CZECH TECHNICAL UNIVERSITY IN PRAGUE FACULTY

OF MECHANICAL ENGINEERING

Department of automation and information technology

Bachelor Thesis

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CZECH TECHNICAL UNIVERSITY IN PRAGUE FACULTY OF

MECHANICAL ENGINEERING

Department of automation and information technology

Identification of turbojet engine Parameters

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- Thermodynamics - theoretical background
- Experiment description
- Choose of the parameters for identification
- Parameter identification based on engine thermal cycle

Bibliography / sources:

- Farokhi, S., Aircraft Propulsion, Wiley, 2014, ISBN 978-1-118-80677-7
- Documentation for CM14 – Axial Flow Gas Turbine, Armfield Ltd.

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Abstract

Gas turbines are one of the most important combustion engines in modern world's transportation, any possible improvement of the performance of the gas turbine engines would help to minimize the world's annual fossil fuel consumption, and hence the emission of the adverse greenhouse gases.

The aim of this study is to identify the thermal parameters of a CM14 – Axial flow gas turbine engine tested in a lab, the Analysis of the Turbo jet engine thermal cycle was performed using mainly first and second law of thermodynamics, and Fourier law for heat conduction

The identification was done using MATLAB Simulink, through 2 methods, the first consists of calculating the arithmetic mean of inputs then calculating the outputs, and the second consists of calculating outputs then their arithmetic mean

Keywords:

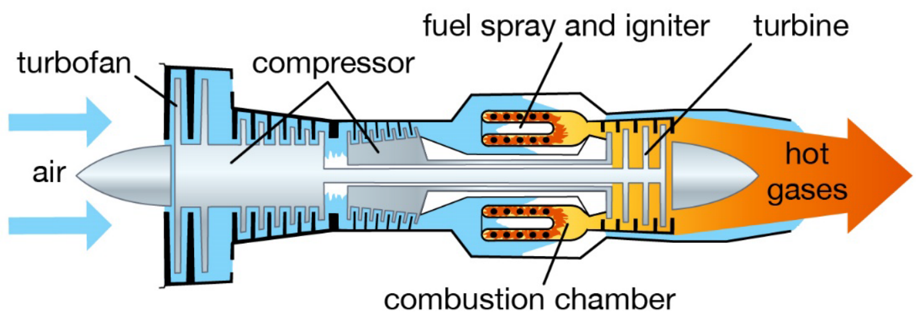
Turbo Jet engines cycles Analysis, Turbo Jet engine Parameter's identification

Chapter 01: Turbine aircraft engines

Introduction : in this chapter we are going to have a briefing of different types of gas turbine engines

Turbofan engine: The turbofan engine has evolved from its introduction to a primary power source for commercial aircrafts, the main difference in its thrust production. is that air moves through two parts of the engine, The air that bypasses the turbine moves around the engine core whereas the air that passes through the turbine flows through engine remaining section depending on the design, an additional thrust is created from the bypass air, which in today’s modern turbofans represents the majority of engine thrust, this bypass air would cool the engine, and make it quitter by blanketing the exhaust air that’s exiting the engine.

Air-breathing engines



Turbofan. Some air taken in by the fan goes to the compressor; the rest bypasses the main engine.

Figure 1.1.1: Turbofan working principle, britannica.com

The turbofan turbine is larger than that of a TJ engine, meanwhile the nozzles are smaller, this is relatively mainly because the turbine has to additionally drive the fans, and the fact that some of the air passes through the combustion chamber outside of the engine, the nozzle size increases the turbofan thrust which results in higher velocity.

