

T.R.
GEBZE TECHNICAL UNIVERSITY
GRADUATE SCHOOL

**PERFORMANCE MODELLING AND ANALYSIS OF A MICRO
GAS TURBINE ENGINE**

MEHMET ALPER AKSOY

**A THESIS OF MASTER OF SCIENCE
DEPARTMENT OF ENERGY TECHNOLOGIES
APPLIED PROPULSION SYSTEM DESIGN ENGINEERING IN
AEROSPACE TECHNOLOGIES PROGRAM**

ADVISOR: ASSISTANT PROFESSOR ONUR SON

JUNE 2024

T.R.
GEBZE TECHNICAL UNIVERSITY
GRADUATE SCHOOL

PERFORMANCE MODELLING AND ANALYSIS OF A
MICRO GAS TURBINE ENGINE

MEHMET ALPER AKSOY

A THESIS OF MASTER OF SCIENCE
DEPARTMENT OF ENERGY TECHNOLOGIES
APPLIED PROPULSION SYSTEM DESIGN ENGINEERING IN
AEROSPACE TECHNOLOGIES PROGRAM

ADVISOR: ASSISTANT PROFESSOR ONUR SON

JUNE 2024

**T.C.
GEBZE TEKNİK ÜNİVERSİTESİ
LİSANSÜSTÜ EĞİTİM ENSTİTÜSÜ**

**BİR MİKRO GAZ TÜRBİN MOTORUNUN
PERFORMANS MODELLEMESİ VE ANALİZİ**

MEHMET ALPER AKSOY

**YÜKSEK LİSANS TEZİ
ENERJİ TEKNOLOJİLERİ ANABİLİM DALI
HAVACILIK VE UZAY TEKNOLOJİLERİNDE UYGULAMALI
İTKİ SİSTEMİ TASARIM MÜHENDİSLİĞİ PROGRAMI**

DANIŞMAN: DR. ÖĞR. ÜYESİ ONUR SON

HAZİRAN 2024



MASTER of SCIENCE JURY APPROVAL FORM

A thesis submitted by Mehmet Alper AKSOY defended on 10/07/2024 before the jury formed with the 08/07/2024 date and 2024/35 numbered decision of the GTU Graduate Administration Board, has been accepted as a MASTER of SCIENCE thesis in the Department of Energy Technologies, Applied Propulsion System Design Engineering in Aerospace Technologies Program.

JURY

MEMBER

(THESIS ADVISOR) : Assist. Prof. Dr. Onur SON

MEMBER

: Prof. Dr. N.L. Okşan Çetiner YILDIRIM

MEMBER

: Assist. Prof. Dr. Fatih USTA

APPROVAL

Gebze Technical University Graduate Administration Board

...../...../..... date and/..... numbered decision.

SIGNATURE/SEAL

ABSTRACT

In today's aviation industry, micro gas turbine (MGT) engines are used in unmanned aerial vehicles (UAVs), remote control (RC) model aircraft, small-scale power generators and as auxiliary power units (APU) in some aircraft. The advantages of these engines are their ability to operate on various fuels, low maintenance costs, simple design and high power-to-weight ratio.

The preference for micro gas turbine engines brings along some costs and risks associated with their design and operation. These engines have design processes and manufacturing methods that require high technical knowledge and precision. Detailed engineering studies are required to optimize the engine and improve its performance. The accuracy and reliability of the mathematical models developed to analyze and improve the performance of micro gas turbine engines must be ensured.

In this thesis, the Olympus HP turbojet engine manufactured by AMT Netherlands is analyzed in detail. This engine has single-stage radial compressor and axial turbine components and is also equipped with an electronic control unit for the control of auxiliary elements such as fuel pump, valves, starter motor. A 1D mathematical model based on the Brayton cycle has been developed to analyze the thermodynamic and performance characteristics of the engine. A published compressor map was used for component modelling and the turbine performance was provided by a compressor-turbine work balance formulation. This model is compared with data and test results published in the literature and by the manufacturer to better understand the characteristics and behavior of the engine.

This study investigates the performance behavior of the engine under different environmental conditions. Firstly, the performance of the engine at certain shaft speeds in sea level standard air conditions was analyzed. The shaft speed dependent exhaust gas temperature, thrust data, station temperature and pressure data were calculated. This data was verified with the manufacturer and experimental test data published in the literature. In the second stage of the study, the model was run at different altitudes and Mach numbers to examine the engine behavior in flight. The results were compared with the data of the same class but different engine.

In conclusion, this study, which includes a comparison of theoretical studies with practical applications, aims to understand the performance characteristics, off-design activities and useful implications for the development of the electronic control unit of micro gas turbine engines.

Keywords: Micro gas turbine, Engine performance, Mathematical model, Mach number, Brayton cycle

ÖZET

Günümüz havacılık endüstrisinde, mikro ölçekli gaz türbin motorları, özellikle insansız hava araçları ve model uçaklar için birçok avantaj sunmaktadır ve bu nedenle sıklıkla tercih edilmektedir. Mikro gaz türbin motorları, genellikle kerosen, JP-4, JP-8 ve dizel gibi farklı yakıt tipleriyle çalışabilirler. Geleneksel jet motorlarına kıyasla daha basit bir tasarıma sahip olmaları ve daha az parça içermeleri, bakım ve onarım maliyetlerini düşürmektedir. Ayrıca bu motorlar yüksek güç-ağırlık oranına sahiptir. Bir mikro gaz türbinin endüstrinin ihtiyaçlarına cevap verebilecek nitelikte tasarlanırken maliyet ve risklerin en düşük seviyede tutulması gerekmektedir. Özellikle, bu motorların yüksek teknik bilgi ve hassasiyet gerektiren tasarım süreçleri ve üretim yöntemleri vardır. Motorun optimize edilmesi ve performansının artırılması için detaylı mühendislik çalışmaları gerekmektedir. Mikro gaz türbin motorlarının performansını analiz etmek ve iyileştirmek için geliştirilen matematiksel modellerin doğruluğu ve güvenilirliği sağlanmalıdır.

Bu tez kapsamında, AMT Netherlands tarafından üretilen Olympus HP turbojet detaylı bir şekilde incelenmektedir. Motor tek kademeli radyal kompresör ve eksenel türbin bileşenlerine sahiptir. Yardımcı sistemler olarak yakıt pompası, valfler, başlatma motoru, egzoz sıcaklık sensörü ve devir sensörü vardır. Motor kontrolü elektronik kontrol ünitesi ile yapılmaktadır. Motorun termodinamik ve performans karakteristiklerinin analizi için Brayton çevrimine dayalı bir boyutlu matematiksel model geliştirilmiştir. Bileşen modellenmesinde yayınlanmış kompresör performans haritası kullanılmış olup türbin performansı ise kompresör-türbin iş dengesi formülasyonu ile sağlanmıştır. Bu model, motorun özelliklerini ve davranışını daha iyi anlamak için literatürde ve üretici firma tarafından yayınlanan veriler ve deneysel test sonuçlarıyla karşılaştırılmıştır. Doğrulama sürecinde, motorun deniz seviyesi statik koşullarındaki performansı belirli shaft hızlarında incelenmiş ve motorun itki üretimi, egzoz gaz sıcaklığı verileri ile istasyon sıcaklık ve basınçları karşılaştırılmıştır. Ayrıca, motorun farklı irtifalarda ve Mach sayılarında nasıl davrandığı teorik olarak incelenmiş ve sonuçlar, benzer bir sınıftaki farklı bir motorun yayınlanmış verileri ile karşılaştırılmıştır.

Mikro gaz türbin motorlarındaki gelişmeler savunma sanayii, sivil ve endüstriyel enerji taleplerini de karşılayabilmesi giderek daha fazla tercih edilmesini sağlamıştır. Bu motorlar, yüksek güç-ağırlık oranına sahip olmaları, farklı yakıt seçenekleriyle ve düşük bakım maliyetleriyle geniş bir kullanım alanına sahiptir. Gelecekte, mikro ölçekli gaz türbin motorlarının daha da yaygınlaşması ve geliştirilmesi beklenmektedir.

Sonuç olarak teorik çalışmaların pratik uygulamalarla kıyaslamasını içeren bu çalışma ile mikro gaz türbin motorlarının elektronik kontrol ünitesinin geliştirilmesi için faydalı çıkarımlar, tasarım faaliyetleri ve performans karakteristiğinin anlaşılması hedeflenmektedir.

Anahtar Kelimeler: Mikro gaz türbini, Motor performansı, Matematiksel model, Mach sayısı

ACKNOWLEDGEMENTS

I would like to express my deep and sincere gratitude to my supervisor, Assistant Professor Onur SON, who not only shared his profound scientific knowledge with me but also taught me great lessons of life. His support, suggestions and encouragement gave me the drive and will to complete this work.

I want to thank my parents and siblings for their encouragement and support.

TABLE OF CONTENTS

	<u>Page</u>
ABSTRACT	v
ÖZET	vi
ACKNOWLEDGEMENTS	vii
TABLE OF CONTENTS	viii
LIST OF SYMBOLS AND ABBREVIATIONS	ix
LIST OF FIGURES	xi
LIST OF TABLES	xiii
1. INTRODUCTION	1
1.1. Ideal Brayton Cycle	1
1.2. Actual Brayton Cycle	5
1.3. Gas Turbine Cycles for Propulsion	6
1.4. Types of Jet Engines	10
1.4.1. Comparison of Jet Engine	14
1.5. Micro Gas Turbine	14
1.6. Literature Review	17
2. METHODOLOGY	20
2.1. AMT Olympus Turbojet	24
2.2. Stagnation Properties	25
2.3. Component Modelling	26
2.3.1. Intake Modelling	26
2.3.2. Compressor Modelling	27
2.3.3. Combustion Chamber Modelling	31
2.3.4. Turbine Modelling	33
2.3.5. Nozzle Modelling	35
3. RESULTS	36
4. CONCLUSIONS	47
REFERENCES	48
BIOGRAPHY	50
PUBLICATIONS AND PRESENTATIONS FROM THE THESIS	51

LIST OF SYMBOLS AND ABBREVIATIONS

f	: Air-Fuel Ratio
A	: Area
ρ	: Density
η	: Efficiency
h	: Enthalpy
F	: Force
Ma	: Mach Number
$\dot{m}$	: Mass Flow Rate
m	: Meter
N	: Newton
P	: Pressure
r	: Pressure Ratio
R	: Specific Gas Constant
γ	: Specific Heat Ratio
c	: Speed of Sound
T	: Temperature
V	: Velocity
v	: Volume
W	: Work
APU	: Auxiliary Power Units
CFD	: Computational Fluid Dynamics
EGT	: Exhaust Gas Temperature
FHV	: Fuel Heat Value
JP	: Jet Propulsion
KRPM	: Kilo Revolutions per Minute
MGT	: Micro Gas Turbine
NPR	: Nozzle Pressure Ratio
NPSS	: Numerical Propulsion System Simulation
OEM	: Original Equipment Manufacturer
RPM	: Revolutions per Minute
SFC	: Specific Fuel Consumption

SOFC	: Solid Oxide Fuel Cell
TJ	: Turbojet
T-MATS	: Toolbox for the Modelling and Analysis of Thermodynamic Systems
UAV	: Unmanned Aerial Vehicle
VIGV	: Variable Guide Vane

LIST OF FIGURES

	<u>Page</u>
Figure 1.1: Open Brayton cycle diagram.	2
Figure 1.2: Closed Brayton cycle diagram.	2
Figure 1.3: P-v and T-s diagrams for ideal Brayton cycle.	3
Figure 1.4: Thermal efficiency of the ideal Brayton cycle as a function of the pressure ratio (Cengel et al., 2011).	4
Figure 1.5: Pressure ratio and specific temperature (Cengel et al., 2011).	5
Figure 1.6: Actual Brayton cycle (Cengel et al., 2011).	6
Figure 1.7: A turbojet engine (Cengel et al., 2011).	7
Figure 1.8: Effect of air fuel ratio on combustion chamber temperature (Saravanamuttoo et al., 1996).	8
Figure 1.9: Section of a turbojet engine (Cengel et al., 2011).	11
Figure 1.10: A turbofan engine (Cengel et al., 2011).	11
Figure 1.11: A turboprop engine (Cengel et al., 2011)	12
Figure 1.12: A turboshaft engine.	13
Figure 1.13: A ramjet engine (Cengel et al., 2011).	13
Figure 1.14: Jet engines altitude and flight Mach number (Mattingly, 2005).	14
Figure 1.15: Performance development of the MGT (Massardo et. al, 2002).	15
Figure 1.16: Parts of a micro turbine engine (Schreckling, 2005).	16
Figure 1.17: AMT Olympus sensor layout for testing (AMT Netherlands, 2019).	16
Figure 1.18: Flight envelope study at different altitudes and Mach numbers (Chapman et al., 2016).	18
Figure 2.1: Ambient pressure vs altitude (Walsh et al., 2004).	21
Figure 2.2: Ambient temperature vs altitude (Walsh et al., 2004)	21
Figure 2.3: Mathematical model schematic.	22
Figure 2.4: Specific heat of air depending on temperature (Kurzke, 2018).	23
Figure 2.5: Comparison of temperature and pressure ratios with Mach number (Mattingly, 2005).	26
Figure 2.6: Intake for MGT (Web 1, 2024).	27
Figure 2.7: Radial compressor and diffuser for MGT (Web 1, 2024).	28
Figure 2.8: General compressor performance map (Lichtsinder, 2006).	29
Figure 2.9: Scaled compressor map (Leylek, 2012).	30
Figure 2.10: Compressor operating line (Leylek, 2012).	30
Figure 2.11: Combustion chamber for MGT (Web 1, 2024).	32
Figure 2.12: Axial turbine and Nozzle Guide Vane for MGT (Web 1, 2024).	33
Figure 2.13: General turbine performance map (Al-Hamdan, 2005).	34
Figure 3.1: Station nomenclature of the mathematical model (Kurzke, 2018)	36
Figure 3.2: Station pressures at 70K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	37
Figure 3.3: Station pressures at 80K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	37
Figure 3.4: Station pressures at 90K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	38
Figure 3.5: Station pressures at 100 KRPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	39

Figure 3.6: Station pressures at 108K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	39
Figure 3.7: Station temperatures at 70K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	40
Figure 3.8: Station temperatures at 80K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	41
Figure 3.9: Station temperatures at 90K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	41
Figure 3.10: Station temperatures at 100K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	42
Figure 3.11: Station temperatures at 108K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.	42
Figure 3.12: EGT vs RPM comparison for the test data of Leylek (2012), OEM from engine manufacturer and our model prediction.	43
Figure 3.13: Thrust vs RPM comparison for the test data of Leylek (2012), OEM from engine manufacturer and our model prediction.	44
Figure 3.14: Thrust vs Mach number and altitude.	44
Figure 3.15: Thrust/max. thrust vs Mach number and altitude.	45
Figure 3.16: Mach number vs Altitude and Thrust.	46

LIST OF TABLES

	<u>Page</u>
Table 1.1: Specifications of reactive engines (Flack, 2005).	14
Table 1.2: General MGT characteristics (Massardo et al., 2002).	15
Table 2.1: List of model parameters.	23
Table 2.2: AMT Olympus turbojet technical data (AMT Netherlands, 2019).	24

1. INTRODUCTION

Gas turbines are used in power generation plants, aircraft, some trains, and ships to provide the required power to the system. The maintenance, operation, and testing costs of gas turbines, which are constantly developing with modern technology, are increasing. Mathematical models that can predict the performance of gas turbine systems under different conditions contribute to the reduction of these costs. In this study, the performance modelling of a micro-scale turbojet is carried out and compared with experimental test data and simulations in the literature. In this section, basic thermodynamic cycles and jet engine types are mentioned. Subsequently, after giving information about the micro gas turbine, which is the subject of this study, the studies in the literature are given.

1.1. Ideal Brayton Cycle

The Brayton cycle, one of the gas-fluid power cycles, defines the operating system of gas turbines. There are two cycles, an open cycle where the exhaust gas does not circulate in the system as shown in Figure 1.1 and a closed cycle where the exhaust gas circulates in the system as shown in Figure 1.2. The main components of the cycle are the compressor, the combustion chamber or heat exchanger and the turbine. As fresh air is drawn into the system by the compressor, the temperature and pressure of the system increases. As the compressor compresses the air (isentropic process), the highest pressure in the system is obtained. The incoming air is burned in the combustion chamber with the fuel fed in at a stoichiometric ratio and the highest temperature in the cycle is obtained. Under ideal conditions, combustion is an isobaric process. The energized air is expanded (isentropic process) to atmospheric pressure by means of the turbine connected on the same shaft with the compressor, providing both the continuity of the cycle and the power required for electrical energy.

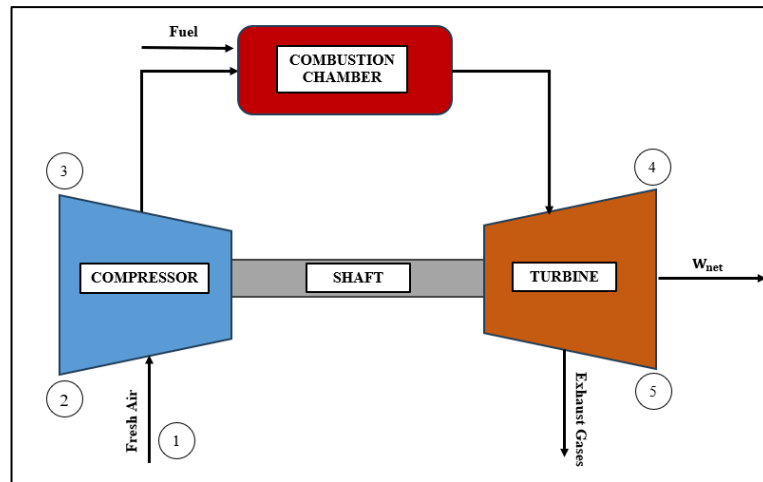


Figure 1.1: Open Brayton cycle diagram.

