzle

  The cross-section area of the duct decreases along the streamline direction that ends with a smaller cross-section area, thus, accelerating the flow. The acceleration occurs due to a higher pressure, at the nozzle entry, than the ambient pressure, discharging the flow until it reaches the ambient pressure, where the gas will fully expand. If this nozzle discharges at sonic velocity, where the mass flow rate is maximum, the nozzle is considered to be choked. It is mostly applied in high-subsonic commercial and military aircrafts [6].

- Divergent Nozzle

  The cross-section area increases reaching it maximum size at the discharge. This causes the fluid stream to spread across the nozzle, decelerating the flow [31].

- Convergent-divergent nozzle

  Characterized by a varying cross-section. At the forward part, the cross-section area decreases, succeeded by a further increase in the cross-sectional area. Controlling the convergent and divergent part makes the nozzle of variable geometry often used to add more thrust[6]. The figure 2.34 is a skecth of the convergent and divergent nozzle.

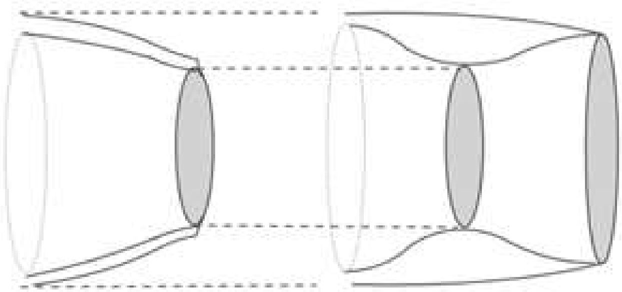

Figure 2.34: Convergent-Divergent Nozzle

2.3.7 Fuel system

2.3.7.1 Fuels

Selecting the appropriate fuel for a small size turbojet engine depends on the maximum available energy per kilograms of fuel [10]. The fuels with a high specific heat of combustion include petrol, kerosene, diesel, JP1 to JP8, which are different types of jet petrol, propane and butane gas. If possible, the use of jet petrol type four kerosene and, petrol, are the best options to use for the combustion process. The heavy presence of volatile hydrocarbons in these two fuels puts them as favorites [29]. However, diesel fuel, which is similar to kerosene, has a high energy density, making it preferable for use because it is easily obtained, being found at every gas station [10]. Furthermore, diesel vapor-air mixture has a lower ignition temperature, roughly, 300 degrees Celsius, than petrol vapor-air mixture that needs a temperature above 600 degrees Celsius to ignite [10].

The most appropriate fuels to small size turbojet engines are the gaseous fuel, particularly, propane or butane. Besides these gases, the use of methanol could be considered, albeit its low energy density plays disadvantageously for its selection. The fuel pump for this fuel type is dispensable since the pressurized gas flows through the engine spontaneously [29]. From the gaseous fuel types stated, the propane is considered more suitable for static testing or development work [10]. In spite of the benefits, this gas requires a pressure tank with twice the volume for the same mass of diesel fuel, owing to its low energy density [10]. The differences of the discussed fuels, can be discerned in the demonstrated table, table 2.2.

Table 2.2: Fuels [29]

SPECIFICATION OF POSSIBLE MODEL JET ENGINE FUELS

	Diesel	Petrol	JP1/Jet A	JP4	Propane	Methanol
Density [kg/l]	0.85	0.76	0.804	0.76	0.5(1)	0.79
H_{0u} [MJ/kg]	42.8	42.5	43.3	>42.6	46.3	19.5
Boiling Range (°C)	190-334	80-130	160-260	60-240	-42	65
Fuel tank Capacity (ml) (5 Minutes, 30 N Thrust)(2)	880	990	920	990	1,380	2,080
Flammability/Fire Hazard	Low	High	Low	High	Very High	High
Price (E/1)	0.8	1.05	1.2	?	0.7	0.6

(1) Liquid Under Pressure
(2) Sufficient for 5 minutes of powered flight at a thrust of 30 Newtons. (Specific Consumption = 0.3 kg/N/h)

2.3.7.2 Fuel Injection Modes

The air and fuel mixture can be carried out by fuel atomization or vaporization. The former is a common technique used in full-size aircrafts, where the quality of the combustion relies, heavily, on the droplet size of the atomized fuel. The complexity of this technique outweighs its benefits for injection in small engines [29].

The fuel vaporization consists of small tubes, known as vaporizers, where the pre-heated gas flows, vaporizing part of the fuel before reaching the primary zone. The fuel is pushed through, by thin tubes, where each is connected to a single hooked tube. The beneficial aspect of this system is the fuel mixes with air before entering the combustion zone. Unfortunately, the effectiveness of this method can only be confirmed through systematic experiments, for example, if the vaporizer is overextended, the temperature tends to climb up [29].
The types of fuel injection can be seen in figure 2.35.

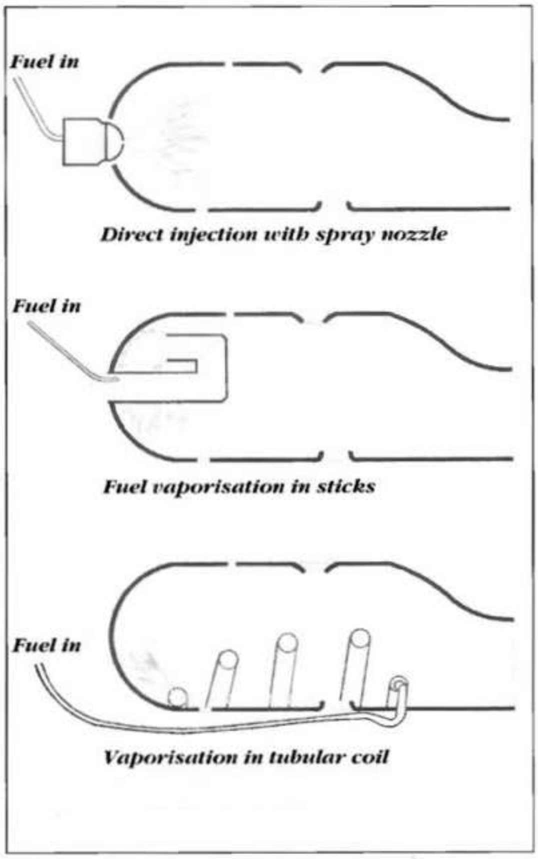

Figure 2.35: Fuel Injection Modes [29]

2.4 Turbocharger Basics

The very beginning of the turbocharger is in 1885 when Gottlieb Daimler created and patented the mechanical process of pre-compression of the air entering the engine. Although, the considered creator of the turbocharger was a swiss engineer, Alfred Büchi, patenting his invention in 1915 demonstrated in the figure 2.36 [32].
This device is composed of a compressor and turbine connected by a common shaft, forcing the

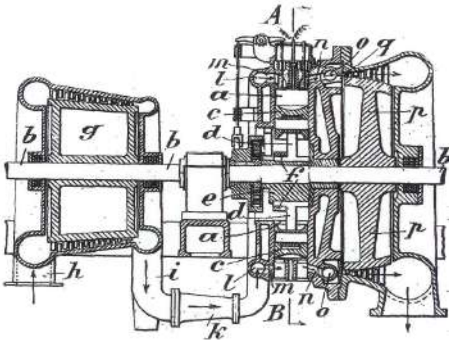

Figure 2.36: Turbocharger, Alfred Büchi's Patent [32]

induction of air to the combustion chamber, of an internal combustion engine. The components of the turbocharger are illustrated in figure 2.37.

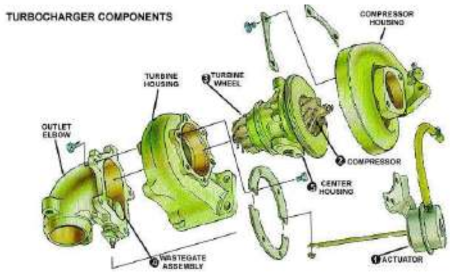

Figure 2.37: Turbocharger Components

The turbine uses the exhaust gases of the engine as a way to produce the necessary work, driving it by the shaft, for the compressor start to rotate. The compressor pressurizes the air, supplying it to the combustion chamber. In the combustion chamber, there is an increase in the fuel-air mixture flow, since the amount of fuel is proportionate to the surplus of air [33].
With more fuel burnt, the power output increases as the internal combustion engine efficiency. This cycle is demonstrated in figure 2.38. These outcomes were firstly discovered by Alfred Büchi in his first and successful application, obtaining a power augment over 40 percent [32]. This boost, as in power, as in efficiency made the turbocharger, a device widely used in vehicles.

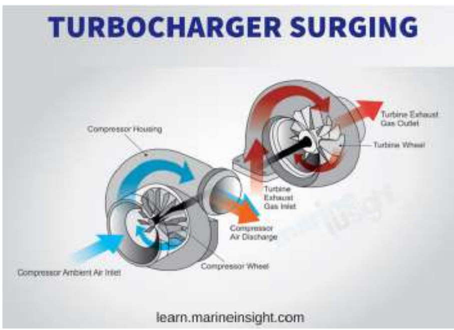

Figure 2.38: Turbocharger Work[35]

The development of a turbocharger involves the design of compressor and turbine blades as their housings. The blades designs are analyzed in CFD, Computational Fluid Dynamics, to know how the air flows. As for the housings design, the gap between the rotor has to be small enough so the rotor is able to conduct the airflow. If not, there is the possibility of the flow slipping between the rotor edge and the housing [33].

In general, if more information is needed about a specific turbocharger, the fabricator has a chart describing the performance of the turbo's compressor, in which includes the pressure ratio, mass flow rate, turbo speed and efficiency regions. The chart, referred to as compressor map, gives important details, allowing to know what would be the airflow rate for, specifically, this compressor, to a given pressure ratio. Figure 2.39 is presented, recognizing the different performance characteristics in the chart. Right below, is a brief explanation of the observed characteristics [34].

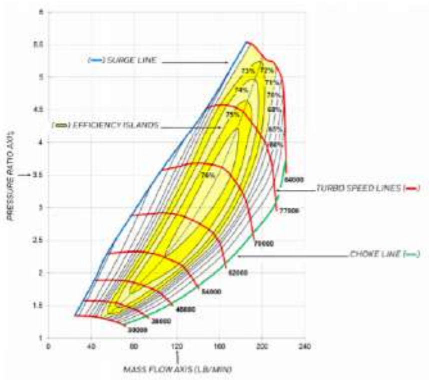

Figure 2.39: Example of a Compressor Map[34]

- Pressure ratio: ratio of absolute outlet pressure by absolute inlet pressure [34].
- Mass Flow Rate: mass of air passing through the compressor during a given period of time [34].
- Surge line: traces the line that distinguishes a stable operation, found at the right side of the line, from a flow instability region that could lead to a precipitate, turbo failure, caused by large thrust loading [34].
- Choke line: is the line separating the maximum mass flow rate that the compressor can process, left side of the line, from the inability of the compressor to process the flow due to sonic speeds achieved, at the rotor, inlet by the flow, impeding increase of the flow rate [34].
- Efficiency Islands: concentric regions on the compressor map that corresponds to the compressor efficiency differentiated by the sizes of the regions. The smaller it is the region, the higher the efficiency is [34].

The compressor map obtained for this thesis is illustrated in figure 2.40, although, it is not the real graphic of the compressor performance, which made it difficult to obtain an exact mass flow rate for a given pressure ratio. The graphic represents the compressor map of the turbo model, RHB31 VZ21, similar to the turbo IHI RHF3 [36].

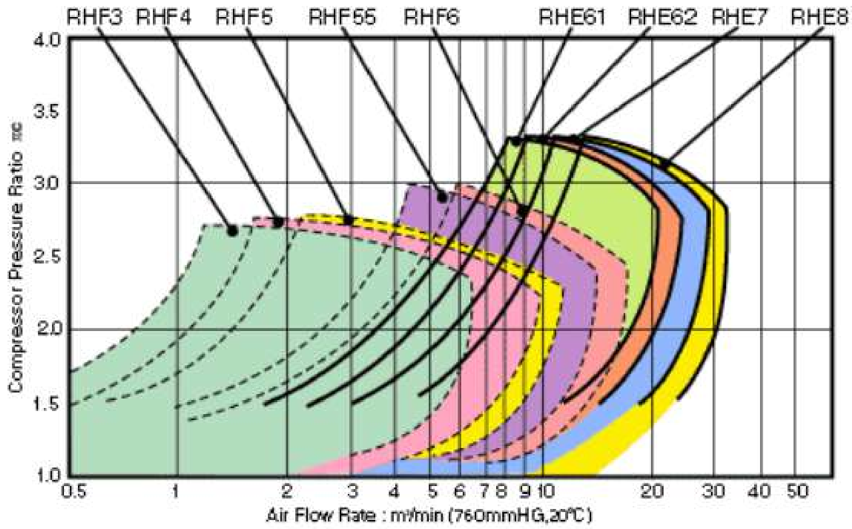

Figure 2.40: RHB31VZ21 Compressor Map [36]

Chapter 3

Methodology

This chapter points out how the design and manufacture process of each component of the mini-turbo jet was carried through. It is designed each component of the engine, explaining how the dimensions were obtained for the design and, what are the materials for each component. The second section describes the manufacturing process chosen. A flowchart of methodology is presented in figure 3.9, found at the end of chapter 3.

3.1 Dimensioning Process

3.1.1 Compressor

The starting point for the dimensioning of this engine is at the compressor. The compressor chosen for this experimental project is from the turbo's company IHI, the RHB31 VZ21 model.

From the literature examination, the compressors used in similar projects, for instance, the Kamps's turbojet or the WPI turbojet, were centrifugal for offering a greater compression ratio and efficiency. Moreover, the turbo or the compressor/turbine set is easily available online and can be purchased at relatively low cost, when compared to other turbos. Since it was already purchased, this compressor was used to develop this thesis.

The design of the compressor should be done with the exact measures, for the design to be accurate. In order to do so, the turbo producers were contacted. Nevertheless, it was not possible, for them, to give these dimensions as it is confidential information. The information through research for the compressor map, was found in ECOTRON technical specifications document [36], show in figure 2.40.

The basic dimensions as, base thickness, impeller/blade height, and, impeller inlet/exit diameter were measured by the use of a caliper. Nonetheless, only the impeller exit diameter was needed.

According to Kamps, a model of a turbojet can be produced using his turbojet dimensions with a scale factor, achieved from a ratio between the compressor diameter, 36.6 millimeters, with the Kamps's compressor diameter, 66 millimeters [29] with a value of, approximately, 0.55. From this value, it was obtained the estimated dimensions of the engine pieces.

3.1.2 Inlet Flange

This piece is the cover of the engine at the compressor side that fixes the diffuser to the outer casing. It was designed based on the Worcester Polythecnic Institute project [37], adapting its size according to the compressor shroud and outer casing dimensions of this engine, in order to the design to fit in the outer casing with a very small clearance. In addition, the bolt holes

were made to be in line with the diffuser and the outer casing holes, for a better coupling of both components.
The aluminium was the material opted for the manufacture of this component. However, it is not possible to specify the type of metal, owing to the fact this material was taken from a spare engine block.

3.1.3 Compressor shroud

This component was designed according to the diffuser vanes diameter, in order to not oversize the shroud and the screws holes to be aligned with the diffuser vane holes. Then, it was proceeded to the part that accommodates the compressor. The gap of the inlet surrounding the compressor must not exceed 0.3 millimeters to attain tolerable efficiencies [29].

However, to succeed in the design of the compressor shroud respecting the tolerance, a 3D scan of the compressor should have been made, to be able to determine the curvature of the rotating compressor. Owing to the fact, that, it was not possible to obtain a 3D scan, the shroud was designed by making circumferences of the exducer and inducer diameter of the compressor. In figure 3.1, is encountered the starting point for this design.

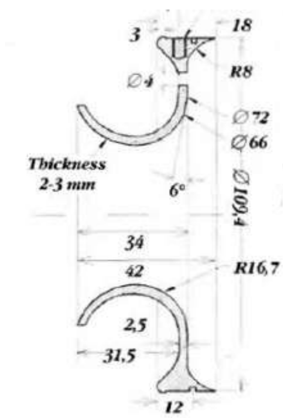

Figure 3.1: Compressor shroud [29]

In the literature examined, the material recommended for this part is aluminum. Therefore, was decided to build using this material. Alas, the specification of the aluminum type is not possible to know, as it was also taken from a spare engine block.

3.1.4 Diffuser

The trickier and challenging piece to design was the diffuser. The first step to take was choosing the diffuser style: bladeless or bladed, and, if it is bladed, decide between straight, forward-curved or wedge-shaped blades. From the examples observed from the literature, was opted to design a wedge-shaped blade diffuser taking into consideration the fixing bolts, that allows the compressor shroud cling to the diffuser and avoid leaks of the gas flow [29].

First, it was dimensioned the diffuser dependent on the Kamps's diffuser. Thomas Kamps's book, [29] indicates the diffuser dimensions and includes the axial blade profile, displayed in figure 3.2 and 3.3.

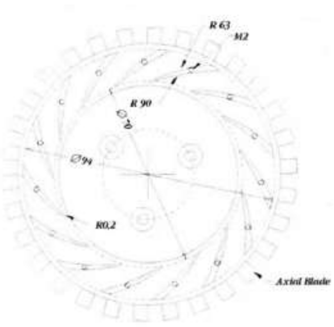

Figure 3.2: Diffuser [29]

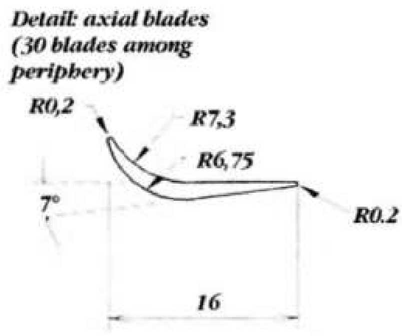

Figure 3.3: Diffuser Blading Profile[29]

3.1.5 Shaft and Shaft Housing

Reviewing mini-turbojets shaft designs as the KJ66, AMT Olympus or the Kamps's engine, it was observed that the designs were, relatively, equal. Therefore, it was dimensioned the shaft by scaling down the dimensions of Kamps's shaft described in figure 3.4.

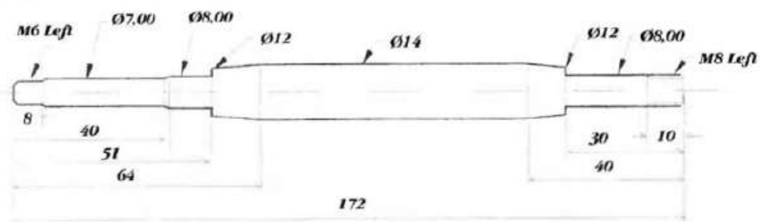

Figure 3.4: Shaft [29]

The shaft accommodates two bearings, located at the bearing seats, positioning the bearing on both sides of the shaft. The steps are for tight-fitting of the bearings to prevent it from displacing. To produce this shaft, a stainless steel rod was decided as the material to be further machined.