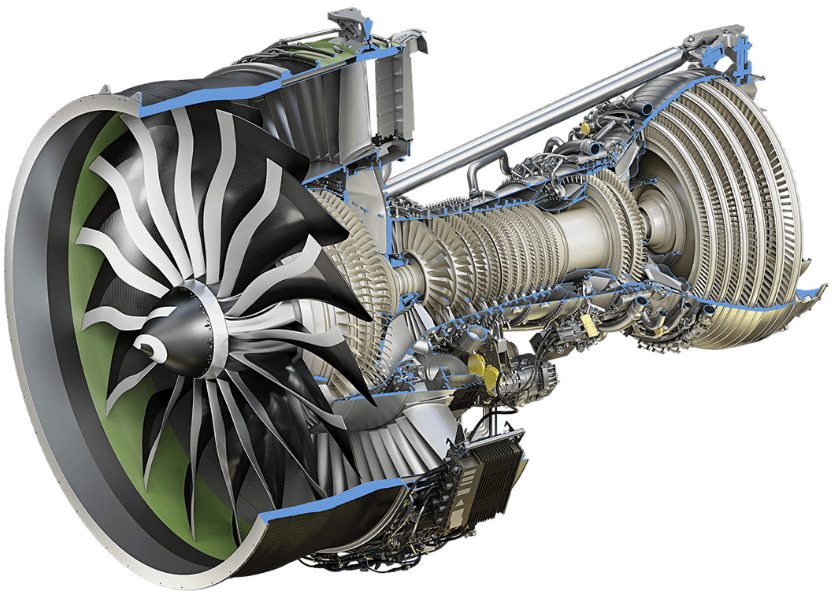


Figure 1.1.2 GE9X turbofan engine, source: geaviation.com

the main parameter that describes the turbofan engine is the bypass ratio, a 5:1 BPR mean that for every 5 kg of air that passes through the bypass duct, only 1 kg of air passes through the core, this parameter plays a key role for identifying the fuel consumption for the same thrust

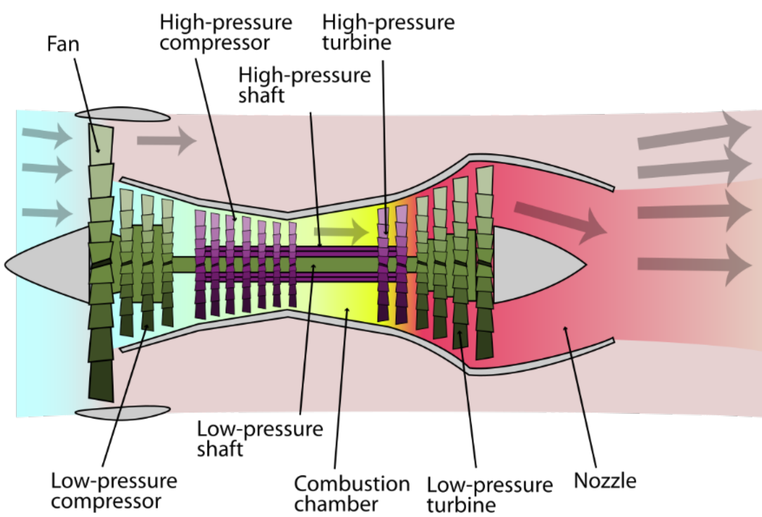


Figure 1.1.3: Bypass ratio, source: en.wikipedia.com

turbo-fan engines extract shaft power and transfer it to a bypass stream which results in extra losses and thus lower efficiency, together with combustor irreversibility, and core exhaust heat loss. These losses would make a large turbofan overall efficiency around 40%, the remaining losses can be overcome by inventing a new design concept, but still for Mach number of 0.75-0.9, turbofan engines are the most common engine configurations, which are mainly used in the commercial sector