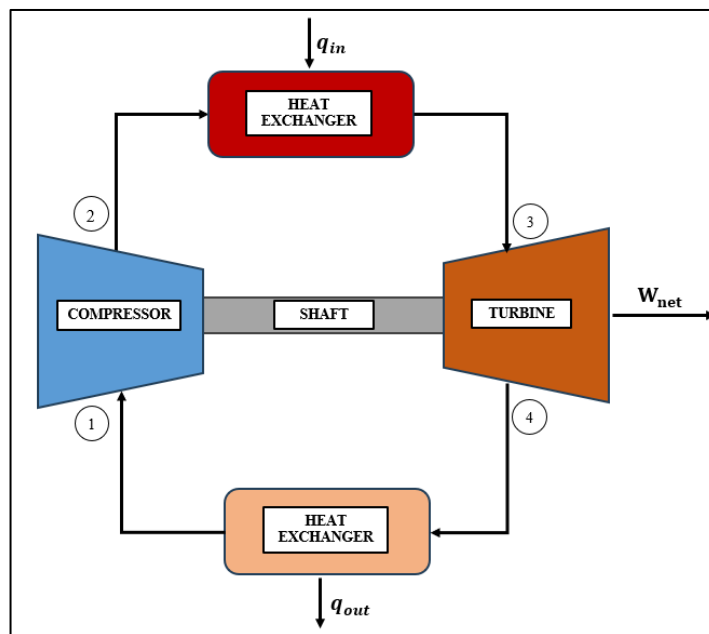


Figure 1.2: Closed Brayton cycle diagram.

In Figure 1.3, Pressure-Volume (P-v) and Temperature-Entropy (T-s) diagrams for the ideal Brayton cycle are shown. These Brayton cycle stages are as follows:

2-3: Compression process is the isentropic (at constant entropy) process where the pressure and temperature of the air increase mechanically.

3-4: The combustion of the fuel-air mixture with an ideal isobaric process is where the air enthalpy increases the most within the cycle.

4-5: It is the expansion process where pressure and enthalpy decrease isentropically, and mechanical energy is extracted from the air.

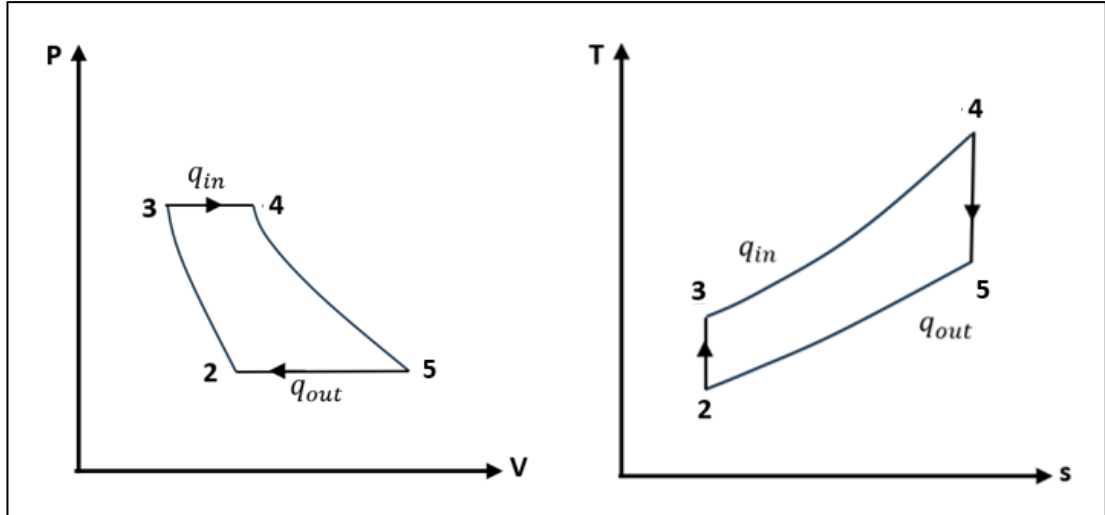


Figure 1.3: P-v and T-s diagrams for ideal Brayton cycle.

When the changes in kinetic and potential energy of a unit mass continuous flow system are neglected, the equation of energy conservation in a Brayton cycle is as follows.

$$(q_{in} - q_{out}) + (w_{in} - w_{out}) = h_{exit} - h_{inlet} \quad (1.1)$$

According to this, heat transfer of working fluids

$$(q_{in} - q_{out}) + (w_{in} - w_{out}) = h_{exit} - h_{inlet} \quad (1.2)$$

and

$$q_{in} = h_4 - h_3 = c_p(T_4 - T_3) \quad (1.3)$$

$$q_{out} = h_5 - h_2 = c_p(T_5 - T_2) \quad (1.4)$$

The thermal efficiency of the Brayton cycle under cold air standard assumption is as follows.

$$\eta_{th,Brayton} = \frac{W_{net}}{q_{in}} = 1 - \frac{q_{out}}{q_{in}} = 1 - \frac{c_p(T_5 - T_2)}{c_p(T_4 - T_3)} = 1 - \frac{T_1(T_5/T_2 - 1)}{T_2(T_4/T_3 - 1)} \quad (1.5)$$

As seen in the Pressure-Volume (P-v) diagram, $P_3 = P_4$ and $P_5 = P_2$ and the processes 2-3 and 4-5 are isentropic, therefore:

$$\frac{T_3}{T_2} = \left(\frac{P_3}{P_2}\right)^{\frac{\gamma-1}{\gamma}} = \left(\frac{P_4}{P_5}\right)^{\frac{\gamma-1}{\gamma}} = \frac{T_4}{T_5} \quad (1.6)$$

When these equations are simplified, the thermal efficiency equation of the Brayton cycle can be found.

$$\eta_{th,Brayton} = 1 - \frac{1}{r_p^{(\gamma-1)/\gamma}} \quad (1.7)$$

Here, the pressure ratio r_p used is the ratio of pressures, and γ is the specific heat ratio at room temperature assuming $\gamma=1.4$. Accordingly, the variation of the thermal efficiency of the Brayton cycle as a function of the pressure ratio is shown in Figure 1.4.

$$r_p = \frac{P_3}{P_2} \quad (1.8)$$

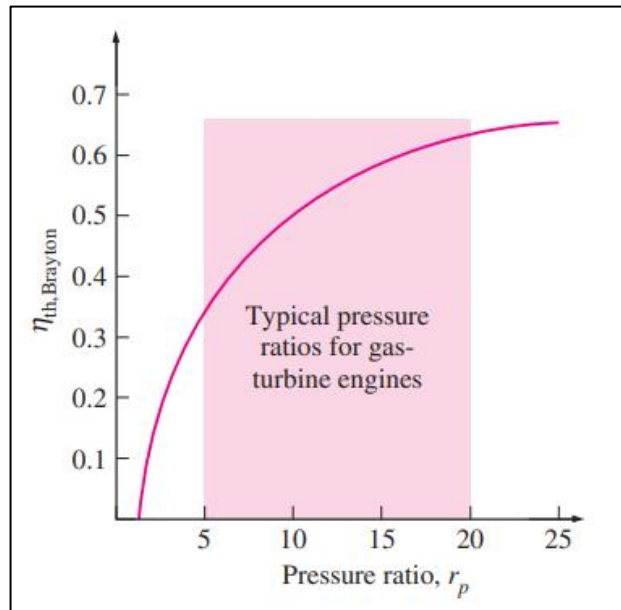


Figure 1.4: Thermal efficiency of the ideal Brayton cycle as a function of the pressure ratio (Cengel et al., 2011).

The turbine is exposed to the highest temperature in the system. Considering the material technology, the temperature limit that the turbine material can withstand is the factor that determines the combustion chamber outlet temperature. The operating profile of all components in the cycle affects each other. For this reason, this limiting

feature in turbines also affects the compressor compression ratio. As can be seen in Figure 1.4, assuming a constant outlet temperature from the combustion chamber, increasing the pressure ratio increases the overall work output of the cycle. However, as shown in Figure 1.5, a continuous increase in the pressure ratio reduces efficiency and subsequently the produced net work decreases. The resulting reduced cycle efficiency requires higher mass flow rates to maintain the same level of performance and poses challenges in the design process.

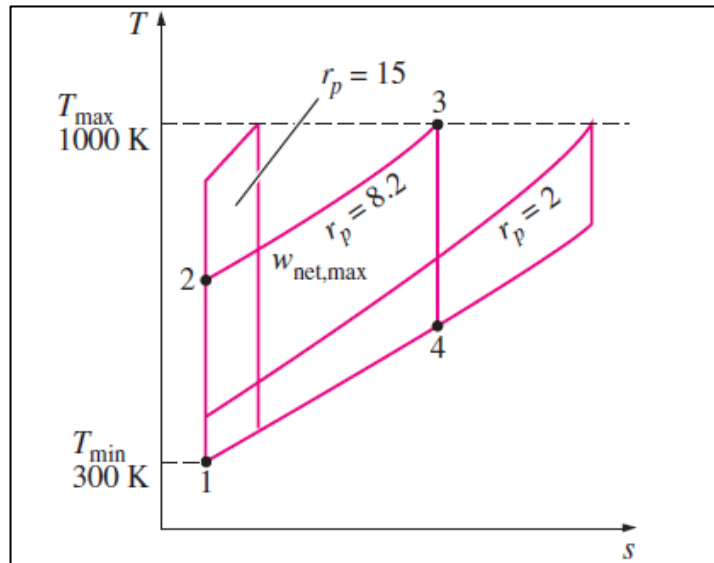


Figure 1.5: Pressure ratio and specific temperature (Cengel et al., 2011).

Air has two important tasks in the system. One of them is to provide the oxygen required for combustion and the other is to cool the components exposed to high temperatures. Due to the cooling task, the system works according to the principle of excess air. In addition, the turbine is the component in the system that is exposed to the highest flow due to the fuel flow in the combustion chamber. Since the fuel flow rate is very low compared to the air flow rate, it can be neglected in the calculations, but it is not neglected in this thesis.

1.2. Actual Brayton Cycle

Calculations made under ideal assumptions state that there will be no loss of components during operation. In this context, two terms appear as adiabatic and isentropic processes. While there is no heat transfer in adiabatic processes, entropy is

constant in isentropic processes. In practice, this situation is different. When fluid frictions are included in the calculations, pressure losses occur at the compressor inlet, the combustion chamber, and the exhaust nozzle respectively. In other words, due to irreversibilities, the actual work transferred to the compressor is greater than that of the ideal cycle and the actual work delivered from the turbine to the system is less. (Figure 1.6)

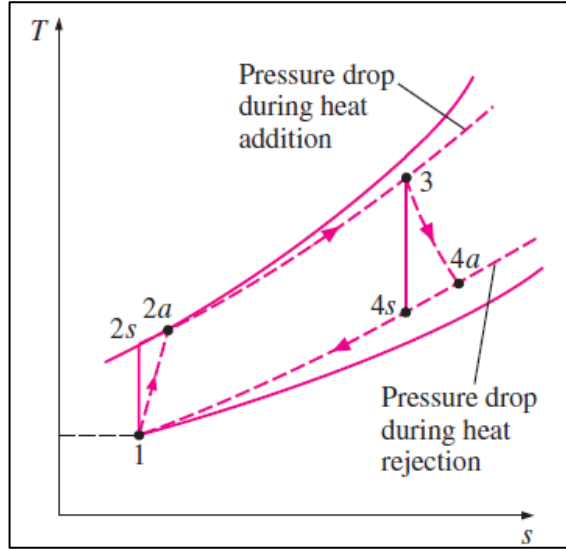


Figure 1.6: Actual Brayton cycle (Cengel et al., 2011).

In an actual Brayton cycle the efficiencies of compressor and turbine can be calculated as below.

$$\eta_{compressor} = \frac{W_s}{W_a} \cong \frac{h_{2s} - h_1}{h_{2a} - h_1} \quad (1.9)$$

$$\eta_{turbine} = \frac{W_a}{W_s} \cong \frac{h_3 - h_{4a}}{h_3 - h_{4s}} \quad (1.10)$$

As shown in Equations 1.9 and 1.10, the symbol “a” represents the actual state while “s” denotes the isentropic (or ideal) state.

1.3. Gas Turbine Cycles for Propulsion

There are two main cycles used in gas turbines. These are the shaft power cycle and the aircraft propulsion cycles. In the shaft power cycle, the high energy gases coming from the combustion chamber are expanded to ambient pressure through the turbine to

absorb all the energy. All the power obtained is transferred to the shaft. In aircraft jet propulsion cycles, the high-energy air coming from the combustion chamber is expanded in the turbine only up to the pressure that will ensure the continuity of the system. In other words, the power required for the compressor and fan on the same shaft with the turbine is generated. In some applications, energy is generated within the electricity requirement of the aircraft. However, this affects the thrust performance of the engine. Subsequently, the air passed through the exhaust nozzle at high-speed producing thrust. In all turbofan and turbojet engines, thrust is generated by this method. Another difference is that in aircraft jet propulsion cycles, the flight envelope, i.e. altitude and Mach number, is one of the important factors affecting performance.

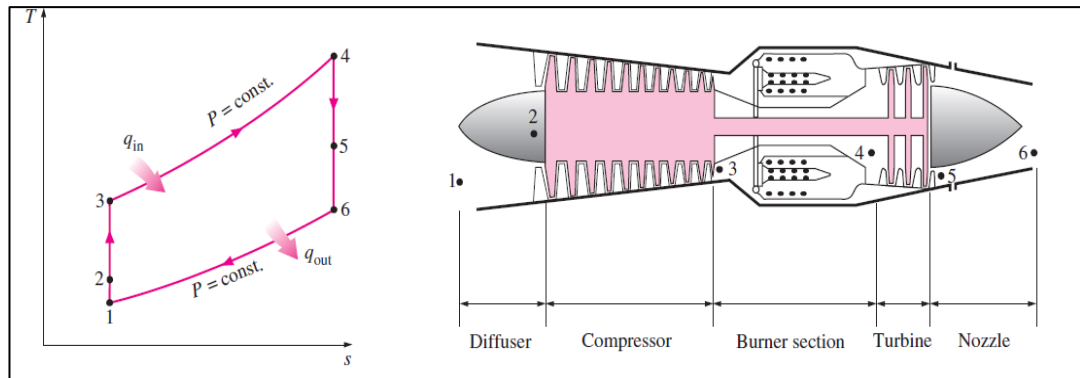


Figure 1.7: A turbojet engine (Cengel et al., 2011).

A cross-sectional view of a turbojet engine and a T-s diagram for the ideal cycle are given in Figure 1.7. The first component of the engine is the diffuser or intake. Its main task is to increase the static pressure of the air by reducing its speed until it reaches the compressor. However, by reducing the pressure loss to the lowest level, the air flow is transmitted to the compressor uniformly in all flight operations of the aircraft.

The task of compressors is to increase the pressure and temperature of the air. Compression ratios directly affect engine performance. There are two different types of compressors: Radial compressors and axial compressors. The compressor type used in this study is the radial compressor (AMT Netherlands, 2019). The first compressor type used in jet engines was the radial compressor. It is still used today in small-scale turbojet and turboshaft engines as well as in auxiliary power units (APU). A single-stage radial compressor consists of 4 parts: inlet, impeller, diffuser (stator) and outlet. In radial compressors, the air flow is perpendicular to the axis of rotation. When air

enters the compressor, its pressure and temperature increase due to the centrifugal effect. In light alloy single-stage radial compressors, the compression ratio is about 4:1 (Saravanamuttoo et al., 1996). The air reaching high speeds is then slowed down by a diffuser (stator) and its static pressure is increased. The air, whose temperature and pressure are increased in the compressor, is burned in the combustion chamber by adding fuel at a certain stoichiometric ratio. Under ideal conditions, the combustion process is isobaric. But there is pressure loss. The combustion chamber outlet temperature is the highest temperature of the cycle. The factor limiting this temperature is the turbine material. There are 3 types of combustors. These are annular combustor, can or tubular combustor and can-annular combustor. The combustor type used in this study is annular combustion (AMT Netherlands, 2019). Figure 1.8 shows the inlet temperature into the combustion chamber and how much the added fuel increases the air temperature.

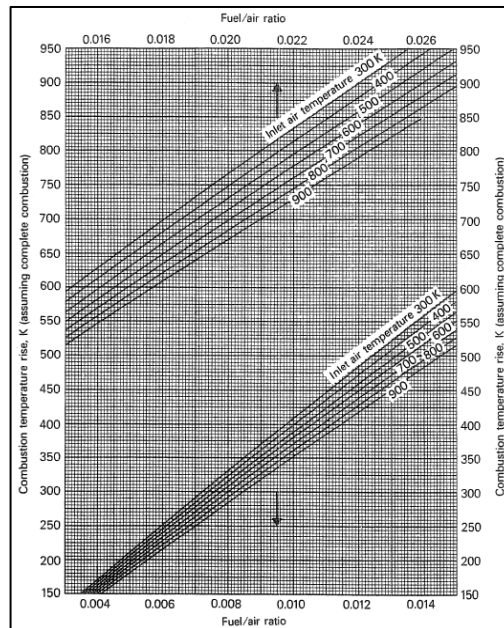


Figure 1.8:Effect of air fuel ratio on combustion chamber temperature (Saravanamuttoo et al., 1996).