The shaft housing was designed, like the shaft, to hold inside the shaft with the two bearings and, couple the diffuser, fixed with bolts to the housing, together with the stator housing, which will also be fixed with bolts to the housing. Despite the variety of shaft housing designs, the determined design would be a simple and straight, widening at both ends of the housing to allow room for the bolts to fix the diffuser and stator housing. The dimensioning was exectuted by adapting to our scale the shaft housing dimensions that are represented in figure 3.5.

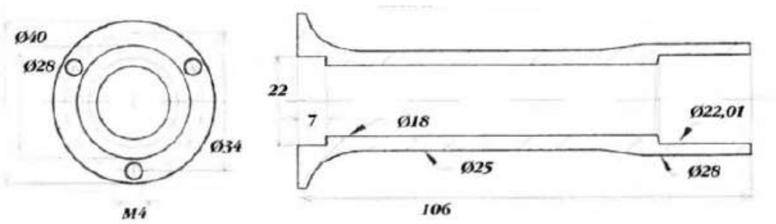

Figure 3.5: Shaft Housing [29]

The aluminum is the material of which is constituted the shaft housing, but its type is not feasible to determine since it was withdrawn from a spare engine block.

3.1.6 Combustion Chamber

The design for the combustion chamber that was considered to be optimal, was an annular chamber. In consonance with the literature reviewed, it seemed the best choice for its simple design and practicality, in other words, would facilitate the manufacturing process as opposed to the other types. Moreover, the combustor design, generally, comes from empirical data and, since the objective is not to improve a design, the choice of a combustion chamber, that had successfully performed its role, was the wisest choice to make. Therefore, it was decided to design it based on the combustion chamber of Kamps, adapting its size and holes for this combustion chamber. Consists of a stainless steel sheet of 0.5 mm width shaped into a tube with the desired diameter and a series of holes of different diameters. Figure 3.6 and 3.7 below demonstrates the combustion chamber design that was relied on [29]. The combustion chamber

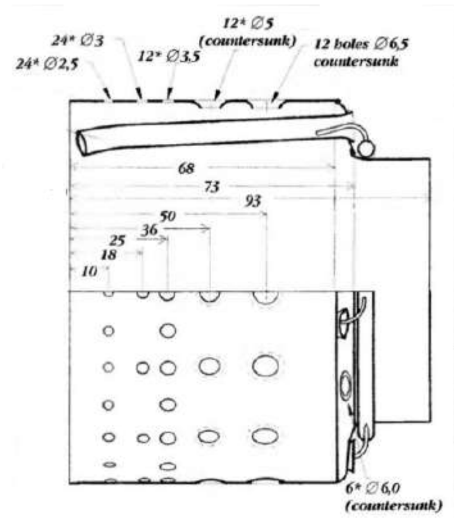

Figure 3.6: Outer Flame Tube [29]

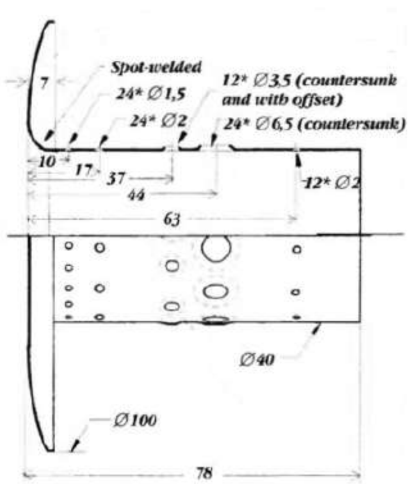

Figure 3.7: Inner Flame Tube[29]

has two tubes, one smaller tube, that goes inside of the larger one. The holes diameters and the distance between holes were scaled down to an adequate size and distance, concerning this mini-turbojet.

3.1.7 Fuel Distributor

The fuel distributor, as the name states, disperses the fuel to the vaporization tubes of the combustion chamber. From the small gas turbines seen, the design is, almost, the same. Therefore, an injector ring was designed, with an adequate diameter, for the combustion chamber designed. The injector ring is set on the inner side of the combustion chamber of the turbine side, and, has various injectors corresponding for each vaporization tube. The fuel comes from an external source that is connected to a tube that crosses the outer casing to the combustion chamber, where it is also linked to the injector ring. The fuel distributor is made of a stainless steel tube with 3 millimeters of diameter [29].

3.1.8 Nozzle Guide Vanes System

The guide vanes or stator, fixed to the housing, redirect the flow to the turbine. The housing, besides holding the guide vanes, is designed to have the necessary space for the turbine to rotate, maintaining the gap between the two components constant, so the efficiency is not

affected. Most of the nozzle guide vanes system designs were made for axial turbines. However, the nozzle guide vanes system was designed based on other designs with use of a radial turbine [38].
The dimensioning was thought, in compliance with the components affected by its design, such as the combustion chamber, fuel distributor and the shaft housing. The idea was to dovetail the outer flame tube with the nozzle guide vanes, where the fuel distributor, would be fixed in a flange, at the inner flame tube.

3.1.9 Turbine

The turbine, like the compressor, was retrieved from the RHB31 VZ21 turbo.

3.1.10 Exhaust Nozzle

This component is responsible for the generation of thrust. Although, this dissertation does not have as a primary objective the optimization of the generated thrust, instead the design and manufacturing of a self-reliant small gas turbine. The nozzle design is a simple, convergent nozzle guaranteeing a straightforward construction that is designed from the dimensioning of this component, that was based on the literature review. The figure 3.8, demonstrates the exhaust nozzle dimensions, although, it was scaled down and adapted to this jet engine. The recommended material for this component is a stainless steel sheet of 0.5 millimeters thickness [29].

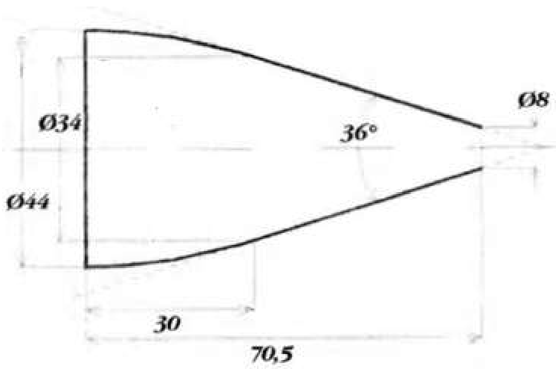

Figure 3.8: Exhaust Nozzle [29]

3.1.11 Bearing, Lubrication, and Fuel Injection

The selection of bearings has to be carried on by considering the purpose of the bearing. The bearing, for a small gas turbine, has to resist to high temperatures conditions and extreme rotational speeds. With these requirements, the bearings manufacturers were looked for the ones that had the set of bearings with the desired measures, to fit properly in the shaft.
The bearings chosen were deep groove ball bearings of stainless steel, from SKF's company, withstanding up to 120000 RPM [39] and are high temperature resistant because, the higher the RPM, the higher temperature of the bearing will be [40].For an adequate operation of the bearings, there has to be a lubrication system, which, should be created by implementing two thin, stainless steel tubes on the shaft housing, connected to an oil pump. Each tube, enters on each side of a bearing, for the transportation of the lubricant. The tubes go through the outer

casing and then, to behind the diffuser, inserted in the shaft housing.
The fuel injection is made by a fuel tube that goes, from the fuel distribution ring through the turbine side of the casing, to the external fuel source. At the distribution ring, the fuel then goes to the vaporization tubes. The combustion chamber was designed to allow the engine to run on different fuels in a gaseous or liquid state.

3.1.12 Outer Casing

The final part of this engine consists of a stainless steel tube with 0.5 millimeters of thickness [29]. At the compressor side, some holes were made to enable the bolts to pass through the casing, reaching the diffuser. The bolts were screwed, fixing the outer casing with the diffuser. At the other end, the outer casing is fixed to the nozzle guide vanes. The design of this component, since it was one of the last to be designed, there was no need for scaling it down based on the literature. Having the other components designed, the outer casing has to cover, adequately, the gas generator.

3.2 Manufacturing Process

The construction guidelines for this thesis, based on the literature reviewed and online videos of the model jet engines manufacture, are divided into two subsections, distinguished by the two main materials used to produce the needed components for the engine. One is aluminum and the second is stainless steel.

The aluminum-based components are the inlet flange, compressor shroud, and the diffuser. The production of these parts was made, with the help of UBI's FABLAB, Fabrication Laboratory, in the 5-axis CNC milling machine. It was considered to do it there due to the precision of the production, which is impossible to equalize if it was handmade, through the use of manual milling machines. For the 5-axis CNC milling machine to produce the desired component, it is required a .stp format file obtained from the design software, in this case, CATIA software, saving options. With this procedure, round blocks of aluminum, are transformed into the expected shape, with an extremely low margin of operating error. However, the shaft housing was created through the manual operation of a lathe machine.

The stainless steel elements were manufactured with the use of a vertical drilling machine, a lathe machine, a roller and a water jet machine. It was firstly used the water jet machine to cut the pieces designed for each component. The pieces were cut from a stainless steel sheet, consequently, the designed components had to be drawn to their plane shape, which, later, were worked to achieve the desired shape. The following action was to drill the holes, in the flat pieces, belonging to the inner and outer flame tube, the combustion chamber. The drills were made using a vertical drilling machine.

“Flat washers” will be fixed, later on, to the respective components, specifically, the nozzle guide vane system, the end-rear of the combustion chamber and the casing. The next step to take was to mold the flat pieces of the combustion chamber, nozzle guide vane system and casing to the required diameter. Coming to an end the molding of the piece, the ”flat washers” were welded to obtain the tubes and flat rings. The last component, the shaft, with the help of a two-dimensional sketch design, was manufactured with a lathe machine.

In general, parts were designed and manufactured using a procedure similar of the one used for the diffuser, seen in figure 3.9.

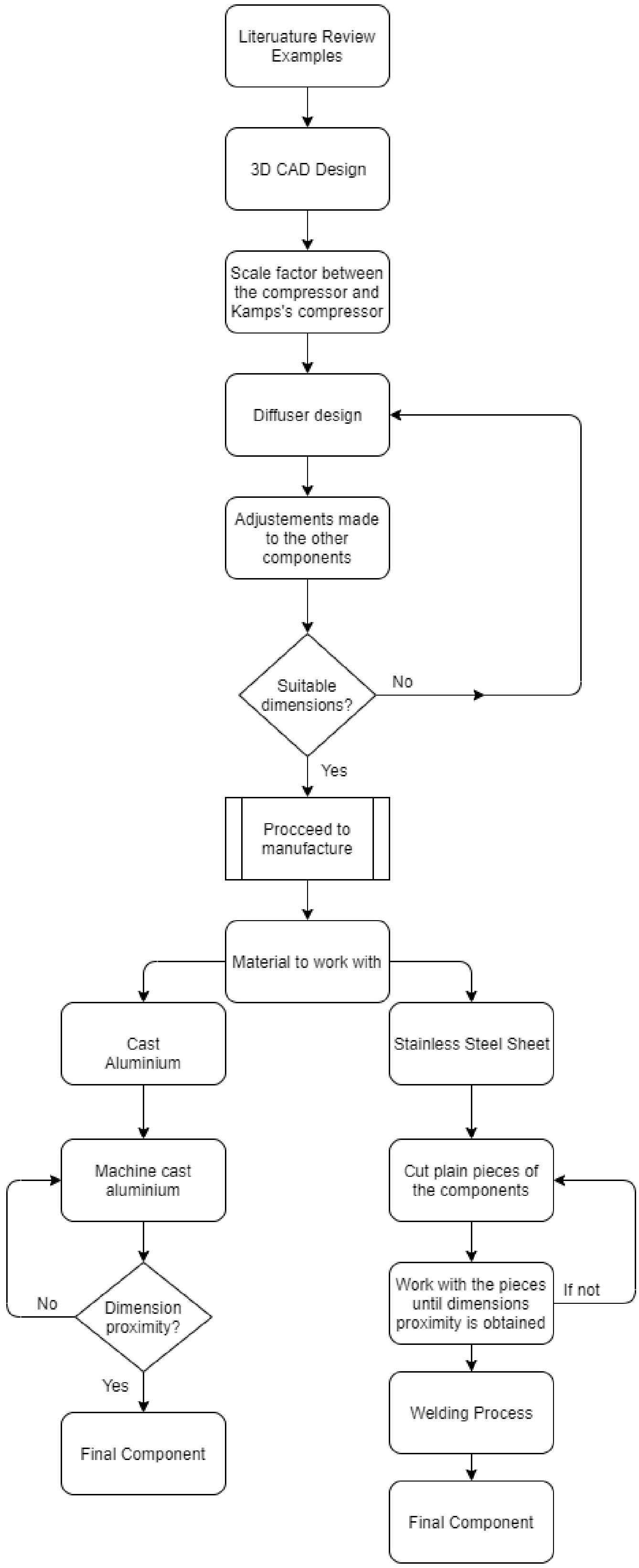

Figure 3.9: Diffuser design procedure

Chapter 4

Practical Case

In this chapter, the results from the dimensioning of the components will be presented, as well as, their design. It will also be explained, in depth, the process of fabrication followed to obtain the components, and how they were machined.

4.1 Dimensioning Results

The final dimensions of the small-turbojet are described along with the assembly design, in the Appendix B and C. The dimensioning, as it was stated previously, was executed by the comparison of empirical data, in which it was compared and adopted to this design. There are some pieces that did not follow the scale factor, precisely, due to adjustments made while designing all the components. The design was finalized when the parts were assembled, to verify if all the components were aligned and, fit properly.

4.2 Mini-Turbojet Prototype's Design

The components' design of this small gas turbine, was briefly explained in the previous chapter 3, whereas, in this section, it will be describe the design techniques applied, to obtain the shape and dimensions of the pieces composing the gas turbine. The blueprint will be outlined by the use of software CATIA V5R18, property of Dassault Systèmes.

4.2.1 Compressor

The compressor design was a challenge to understand and was tried to duplicate it in CATIA software, diverting the least possible from its physical dimensions. To start it was drawn the basis of the compressor as shown in figure 4.1. Followed by the shaft CAD operation, was drawn, from a visual perspective, the compressor vanes profile encountered in figure 4.2. Finally, it was applied the multi-section solid to create the solid vanes from the sketch, concluding the overall design with a hole, through the compressor, to fit the shaft. The final product is shown in figure 4.3.

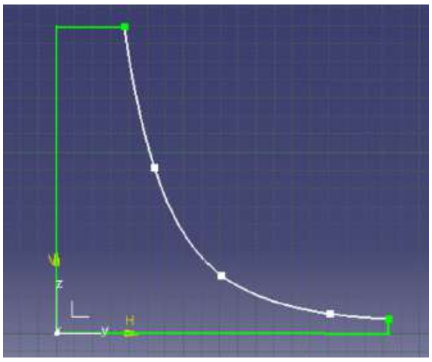

Figure 4.1: Compressor CAD shaft operation

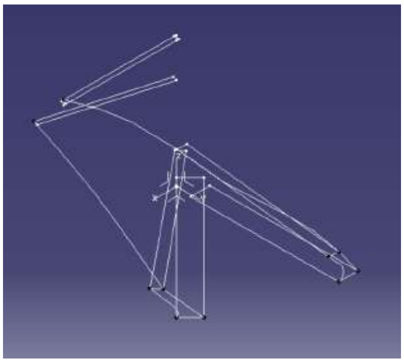

Figure 4.2: Compressor CAD Vanes Sketch

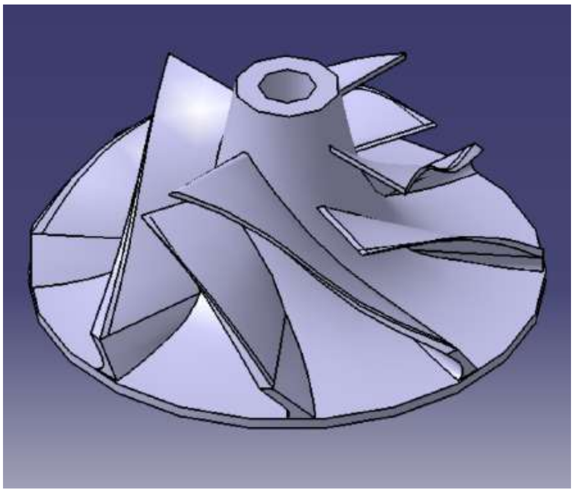

Figure 4.3: Compressor CAD view

4.2.2 Inlet Flange

The design for this component was simple and direct. To start, it was designed the sketch, respecting the compressor shroud dimension, so it would fit properly. Moreover, the sufficient space was given for the accoupling of the diffuser and outer casing. The inlet flange sketch, figure 4.4, was applied in a shaft operation, shaping it to the solid demonstrated in figures 4.5,4.6 and 4.7, accompanied with holes to fix it to the diffuser (front side) and outer casing (side view). An isometric view of the designed piece is illustrated in figure 4.8.

Figure 4.4: Compressor CAD view

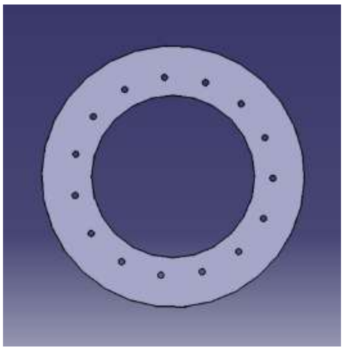

Figure 4.5: Inlet Front View

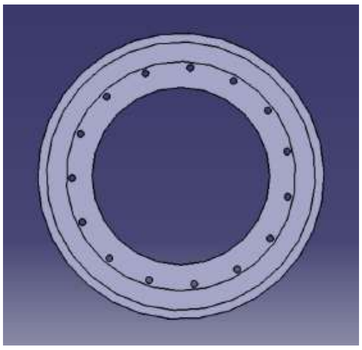

Figure 4.6: Inlet back view

Figure 4.7: Inlet side view

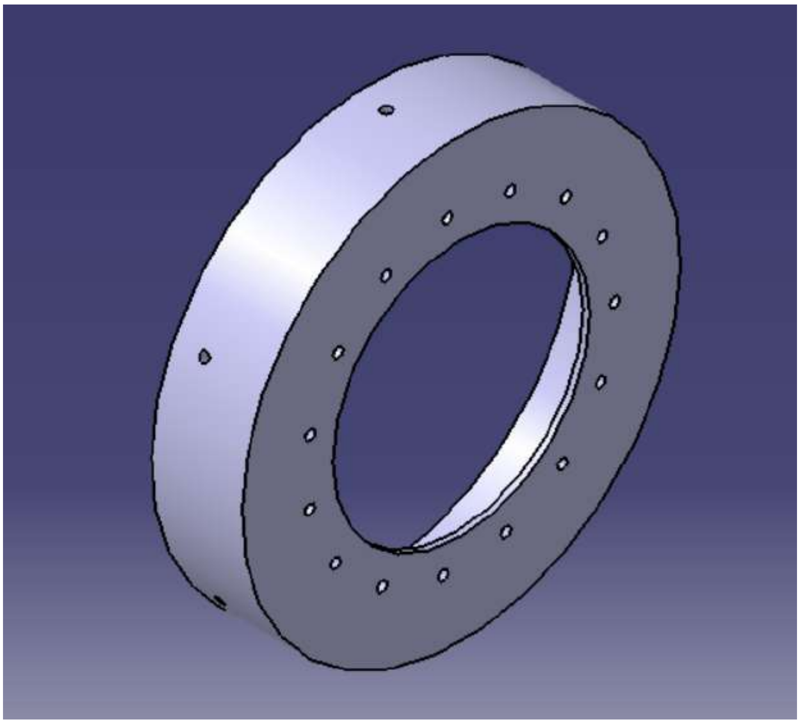

Figure 4.8: Inlet isometric view

4.2.3 Compressor shroud

This design is based upon the exducer and inducer diameters of the compressor. It was designed applying the multi-section solid, utilizing the sketches presented in figure 4.9. The inner circumferences were drawn taking into account the 0.3 millimeters of tolerance gap between the rotor and the inside walls of the compressor shroud. The outside circumferences were designed, when applying the multi-section solid and removing solid function, to grant the 2 millimeters of thickness as recommended [29]. The circumferences height was drawn in relation to the compressor blades height and the exducer height. The figure 4.10, describes the circumferences sketch with height measures. Furthermore, the holes were made after the solid product was designed, with the correct diameter for the fixing to the diffuser. The design of the compressor shroud is shown in figure 4.13 along with the side view, figure 4.11 and back view, figure 4.12.