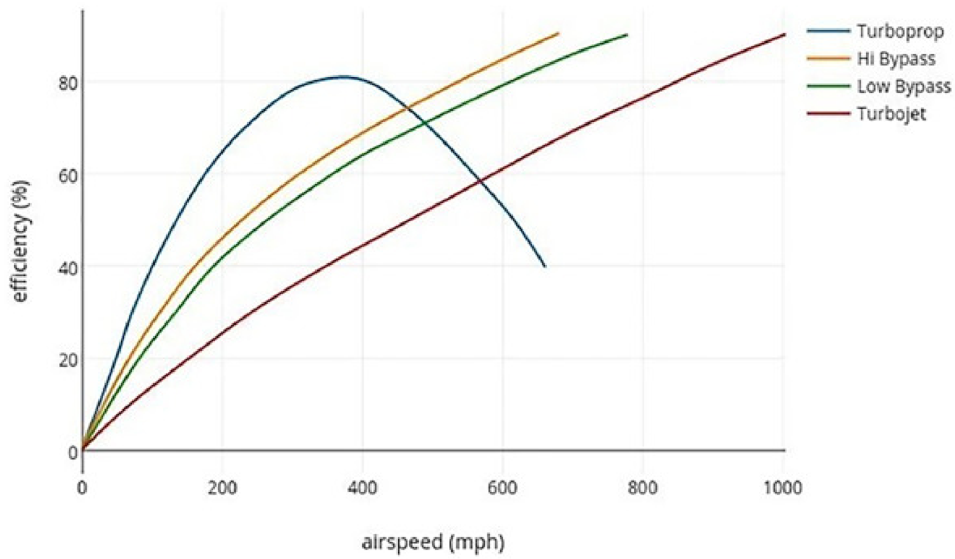
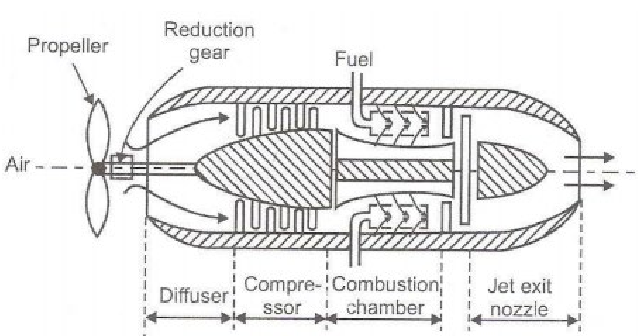


Figure 1.1.4 Propulsive efficiency versus airspeed for different propulsion system types, source: [ars.els-cdn.com]()

Turboprop engine

Working principle: It uses the same principle as turbojet to produce energy, but in this engine 80 to 90% of the total propulsive thrust is generated by the gas turbine and the remainder is developed by the expansion of the gases in nozzles, the primary difference is that it has additional turbines, a power shaft, and a reduction gearbox. The gearbox maybe driven by the same turbines and shaft, mechanically linking the propeller and the engine, the majority of the turboprop power is used to drive the propeller, and thus the engine gases doesn't have enough energy to generate the necessary thrust, the propellers on the other hand develop the necessary thrust by moving a large mass of air through a small change in velocity, It converts the high rpm low torque output to low Rpm high torque, it can accelerate a large volume of air due to its large diameter which permits a lower airstream velocity for a given amount of thrust. For Mach number 0.2-0.7, the propeller is the most efficient and most common used gas turbine engine, in which the most popular model is Pratt and Whitney PT6



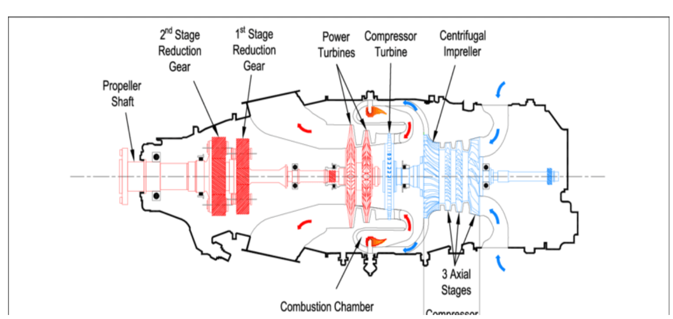


Figure 1.2.1: on the left 2D sketch of turboprop engine, source: [mechdiploma.com]()

Figure 1.2.2 working principle of pratt and whitney PT6, source: [reseatchgate.net]()