The air energized in the combustion chamber comes to the turbine. The turbine is on the same shaft as the compressor. For the continuity of the system, the power generated by the turbine must be equal to the power required by the compressor. The turbine expands the incoming air in the combustion chamber up to ambient pressure to ensure this continuity. The high velocity air going to the exhaust nozzle then generates the thrust.

The basic principle for calculating thrust (F) is the momentum change of the air. Air enters the engine at a speed (V_{inlet}) equal to the forward motion of the engine. The air is then accelerated inside the engine and leaves through the exhaust nozzle. Here the fuel flow rate ($\dot{m}_f$) can be neglected as it is very small compared to the air flow rate. The net thrust formulation was given in Equation 1.11. Here it is assumed that the pressure of the exhaust gases (P_{exit}) is fully expanded to atmospheric pressure (P_0).

$F = \text{Momentum change in the air}$

$$F = (\dot{m}V_{exit}) - (\dot{m}V_{inlet}) = \dot{m}(V_{exit} - V_{inlet}) \quad (1.11)$$

As shown in equation 1.11, $\dot{m}V_{exit}$ represents the equation for gross thrust.

If the exhaust gases are not fully expanded to atmospheric pressure, the exhaust pressure P_{exit} will be higher than the atmospheric pressure. This pressure difference over the exhaust outlet area (A_{exit}) will provide additional force to the thrust.

$F = \text{Momentum change in the air} + \text{pressure force}$

$$F = (\dot{m}V_{exit} - \dot{m}V_{inlet}) + A_{exit}(P_{exit} - P_0) \quad (1.12)$$

The thrust efficiency is given by the following Equation 1.13. According to this formulation, the engine achieves the highest thrust in static conditions. However, in this case the thrust efficiency is equal to 0. In the case where the inlet and outlet velocities of the air are equal, the thrust efficiency is 1 while the thrust is 0. The result obtained here is that the ratio between the inlet and outlet velocities of the air does not need to be very large (Saravanamuttoo et al., 1996).

$$\eta_p = \frac{\dot{m}V_{inlet}(V_{exit} - V_{inlet})}{\dot{m} \left[V_{inlet}(V_{exit} - V_{inlet}) + \frac{(V_{exit} - V_{inlet})^2}{2} \right]} = \frac{2}{1 + \left(\frac{V_{exit}}{V_{inlet}} \right)} \quad (1.13)$$

The thermal efficiency of the system is calculated by Equation 1.14.

$$\eta_e = \frac{\dot{m}(V_{exit}^2 - V_{inlet}^2)}{2\dot{m}_f Q_{net}} \quad (1.14)$$

The energy given to the system by the fuel consumed for the aircraft to overcome air friction and reach the required speed is calculated by Equation 1.16. The total efficiency equation is obtained by multiplying propulsion efficiency by thermal efficiency. However, this efficiency term is not used very often. Because when the efficiency is 0, the airplane is not moving according to the formulation.

$$\eta_{overall} = \eta_e * \eta_p \quad (1.15)$$

$$\eta_{overall} = \frac{FV_{inlet}}{\dot{m}_f Q_{net}} \quad (1.16)$$

The most used efficiency in jet engines is the amount of fuel per thrust. Its unit is Kg/hN. Specific fuel consumption (SFC) equation is given in 1.17.

$$SFC = \frac{\dot{m}_f}{F} \quad (1.17)$$

1.4. Types of Jet Engines

Although all jet engines share essentially the same operating principles, the different types differ from each other with significant advantages and disadvantages. In this section, turbojet, turbofan and ramjet engines that produce thrust, turboprop and turboshaft engines that produce shaft power will be discussed.

a) Turbojet

The turbojet engine is conceptually simple. Structurally, it is a gas generator based on the Brayton cycle with the addition of a diffuser and nozzle. The air first comes into the diffuser, where its speed is slowed down, and its pressure increases. It is then compressed by the compressor and burned with fuel in the combustion chamber, then expanded in the turbine to a pressure sufficient to feed the system, and then leaves the system through the nozzle at high speeds, producing the thrust required for the aircraft. Although turbojet engines perform well at high altitudes and speeds, fuel consumption is high compared to other types of jet engines (Figure 1.9).

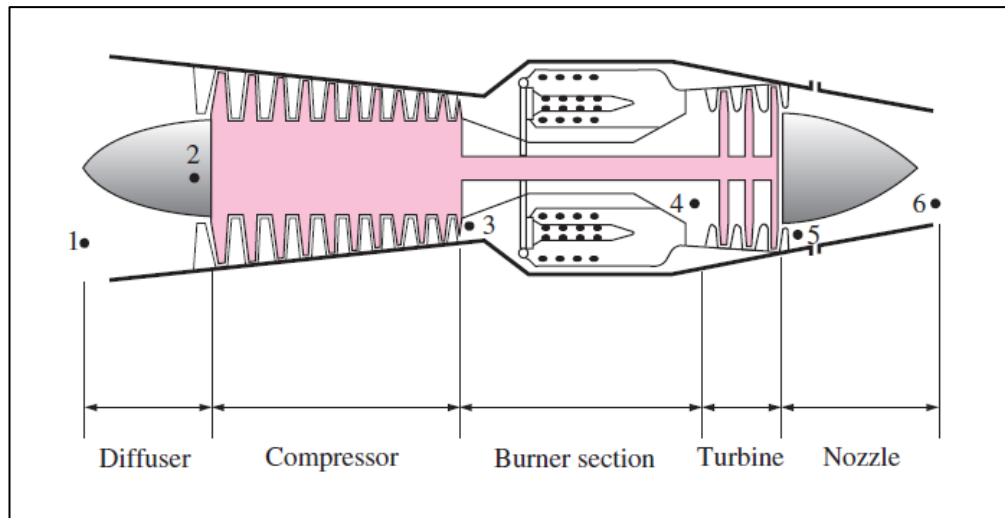


Figure 1.9: Section of a turbojet engine (Cengel et al., 2011).

b) Turbofan

Today, turbofan engines (Figure 1.10) are the most widely used engine type due to their many advantages. The most obvious structural difference from turbojet engines is the fan located in front of the compressor. Turbofan engines work on the principle of producing more thrust by combining a higher volume of air at low speeds with a lower volume of air at high speeds. Thanks to this principle, it works quieter, more efficiently and economically than turbojet engines. However, due to the fan, the front area is larger than the turbojet engine. The turbine also generates power for the fan. In a turbofan engine, the ratio of the mass flow rate of the air passing outside the combustion chamber to the mass flow rate of the air passing through the combustion chamber is called the bypass ratio. Turbofan engine with afterburner is used in F-15 Eagle and F-16 Falcon fighter aircraft.

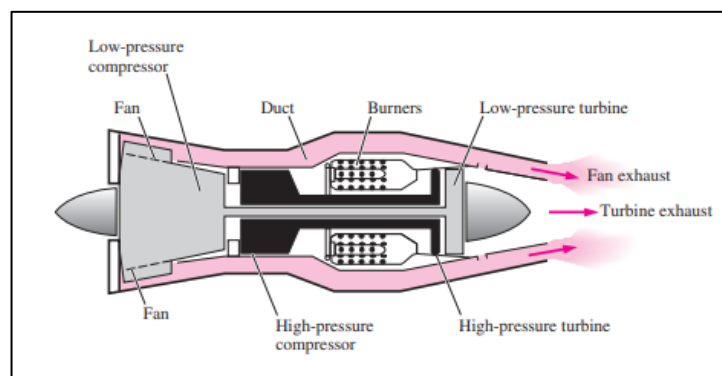


Figure 1.10: A turbofan engine (Cengel et al., 2011).

c) Turboprop

It is a type of jet engine used in propeller aircraft. Structurally, it consists of two main parts. These are the core parts consisting of the compressor, combustion chamber and turbine, and the propeller that produces thrust. Unlike turbofan and turbojet engines, thrust is not produced by high-speed exhaust gases. The turbine absorbs the high temperature and pressure air from the combustion chamber and transmits this power to the gearbox connected to the shaft. After making the necessary speed adjustment, the gearbox transmits this power to the propellers where the thrust is generated. Since the efficiency of the propellers decreases at high speeds, they are generally used in low-speed cargo aircraft. In addition, it is used as an unmanned aerial vehicle in the military field due to its low unit cost in targets and kamikazes that will be suitable for its purpose (Walsh et al., 2004) (Figure 1.11).

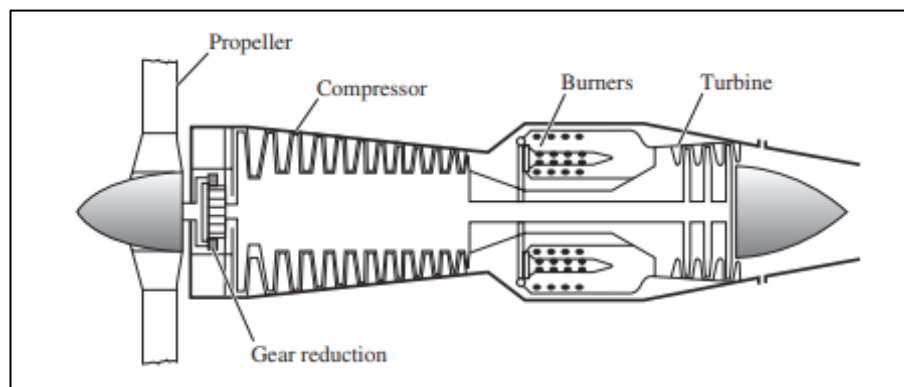


Figure 1.11: A turboprop engine (Cengel et al., 2011).

d) Turboshift

Turboshift engines are designed to provide horsepower to the system they are involved in. The engine core is like turbojet engines. High-energy air from the combustion chamber is absorbed by the power turbine. This power is transferred to the system through the shaft. They are used in helicopters, auxiliary power units, tank and ship engines due to their ability to provide continuous high power and compact structure (Figure 1.12).

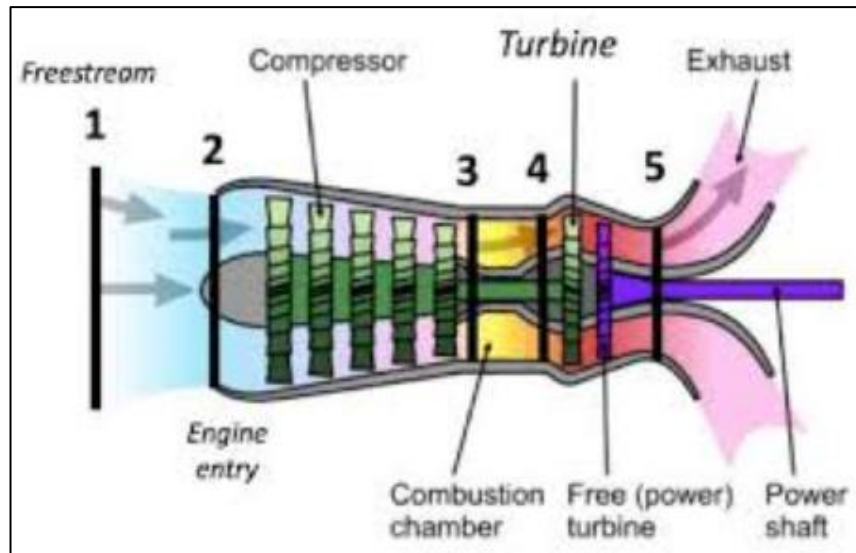


Figure 1.12: A turboshaft engine.

e) Ramjet

Ramjet engines are structurally the engines with the fewest components compared to other engine types. There is no compressor and turbine in these engines, so they have a light and simple structure. For the system to fulfil the task of the compressor, it must be accelerated to high speeds by means of an external system. Thus, the air reaches a certain speed. This is called ram effect. For this reason, the system cannot produce thrust under static conditions. The air coming into the combustion chamber at high speed is slowed down to subsonic speed and burned with fuel, then it is passed through the nozzle and the air whose speed increases produces thrust. As the name suggests, ramjet engines work according to the principle of ram effect (Figure 1.13).

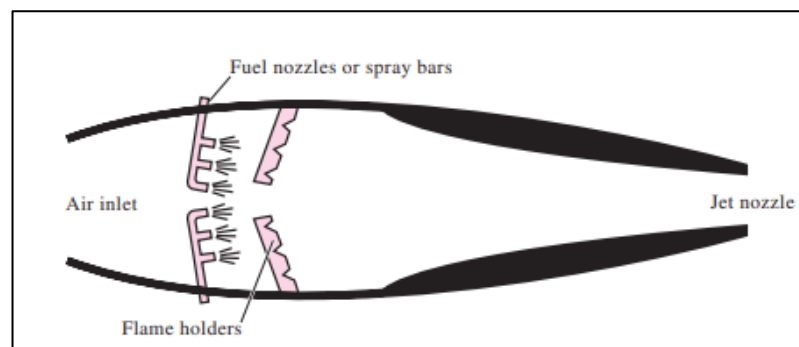


Figure 1.13: A ramjet engine (Cengel et al., 2011).

1.4.1. Comparison of Jet Engine

As can be seen in Figure 1.14, ram jet has operational capability at the highest altitude and Mach numbers among jet engines. Although they perform better than piston engines, turboprop engines (Table 1.1), which operate at lower performance compared to others, stand out with their low fuel consumption. Turboprop engines have the lowest speed among jet engines. Turbofan engines are more advantageous than turbojet engines due to their lower fuel use/thrust and because they are faster than a turboprop engine. One of the main differences between turboprop and turbofan engines is the bypass ratio. While it is approximately 100 in turboprop engines, this ratio is 5-6 in turbofan engines (Cengel et al., 2011).

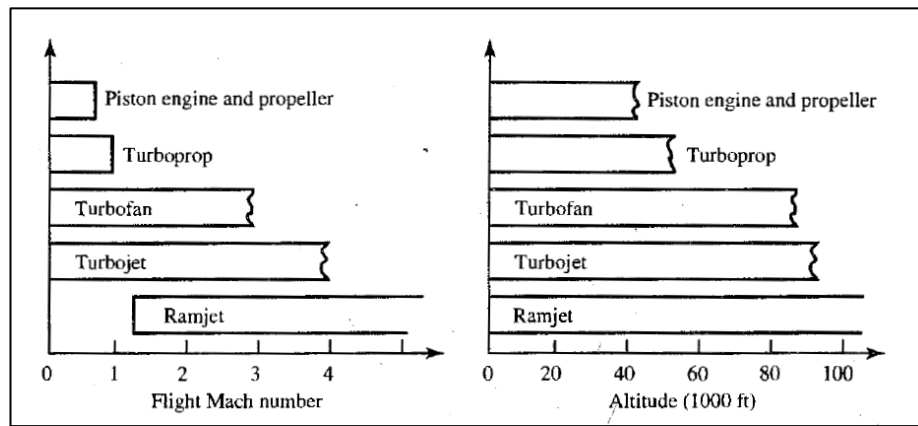


Figure 1.14: Jet engines altitude and flight Mach number (Mattingly, 2005).

Table 1.1: Specifications of reactive engines (Flack, 2005).

	Ramjet	Turbojet	Turbofan	Turboprop
Speed	Very High	High	Moderate	Low
Thrust	Moderate	High	Moderate/Low	Low
Fuel Use/Thrust	High	High	Moderate	Low
Thrust/Air Flow	High	High	Moderate	Moderate
Ground Clearance	NA	Good	Moderate	Poor

1.5. Micro Gas Turbine

In this study, a small-scale turbojet engine was selected to validate the mathematical model developed to study engine performance and thermodynamic properties. This choice was made because it provides easier access to the required performance and test data compared to other engines. Compact design, high power output, low weight,

ability to run on various fuels (kerosene, JP-4, JP-8, paraffin, diesel, etc.) and low maintenance costs are important advantages of this engine. Thanks to these features, micro gas turbine engines play a critical role in applications in the defense industry, including target drones, kamikaze drones, and reconnaissance UAVs (unmanned aerial vehicles). In addition, their ability to serve as generators in areas with limited access or infrastructure is a significant advantage. MGTs can generate 20-230 kW of power (Asgari, 2014). Figure 1.15 shows the performance development of the micro-scale jet engine. Long term goals for micro turbine engines include lower specific fuel consumption and increased thermal efficiency. Table 1.2 shows the characteristics of a standard MGT. Vannoy et al. (2015) conducted experimental tests of engine performance components according to the sensor layout shown in Figure 1.17.

Table 1.2: General MGT characteristics (Massardo et al., 2002).

Component	Feature
Turbomachine type	Variable speed single shaft rotor
Rotational speed, RPM	Approx. 100,000
Thermal efficiency	28% - 30%
Compressor	Single stage radial flow
Compressor pressure ratio	4,0
Turbine	Single stage radial flow
Turbine inlet temp., °C	900
Combustor	Low emission annular or can type
Machine start	Generator run as motor from battery
Technology status	State-of-the-art

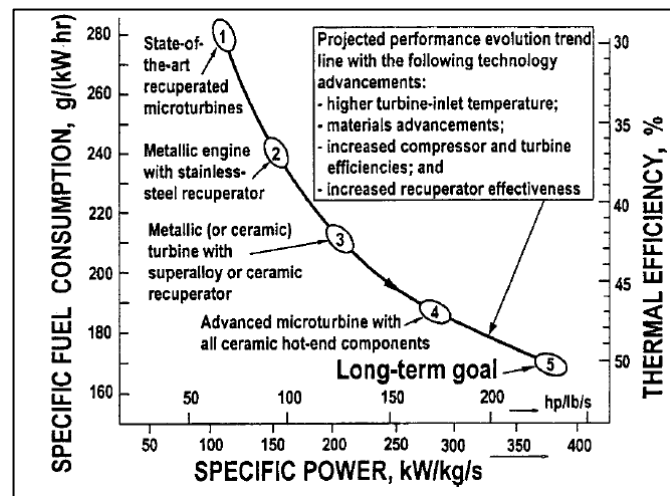


Figure 1.15: Performance development of the MGT (Massardo et. al, 2002).