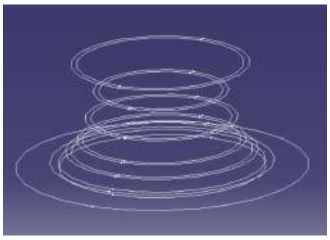

Figure 4.9: Compressor Shroud sketch

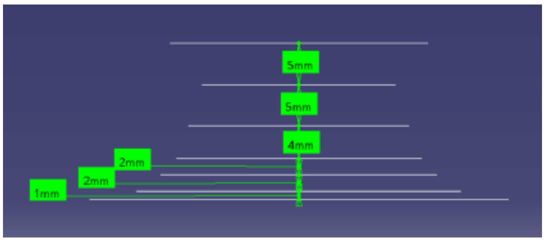

Figure 4.10: Compressor shroud design planes height

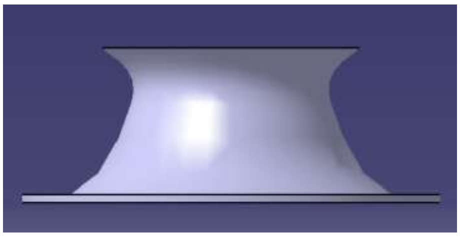

Figure 4.11: Compressor Shroud side view

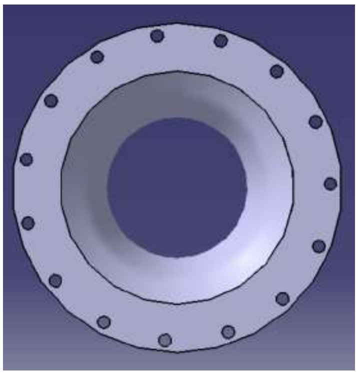

Figure 4.12: Compressor Shroud back view

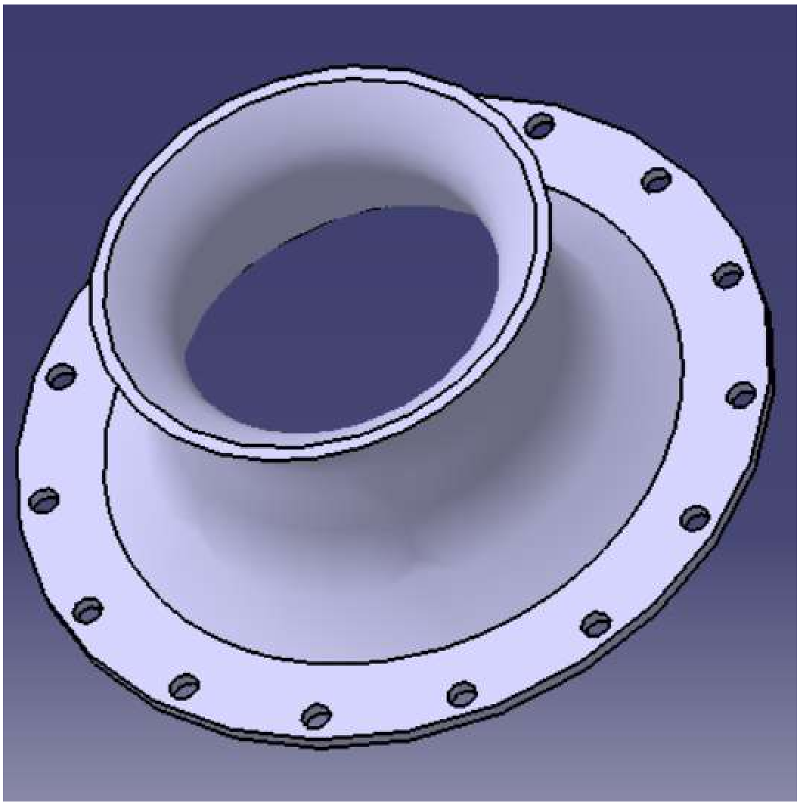

Figure 4.13: Compressor Shroud isometric view

4.2.4 Diffuser

The starting point for this piece is a small cylinder, where the axial and wedged-shape blades are created, respecting the dimensions and angles advised from the previously referred Thomas Kamps's book, in chapter 3. The basis was designed with the space to allocate the compressor, via shaft operation, followed by the wedged-shape and axial blades design, as demonstrated in figure 4.15 and figure 4.16.

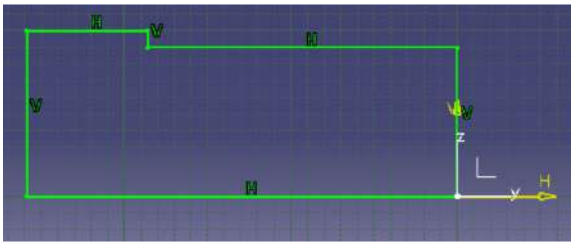

Figure 4.14: Diffuser Base

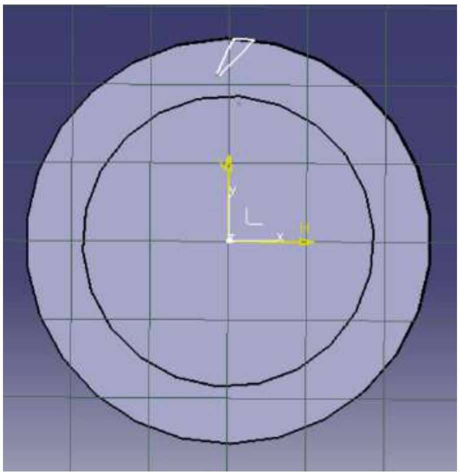

Figure 4.15: Diffuser Wegded-shape blade

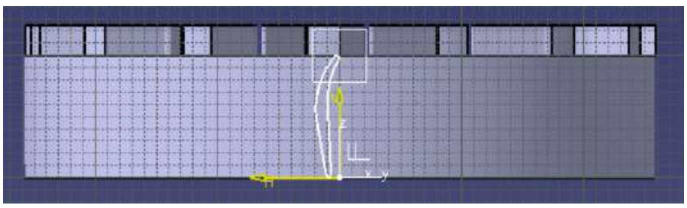

Figure 4.16: Diffuser axial blade airfoil

The wedged-shape blades, demonstrated in figure 4.15, were given a gap of 1.15 times the compressor wheel diameter [29]. The diffuser, figure 4.17 was finalized by creating the holes to fix it to the shaft housing and outer casing.

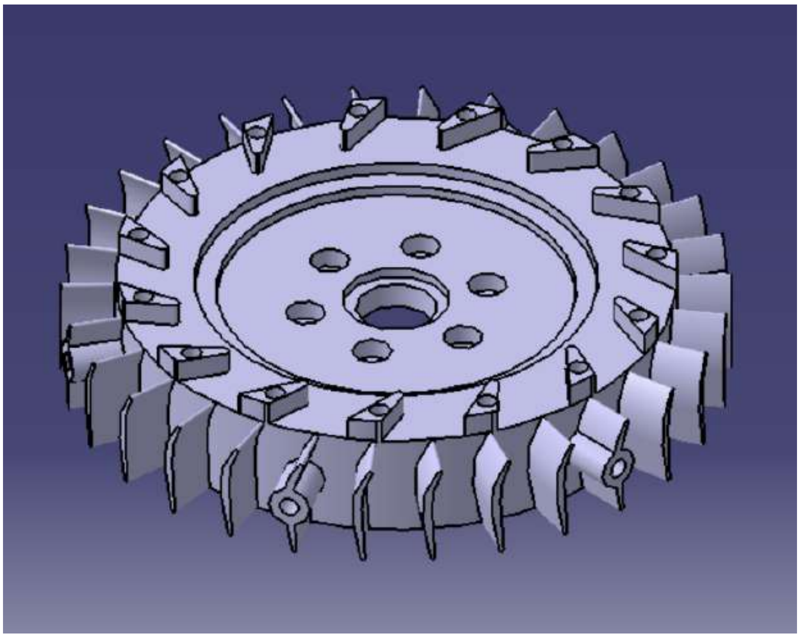

Figure 4.17: Diffuser isometric view

4.2.5 Shaft and Shaft Housing

The shaft housing was the first to be designed as a consequence of the completed designs of the NGV, CC, and diffuser. The objective was only to design a rigid structure connecting the NGV with the diffuser, obeying to the general design of other housings structures, used for the construction of a small dimensions' turbojet. First, was opted for a straighter design, since it was a more convenient design to fix to the diffuser or NGV, using screws tightened by nuts, as the affixing elements. Having the design thought through, the draft was carried out. It consists of two thick rings, joined by a compact rod, designed by the use of shaft operation. This operation was implemented to the sketch demonstrated in figure 4.18.

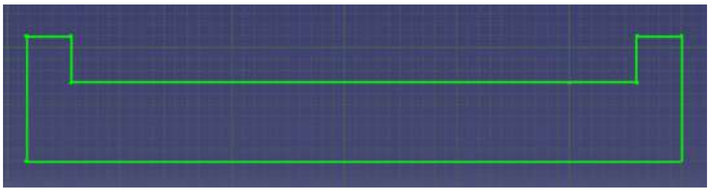

Figure 4.18: Shaft Housing sketch

Subsequently, holes were made, to fix the housing to the diffuser and NGV, as well as, to give the space for the shaft to permeate with the bearings. The inside of the housing, at the turbine side, was extended to confer space for a string and a sleeve. The second part, the shaft, was also created through the use of the shaft operation, applied to the following sketch, illustrated in figure 4.19.

Figure 4.19: Shaft sketch

The length of the shaft was obtained, when all the parts, traversed by the shaft, were designed and assembled at the correct positions. The assembly mode was used and the necessary length for the shaft was measured. Afterward, the design of the actual component was completed, shown in Appendix C.

4.2.6 Combustion Chamber

This part is simple to outline, consisting of two tubes with a total of 132 holes, where 72 holes are distributed in the region of the primary zone, 48 holes in the intermediate zone and the remaining 12 holes in the dilution zone. The first step to take was to obtain the outer flame tube diameter and its length. For the inner flame tube, the same procedure was followed, although, to its design was added the cover for one end of the combustion chamber, as well as, support for the fuel distributor ring. The design was completed through the use of shaft operation, based on their initial sketches, demonstrated in figure 4.44 and figure 4.43. The support for the fuel ring distributer was further added to the design, with a suitable diameter to be later welded, demonstrated in figure 4.22 with an overview of the outer flame tube, figure 4.23.

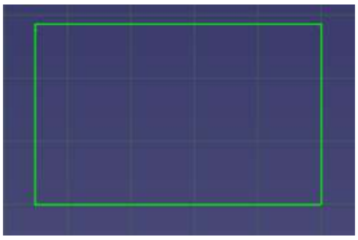

Figure 4.20: Outer Flame tube sketch

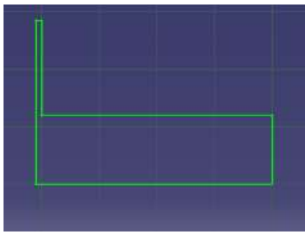

Figure 4.21: Inner Flame tube sketch

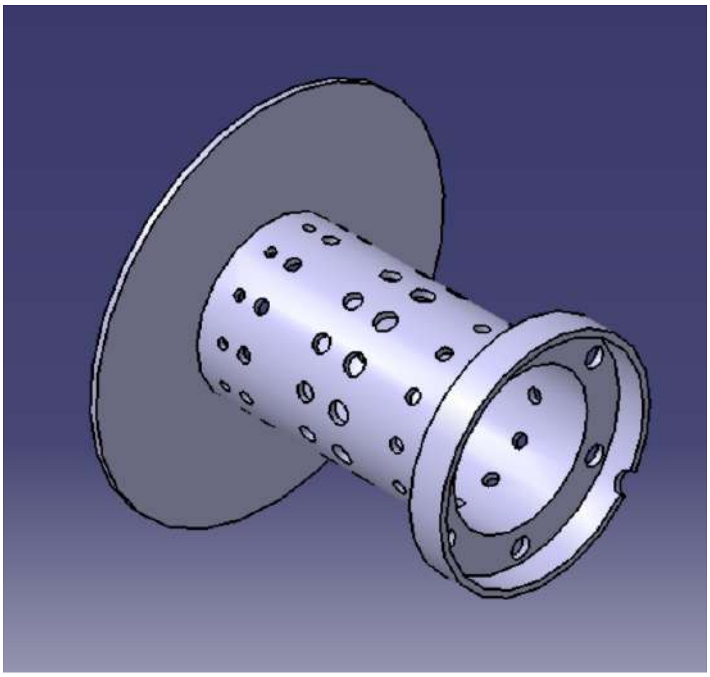

Figure 4.22: Inner Flame tube isometric view

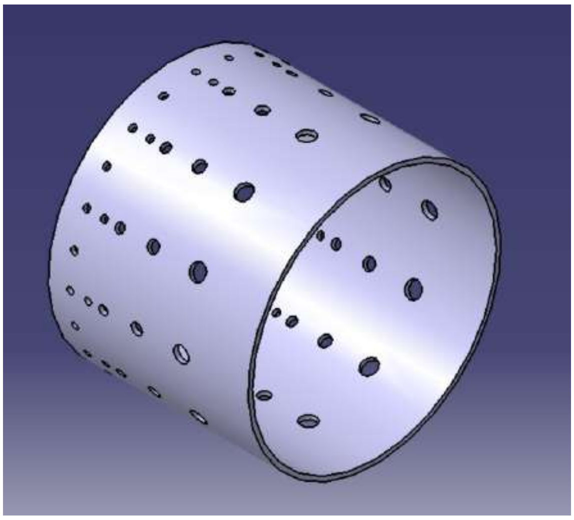

Figure 4.23: Outer Flame tube isometric view

4.2.7 Fuel Distributer

This design was started by drawing a tube, with 3 millimeters of diameter. Having the tube, were created 6 needles, using the pad and pocket function of the CAD software, for the distribution of fuel to the vaporization tubes. The needles are a similar representation of the real ones, further used for the construction of the small turbojet.An image of the fuel ring is presented in figure 4.24.

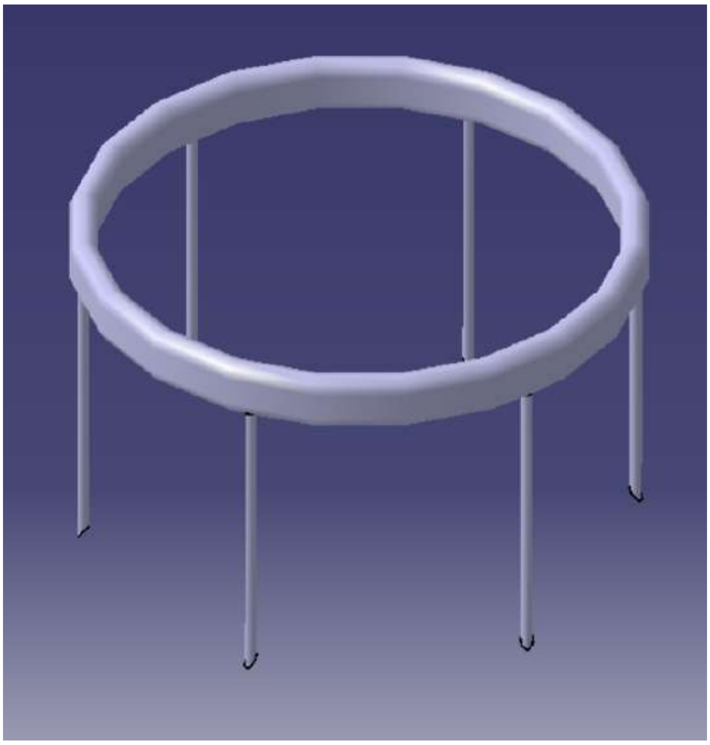

Figure 4.24: Fuel Ring

4.2.8 Nozzle Guide Vane System

This part was designed according to the NGV system observed from an online video [38]. This design was made as a single part rather than in different parts with an assembly. First, it was created the base, where one end of the shaft housing will be fixed to. The part that surrounds the base, and the guide vanes, was created through the use of the multi-section and removing solid function, in the sketch presented in the figure 4.25.

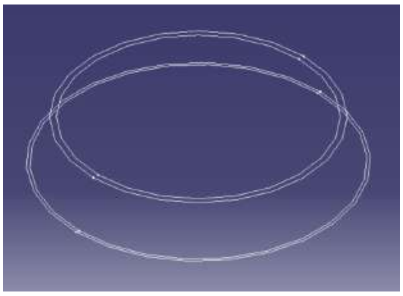

Figure 4.25: NGV system

The design of the turbine blades was similar to the blades design process of the compressor and turbine blades. The only alteration was in the planned positioning chosen to draw the sketch. There are two sketches, in which, one, is connected to the turbine's base, whereas, the second, is connected to a washer. The washer is the final piece designed, positioned to be aligned with the surrounding, top part of the NGV system. Having the solid design, the holes were made to allow the bolts to fix to the shaft housing, as well as, the spacer at the turbine side. The end-piece is demonstrated in figure 4.26 and figure 4.27.