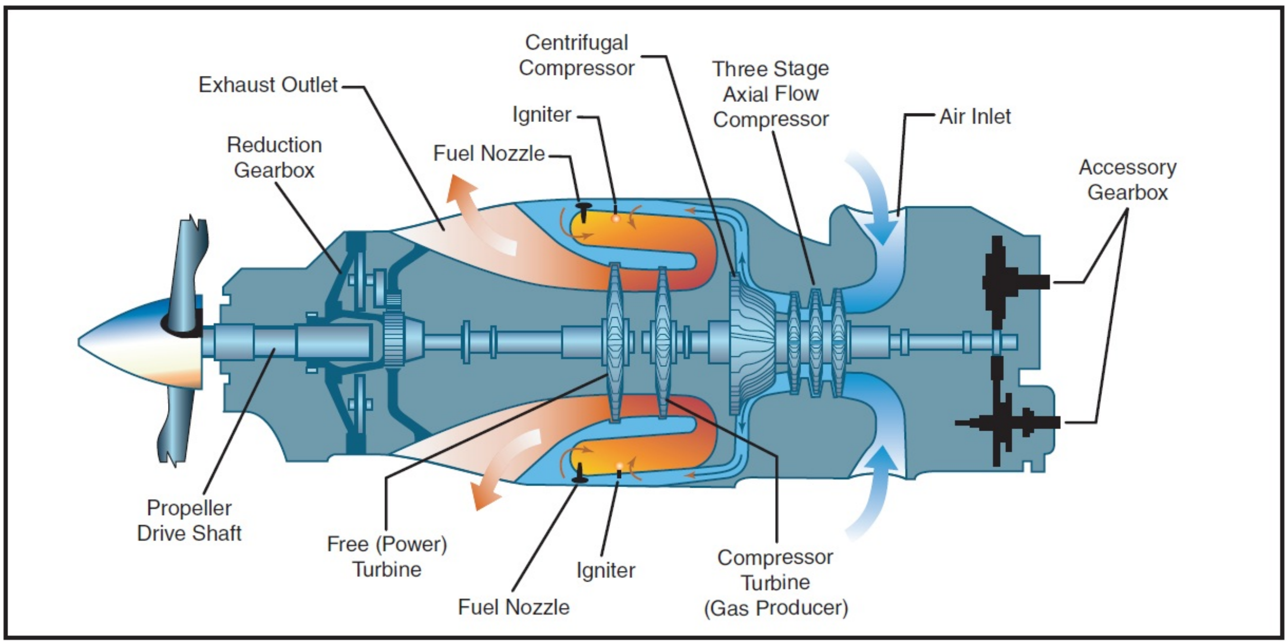


Figure1.2.3: working principle of turboprop engine, source: i.stack.imgur.com

A turboprop is quite similar to turboshaft, but the main difference is that turboprop must be designed to support the loads, of the attached propeller meanwhile turboshaft doesn't need to be as robust as it normally drives a transmission which structurally supported by the vehicle and not by the engine itself

Efficiency: Turbo prop has a bypass ratio 50-100, it burns less fuel per seat-km and requires les runway than turbojet and turbofan for takeoff and landing, it works well until the flight speed of the aircraft reaches the speed of sound, beyond this speed the proportion of the engine power that that is converted to propeller thrust falls dramatically.