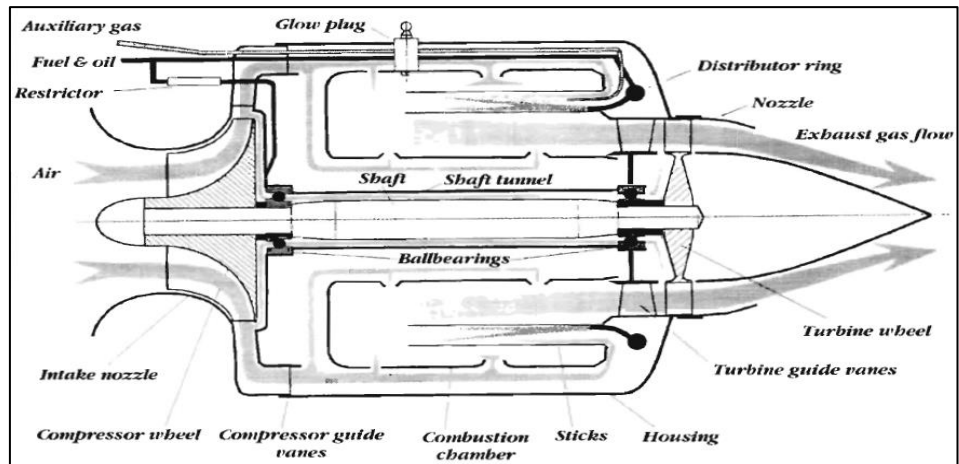


Figure 1.16: Parts of a micro turbine engine (Schreckling, 2005).

As can be seen in Figure 1.16, a cross section of a standard MGT is given. The main parts are air intake, compressor (rotor)-diffuser (stator), combustion chamber, shaft, shaft tunnel, turbine guide vanes (stator)-turbine (rotor) and nozzle. Auxiliary systems are two bearings (for bearing the turbine and compressor), glow plug (to start the first ignition), lubrication and fuel line. Bearing lubrication is done by fuel. There are also two types of starting systems. Gas (propane) or electric engine start. Electric systems have a brush or brushless starting motor. Figure 1.17 shows the sensor layout of the AMT Olympus turbojet used for the experimental tests conducted by Vannoy and Cadou (2015). Accordingly, the total temperature and total pressures at the stations were measured.

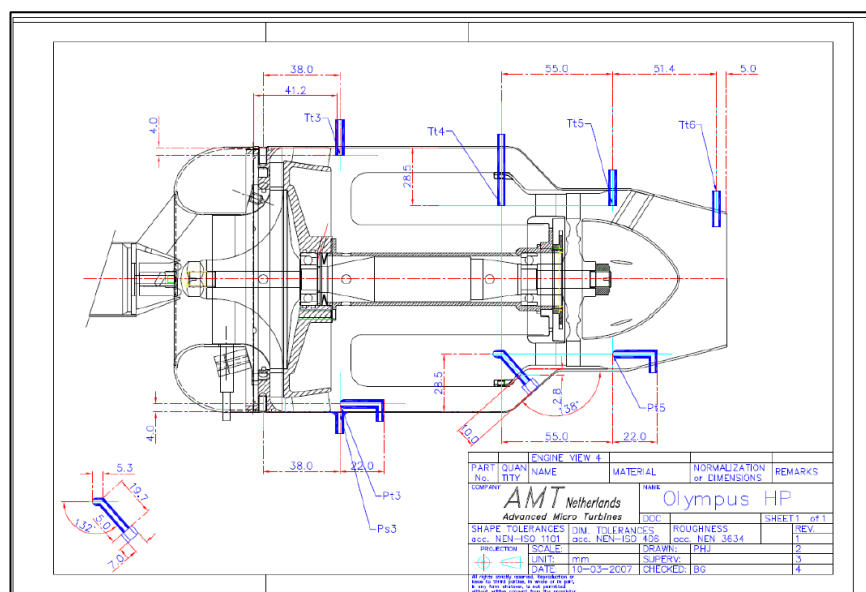


Figure 1.17: AMT Olympus sensor layout for testing (AMT Netherlands, 2019).

1.6. Literature Review

In this section, studies on the performance of gas turbine engines are presented.

The study conducted by Leylek (2012) focused on the performance modeling of the AMT Olympus turbojet engine. In this context, the necessary modeling tool for the engine was developed. Performance maps were used for the compressor and turbine models. The compressor performance map was obtained by scaling a previously published compressor map through testing. The turbine performance map was determined through CFD analysis. The developed model was compared with test results and data published by the engine manufacturer. As a result of these studies, investigations into thrust augmentation options, modeling and analysis of infrared suppression technologies, and research into the development of alternative fuels were carried out.

Vannoy and Cadou (2015) developed an engine performance model to integrate a solid oxide fuel cell (SOFC) system into a micro turbine engine platform. They used AMT Olympus turbojet engine as the engine choice and Numerical Propulsion System Simulation (NPSS) as the modelling program. In the modelling, they used a ready-made low pressure turbine map embedded in the NPSS software since the turbine performance map was not published while the compressor performance map was adhered to. The modelling was validated through experimental tests.

Opping (2016) investigated the performance of the BMT 120 KS micro gas turbine engine. Turbine and compressor performance maps were used in the model. Analytical and numerical calculations were used to evaluate the thermodynamic characteristics of the engine and to investigate component losses. The engine performance has been modelled using commercial software GasTurb, Flownex SE and one-dimensional mathematical code for in-house. The results are validated by experimental tests. In addition, an analysis of the mean-line turbine design was performed to improve the performance of the axial turbine.

Al-Hamdan and Ebaid (2006) have developed a real-time performance modelling of a gas turbine engine for power generation based on performance maps. They used the turbine-compressor matching method.

Chapman, Lavelle, & Litt (2016) used Toolbox for the Modelling and Analysis of Thermodynamic Systems (T-MATS), a specialized program for modelling and analyzing the systems of a turbojet engine. The paper focuses on 3 main topics: component level modelling, system level modelling using limited data sets and performance analysis. The system inputs are fuel flow rate, Mach number and standard day delta temperatures while the output data is thrust. System modelling is based on the Brayton cycle and component modelling is performed using performance maps. (Figure 1.18)

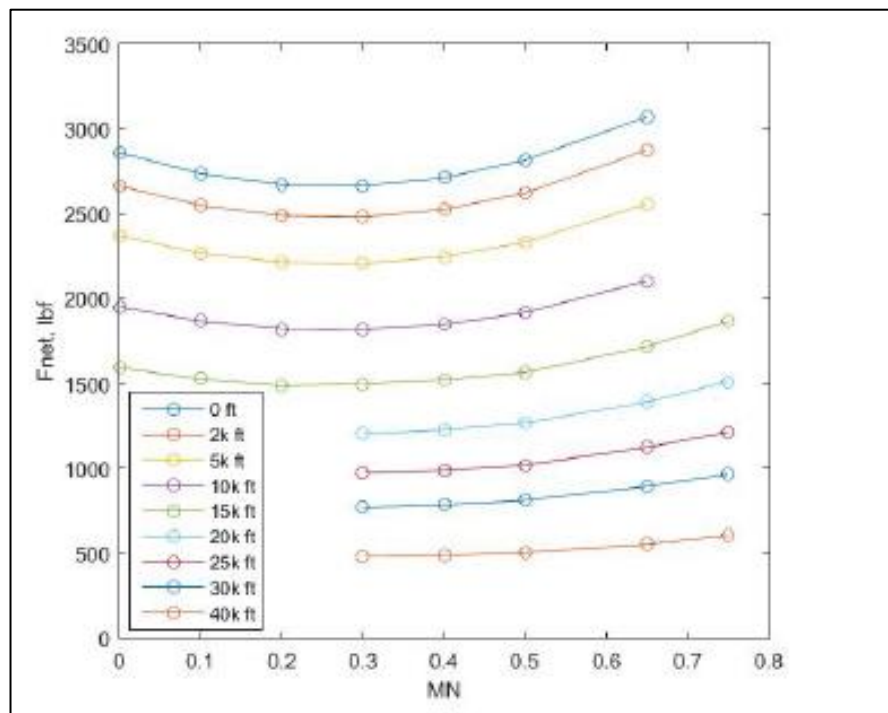


Figure 1.18 Flight envelope study at different altitudes and Mach numbers (Chapman et al., 2016).

In the study by Rahman (2009) the dynamic modelling and overall performance of the AMT Olympus turbojet engine under variable bleed conditions were investigated. The modelling approach focuses on creating a mathematical model of the turbine and compressor using performance maps and relating all components to each other. Two approaches were considered in performance modelling. These are inter-component volume method and constraint satisfaction iterative method. As a result, the inter-component volume method was chosen for real-time performance analysis due to its simplicity. The study simulates both user-defined fuel flow demand (open loop) and

flow demand (closed loop) responses set by the RPM control. The modelling is validated by experimental tests.

In his study, Lazzaretto (2001) modelled a gas turbine suitable for power generation by scaling the performance maps of generalized compressors and turbines in accordance with his own study. He validated his model with test data taken from real plants. One of the important parts of his study is to obtain off-design performance with compressor map modifications according to variable guide vane (VIGV) closure. In his study, he used a commercial package for analyzing neural net work and analytical equation systems.

2. METHODOLOGY

The engine model is expressed by equations representing the aerothermal process of each engine component and appropriate to the component character. Within the scope of this thesis, the performance behavior of the engine is investigated both at sea level and at different altitudes and Mach numbers. In the first part of the study, the engine was calculated under standard sea level conditions. In the model, the engine was operated from idle speed to full power. The initial conditions are P_{t1} (101,325 kPa), T_{t1} (288,15 K) and $Ma=0$. Since the air intake is designed adiabatically, $T_{t1} = T_{t2}$ and pressure loss is assumed constant at all shaft speeds. Compressor efficiency (η_c), compression ratio (r), mass flow rate ($\dot{m}_{air}$) are extracted from the compressor performance map depending on shaft speed. Subsequently, the air coming from the compressor to the combustion chamber is burned with fuel ($\dot{m}_{fuel}$). The efficiency and pressure loss in the combustion chamber are constant at all shaft speeds. The high-energy air is expanded in the turbine to the power required for the compressor operation. Since there is no turbine performance map, turbine output conditions (P_{t5}, T_{t5}) are obtained by including mechanical efficiency with compressor-turbine work balance formulations. Since the nozzle is modelled adiabatically, $T_{t5} = T_{t8}$. There is a pressure loss between the turbine and the nozzle, so this ratio is taken constant. Since all the pressure in the nozzle expands to the atmosphere, the thrust is found by the momentum equation. Since the fuel flow rate is very low compared to the air flow rate, it is not included in the thrust formulation. In the model, the engine is divided into two parts as cold zone (intake, compressor) and hot zone (combustion chamber, turbine and nozzle) and the specific heat ratio ($\gamma_{hot}, \gamma_{cold}$) and specific heat of the air (c_{hot}, c_{cold}) are calculated accordingly. In addition, the data considered constant in the model are given in Table 2.1.

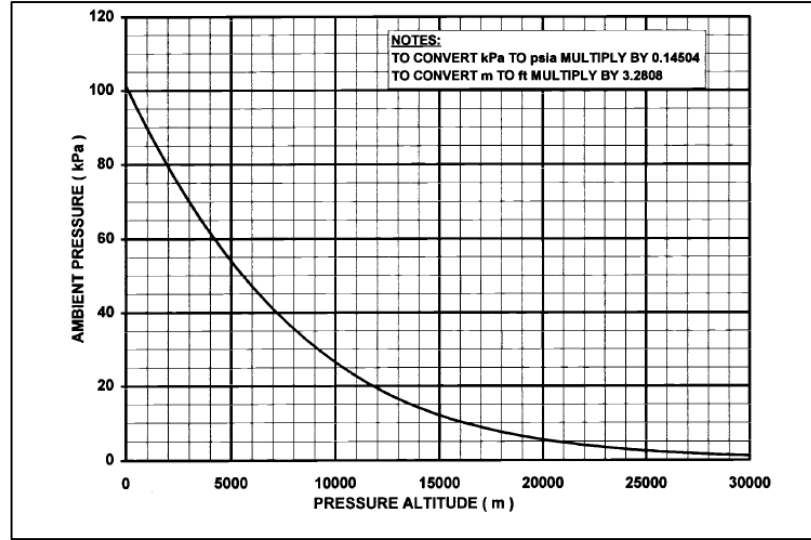


Figure 2.1: Ambient pressure vs altitude (Walsh et al., 2004).

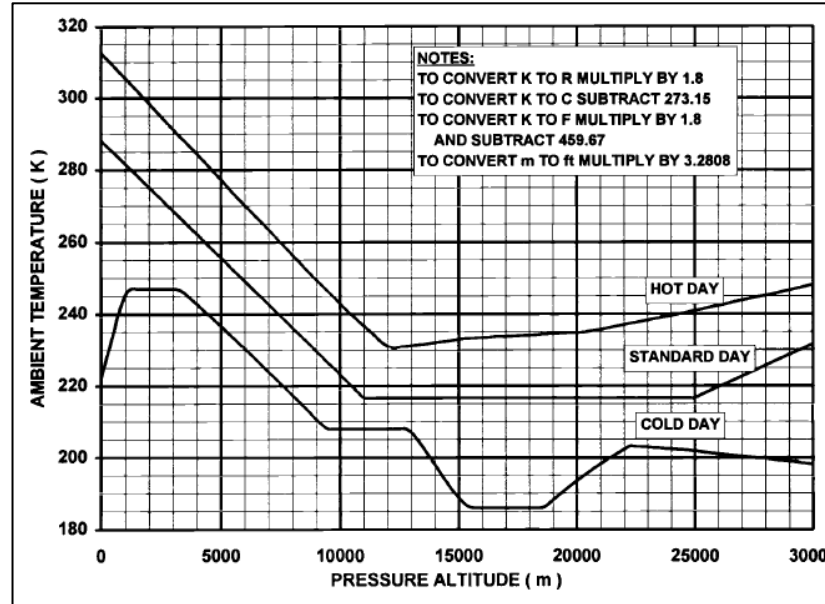


Figure 2.2: Ambient temperature vs altitude (Walsh et al., 2004).

In the second stage of the study, to analyze the performance behavior of the engine in flight, the input conditions varied according to altitude and Mach number. As can be seen in Figure 2.1 and Figure 2.2, the ambient temperature (T_{t1}) and pressure (P_{t1}) were varied according to the altitude. The model was operated between 0-8000 meters and 0-0.9 Mach and at the speed corresponding to the maximum power. With the increase of Mach number, inlet temperatures and pressures of the air were calculated with stagnation formulations. A summary of the mathematical model is given in Figure 2.3.