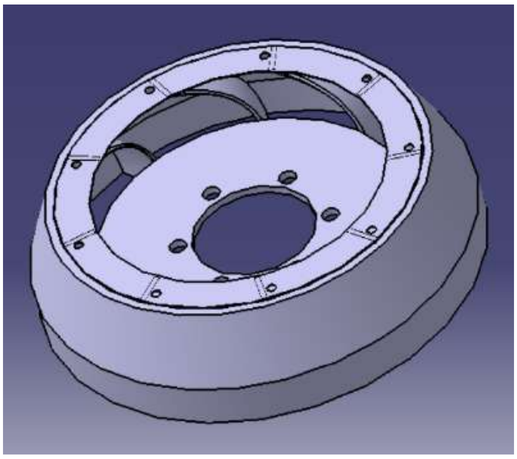

Figure 4.26: Nozzle Guide Vanes system view

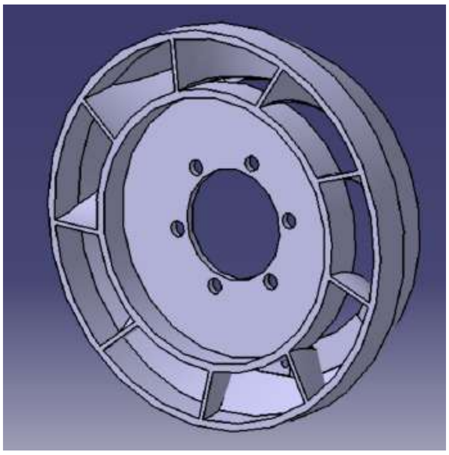

Figure 4.27: Nozzle Guide Vanes system view

4.2.9 Turbine

The procedure to attempt to reproduce the turbine, of the turbo RHB31 VZ21, was similar to the compressor design. Firstly, it was designed a basis for the turbine vanes, respecting the inducer diameter, as shown in figure 4.28. The second step to take was to design the sketch of the vanes from a visual perspective. The design of the vanes was divided into two sketches, attending to two different heights, the inducer blade height and exducer blade height, as it is demonstrated in figure 4.29.

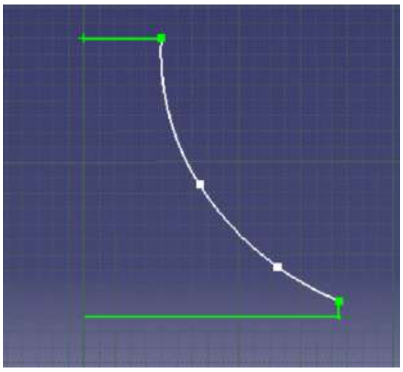

Figure 4.28: Turbine shaft operation sketch

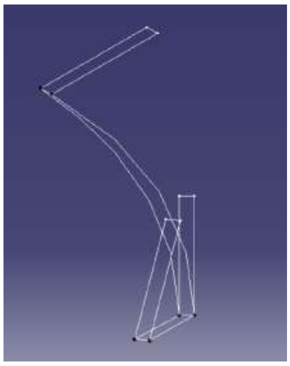

Figure 4.29: Turbine vane sketch

The design was concluded by making use of the shaft and multi-section solid function, in the respective order. The final product is illustrated in figure 4.30.

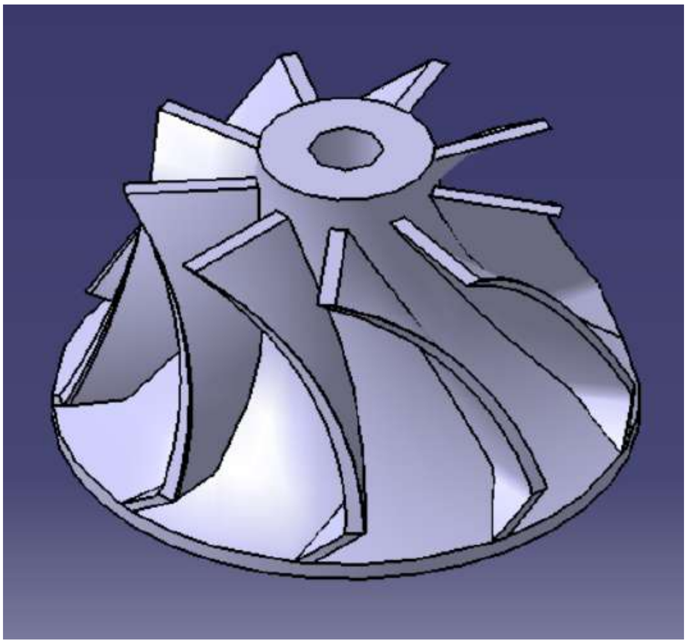

Figure 4.30: Turbine isometric view

4.2.10 Exhaust Nozzle, Outer Casing, and Bearings

The turbine was measured from the exducer and inducer diameter, as, the height of the blade, in order to design an exhaust nozzle based on the referred diameters and height, starting with the sketches of the circumferences. Succeeded by the use of the remove and multi-solid function, applied to the sketches represented in figure 4.31. Moreover, a round piece was added to the base of the exhaust nozzle to act as a connector, allowing the bolts to pass and fix the set NGV system, exhaust nozzle, and outer casing together. The end product is illustrated in figure 4.32.

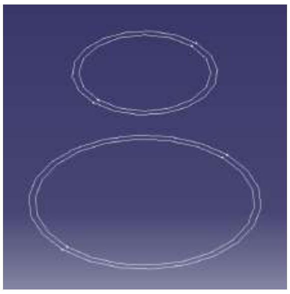

Figure 4.31: Exhaust Nozzle sketch

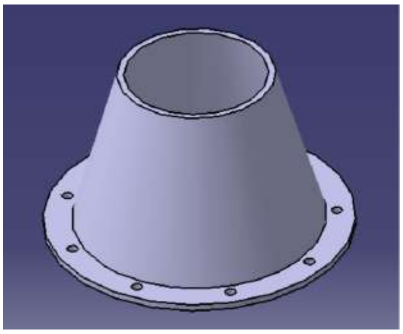

Figure 4.32: Exhaust Nozzle isometric view

The outer casing was the easiest component to design since it has the same design procedure as the combustion chamber. It is demonstrated in figure 4.33.

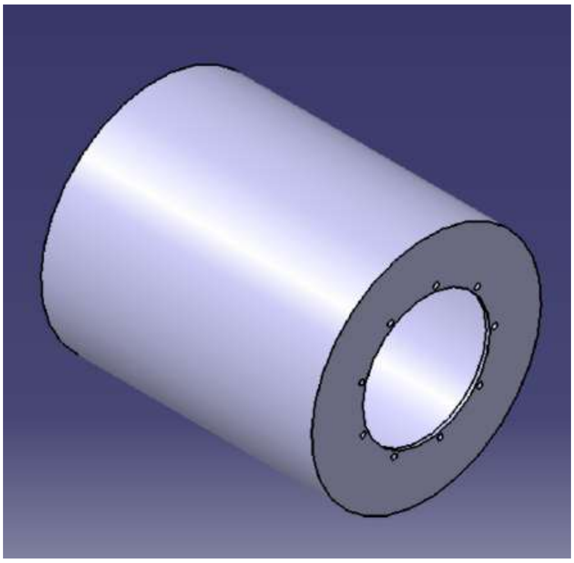

Figure 4.33: Outer casing isometric view

The final component, bearings, was designed in agreement with the given dimensions by the SKF's company site in regard of the model 618/5 [39].The first step taken was the design of the rings surrounding the spheres, using the sketch described in figure 4.34 for a shaft operation. The next and final step was the design of the spheres. This was, as well, a design for a shaft operation applied to the sketch in figure 4.35, followed by a circle pattern, that multiplies the spheres to nine, rightly positioned. The reproduction of the bearing 618/5 is displayed in figure 4.36.

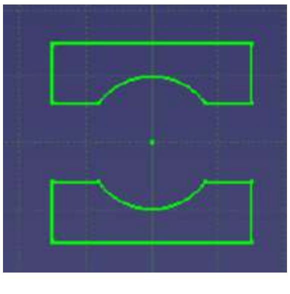

Figure 4.34: Bearing casing sketch

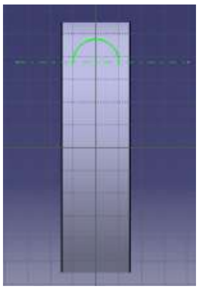

Figure 4.35: Bearing spheres sketch

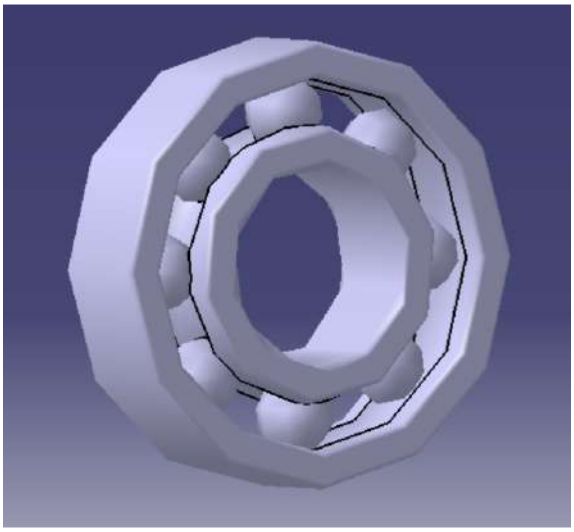

Figure 4.36: Bearing 618/5 example

4.3 Components Manufacturing

At this stage, with the concept design of the several components accomplished, it was initiated the manufacturing process. This process involves a series of different machines and steps taken, to produce the desired pieces. The raw material adopted, for the majority of the components, was stainless steel, in sheet form, owing to the physical properties of this type of steel, like resistance to corrosion, extreme temperatures and high strength [41]. Additionally, the use of this material was often found in the literature reviewed, as the material for certain components of small-scale turbojets. Other raw material opted for, was cast aluminum, characterized by its lightweight, high strength-to-weight ratio, corrosion-resistant, robust, easy to fabricate and assemble designs [42].

The production phase was started by casting aluminum from the remains of an engine's block that was available for use. The first step was to cut off sufficient aluminum for the casting process. Some pieces of steel scraps were cut and welded to make a cup for the aluminum to be melted, as well as, to make molds and fill them after the forging of the aluminum was completed. For the forging, the furnace was heated as illustrated in figure 4.37, with a torch placed in the bottom hole of it. In order to obtain the lowest, possible quantity of impurities, while the aluminum was being melted, it was applied sodium carbonate, a chemical substance that, when mixed with the aluminum at the liquid phase, pushes most of the material impurities to the top of the casting cup. However, this purification method is not completely efficient. Afterward, the cast aluminum was poured into the molds of steel, previously obtained. The molds were relatively simple to produce, consisting of thick, large, tubular steel with a base under. The molds were chosen with large diameters to give margin for the machining process. When the aluminum cooled off, it was extracted by cutting the molds with a grinding wheel. The cast aluminum is demonstrated, below, in figure 4.38.

Figure 4.37: Furnace

Figure 4.38: Cast aluminium

The manufacturing process is organized in different sections, each one is categorized by the machining procedure used to manufacture the components, explaining, then, how and what pieces were manufactured through the use of the corresponding process.

4.3.1 Shaping Process

The stainless steel was acquired from a metalwork shop, in the form of a sheet with one millimeter. The first attempt working with the stainless-steel sheet was with a thickness of 0.5mm, as it was recommended [29]. Nonetheless, the thickness of the sheet was insufficient for the necessary manufacture methods to be applied, hence the choice for the one-millimeter sheet.

Firstly, the pieces were designed, in their planar shape, according to the dimensions, utilizing CATIA V5. The draft or blueprint was realized due to a saving option that has the file format specification, .dxf. The file format .dxf is necessary, to the software of the water jet cutter, to read the delineated cuts to be made in the stainless steel sheet. A sample of pieces is shown below, in figure 4.39.

Figure 4.39: Samples of the pieces cut with the water jet

The following step was drilling the holes for the combustion chamber, proceeded by the cover of the two rectangular shape pieces, at the left end of figure 4.39, applying a paper glue tape on top of the two pieces. Lines and points were traced in the paper glue tape, with the help of a rule and set square. When the pointing of the drilling holes position was finished, the holes were drilled with a vertical drilling machine, using borers of the following diameters, in millimeters: 0.75, 1, 1.5, 2,2.5,3,3.5. In the first attempt, with the 0.5-millimeter sheet, it was possible to drill the holes but the borers were too fragile due to their reduced thickness. Consequently, throughout the drilling, the less thicker borers broke, despite the appropriate approach when drilling, such as, the use of a puncture to mark the holes in the sheet, for the borer to have a more stable and precise drill. When the borer was showing some resistance to drill the holes, oil was applied to help the drilling. Even with caution while drilling, due to the increased thickness, to one millimeter, the diameter of the holes was altered for 2,2.5,3 and 3.5 millimeters. Figure 4.40 and 4.41 demonstrate the method used to drill the two small sheets and the executed drills.

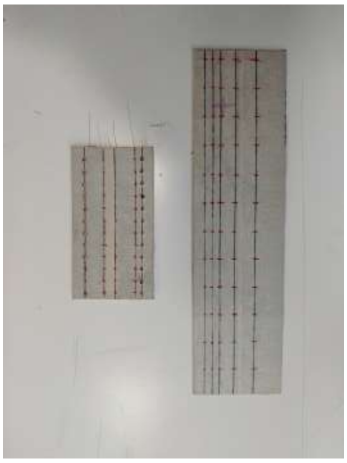

Figure 4.40: Holes pointed in the plain flame tubes

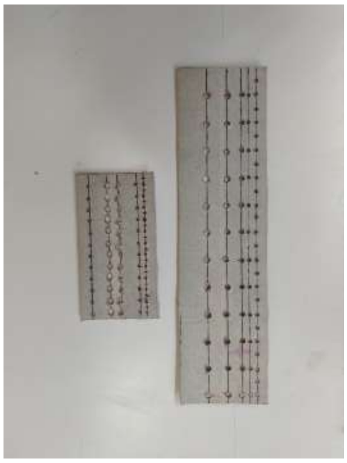

Figure 4.41: Plain flame tubes drilled

The parts made of the stainless steel were molded with the use of a wooden tapping block, wider than the pieces that were worked. This is a requirement for the formation of the piece to be even, without folding them. The pieces must be molded around a tube, resistant enough to not fold with the taps against it. The tube diameter must be slightly smaller than the desired diameter, although, this technique is not completely accurate, which caused the pieces' diameter to differ from the designed components measures. Before the use of a tapping block, a roller was used to form the pieces but, unfortunately, it was not possible to make the pieces round, due to the thickness of the sheet, which did not fit between the rollers. Below, it is demonstrated the hand-shaped parts.

Figure 4.42: Exhaust Nozzle

Figure 4.43: Inner Flame Tube

Figure 4.44: Outer Flame Tube

Figure 4.45: Outer Casing

4.3.2 Milling Process

Turbine

The next step was to dismantle the core of the turbocharger used for this thesis, by detaching the compressor, and, the turbine, from the shaft. The compressor was easily taken off since it was only tightened by a nut. On the other side of the shaft, the turbine was welded by friction to the shaft, which was removed along with the top nut from the turbine, using a grinding wheel. Afterward, the center of the turbine was drilled, performed in a milling machine to obtain a well, executed drill, to fit the designed shaft. Since the turbine material is hardened, a drill bit was used, to pierce the turbine from one end to the other, illustrated in figure 4.46. However, due to ineffective means chosen to drill the turbine, the drill was not centered with the turbine, discarding the future use of this turbine, since the use of this turbine, at high speeds, would turn the small gap into an enormous, leading to a serious accident, compromising the lives around the turbine. The optimal drill method would be to preserve the shaft, attached to the turbine, and, fix it, in the correct placement of a lathe machine, assuring, in this way, the drill will be centered.

Figure 4.46: Turbine drilled

Nozzle Guide Vane system

The nozzle guide vane system was based on the procedure of an experienced person in making small model turbojets, found online at [38]. The surrounding parts of the NGV were cut and welded. Although, to complete this system blades were cut from another turbine, in which a grinding wheel was used to separate, individually, the blades from the rotor. Later, some width of the blades was removed to fit in the between in the surrounding parts and the turbine's base. The blades taken from the other turbine are shown in figure 4.47.

Figure 4.47: Blades of the nozzle guide vane system

The final part to fabricate this component was to weld the blades between the two metal pieces, demonstrated in figure 4.48 and figure 4.49. Unfortunately, it was not possible to fix the blades in this component due to the welding technique that released too much heat considering the size and thickness of the component.

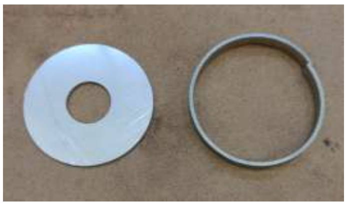

Figure 4.48: Turbine's base

Figure 4.49: Surrounding and upper part of the nozzle guide vane system

The washer in figure 4.48 is the turbine's base that with the rectangular next to it, forms the inner part of the nozzle guide vane system, where a segment of the blades would be welded to. Figure 4.49 illustrates the three external parts of the system, a lower part that surrounds the blades, a conic piece, that it is put on top of the lower piece, and, then, the washer welded on the upper base of the conic piece.

4.3.3 Lathe Process

Aluminum

The aluminum components were worked differently from the rest of the components. This material was machined with the use of a lathe machine. The blueprints of the aluminum components were printed, and only then the cast material was machined. The blocks were put in the lathe machine, where they were fixed in the appropriate setting. Afterward, the cast aluminum was machined, removing the necessary material to acquire the design with the correct form and dimensions. However, it is extremely difficult to work the pieces and obtain an exact, physical copy of the design, because it is not possible to know if the lathe machine is removing the right amount of material, even with the most caution while working with it. The components worked with the lathe machine were the shaft housing and the inlet flange, indicated in figure 4.50 and figure 4.51, respectively. The compressor shroud and diffuser would have been made in the vertical machining center, at UBI's FABLAB. Unluckily, it was not possible to produce these two pieces due to their small-scale dimensions, requiring extremely precise machinery.

Figure 4.50: Shaft Housing

Figure 4.51: Inlet Flange

Stainless steel

Another piece worked with the lathe machine was the shaft. This component is made of a stainless-steel, solid cylinder, that was machined, respecting the dimensions obtained for the shaft manufacture.However, the recommended material is a steel screw with a tensile grade of 12.9 or 10.6 because this component has to sustain immense tension when the turbine and compressor are working [29]. Figure 4.52 represents the shaft obtained after the explained manufacture process.

Figure 4.52: Shaft

4.3.4 Brazing Process

The ring support was drilled six times, with the help of a transfer and a puncture, the position of the holes was marked. Subsequently, the drill was made with a three-millimeter borer, so the vaporization tubes could fit within the holes. These tubes were cut from stainless steel with three millimeters of diameter. The small tubes were brazed to the fuel ring support, for the posterior placement of the fuel ring distributor, as shown in figure 4.53. The distributor is made of the same tube as the vaporization tubes, which had to be shaped into a small circle to fit in the ring support. A steel wire was put across the whole length of the tube to not crease the tube, compromising an equivalent distribution of fuel. The methods used to shape the tube were applied with extreme caution to not crease the tube. In spite of the caution taken while working with the tube, it could not be molded to the adequate dimensions, as figure 4.54 shows.