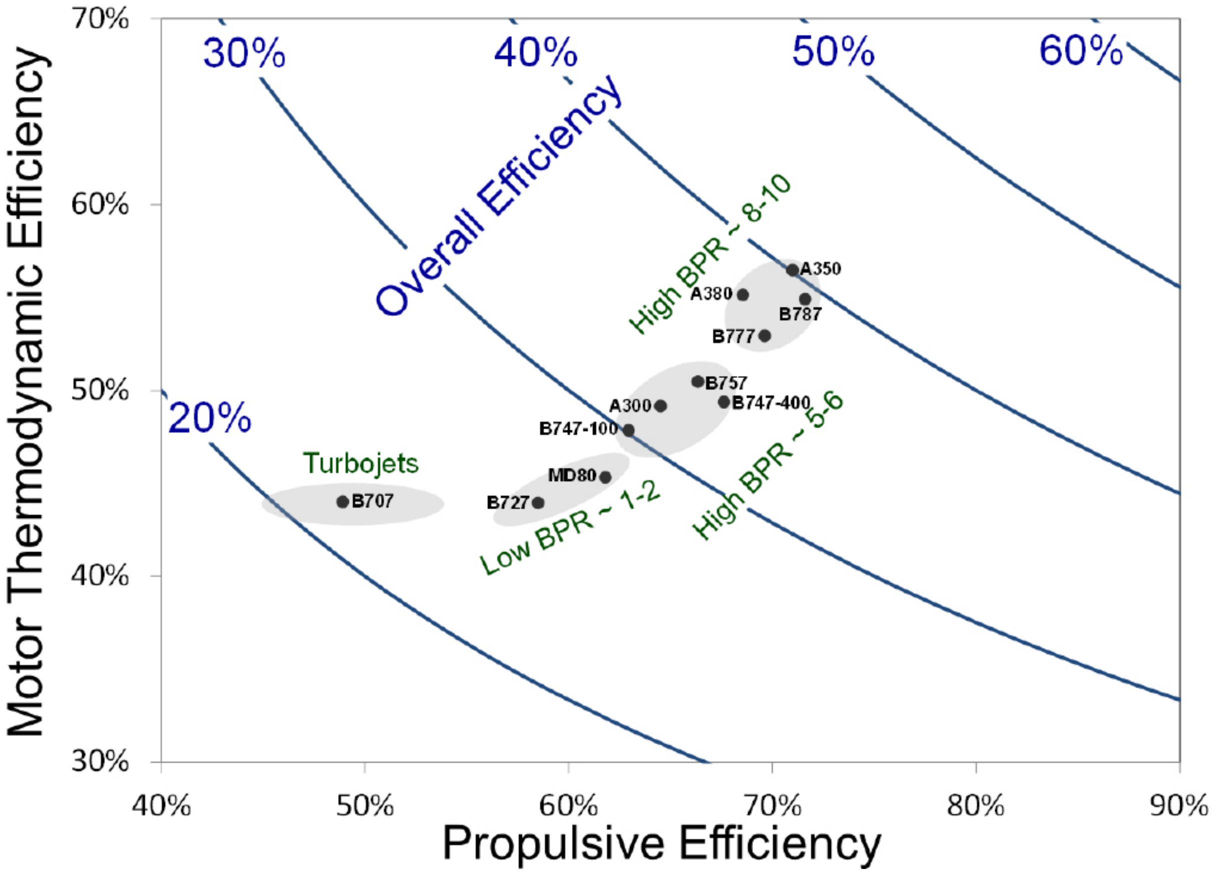


Figure 1.2.4 shows propulsive efficiency in respect to motor thermodynamic efficiency, source: commercial aircraft propulsion and energy system research

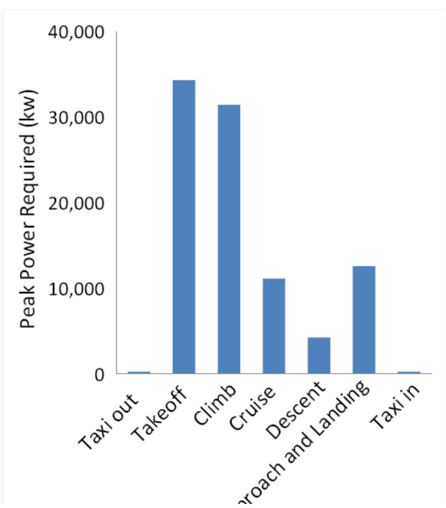


Figure 1.2.5 Single-aisle aircraft power by mission segment; dimensional and percent available powers are shown, source: commercial aircraft propulsion and energy system research

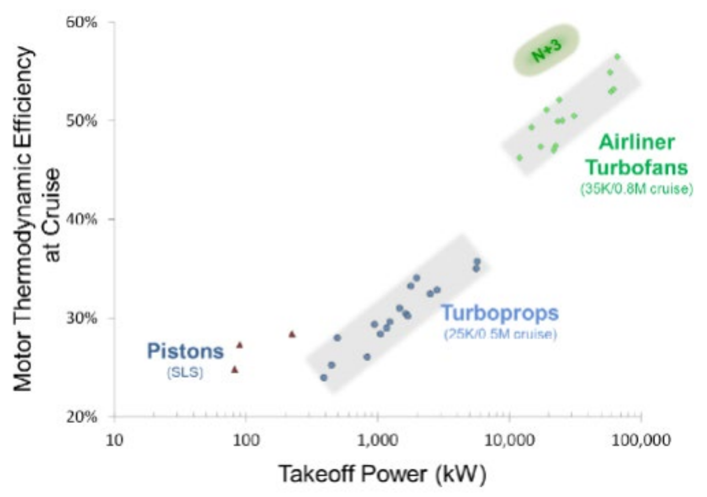


Figure 1.2.6 Variation of motor thermodynamic efficiency at cruise, source: commercial aircraft propulsion and energy system research