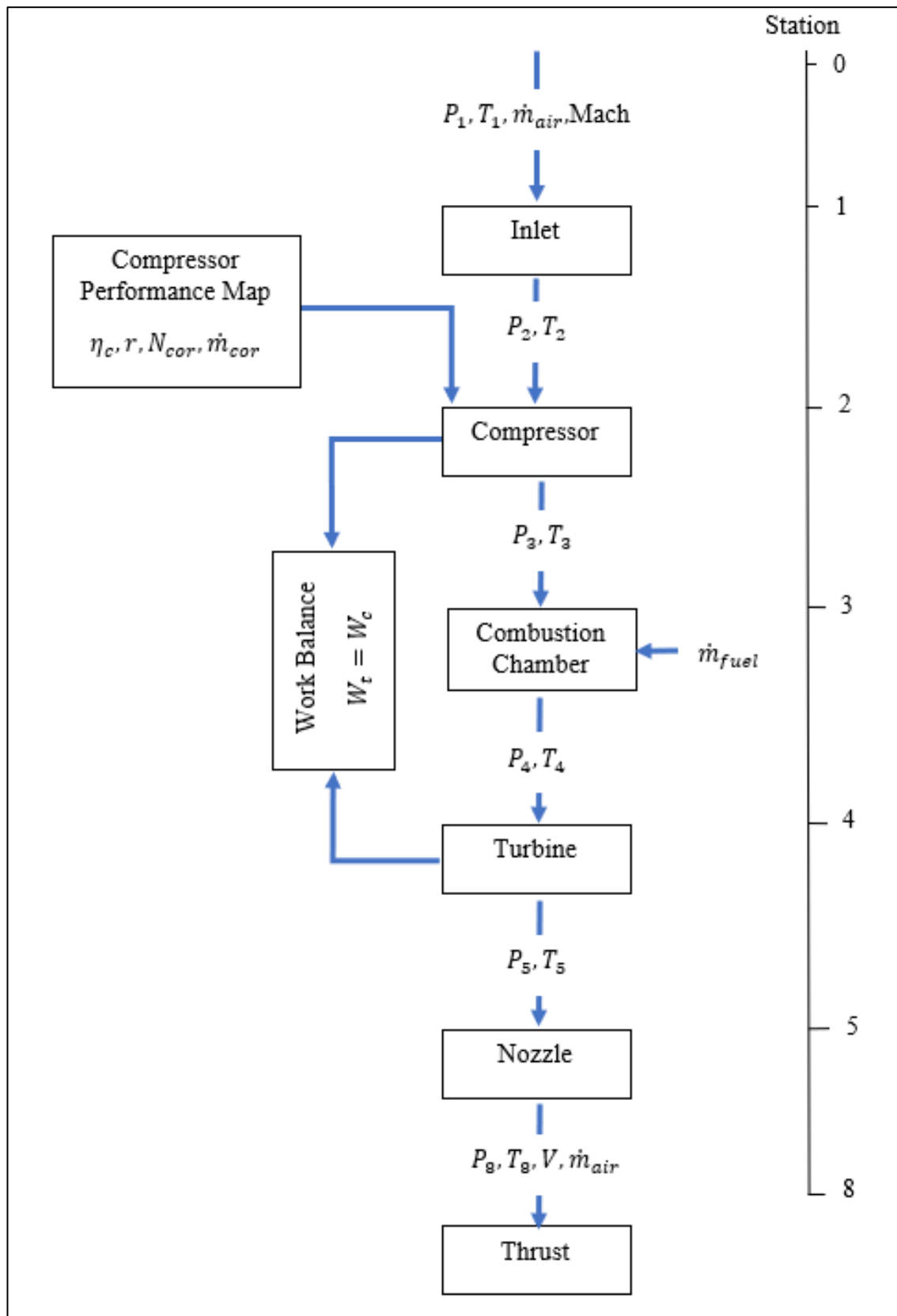
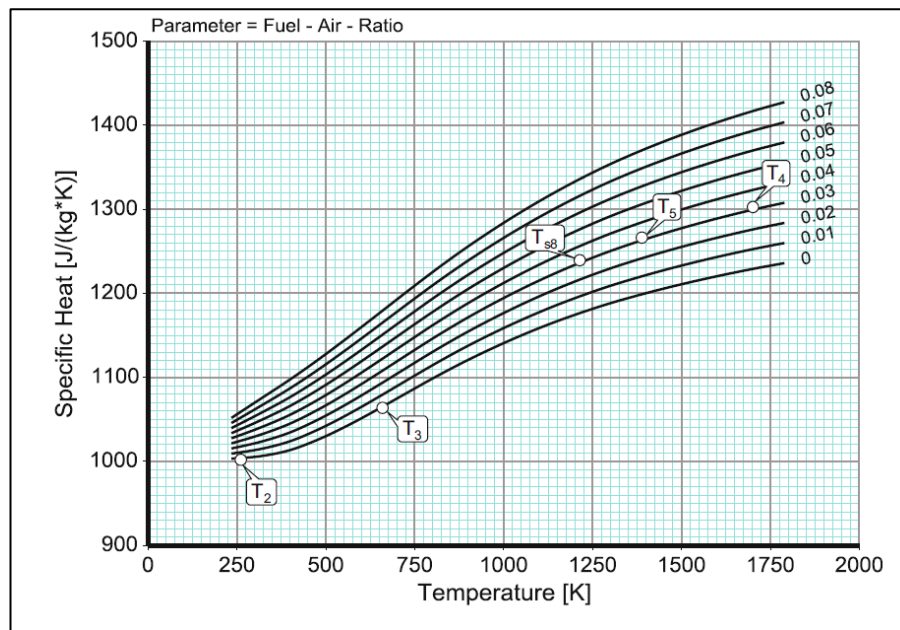


Figure 2.3: Mathematical model schematic.

Table 2.1: List of model parameters.

Parameters	Value
Intake pressure ratio	98%
Intake efficiency	100%
Combustion chamber Pressure ratio	98%
Combustion efficiency	98%
Fuel heat value (Mj/kg)	43,25
Turbine efficiency	80%
Exhaust Pressure ratio	99%
Exhaust efficiency	100%
γ (Cold section)	1,4
γ (Hot section)	1,33
R (J/(kg/K)	287
C_p (Cold section) (J/(kgK)	1005,6
C_p (Hot section) (J/(kgK)	1148,36
Mechanical Efficiency	98%

**Figure 2.4:** Specific heat of air depending on temperature (Kurzke, 2018).

The data considered constant in the mathematical model are given in Table 2.1. The cold parts of the engine are intake and compressor while the hot parts are combustion chamber and exhaust. The specific heat of the air in these regions is taken according to Figure 2.4.

2.1. AMT Olympus Turbojet

Technical information provided by the engine manufacturer is shown in Table 2.2. All modelling was done within these data and limitations. Figure 2.1 shows a cross-sectional view of the AMT turbojet engine. The engine structures are radial compressor diffuser, annular combustion chamber, turbine nozzle guide vane and exhaust. There is also an exhaust gas temperature (EGT) sensor and a speed sensor for engine control. An electric starter is used to start the engine. The fuel system includes a fuel pump and valves. A glow plug is used to ignite the air-fuel mixture in the combustion chamber.

Table 2.2: AMT Olympus turbojet technical data (AMT Netherlands, 2019).

Technical data for AMT Olympus Turbojet	
Thrust	230 N
Pressure Ratio	4:1
Amount of air intake	450 gr/sec
Maximum revolution	108,000 RPM
Maximum allowed revolution	112,000 RPM
Maximum exhaust gas temperature	700 °C
Maximum allowed exhaust gas temperature	750 °C
Fuel usage	640 gr/min
Fuel type	Kerosene/paraffin/A-1/white spirit
Engine diameter	130 mm
Engine length	374 mm
Maximum operating altitude	8000 M
Maximum operating speed	1000 Km/hour
All data at S.T.P: 15°C and 1013 mBar	

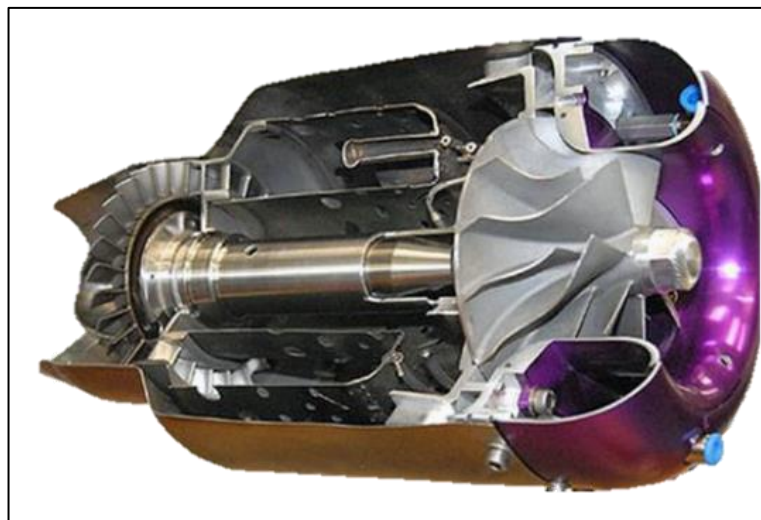


Figure 2.1: AMT Olympus turbojet (AMT Netherlands, 2019).

2.2. Stagnation Properties

When an aircraft moves in the atmosphere, the surrounding air molecules are affected and move around the aircraft. This interaction causes aerodynamic forces between the aircraft and the air. The speed of the aircraft (V) is determined by the Mach number, which is proportional to the speed of sound (c) in air.

$$c = \sqrt{\gamma RT} \quad (2.1)$$

In Equation (2.1), γ is the specific heat, R is the gas constant of air and T is the total temperature.

$$Ma = \frac{V}{c} \quad (2.2)$$

Temperature in moving airflow has the concepts of static and dynamic. Static temperature refers to the temperature measured when the air is not moving. Dynamic temperature is related to the kinetic energy of the air movement. The total temperature is the sum of the static temperature and the dynamic temperature and varies depending on the Mach number of the air flow. Stagnation temperature is the temperature measured when the airflow is isentropically stopped.

The isentropic total temperature equation for a perfect gas is given by the following equation 2.3.

$$\frac{T_0}{T} = 1 + \left(\frac{\gamma - 1}{2}\right) Ma^2 \quad (2.3)$$

In the process of slowing down the air to the stagnation temperature, the air heating depends on the specific heat capacity of the air. If this capacity is constant, the air is called thermally perfect.

Depending on the total temperature formulation, the pressure and density of the fluid reach the stagnation pressure and stagnation density pressure when it is isentropically brought to the stagnation point.

The isentropic total pressure equation (2.4) and density equation (2.5) for a perfect gas are given below.

$$\frac{P_0}{P} = \left[1 + \left(\frac{\gamma - 1}{2} \right) Ma^2 \right]^{\frac{\gamma}{\gamma - 1}} \quad (2.4)$$

$$\frac{\rho_0}{\rho} = \left[1 + \left(\frac{\gamma - 1}{2} \right) Ma^2 \right]^{\frac{1}{\gamma - 1}} \quad (2.5)$$

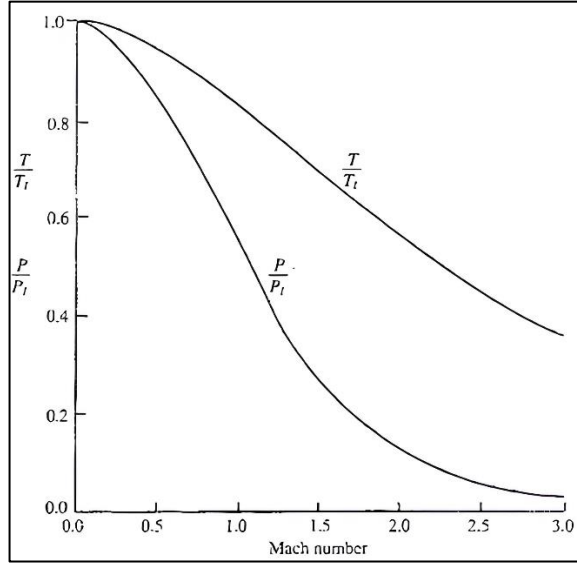


Figure 2.5: Comparison of temperature and pressure ratios with Mach number (Mattingly, 2005)

As can be seen in Figure 2.5, the changes of P/P_t and T/T_t ratios at the speed of sound ($Ma = 1$), subsonic ($Ma < 1$), supersonic ($Ma > 1$) and hypersonic ($Ma \gg 1$) speeds are shown. In these calculations, $\gamma = 1.4$. In this study, the engine operates at subsonic speeds.

2.3. Component Modelling

2.3.2. Intake Modelling

The air intake (Figure 2.6) is modelled as an adiabatic system. In the first part of this study, the air is calculated under static conditions ($Ma_1 = 0$), while the other part is dynamic ($Ma_1 \neq 0$). As a result of the tests and researches conducted on similar intake efficiencies in the literature research, the air intake was modelled by accepting the total

pressure loss as constant. In the study of Rahman (2009) accepted 100% (no pressure loss), Leylek (2012) 98%, Oppong (2016) 100% in his study.

$$V_1 = Ma_1 \sqrt{\gamma_c R T_{amb}} \quad (2.6)$$

$$\left(\frac{T_{t_1}}{T_0}\right) = \left(1 + \frac{\gamma_c - 1}{2} Ma_1^2\right) \quad (2.7)$$

$$\left(\frac{P_{t_1}}{P_0}\right) = \left(1 + \frac{\gamma_c - 1}{2} Ma_1^2\right)^{\frac{\gamma_c}{\gamma_c - 1}} \quad (2.8)$$

P_{t_2} used as inlet pressure to the compressor is provided by following formulation 2.9. Efficiency is accepted as 98%.

$$P_{t_2} = \eta_i * P_{t_1} \quad (2.9)$$

In the formulations of the air intake, which is accepted as the cold zone in the engine, the specific heat ratio is $\gamma_c = 1,4$ and the specific heat is $c_p = 1,005 \frac{kJ}{kg} K$.

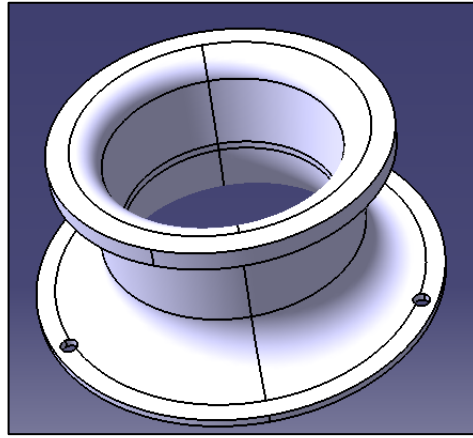


Figure 2.6: Intake for MGT (Web 1, 2024).

2.3.3. Compressor Modelling

The AMT Olympus turbojet engine uses a radial compressor to compress the air flow and feed it to the combustion chamber of the engine. At the same time, a radial diffuser is used, which reduces the velocity of the air at the compressor outlet and increases the static pressure. Since radial compressors have one stage higher compression ratio compared to axial compressors, they are preferred in such simple engines. Figure 2.7

shows an example of a compressor and diffuser used in a micro turbine engine. The diffuser increases the static pressure while reducing the velocity of the high velocity air coming from the compressor. According to Saravanamuttoo et al. (1996), half of the pressure behind the compressor in design applications is provided by the diffuser.



Figure 2.7: Radial compressor and diffuser for MGT (Web 1, 2024).

Compressor performance is represented by a compressor performance map, which includes corrected mass flow rate, shaft speed, compression ratio and efficiency parameters. This map shows the behavior of the compressor under different operating conditions and is used in modelling. A 1% loss in compressor efficiency corresponds to a 0.5% increase in specific fuel consumption. This rate cannot be neglected in calculations (Kurzke,1996).

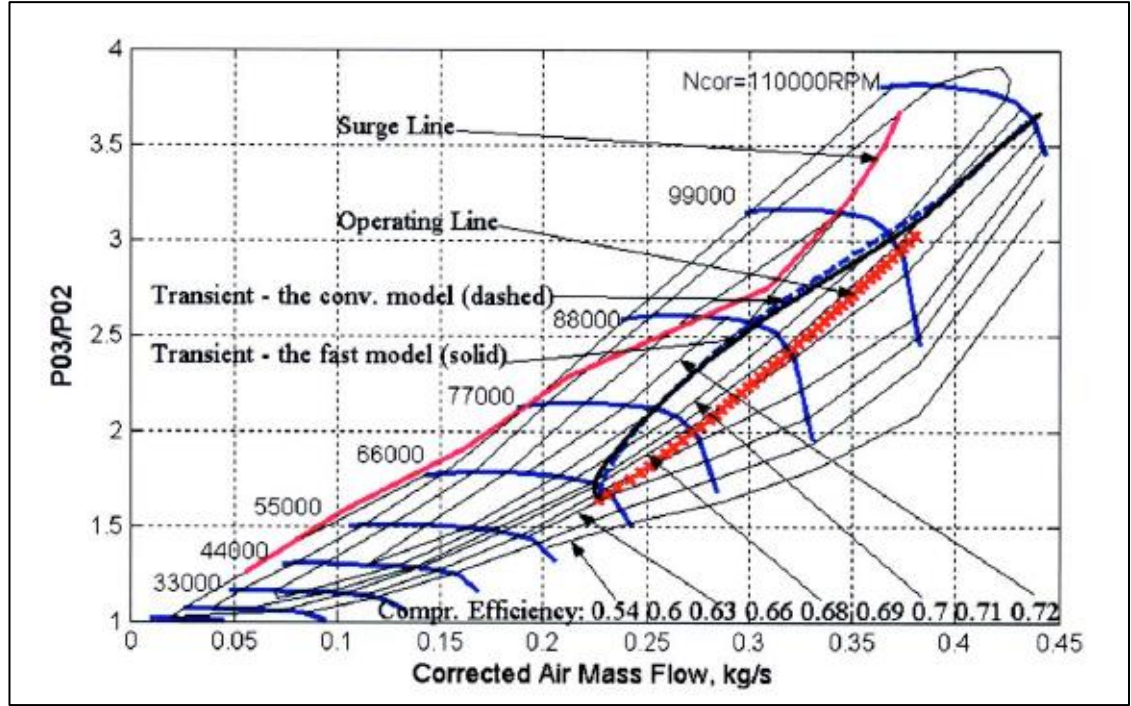


Figure 2.8: General compressor performance map (Lichtsinder, 2006).

As can be seen in Figure 2.8, x-axis is corrected air mass flow, y-axis is compression ratio, contour lines are drawn to correspond to efficiency and corrected speed lines to x and y axes. The compressor operates on an operating line depending on the operating conditions. The following corrected mass flow equation (2.10) and dimensionless corrected speed equation (2.11) are given.

$$\text{Corrected Flow Rate } (\dot{m}_{corr}) = \frac{\dot{m} * \sqrt{\frac{T_2}{288,15 \text{ K}}}}{\frac{P_2}{101,325 \text{ kPa}}} \quad (2.10)$$

$$\text{Corrected Speed} = \frac{\text{Speed}}{\sqrt{\frac{T_2}{288,15 \text{ K}}}} \quad (2.11)$$

Published performance maps of compressors with similar capacity can be scaled and adapted to the existing compressor. With this methodology, the compressor performance map scaled by Leylek (2012) was used in this study. In addition, Leylek

(2012) determined that the mass flow rate of the engine is 10% less than the manufacturer's values between the data obtained by his own tests. For this reason, the mass flow rate of 0.45 kg/s at 108K RPM is modelled as 0.49 kg/s.