 

Figure 4.53: Fuel ring support

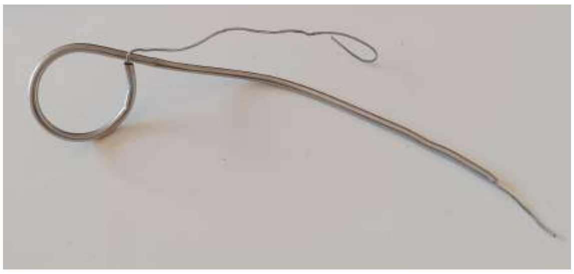

Figure 4.54: Fuel ring

4.3.5 Welding Process

Both ends of the molded pieces were welded, where along this process, only some were put into their definitive shape. The type of welding utilized was the Gas Tungsten Arc Welding or TIG. The welding process was one of the reasons to decide for a thicker stainless-steel sheet. There was the possibility of the heat, provided from the welding, melt part of the pieces, becoming more probable with a thinner sheet. However, even with an increased thickness, it was only possible to weld the components shown in figures below. The remaining components could not be welded because both ends of the pieces had to been put together without a single air breach. Sandpapers were used to the pieces' ends, wearing out the uneven parts, to prevent an air breach when joining the opposites. Nonetheless, it was not feasible to weld all the pieces, as it is demonstrated.

Welded pieces:

Figure 4.55: Exhaust Nozzle

Figure 4.56: Outer Casing

Figure 4.57: Inner Flame Tube Front View

Figure 4.58: Inner Flame Tube Back View

Figure 4.59: Outer Flame Tube

Figure 4.60: Nozzle Guide Vane external parts

Figure 4.61: Internal part of the nozzle guide vane system

Chapter 5

Conclusion

This thesis had the main objective of building a small turbojet engine, in which a study of the turbojet's components, and working cycle, was made at first. The study of the jet engine was already a challenge because the vast information that is available, makes it difficult to choose what is the most important concepts that one should know before developing a turbojet.
This dissertation gave the author the opportunity to learn part of the science behind this engine, as well as, the practical knowledge of how to build a turbojet, which is something it is not learnt everyday, and surely, not possible to learn in a classroom. This dissertation achieved one of the two objectives proposed. The design was concluded, in which an explanation is given to, so a future work could be realized, based on the steps taken. Sadly, the construction of this engine was not accomplished for lack of means to proceed with an adequate manufacture.

5.1 Drawbacks

The design of this project was mostly based on Thomas Kamps' book [29] and the author is grateful for its existence. Information about how to build a turbojet, especially with radial compressor/turbine set, was not possible to find, except in the books of Kurt Shreckling and the other stated previously. Although, their engines did not involve a radial turbine, but an axial one. These two books give details of the manufacture process of a turbojet engine, with all the procedures explicitly described. Since this engine is based on empirical data, the dimensions are not definitive and one can not assure if the engine is with the proper dimensions for the given compressor. Only with experience in modeling jet engines and testing them, that one could assure the dimensions of the components allow the engine to run without problems.

Throughout the experimental phase many obstacles were faced in the attempt of building this turbojet. The first one was the thickness increase of the stainless steel sheet. This had a significant impact on the manufacture of the pieces, since it made harder to hand-shape the pieces, being impossible to adjust them to the appropriated dimension of the designed components. As a consequence of the sheet's thickness, a roller could not be used. This would had been extremely helpful due to a more accurate process of rolling the pieces, as well as, to facilitate the welding, since with use of a roller would result in consistent, aligned pieces.
In regard to the welding process, the most critical obstacle, could not be adequately executed, because the technique, previously referred in chapter 4.3, did not allow the pieces to be put as desired, preventing the achievement of the main goal for this thesis. Despite of the technique, the size of the pieces required an extraordinary precision of manufacture, in which, for a beginner in experimental tasks, was really difficult to comply with.

5.2 Future works and recommendations

Further works based on this dissertation can be developed due to the wide range of themes the turbojet involves. Since the manufacture of this engine was not concluded, the first possible work could be the finalization of this project and, if possible, a stereolithographic file of the compressor should be generated for an accurate design of the compressor shroud, respecting the compressor curvature, which is an important factor for an efficient induction of air, thus, compression and engine's functioning.
For the testing of the engine, the author recommends a shaft calibration, if possible. Additionally, an axial turbine should be developed for this type of engine. It would facilitate the process of manufacturing, such as,the nozzle guide vanes system production.

The future works following this dissertation could be:

- Finish the development of this engine with the appropriate techniques.
- Study and fabrication of an axial turbine to match the compressor.
- Realize a computational fluid dynamics study of the jet engine's airflow and check for improvements to be made in the designed components.
- Develop a testing workbench.

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Appendixes

Appendix A - Part List

Table 1: Part List

Part No.	Description	No. Off	Material/Type	Blank Dimensions/Notes
1	Compressor Shroud	0	Aluminium	
2	Compressor Wheel	1	Cast Aluminium	Possible alloys: Al-Si-Cu-Mg
3	Diffuser	0	Aluminium	
4	Spacer Disk	2	Stainless Steel/Aluminium	
5	Bearing	2	618/5	Deep groove ball bearings
6	Engine shaft	1	Stainless Steel	Turned
7	Shaft Tunnel	1	ALuminium	Turned
8	Combustion Camber	1	Stainless Steel	TIG Welding
8.1	Outer section	1	Stainless Steel	Sheet, 1mm thick
8.2	Inner section	1	Stainless Steel	Sheet, 1mm thick
8.3	Front Section	1	Stainless Steel	Sheet, 1mm thick
8.4	Rear Section	1	Stainless Steel	Sheet, 1mm thick
8.5	Fuel Ring Support	1	Stainless Steel	Sheet, 1mm thick
8.6	Vaporization Tubes	6	Stainless Steel	ø3 mm tube
9	Injector Ring	1	Stainless Steel	Soldered
9.1	Injector Ring	1	Stainless Steel	ø3 mm tube
9.2	Injector Needle	6	Syring needle	ø0,8 mm
10	Turbine NGV	1	Stainless Steel, Cast Aluminium	Possible alloys: Al-Si-Cu-Mg
10.1	Turbine's base	1	Stainless Steel	Sheet, 1mm thick
10.2	Blades	9	Cast Aluminium	Possible alloys: Al-Si-Cu-Mg
10.3	Blade jacket	1	Stainless Steel	Sheet, 1mm thick
10.4	Turbine jacket	1	Stainless Steel	Sheet, 1mm thick
11	Turbine	1	Cast Alumium	Possible alloys: Al-Si-Cu-Mg
12	Exhaust Nozzle	1	Stainless Steel	Sheet, 1mm thick
12.1	Washer	1	Stainless Steel	Sheet, 1mm thick
13	Casing	1	Stainless Steel	Sheet, 1mm thick
13.1	Casing Rear Section	1	Stainless Steel	Sheet, 1mm thick
14	Inlet Flange	1	Aluminium	Turned
16	Pre-Load Spring	1	Steel	
17	Sleeve	1	Stainless Steel	Sheet, 1mm thick, welded

Appendix B - 2D Draws of the turbojet's components

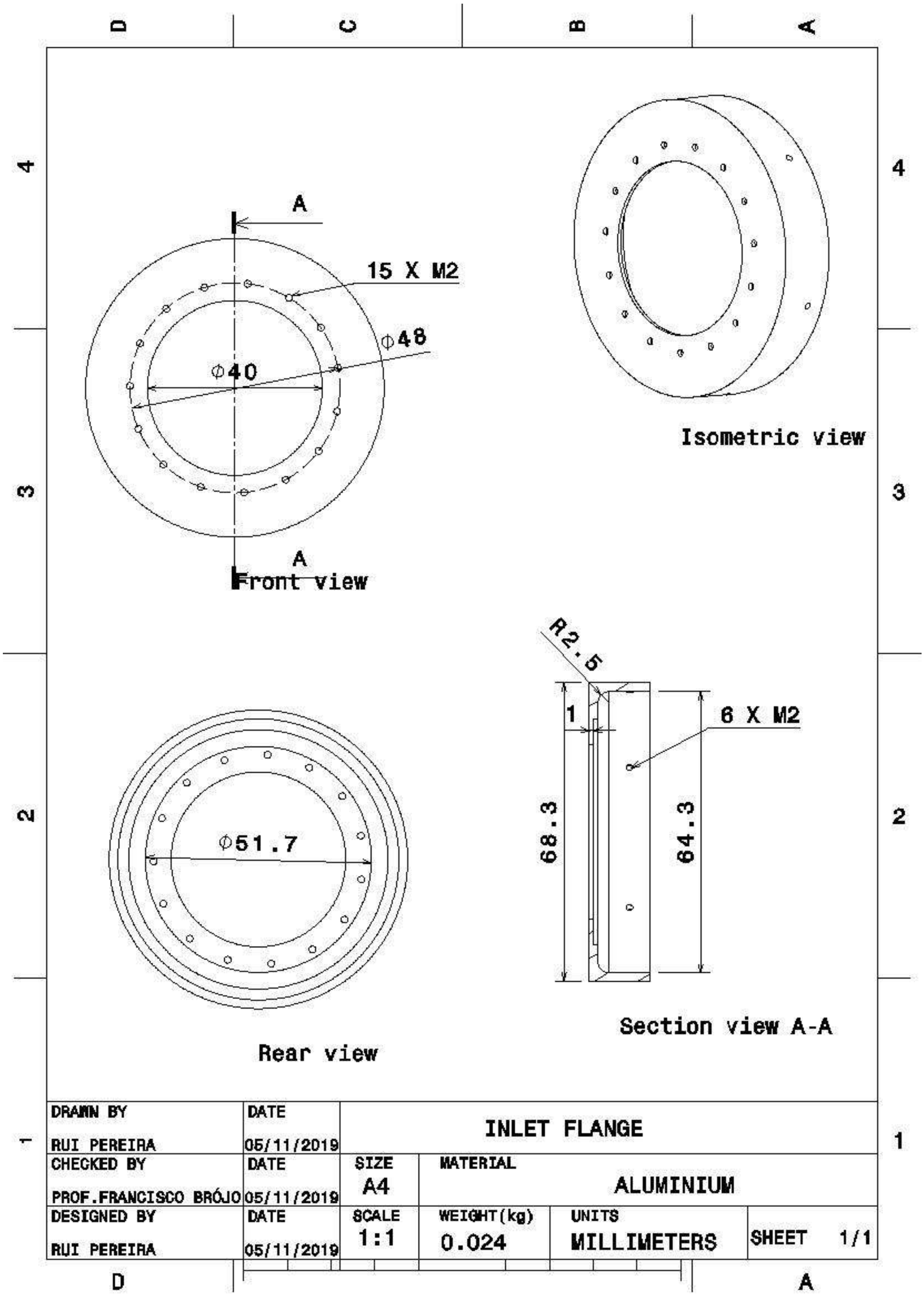

Figure B.1: Inlet Flange

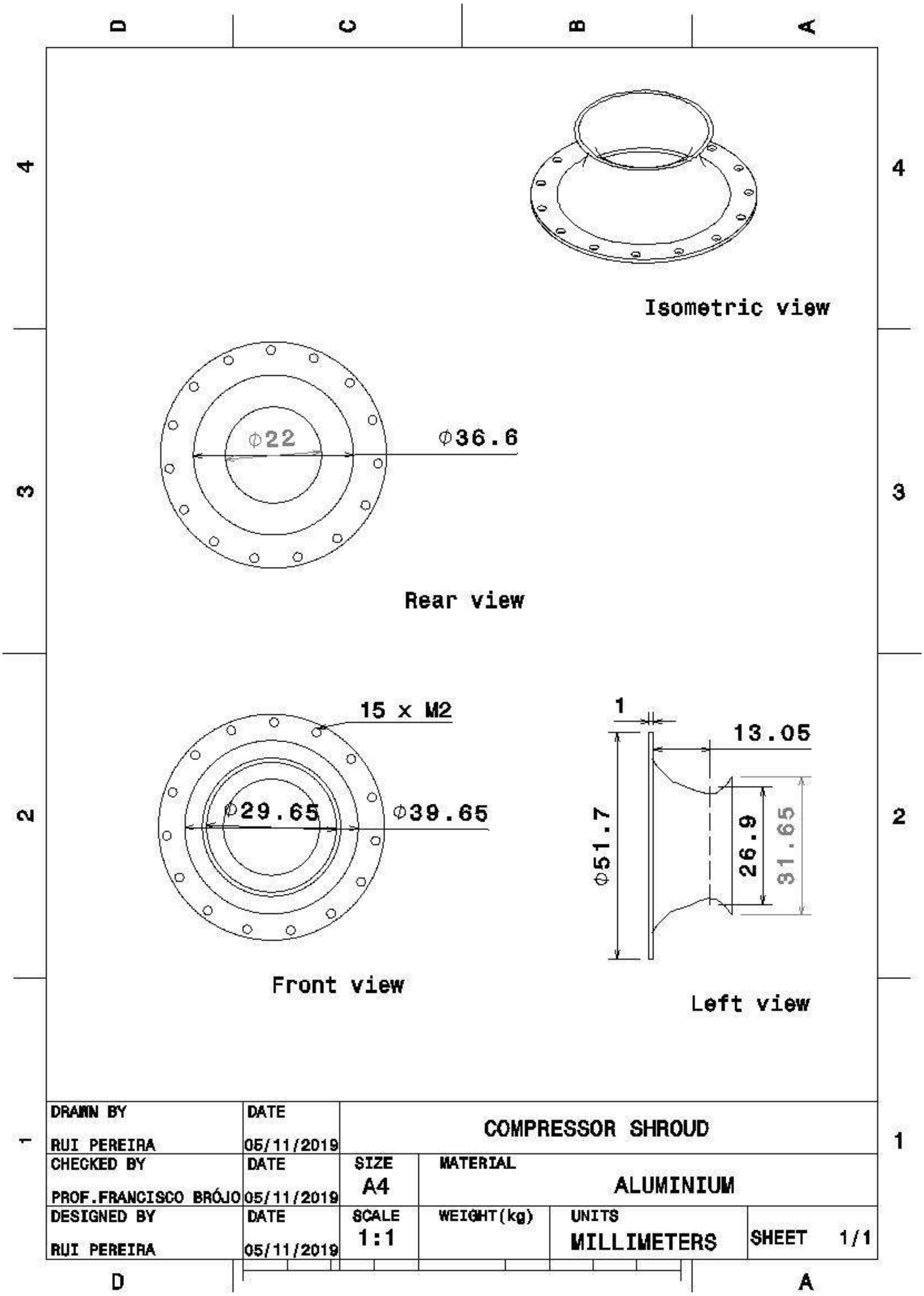

Figure B.2: Compressor Shroud

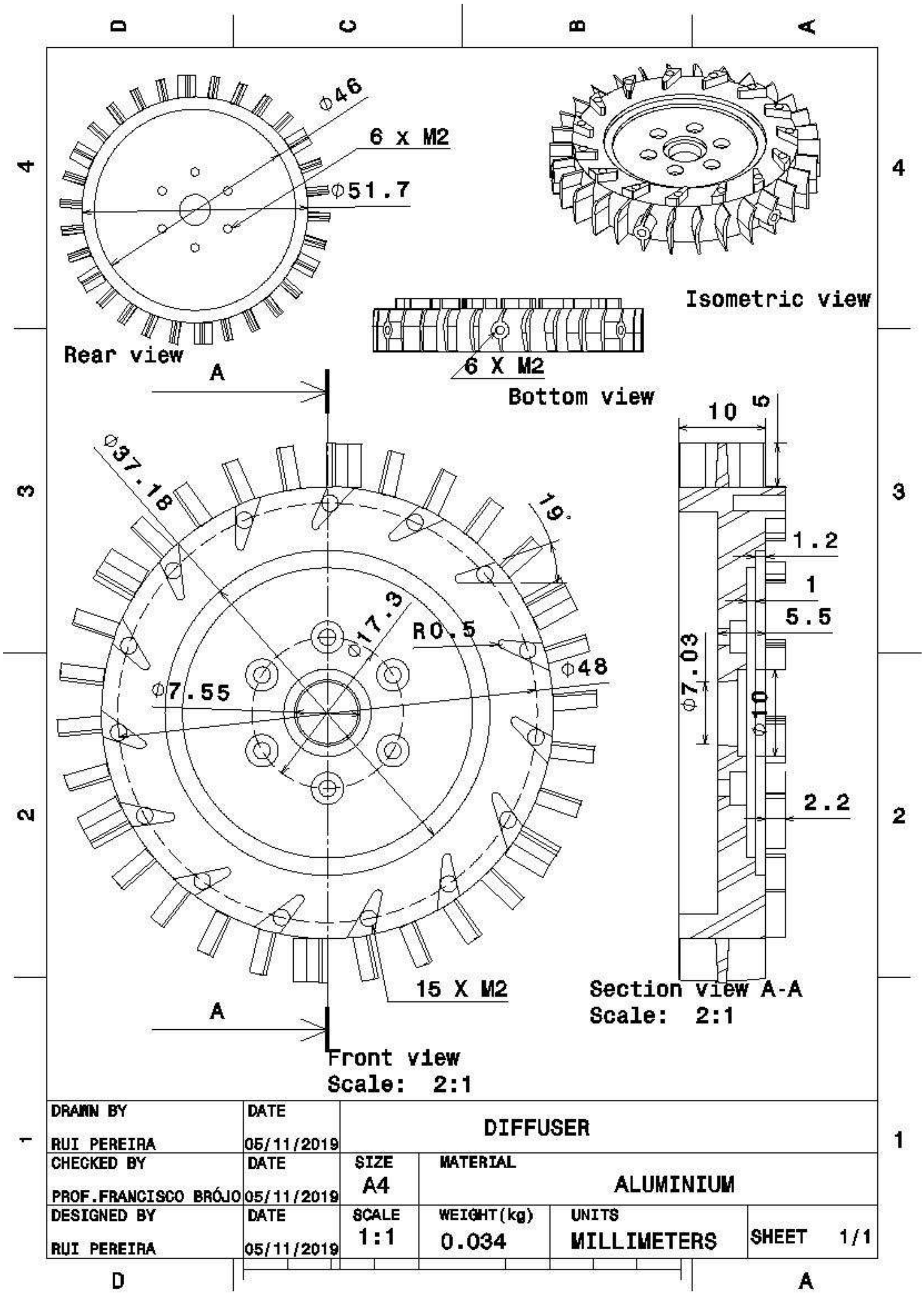

Figure B.3: Diffuser

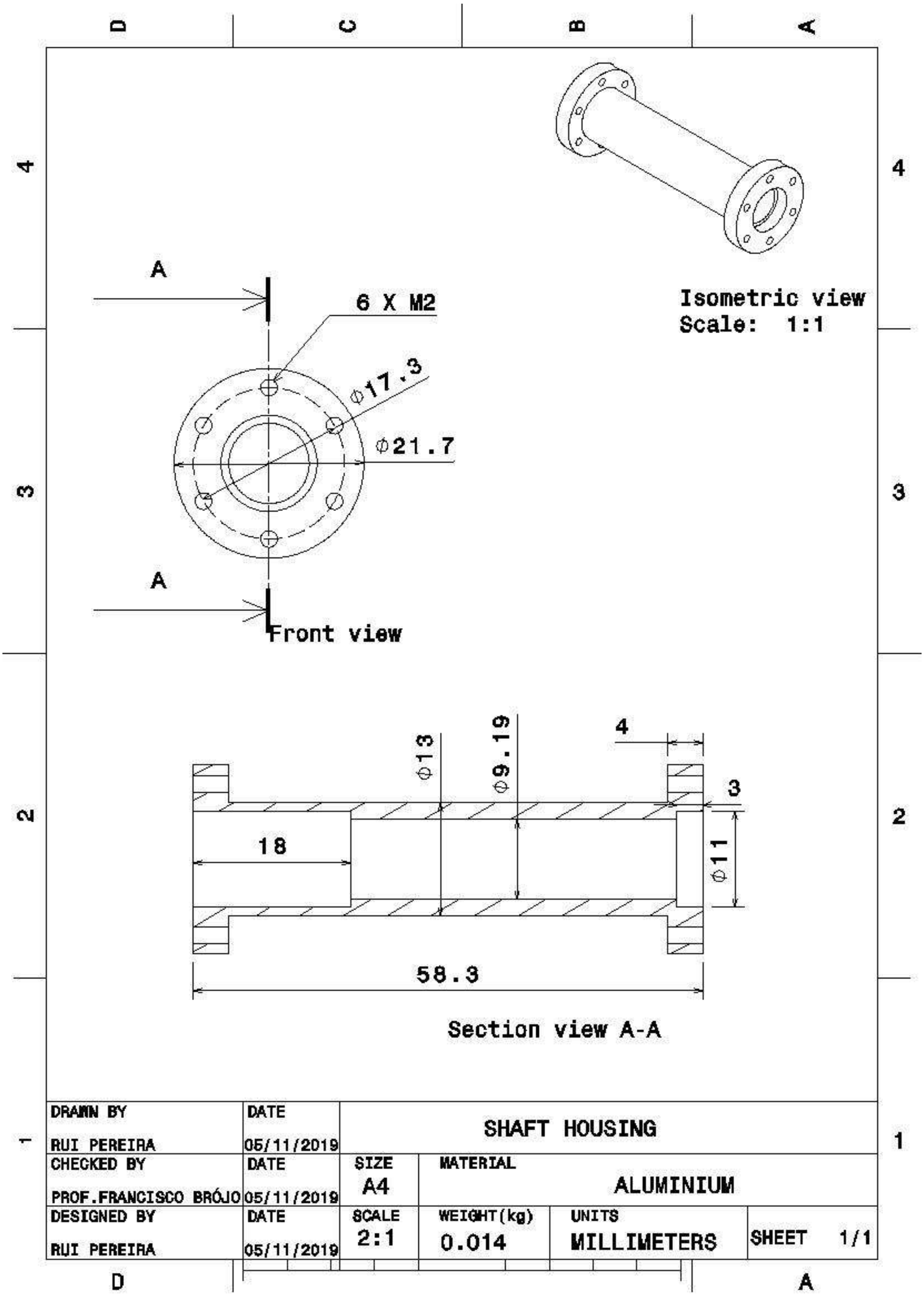

Figure B.4: Shaft Housing

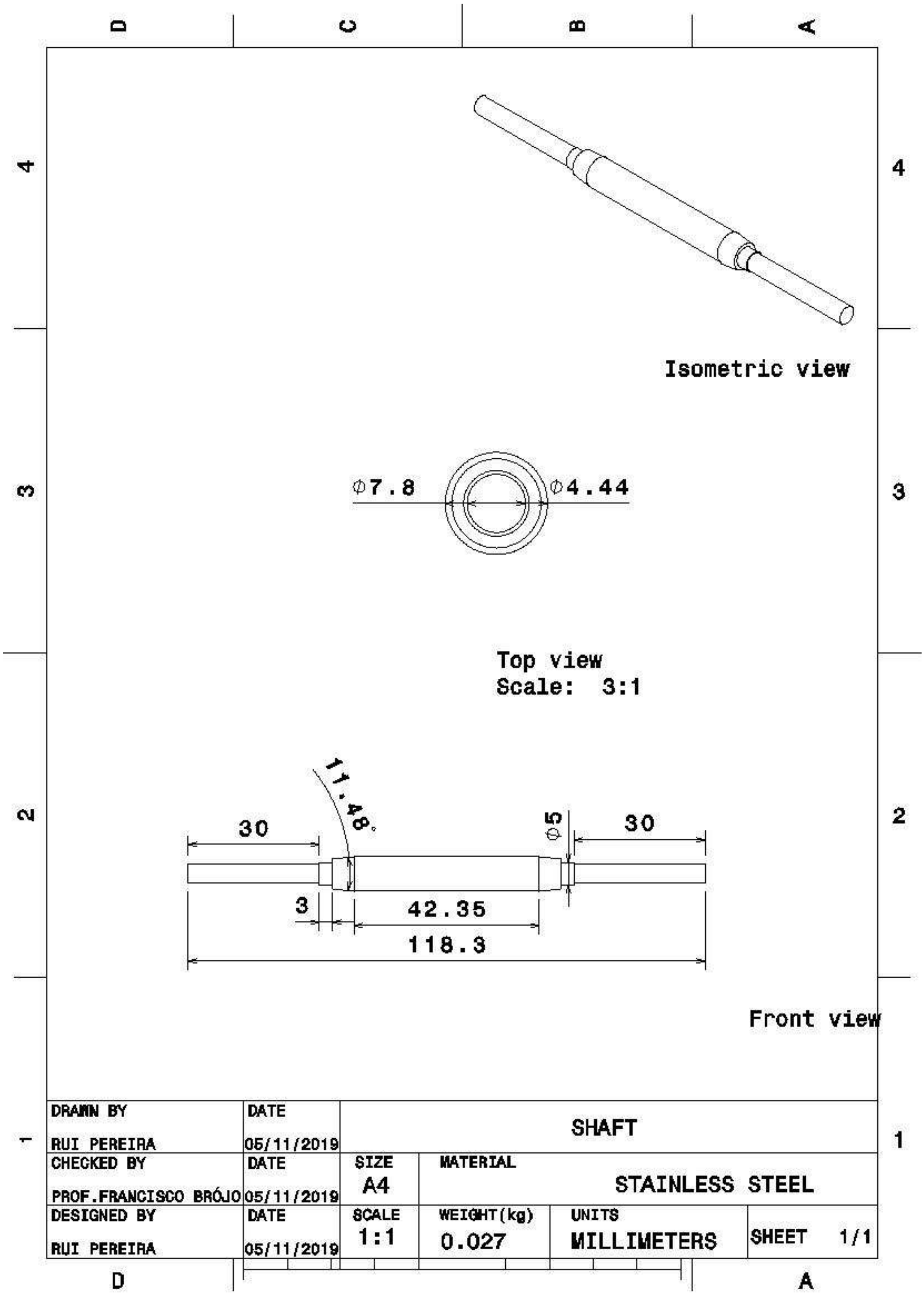

Figure B.5: Shaft

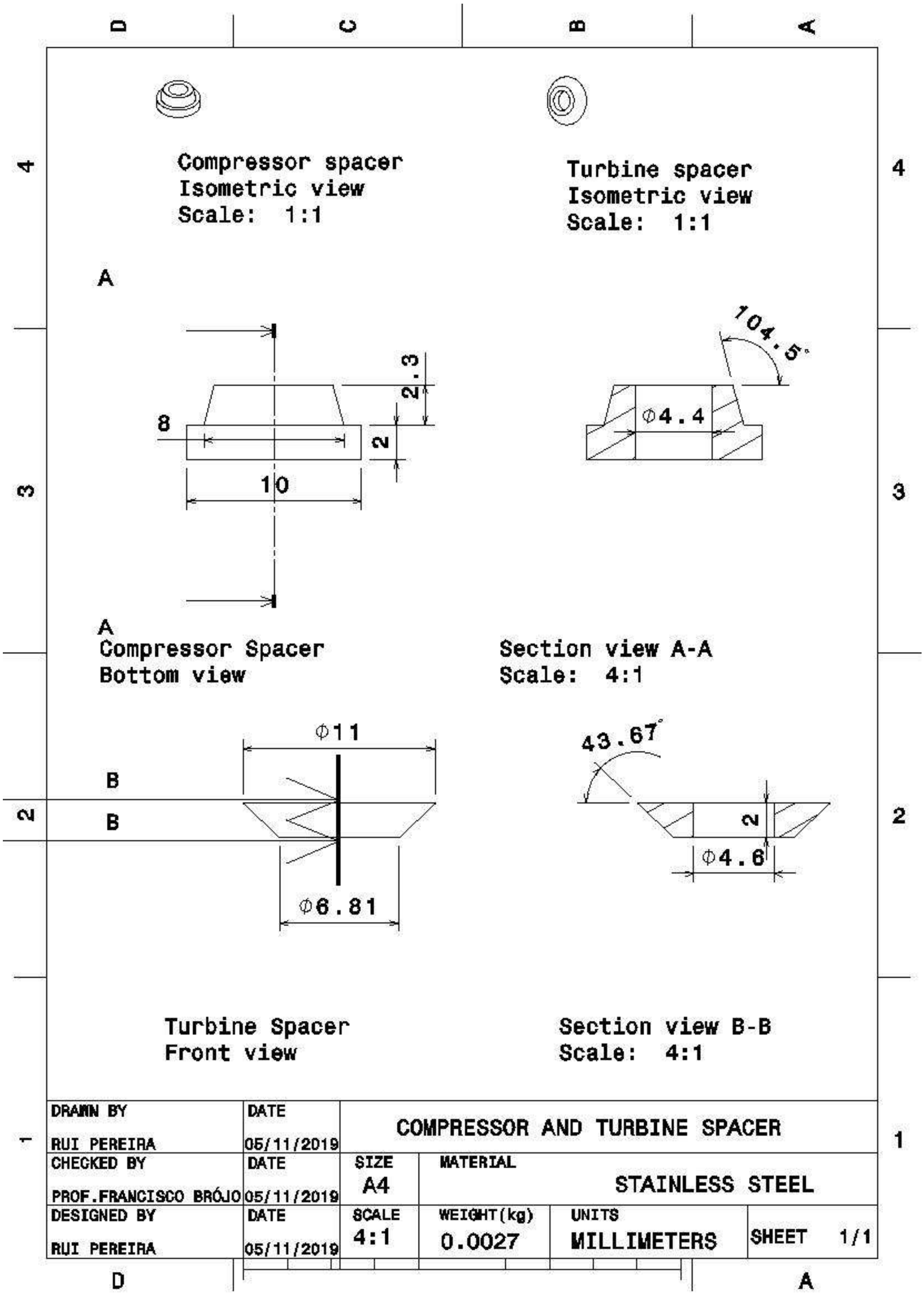

Figure B.6: Spacers

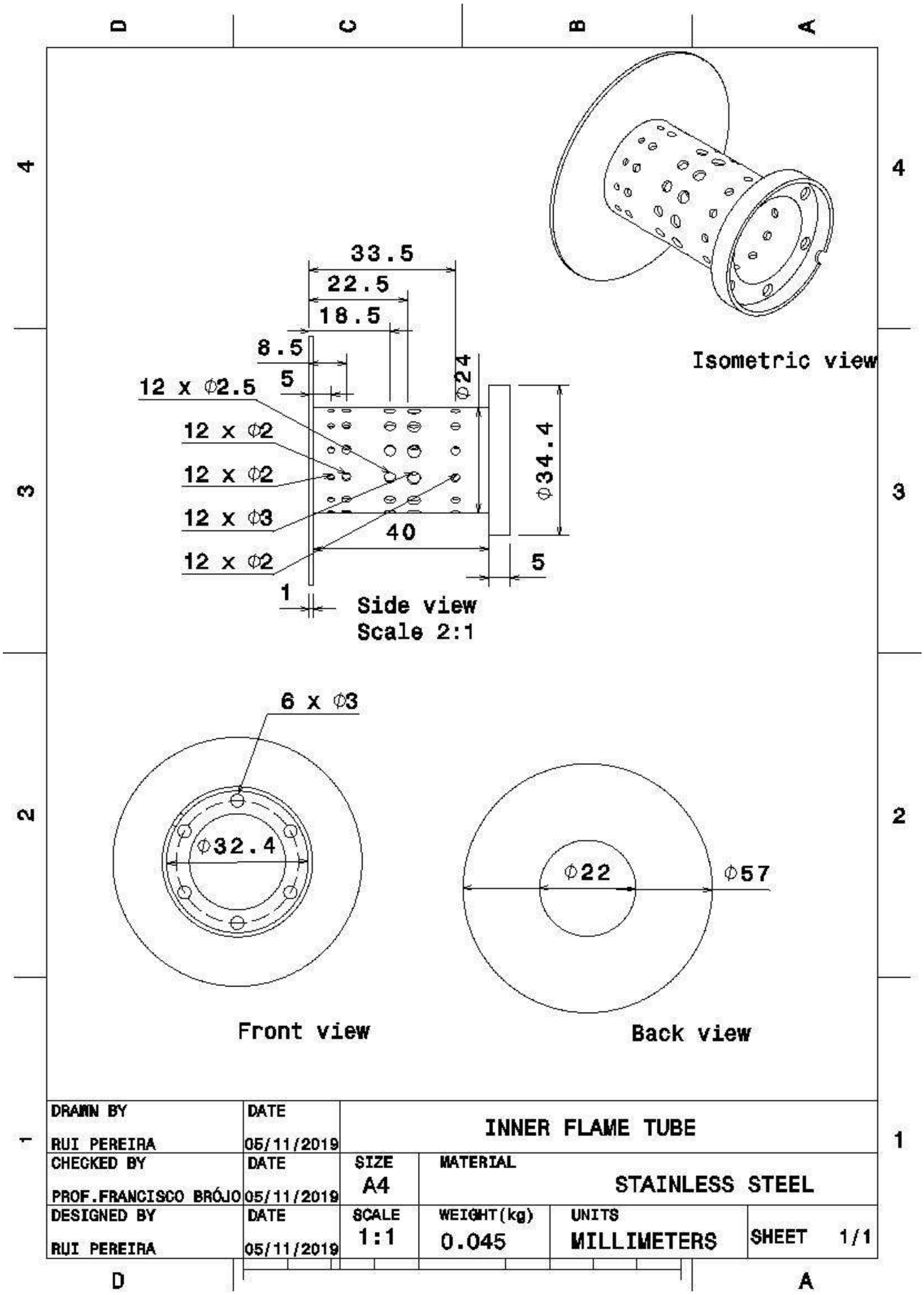