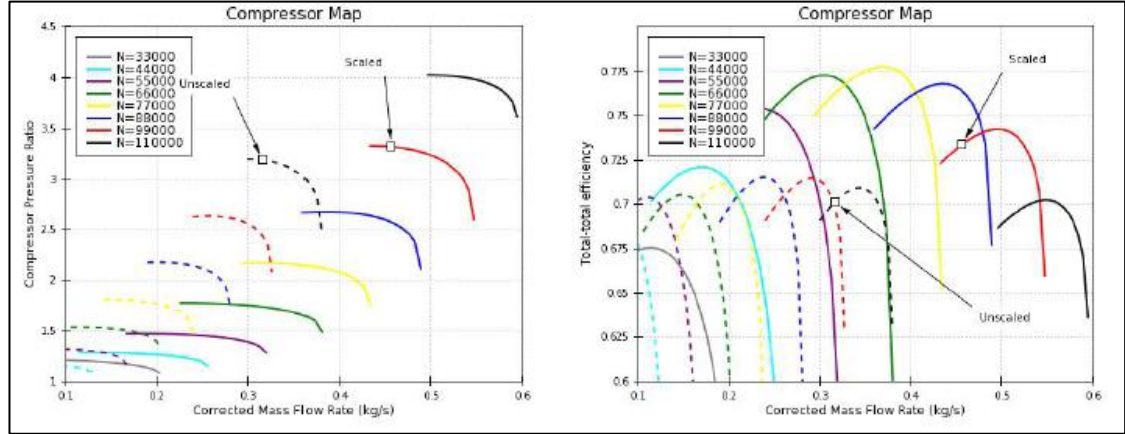


Figure 2.9: Scaled compressor map (Leylek, 2012).

As can be seen in Figure 2.9, a compressor performance map with a similar capacity was scaled and the compressor map used in the AMT Olympus turbojet was obtained.

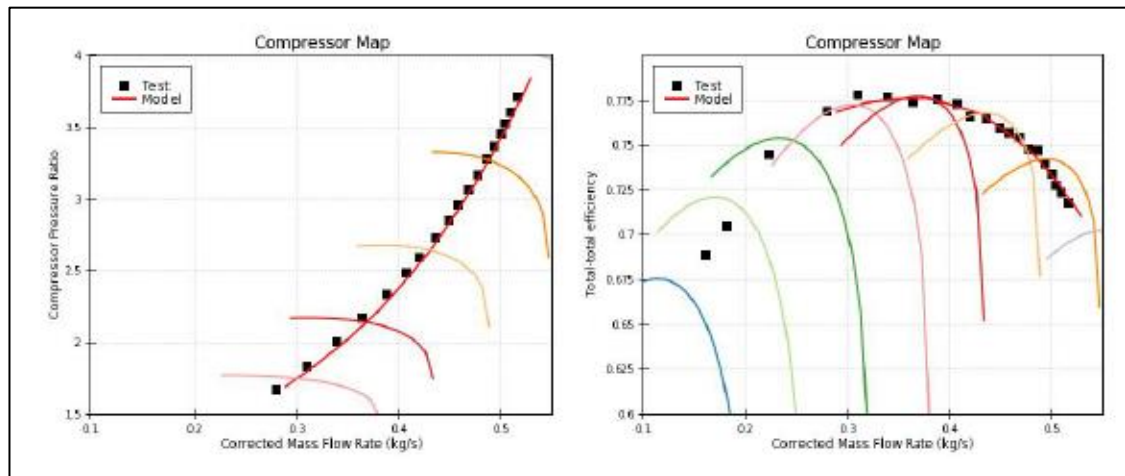


Figure 2.10: Compressor operating line (Leylek, 2012).

Figure 2.10 shows the operating line and efficiency lines used in the compressor modelling obtained from the experimental tests carried out by Leylek (2012).

In the model, station 2 represents the inlet conditions of the air into the compressor, while station 3 represents the outlet conditions.

The shaft speed dependent mass flow rate, compression ratio and efficiency are input data for compressor modelling. Temperature (T_3) and pressure (P_3) are obtained as

output data. In the compressor formulations, which are considered as the cold zone in the engine, the specific heat ratio is $\gamma_c = 1,4$ and the specific heat is $c_p = 1,005 \frac{kJ}{kg} K$.

With the compression ratio taken from the compressor map, the following equation (2.12) compressor outlet pressure is obtained.

$$r = \frac{P_{t3}}{P_{t2}} \quad (2.12)$$

The data T_{t3} , which will be the inlet temperature of the combustion chamber, is obtained by the following equation (2.13).

$$T_{t3} = T_{t2} \left\{ \frac{1}{\eta_c} \left[\left(\frac{P_{t3}}{P_{t2}} \right)^{\frac{\gamma_c-1}{\gamma_c}} - 1 \right] + 1 \right\} \quad (2.13)$$

2.3.4. Combustion Chamber Modelling

The engine uses an annular combustion chamber. (Figure 2.11) This is where the heated and pressurized air from the compressor is combusted with fuel. The main function of the combustor is to raise the air temperature to the maximum level allowed by the turbine metallurgy. The amount of fuel increases or decreases according to the stoichiometric ratio. Since the air also acts as a coolant, these engines operate with excess air. The fuel heat value (FHV) of the fuel is 43.25 MJ/Kg. In the model the combustion chamber inlet data are T_{t3} and P_{t3} and the outlet data are T_{t4} , P_{t4} and $\dot{m}_f$. The fuel quantity is taken from the Leylek (2012) experimental test data. While ideal combustion occurs at constant pressure, in reality there is a pressure drop between 1% and 5%. (Chapman et al., 2016) Given the geometric conditions in a good design, the total pressure loss at the design point should be 2%-4%. (Walsh et al., 2004) Therefore, the pressure drop is assumed to be constant at 2% in the model. Burner efficiency is the ratio of the amount of unburned fuel to the total amount of fuel supplied to the system. In Rahman (2019) study, combustor efficiency is assumed to be 98%. At design points, combustor efficiency is close to 100% (Kurzke, 2018). In GasTurb, a commercial engine performance analysis software, 100% is defined as the design

point. In practical studies, combustion chamber efficiency can be taken as 98-99% (Saravanamuttoo et al., 1996). In this study, combustion chamber efficiency is assumed to be 98%. In addition to the fuel, the AMT Olympus engine fuel is blended with 4.5% Aeroshell 500 turbine oil to lubricate the turbine and compressor bearings. (AMT Netherlands, 2019) The effect of the turbine oil on the system is ignored.

$$P_{t4} = P_{t3}(1 - \Delta P) \quad (2.14)$$

$$f = \frac{\dot{m}_f}{\dot{m}_a} \quad (2.15)$$

$$T_{t3}C_{p,cold}\dot{m}_a + \dot{m}_f FHV\eta_{combustion} = T_{t4}C_{p,hot}(1 + f) \quad (2.16)$$

In the combustion chamber formulations, which is considered as the hot zone in the engine, the specific heat ratio is $\gamma_h = 1,33$ and the specific heat is $c_h = 1,148 \frac{kJ}{kg} K$.



Figure 2.11: Combustion chamber for MGT (Web 1, 2024).

2.3.5. Turbine Modelling

As can be seen in Figure 2.12, an axial turbine is also used in this engine. Like the compressor modelling, the mathematical model of the turbine was modelled based on the performance map. Since the turbine map is not published, performance data were obtained by using the component matching method, which is a common method. The basic logic of turbine-compressor matching is that the power generated from the turbine should be equal to the power required by the compressor since they are connected to each other through the same shaft. An example of a general turbine performance map is given in Figure 2.13. In this engine, the turbine does not generate additional power for any auxiliary system. Turbine efficiency is taken as constant 80% and mechanical efficiency of the system is taken as 98%. In the model, turbine inputs are T_{t4}, P_{t4} and $\dot{m}_e$ while output parameters are T_{t5} and P_{t5} .



Figure 2.12: Axial turbine and Nozzle Guide Vane for MGT (Web 1, 2024).

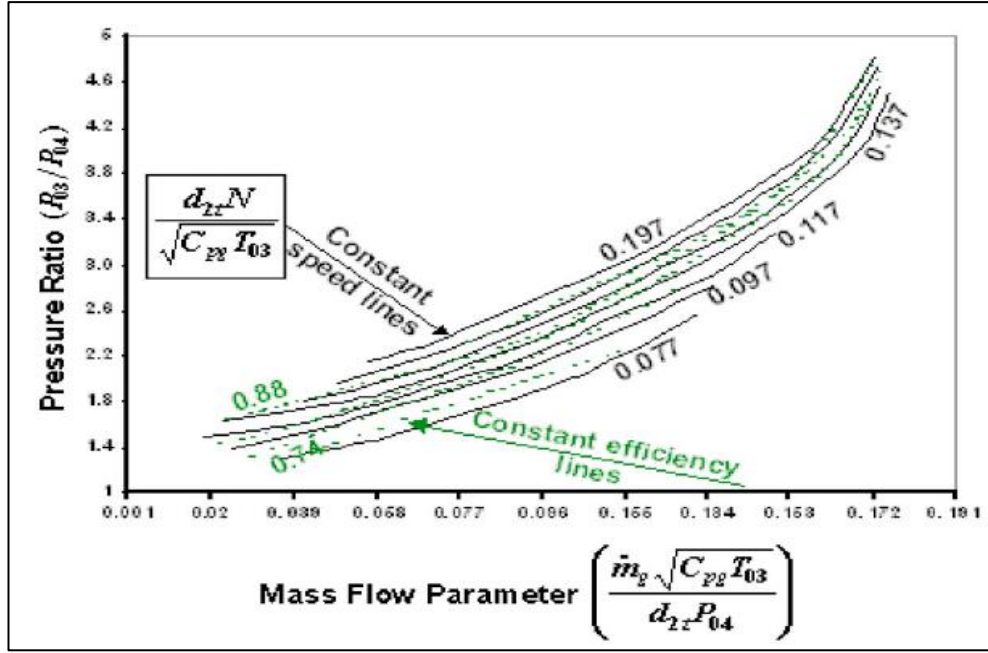


Figure 2.13: General turbine performance map (Al-Hamdan, 2005).

$$\dot{m}_e = \dot{m}_{fuel} + \dot{m}_{air} \quad (2.17)$$

$$(T_{t3} - T_{t2})C_{pcold}\dot{m}_a = (T_{t4} - T_{t5})C_{phot}\dot{m}_a\eta_{mech} \quad (2.18)$$

$$T_{t5} = T_{t4} - \frac{(T_{t3} - T_{t2})C_{pcold}\dot{m}_a}{C_{phot}\dot{m}_a\eta_{mech}} \quad (2.19)$$

$$P_{t5} = P_{t4} \left\{ \left[1 - \frac{1}{\eta_t} \left(1 - \frac{T_{t5}}{T_{t4}} \right) \right]^{\frac{\gamma_h}{\gamma_h - 1}} \right\} \quad (2.20)$$

In the turbine formulations, which is considered as the hot zone in the engine, the specific heat ratio is $\gamma_h = 1.33$ and the specific heat is $c_h = 1,148 \frac{kJ}{kg} K$.

2.3.6. Nozzle Modelling

Since no thermodynamic work is done at the nozzle, the turbine outlet temperature T_{t5} is taken directly equal to the exhaust gas temperature T_{t8} . Between the turbine outlet and the nozzle throat, P_{t5}/P_{t8} is close to 1. (Kurzke, 2004) The pressure drop across the nozzle is assumed to be 1%. Since $Mach < 1$ at the nozzle outlet, the pressure inside the nozzle is expanded to atmospheric pressure. Nozzle efficiency is assumed to be 1.

$$P_{t5} = P_{t8}(1 - \Delta P) \quad (2.21)$$

$$V_8 = \left(\sqrt{2c_h\eta_n T_{t8} \left[1 - \left(\frac{1}{NPR} \right)^{\frac{\gamma_h-1}{\gamma}} \right]} \right) \quad (2.22)$$

With the data obtained from the nozzle outlet, the thrust calculation was made with the following formulation (2.23).

$$F = V_8 * \dot{m}_a \quad (2.23)$$

In the nozzle formulations, which is considered as the hot zone in the engine, the specific heat ratio is $\gamma_h = 1,33$ and the specific heat is $c_h = 1,148 \frac{kJ}{kg} K$.

3. RESULTS

The output of the theoretically prepared model is presented in this section. In this study, test data published in the literature and by the manufacturer were used for verification of the results. The performance data of the engine is obtained at different stations by NPSS software in the study of Vannoy and Cadou (2015). In this context, $T_{t2}, T_{t3}, T_{t4}, T_{t5}$ and $P_{t2}, P_{t3}, P_{t4}, P_{t5}$ data obtained at certain shaft speeds in the thesis study were verified.

The shaft speed dependent exhaust gas temperature and thrust data published by the manufacturer and taken from the experimental tests conducted by Leylek (2012) were verified.

The performances of the engine at different altitude Mach numbers were compared with the data shared in the technical data package of the Stargate-TJ-40 engine of Rafael company.

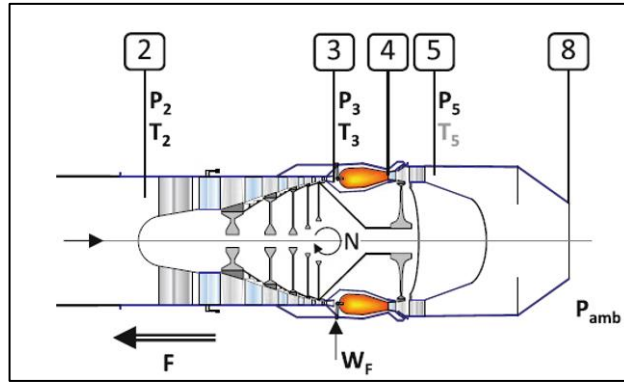


Figure 3.1: Station nomenclature of the mathematical model (Kurzke, 2018).

The station numbers of interrelated motor components in an axial plane are given in Figure 3.1.

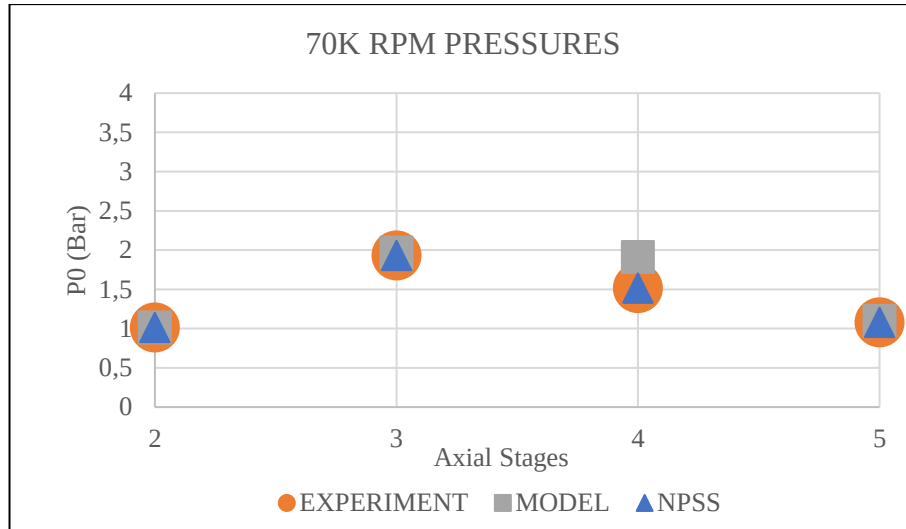


Figure 3.2: Station pressures at 70K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

Figure 3.2 shows the comparison of the station pressures of the engine at 70K RPM. In this context, the system starts with standard sea level conditions at station 2, the pressure ratio from the compressor performance map is obtained at station 3. At station 4, there is a deviation since the combustion chamber is modelled assuming constant pressure loss in the mathematical model. This deviation is corrected at station 5 at the turbine output.

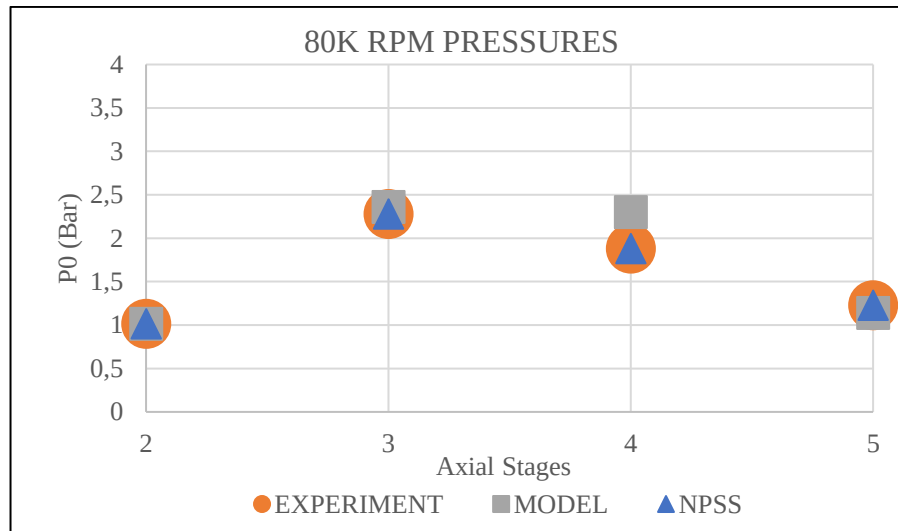


Figure 3.3: Station pressures at 80K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

As can be seen in Figure 3.3, the pressure loss differences at the combustion chamber outlet started to decrease with the increase in engine speed. Since the pressures are equalized at the turbine outlet, it does not give erroneous results.