Figure B.7: Inner Flame Tube with fuel ring support

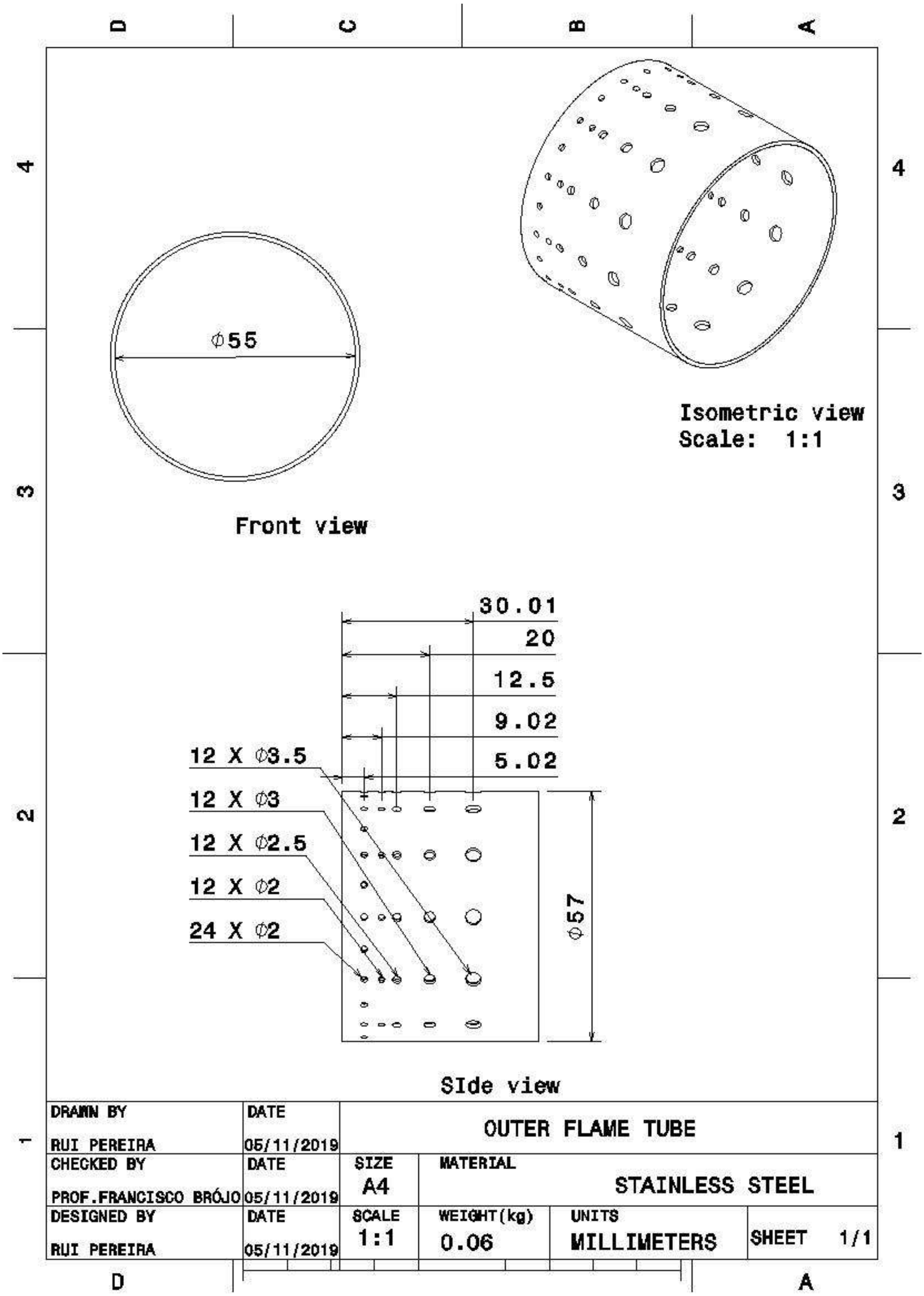

Figure B.8: Outer Flame Tube

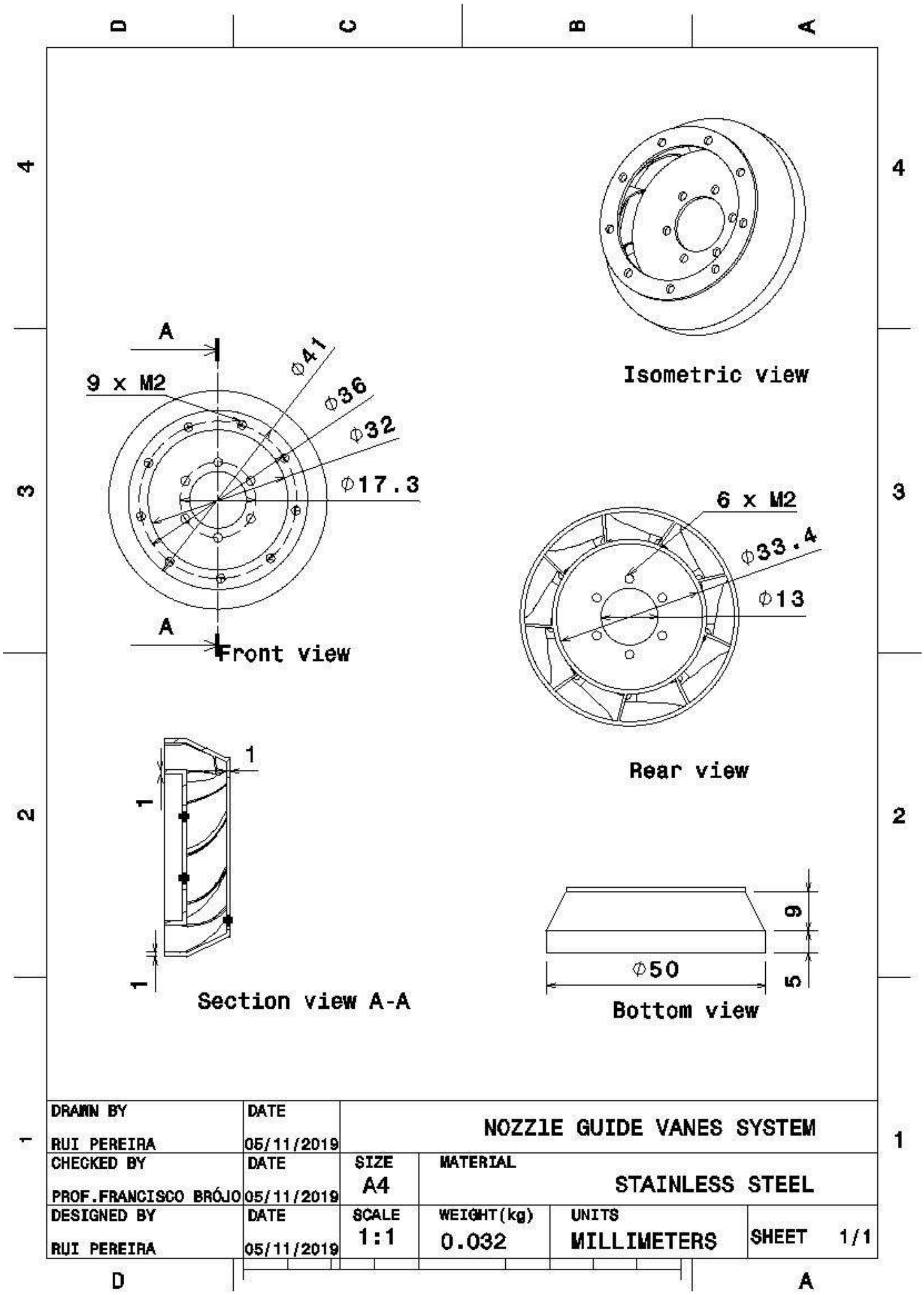

Figure B.9: Nozzle Guide Vane system

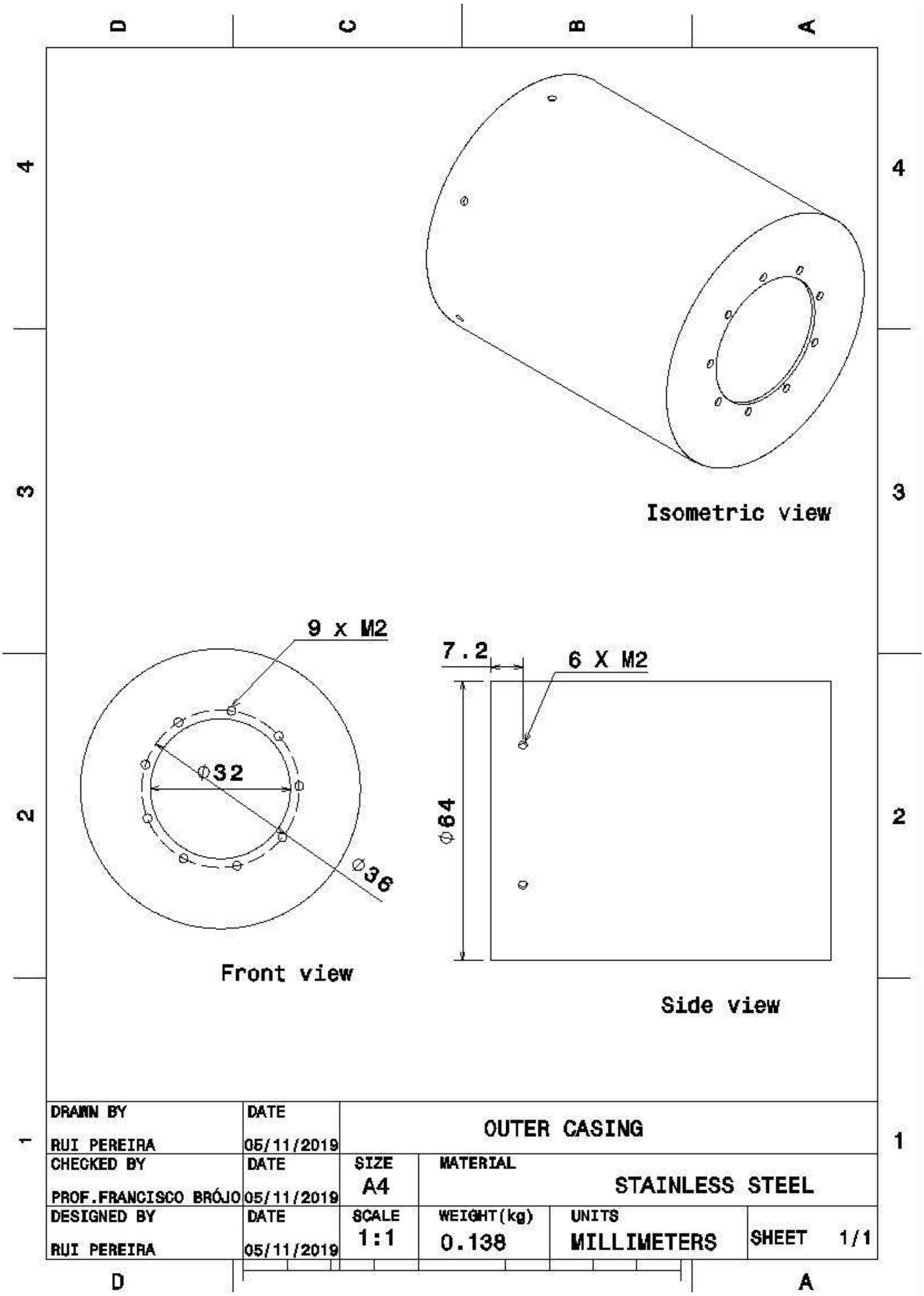

Figure B.10: Outer Casing

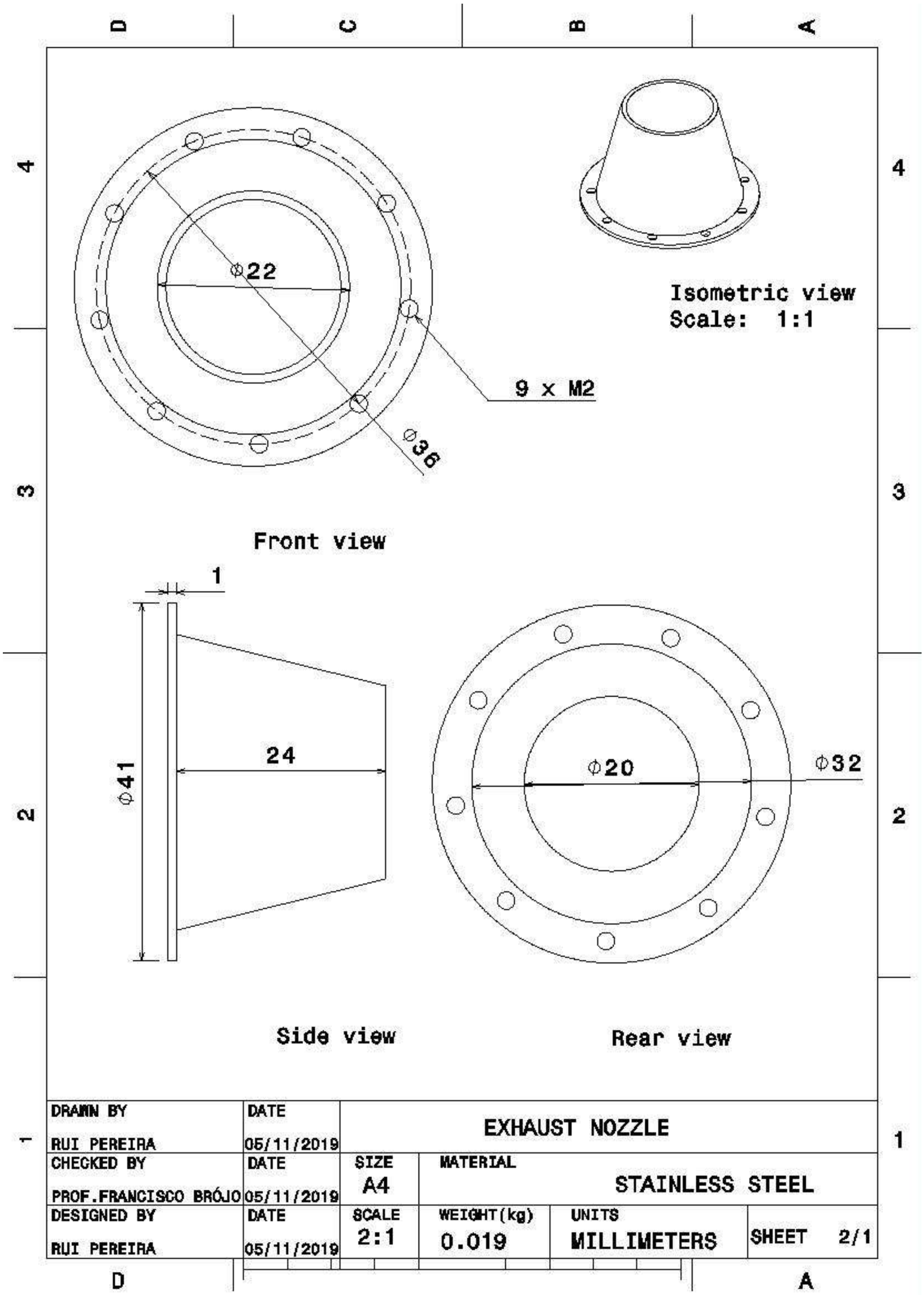

Figure B.11: Exhaust Nozzle

Appendix C - 2D, 3D Views of the turbojet

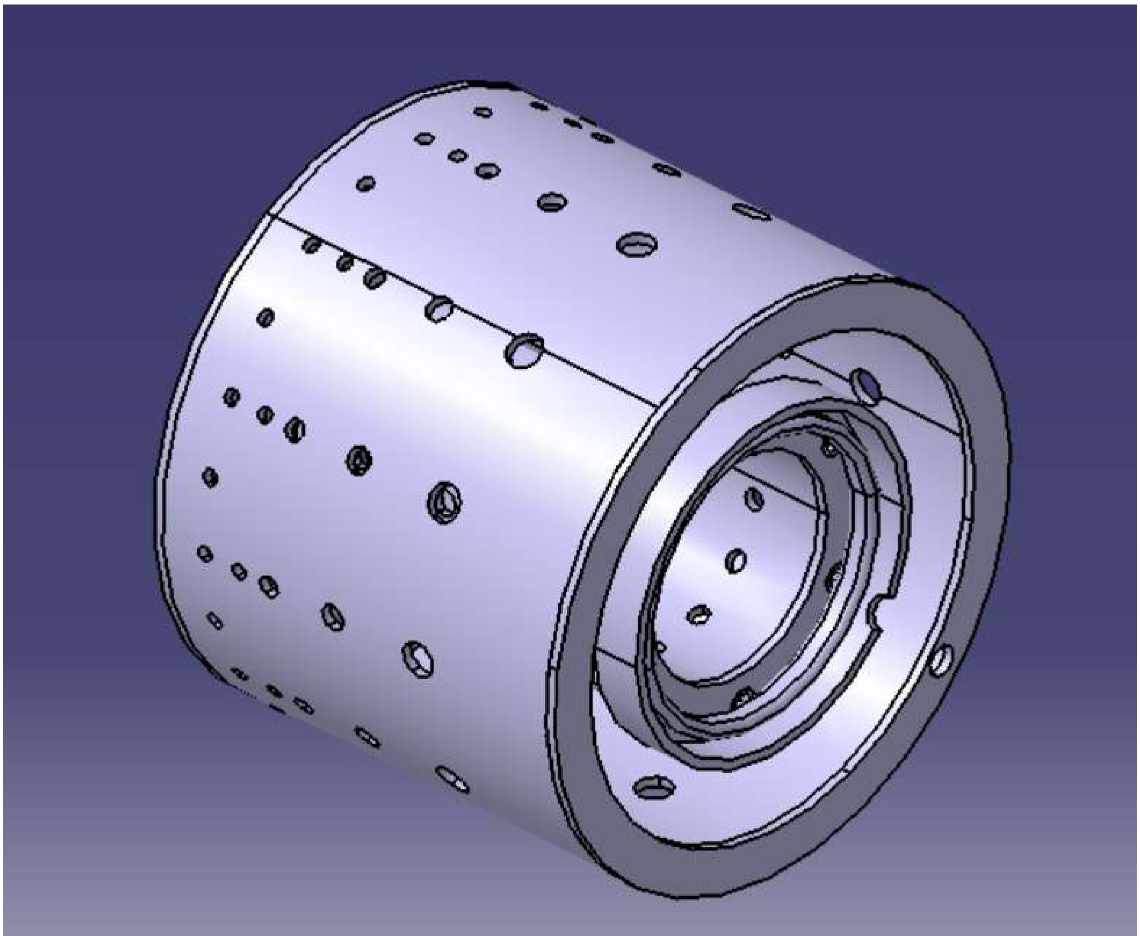

Figure C.1: "Combustion Chamber"

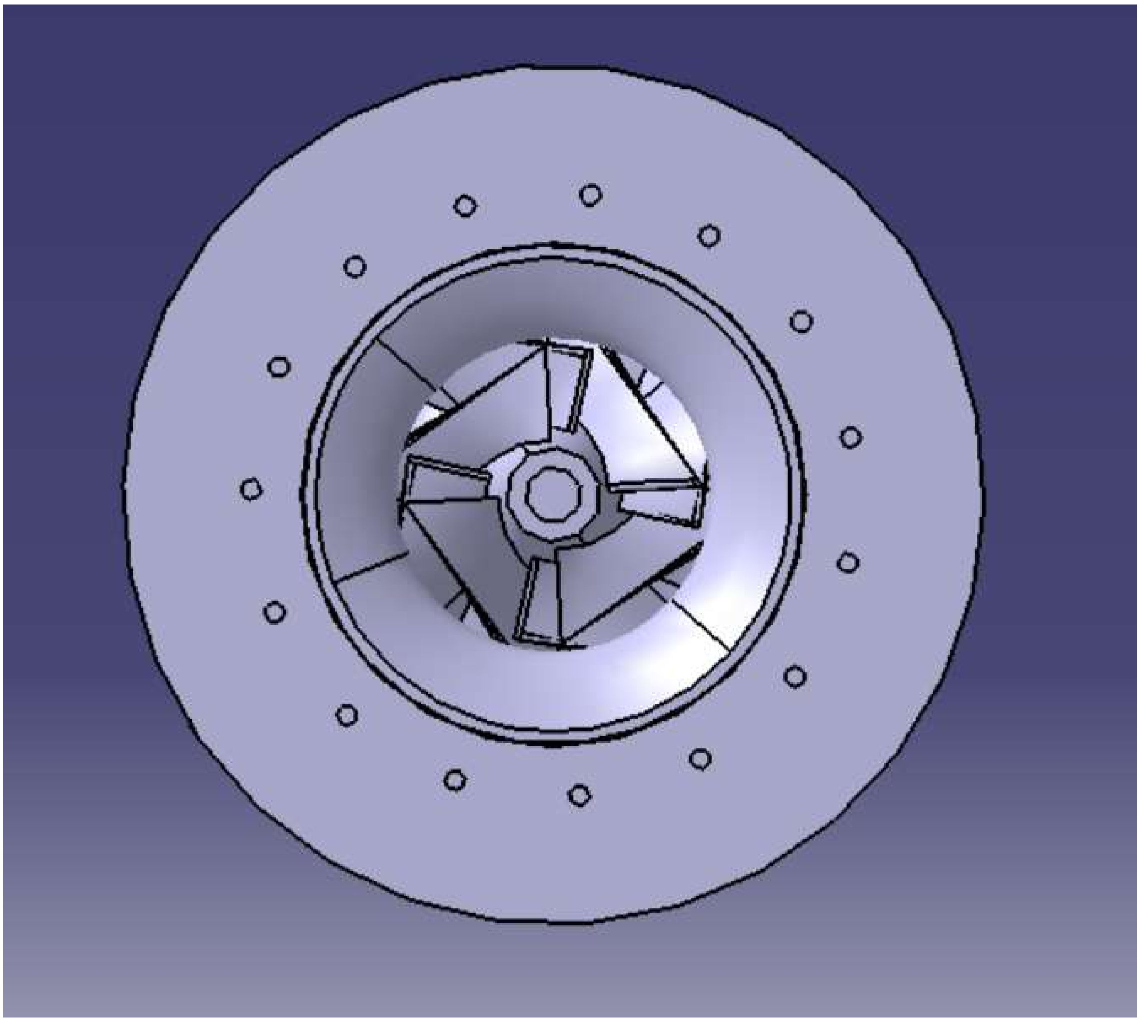

Figure C.2: ”Turbojet Front View”