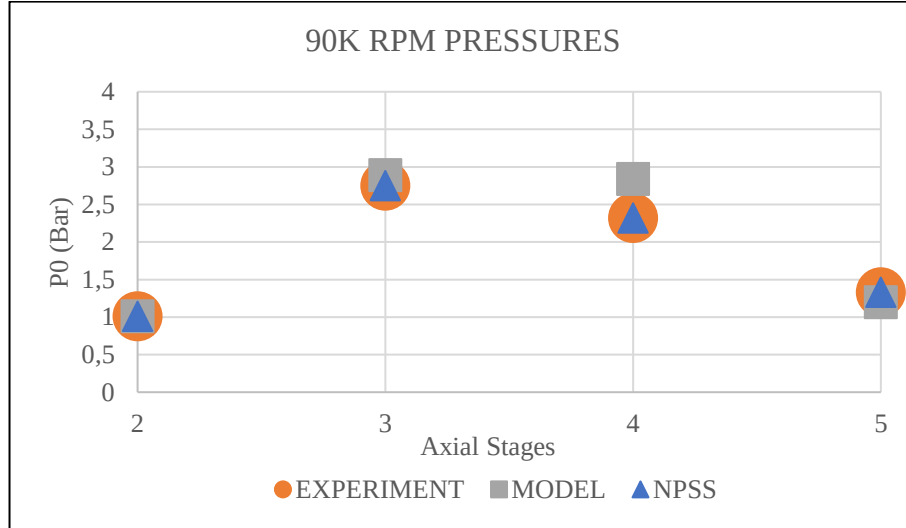


Figure 3.4: Station pressures at 90K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

As can be seen in Figure 3.4, there is a deviation in the compressor output at 90K RPM. The possibility that this deviation is due to the compressor performance maps used should not be ignored. Although the combustion chamber outlet is isobaric under ideal conditions, there are pressure losses in the real cycle. Since the pressure losses are assumed constant at all cycles in this model, this might be one of the reasons for deviation of data. This deviation is compensated for the turbine outlet pressure.

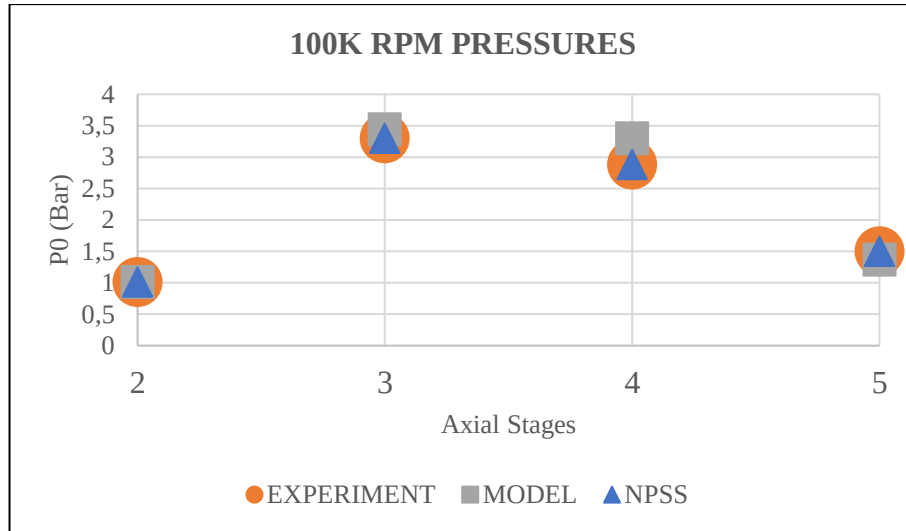


Figure 3.5: Station pressures at 100 K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

In the comparison given in Figure 3.5, it is observed that the combustion chamber pressure losses approach the experimental data and NPSS software package results due to the increase in engine speed.

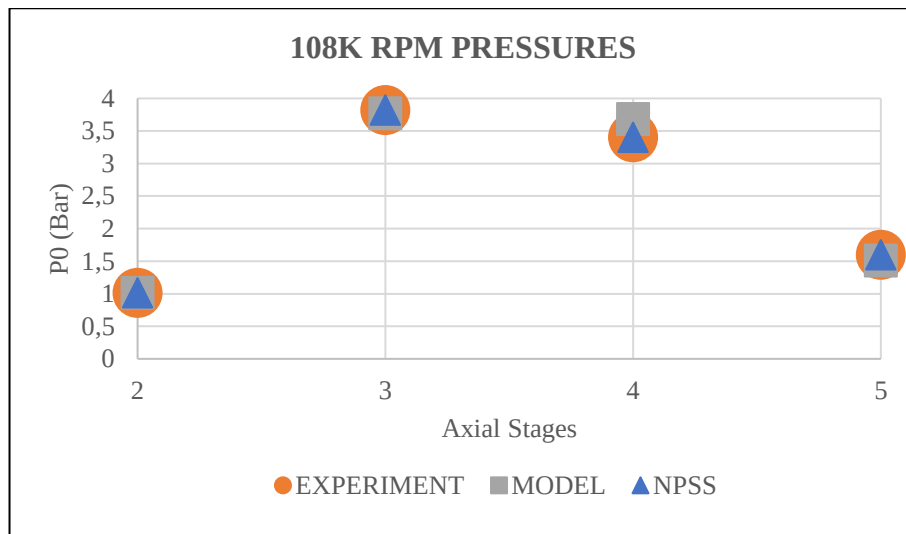


Figure 3.6: Station pressures at 108K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

At maximum engine speed, the difference at station 4 was minimized. As a result of the comparison of station pressures depending on engine speed, it was observed that the pressure losses accepted in the mathematical model were consistent with both test and NPSS software package results. Station 3 compressor outlet pressures were obtained from different performance maps. The air pressure ratios obtained at Station

5 have provided a practical approach to the modelling process because of the turbine performance map using the component matching method. (Figure 3.6)

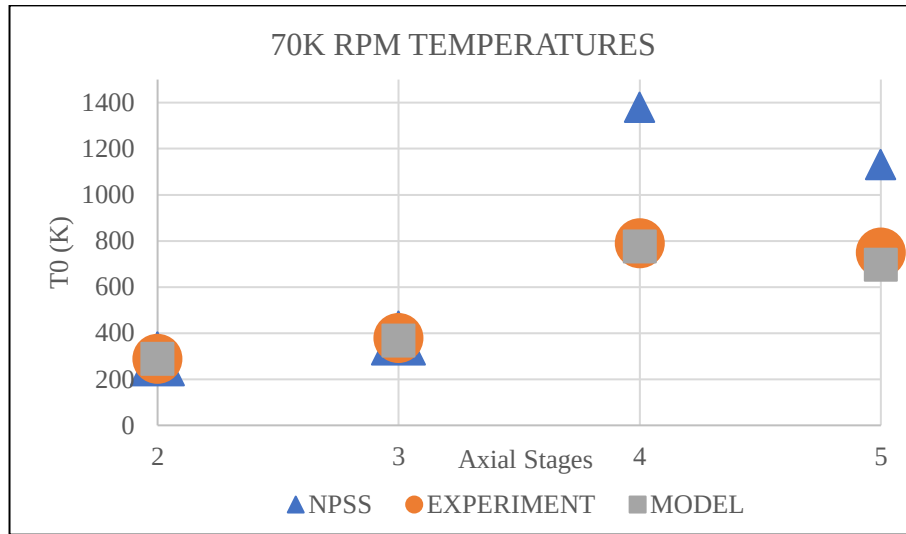


Figure 3.7: Station temperatures at 70K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

Temperature measurements are constant in the adiabatically modelled air intake. From the compressor performance map, the efficiency contour intersecting the mass flow rate and compression ratio as well as the shaft speed confirmed the temperature. In the NPSS model, the team conducting the study mentioned that there may be an error due to the combustion chamber modelling (Vannoy and Cadou., 2015). In addition, since the combustion chamber outlet temperatures were found to be too high, the turbine outlet temperatures were also high. The high temperature obtained by combustion of air-fuel mixture is also consistent with the experimental results. The turbine outlet temperature determined by the component matching formulation matches. (Figure 3.7)

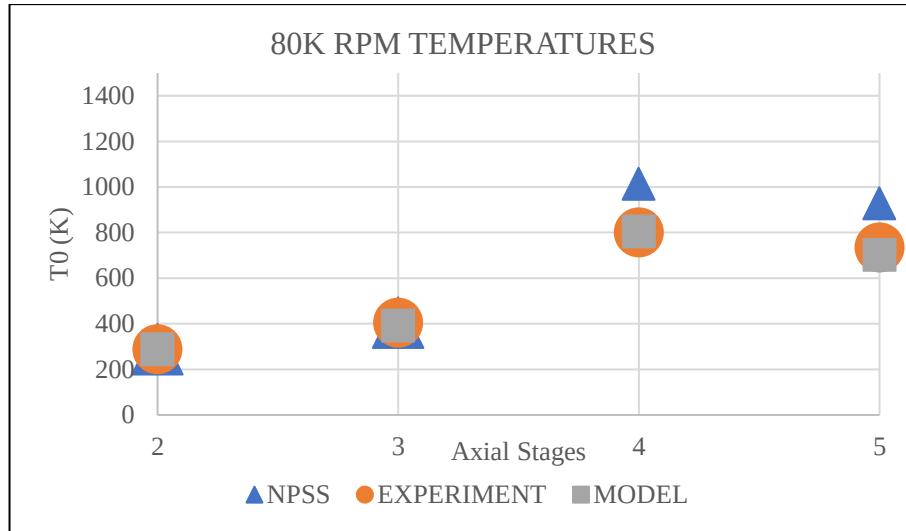


Figure 3.8: Station temperatures at 80K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

A turbojet engine in the MGT class operates more stable at higher speeds. For this reason, the deviations can be corrected during 1D mathematical calculations since the increase in the number of revolutions will stabilize the engine. At all stations, the model prepared for the thesis is consistent with the experimental results. Station 4-5 NPSS results started to converge compared to 70K RPM (Figure 3.8).

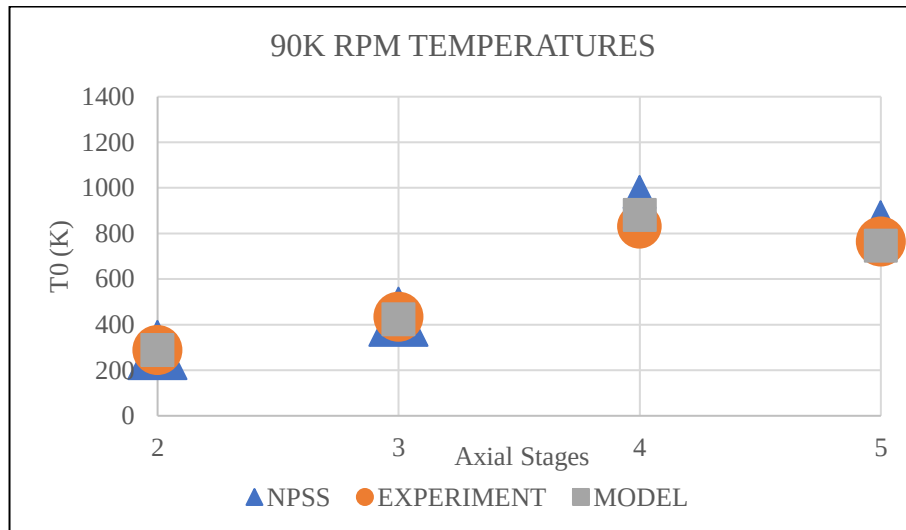


Figure 3.9: Station temperatures at 90K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

The combustion chamber is one of the most difficult engine components to model and test. At station 4 in Figure 3.9, there is a slight deviation from the experiment. The amount of fuel at this shaft speed is too much according to the experiment result.

However, the turbine outlet temperature coincides with the experimental results. (Figure 3.9)

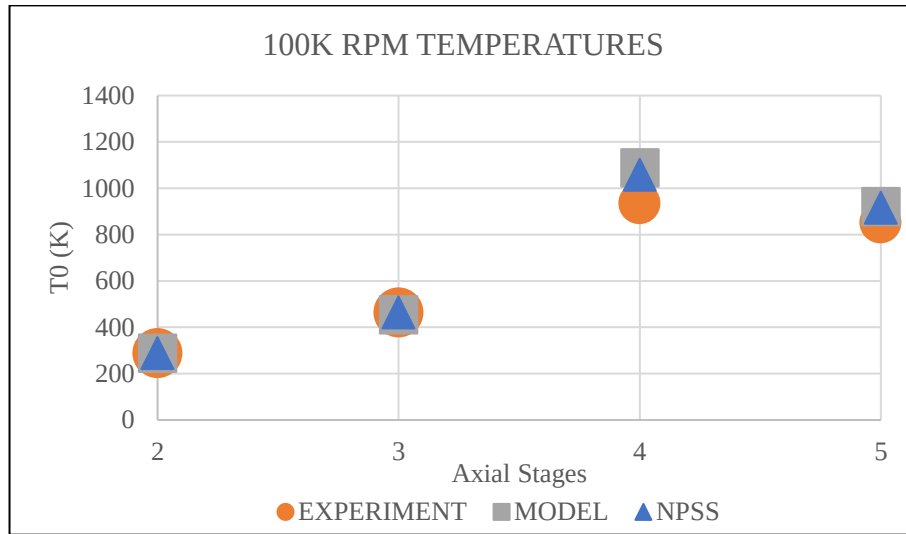


Figure 3.10: Station temperatures at 100K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

Combustion chamber outlet temperatures deviated from the experiment in 2 models. However, this deviation was corrected at station 5. (Figure 3.10)

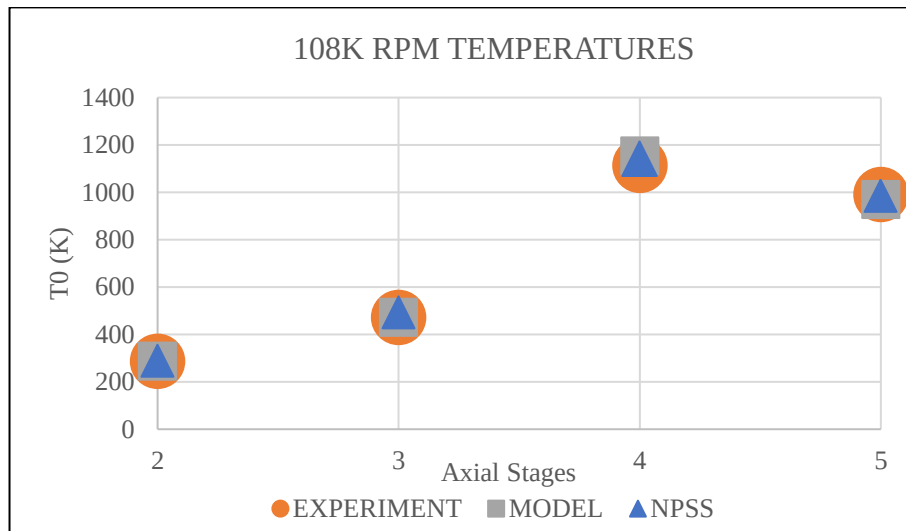


Figure 3.11: Station temperatures at 108K RPM experiment and NPSS simulation data from Vannoy and Cadou (2015) and our prediction model.

The intake and compressor outlet temperatures were consistent with the experiment at all shaft speeds compared. The highest deviation in the model was observed at 100K RPM. This deviation approached the experiment at 108K RPM. (Figure 3.11)

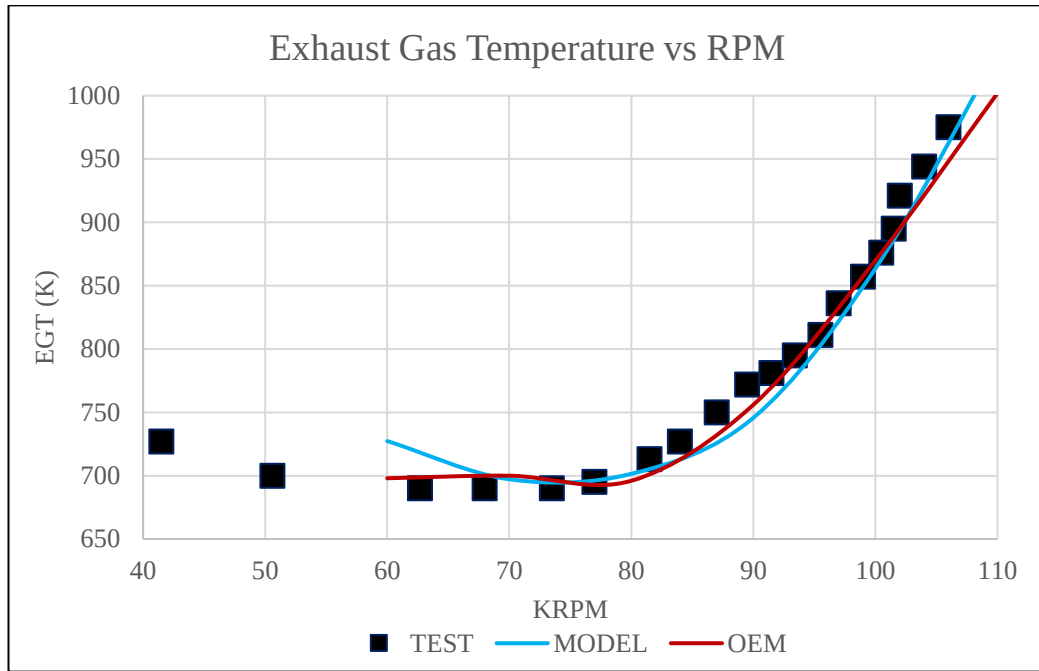


Figure 3.12: EGT vs RPM comparison for the test data of Leylek (2012), OEM from engine manufacturer and our model prediction.

As can be seen in Figure 3.12, the air temperature, which was around 740 K at 40K RPM, decreased approximately 700 K at 50K RPM and then moved horizontally until 80K RPM. Air also has a cooling function in jet engines. As the revolutions increased, the excess air entering the system increased, allowing the hot zone to cool down. However, after 80K RPM, the air temperature increased exponentially. The model calculated a higher temperature at 60K RPM compared to other parameters. However, the results converged with increasing speed.

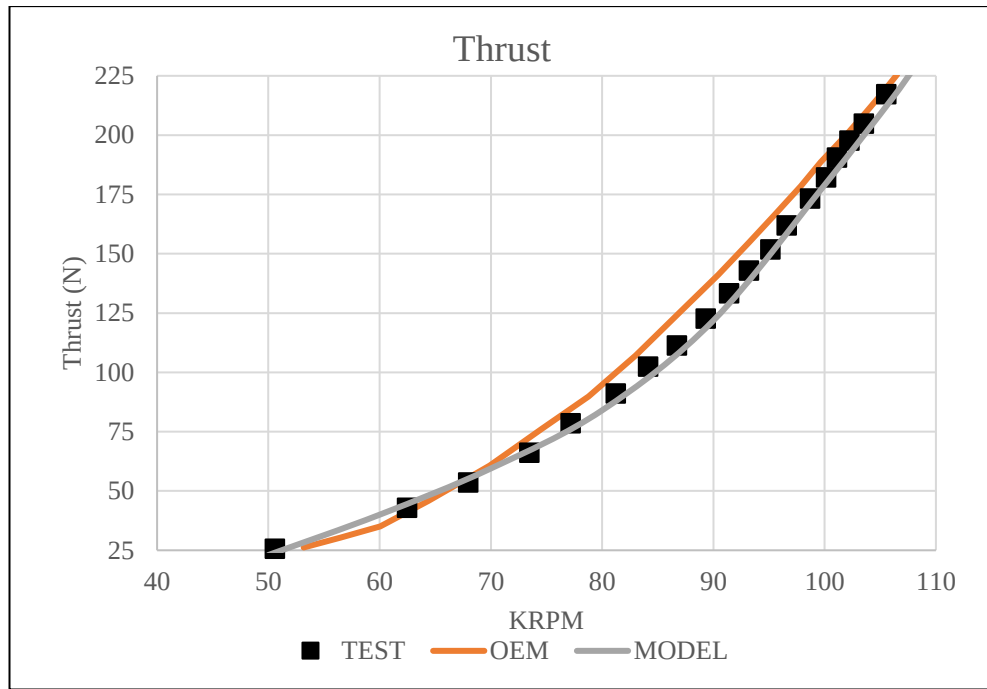


Figure 3.13: Thrust vs RPM comparison for the test data of Leylek (2012), OEM from engine manufacturer and our model prediction.

The thrust data is obtained by our model for different RPMs and compared to the experimental tests carried out by the engine manufacturer (OEM) and Leylek (2012). As can be seen in Figure 3.13, the model results are consistent and agree well with the published data.

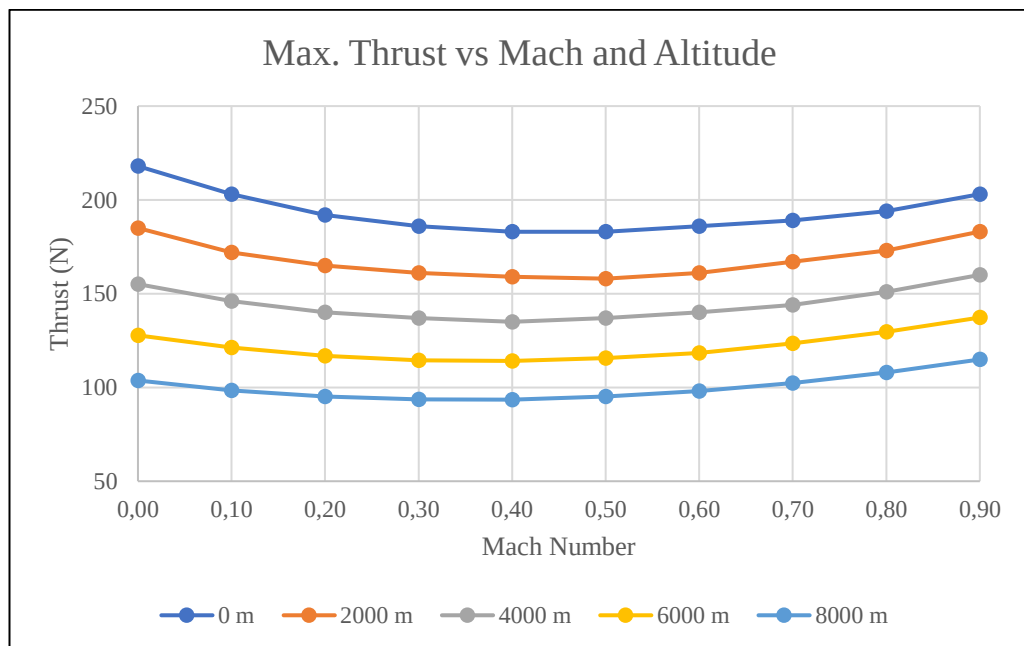


Figure 3.14: Thrust vs Mach number and altitude.

The technical data of the engine were taken into consideration and calculated at different altitudes and Mach numbers in the prepared flight envelope. The engine is at full power in the modelling. During the modelling, the amount of fuel at full power was used in standard sea level conditions. As can be seen in Figure 3.14, the thrust decreases up to 0,3 Mach. The reason for this is the ram drag effect. After Mach 0,3 due to the speed of the air, the temperature and pressure parameters of the engine inlet conditions make a positive contribution to the thrust. These results provide more thrust at low altitudes and high speeds, while providing better fuel economy at high altitudes and low speeds (Chapman, 2016).

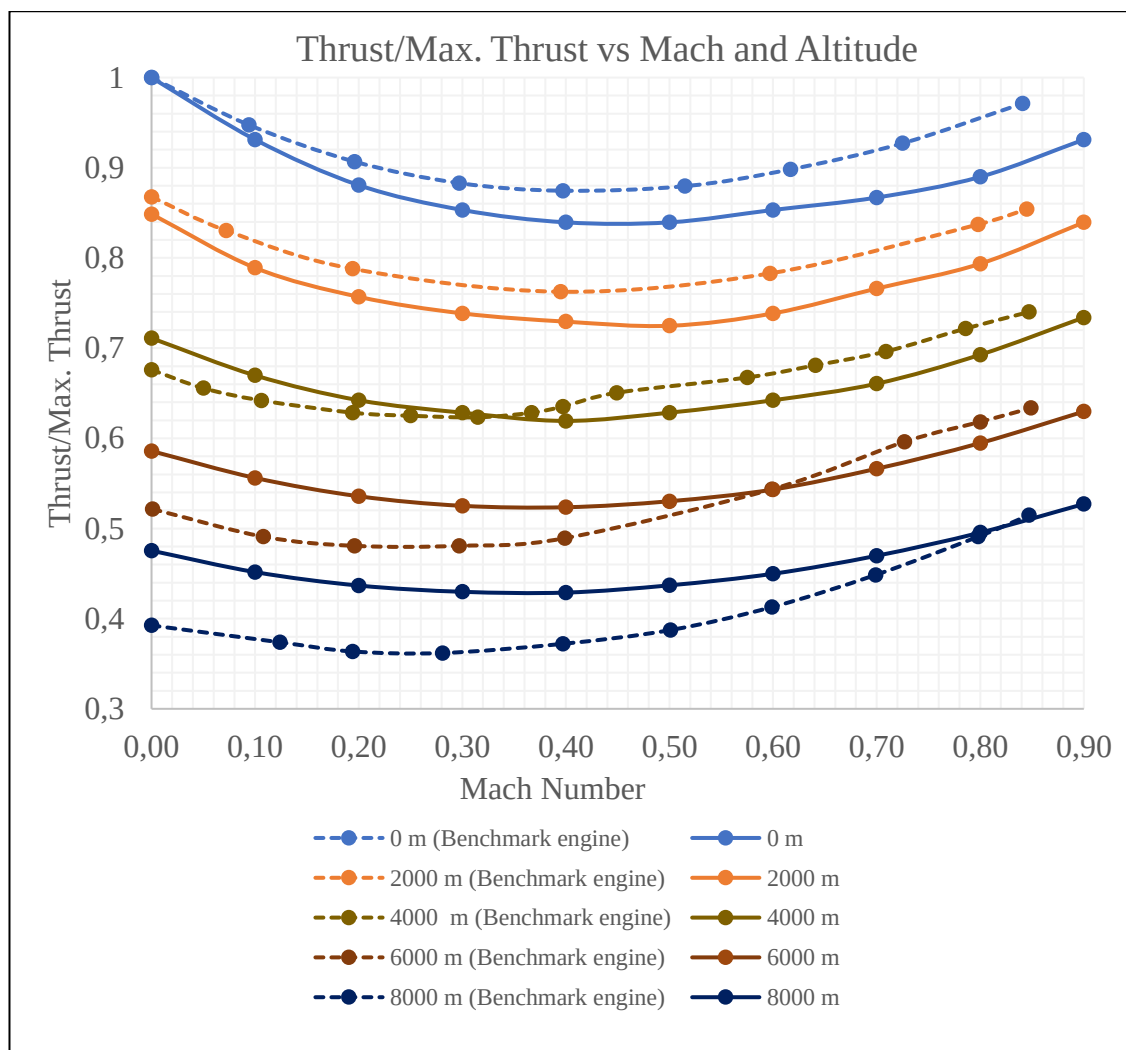


Figure 3.15: Thrust/max. thrust vs Mach number and altitude.

There is no test in the literature to verify the performance characteristics of the engine in the flight envelope under altitude conditions. For this reason, the theoretical

behavior of the engine was compared with the data of the Stargate TJ-40 engine of Rafael company in a similar class. Since the thrust of the Stargate TJ-40 engine is different, the thrust data obtained for comparison was divided by the thrust obtained at sea level maximum power. As can be seen in Figure 3.15, considering the differences between the engines, it is seen that they show similar behavioral responses.

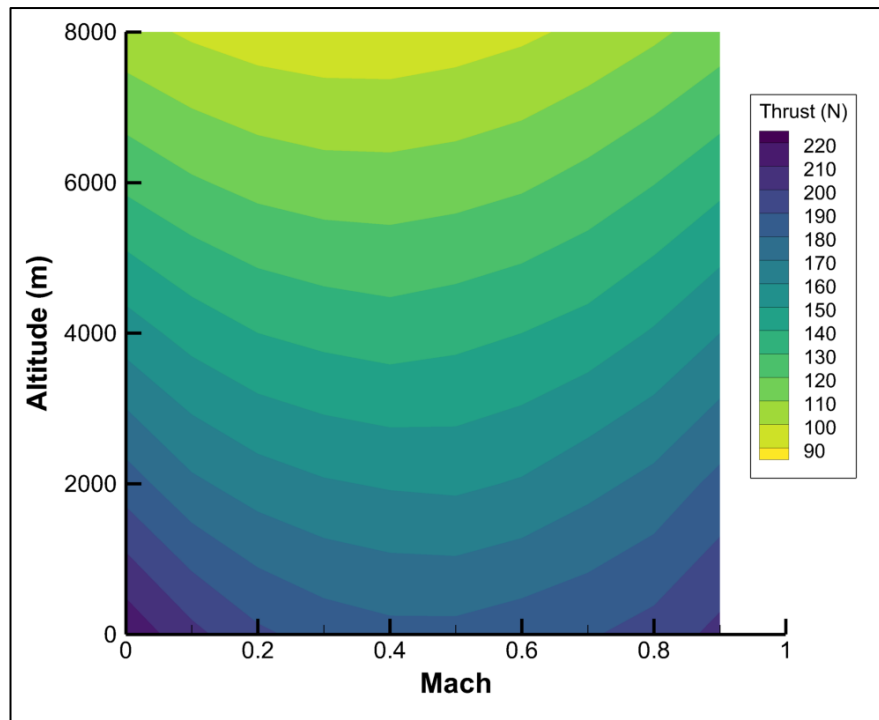


Figure 3.16: Mach number vs Altitude and Thrust

One of the most important factors affecting engine performance is environmental conditions. Determining the flight envelope of the engine affects the intended use of the aircraft. The constraints used for the theoretical study are based on the technical data shared by the engine manufacturer. Figure 3.16 shows the thrust graph depending on Mach number and altitude. The engine was operated at full power. The amount of fuel used at full power in static conditions at sea level was not changed. In static conditions, only when the altitude increases, the thrust decreases due to air density. Therefore, the net power required for the compressor also decreases. As a result, while the engine gives the highest thrust at sea level static conditions, it gives the least thrust at 8000 m 0,3 Mach. High thrust is obtained on the bottom left of the graph where Mach number and altitude are low.

4. CONCLUSION

In this study, the AMT Olympus turbojet engine was selected for performance analysis due to the ease of access to experimental test data compared to other jet engines. A 1D mathematical model based on the Brayton cycle was run under different environmental conditions and power levels. Variable and fixed parameters are used in the developed model. Scaled compressor performance map and turbine performance are calculated by component matching formulae. The thermodynamic and performance characteristics of the engine under the specified conditions are analyzed in detail. In the first stage, the model was run under sea level standard air conditions. The temperature and pressure variations at different stations in jet engine is obtained. Also, the exhaust gas temperature and thrust at different RPMs are calculated. Subsequently the model was run at different altitudes and Mach numbers, which affected the engine inlet conditions. Thus, the effect of different environmental conditions on engine performance was obtained. The results were compared with the published literature data and a good agreement is found. It is shown that turbine-compressor component matching simplifies the model yet gives compatible data with the literature.

As a result of this modelling study, useful implications for the development of the electronic control unit of micro gas turbine engines, off-design studies and performance characteristics of micro class engines are aimed to be understood.

REFERENCES

- Al-Hamdan Q. Z., Ebaid M. S. Y., (2005), “Modeling and Simulation of a Gas Turbine Engine for Power Generation”, *Journal of Engineering for Gas Turbines and Power*, 128(2), 302–311.
- AMT Netherlands (2019), https://amtjets.com/pdf/Olympus_HP_specification.pdf, (Date accessed: 05.05.2024)
- Asgari H., (2014), “Modelling, Simulation and Control of Gas Turbines Using Artificial Neural Networks”, Doctoral Dissertation, University of Canterbury.
- Cengel Y. A., Boles M. A., (2011), “Thermodynamics: An Engineering Approach”, 5th Edition, McGraw-Hill.
- Chapman J. W., Lavelle T. M., Litt J. S., (2016), “Practical Techniques for Modeling Gas Turbine Engine Performance”, In 52nd AIAA/SAE/ASEE Joint Propulsion Conference (p. 4527).
- Flack D. R., (2005), “Fundamentals of Jet Propulsion with Applications” 1st Edition, Cambridge University Press.
- Kurzke J., (1996), “How to Get Component Maps for Aircraft Gas Turbine Performance Calculations” ASME 1996 International Gas Turbine and Aeroengine Congress and Exhibition, Birmingham, UK, 10–13 June.
- Kurzke J., Halliwell I., (2018), “Propulsion and Power: An Exploration of Gas Turbine Performance Modeling”, 1st Edition, Springer International Publishing.
- Lazzaretto A., Toffolo A., (2001), “Analytical and neural network models for gas turbine design and off-design simulation”, *International Journal of Thermodynamics*, 4(4), 173-182.
- Leylek Z., (2012), “An Investigation into Performance Modeling of a Small Gas Turbine Engine”, Technical Report No: DSTO-TR-2757, Australian Government Department of Defense, Australia.
- Lichtsinder M., Levy Y., (2006), “Jet Engine Model for Control and Real-Time Simulations” *Journal of Engineering for Gas Turbines and Power*, 128(4), 745-753.
- Massardo A. F., McDonald C. F., Korakianitis T., (2002), “Microturbine/Fuel-Cell Coupling for High-Efficiency Electrical-Power Generation”, *ASME J. Eng. Gas Turbines Power*, 124(1), 110-116.
- Mattingly D. J., (2005), “Elements of Gas Turbine Propulsion”, Tata McGraw-Hill Edition, Tata McGraw Hill Companies.
- Opping F., (2016), “Micro Gas Turbine Performance Evaluation”, Doctoral Dissertation, Stellenbosch University.

Polat C., (2010), “An Electronic Control Unit Design for a Miniature Jet Engine”, Master’s Thesis, Middle East Technical University.

Rafael Advanced Defense Systems LTD., (2024), [https://www.rafael.co.il/wp-content/uploads/2019/04/STARGATE TJ-40 60 120.pdf](https://www.rafael.co.il/wp-content/uploads/2019/04/STARGATE_TJ-40_60_120.pdf), (Date accessed: 05.05.2024)

Saravanamuttoo H. I., Rogers G. F. C., Cohen H., (1996), “Gas Turbine Theory”, 4th Edition, Longman Group Limited.

Schreckling K., (2005), “Home Built Model Turbines”, 1st Edition, Traplet Publications Ltd.

Ur Rahman N., (2009), “Propulsion and Flight Controls Integration for The Blended Wing Body Aircraft”, Doctoral Dissertation, Cranfield University.

Vannoy S., Cadou C., (2015), <https://tfaws.nasa.gov/wp-content/uploads/TFAWS2015-AE-04.pdf>, (Date accessed: 05.05.2024)

Walsh P. P., Fletcher P., (2004), “Gas Turbine Performance”, 2nd Edition, Blackwell Publishing.

Web 1, (2024), <https://czarn.v.prz.edu.pl/gt60-engine-project>, (Date accessed: 21.06.2024)

BIOGRAPHY

Mehmet Alper AKSOY graduated from Yıldız Technical University, Department of Mechanical Engineering in 2018. In 2019, he started working as a design engineer at Turkish Aerospace.

PUBLICATIONS AND PRESENTATIONS FROM THE THESIS

Aksoy A. M., Son O., (2024) “An Investigation Into Performance Modelling Of Micro Turbine Engine” 8th GTU Graduate Research Symposium, Istanbul, Turkey, 30-31 May.