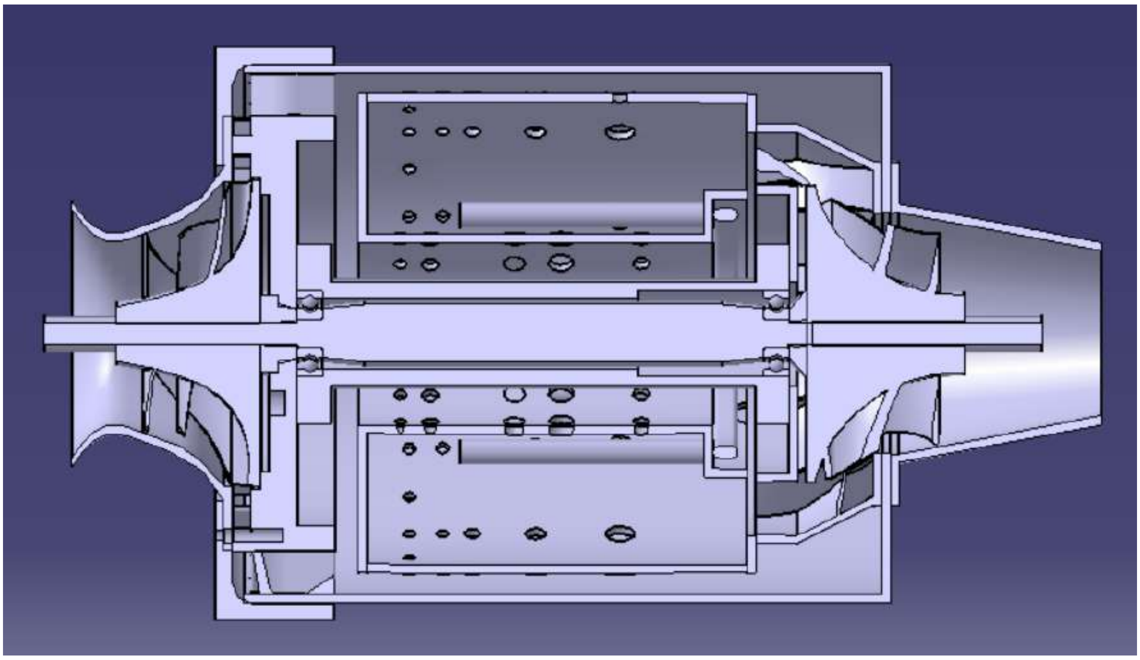

Figure C.3: ”Turbojet Midsection View”

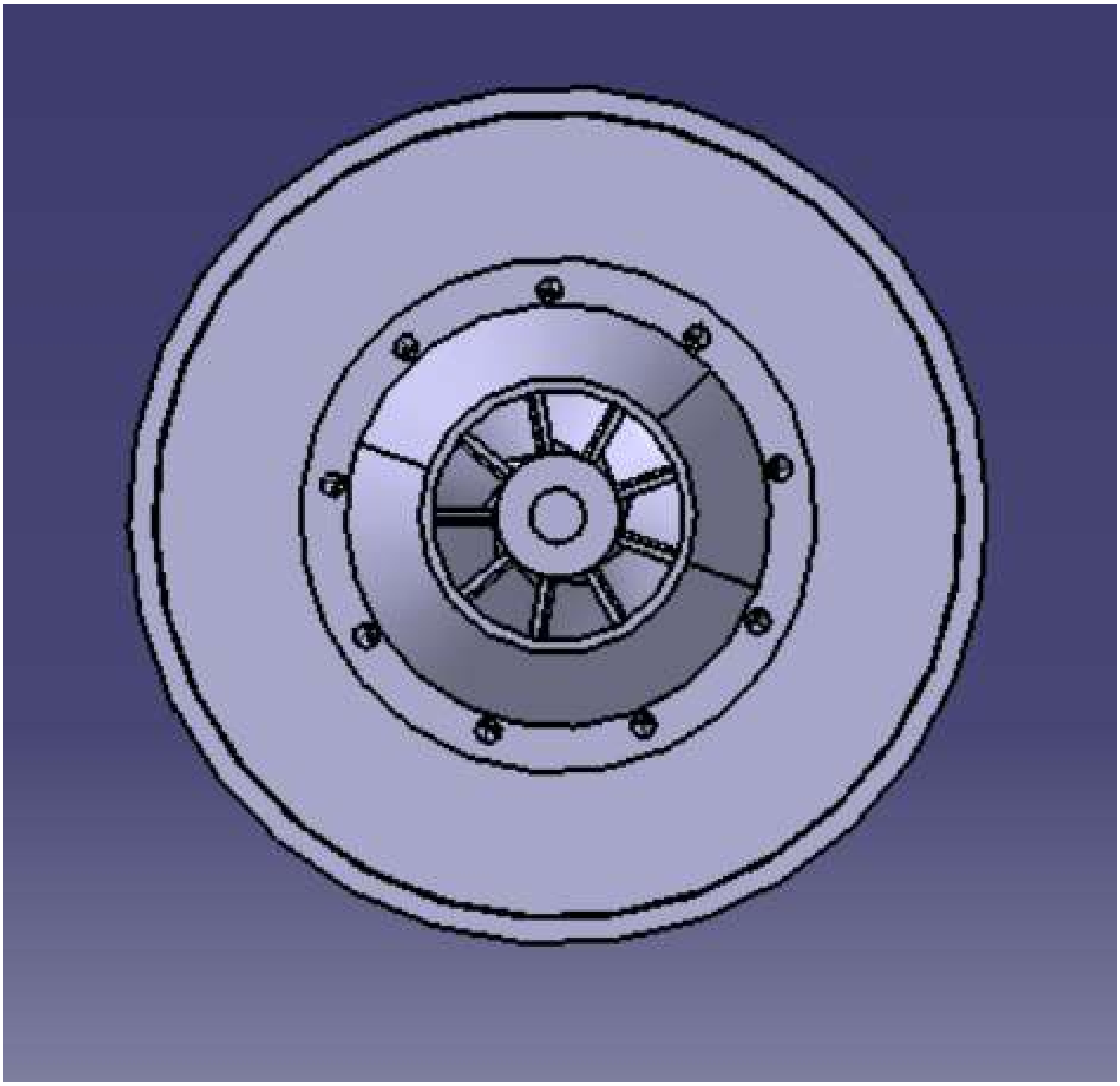

Figure C.4: "Turbojet Back View"

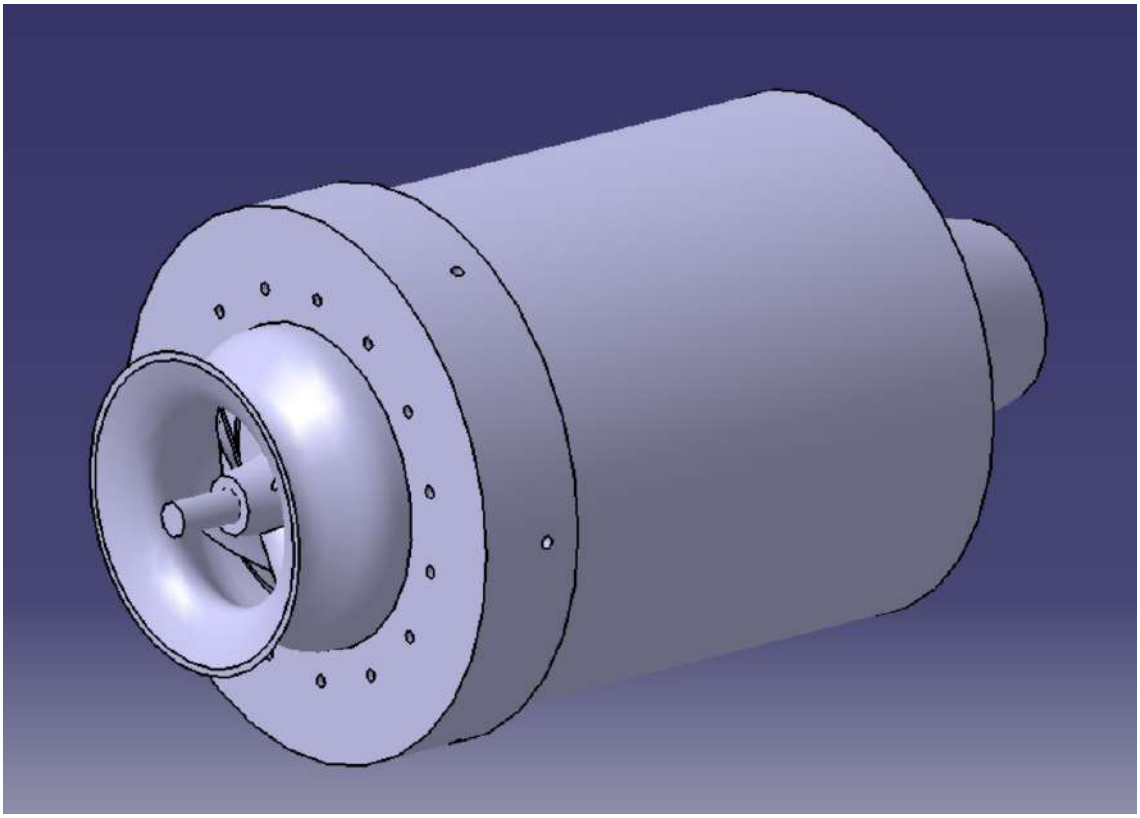

Figure C.5: ”Turbojet isometric 3D view 1”

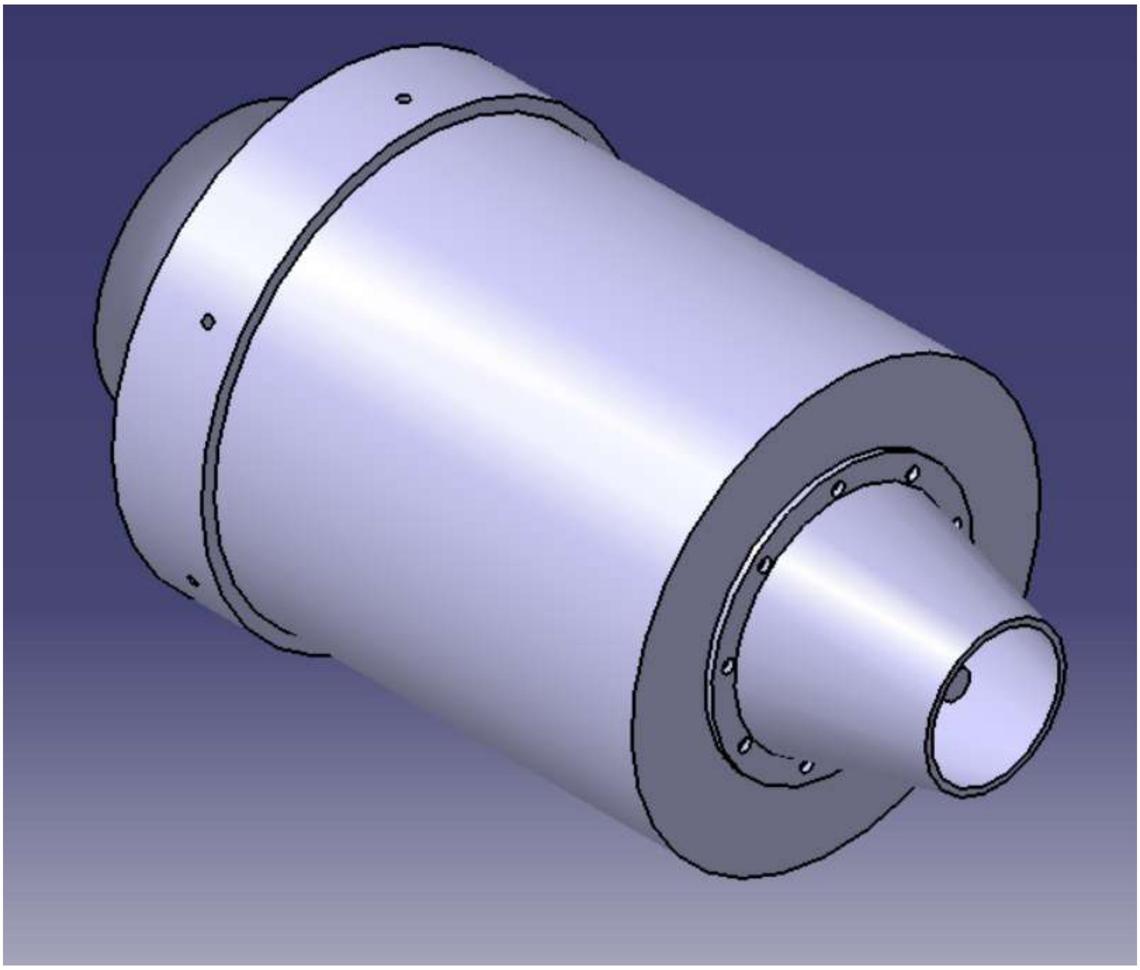

Figure C.6: ”Turbojet isometric 3D view 2”

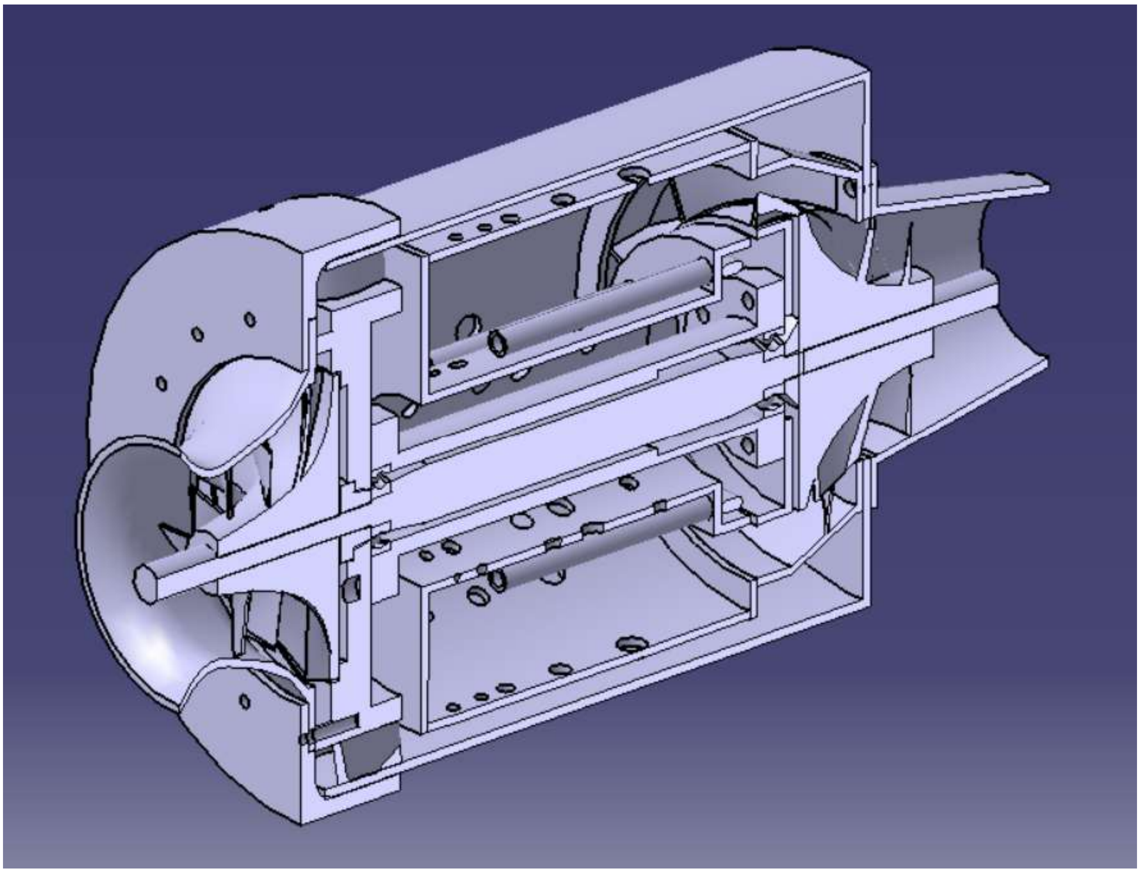

Figure C.7: ”Turbojet midsection isometric 3D view ”

Appendix D - Turbojet Actual Cycle

Table 2: Turbo VZ21 technical specifications [36]

		Type	RHF3	RHF4	RHF5	RHF55	RHF6	RHE61	RHE62	RHE7	RHE8
Item	Air Flow Rate (at: πc=2.0)	m³/min	0.75~6.3	1.0~9.4	1.4~10.8	2.4~13.5	3.1~16.5	3.3~17.5	4.5~20.0	5.5~24.0	7.0~30.0
		ft³/min	26.5~222.5	35.3~332.0	49.4~381.4	84.8~476.8	109.5~582.7	116.5~618.0	158.9~706.3	194.2~847.6	247.2~1059.5
	Maximum Pressure Ratio		2.7	27	2.8	3.0	3.0	3.3	3.3	3.3	3.3
	Maximum Speed	X10³rpm	250	190	180	168	140	140	131	120	107
	Maximum Allowable Gas Temperature	°C	950	950	950	950	950	750	750	750	750
		°F	1742	←	←	←	←	1382	←	←	←
	Weight (without waste-gate valve)	kg	(1.9)	(2.6)	(3.2)	(5.1)	(6.7)	7.8	10.5	(11.4)	(15.0)
		lb	(4.2)	(5.7)	(7.1)	(11.2)	(14.8)	17.2	23.2	(25.1)	(33.1)
	Diesel Engine Application	Ps	20~100	46~130	54-154	85~200	100~40	50.260	70-300	80-360	105~450
	Gasoline Engine Application	Ps	27~135	62~177	73~208	115~270	135~322				
External Dimensions	A	m/m	135	167	170	207	223	238	253	231	251
	B		22	37	27	43	43	47	45	50	52
	C		133	163	167	201	222	234	245	253	270
	D		88	102	110	124	124	136	146	161	175
	E		58	65	70	75	75	78	76	86	95
	F		65	74	80	102	102	100	125	120	130
	G-1 with Waste-Gate Valve		162	180	211	237	237	256	291		
	G-2 without Waste-Gate Valve		(136)	(155)	(167)	(200)	(196)			(246)	(265)
	H-1 with Waste-Gate Valve		135	125	138	156	156	175	186		
	H-2 without Waste-Gate Valve		(127)	(125)	(138)	(156)	(156)			(195)	(215)

An actual cycle analysis is made considering the compressor is rotating at 200000 rpm. From the table 2, it was obtained a compressor ratio π_c of 2.2 with an air mass flow rate, $\dot{m}_a$, of 0.1047 kg/s.

Assumed parameters:

M_0	0.05
V_r	17.35944 m/s
R	0.287 kJ/kgK
P_r	101.325 kPa
T_r	300 K
γ_c	1.4
γ_t	1.333
c_{pc}	1.005 kJ/kgK
c_{pt}	1.148 kJ/kgK
T_{04}	873.15 K
FHV	46300 kJ/kg [29]
η_i	0.7
η_c	0.7
π_c	2.2
λ	0.8
η_b	0.9
$\%\Delta P_c$	2
η_t	0.8
η_n	0.8

- Inlet

P_{02}	101.4492 kPa
T_{02}	300.15 K
π_i	1.001226

- Compressor

P_{03}	223.1882 kPa
T_{03}	408.4893 K
Q_c	11.39 kJ/s

- Burner

P_{04}	218.7244 kPa
f	0.01455321

- Turbine

P_{05}	147.534 kPa
T_{05}	791.3519 K
W_t	-9.824 kJ/s

- Nozzle Critical Pressure Check

P_{06}/P_c	2.196106 kPa
P_{06}/P_a	1.456047 K

$P_{06}/P_c > P_{06}/P_a$ (Unchoked)

V_7	22.17492 m/s
T_{07}	577.1850 K

Jet Engine Performance Parameters

$Specific$	$Thust$	$F/\dot{m}$	5.138204 Ns/kg
$TFSC$	10.19647 kg/Nh		
η_{th}	Thermal Efficiency		0.1465787
η_p	Propulsive Efficiency		0.88348601
η_o	Overall Efficiency		0.1319971