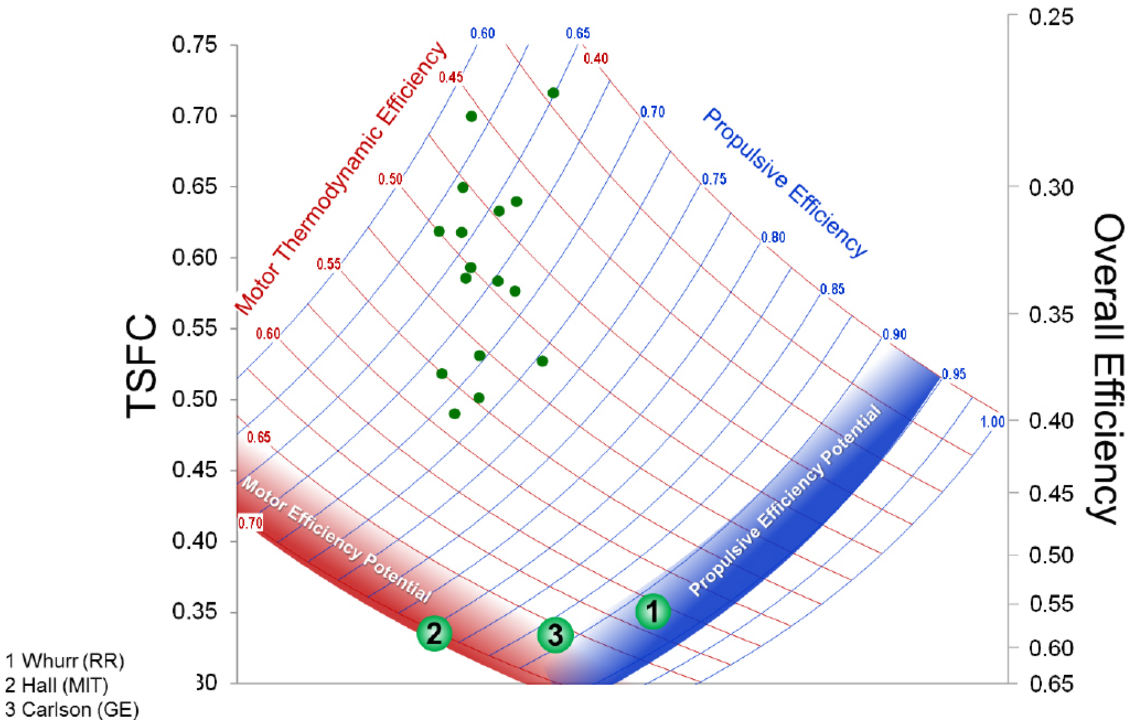


Figure 1.2.7 Motor thermodynamic and propulsive efficiencies at cruise engines (dots) along with literature projections of practical limitations for a simple cycle gas turbine. Also shown are thrust specific fuel consumption (TSFC) and overall efficiency, source: commercial aircraft propulsion and energy system research

Turbojet engine

Working principle: air at the front is sucked by the engine fan. A compressor which has many blades attached to a shaft raises the pressure of the air. The blades spin at high speed and squeeze the air. The compressed air is then sprayed with fuel and an electric spark lights the mixture. The burning gases expand then pass through the turbine and blast out through the nozzle. It is used for both commercial and military purposes

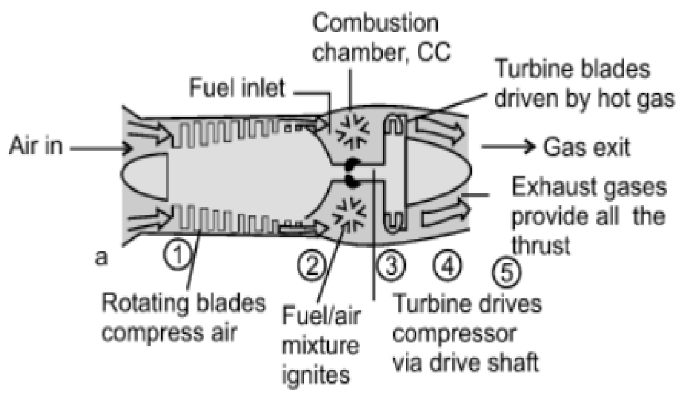


Figure 1.3.1 D Sketch that shows the working principle of turbojet engine, source: [mechdiploma.com]

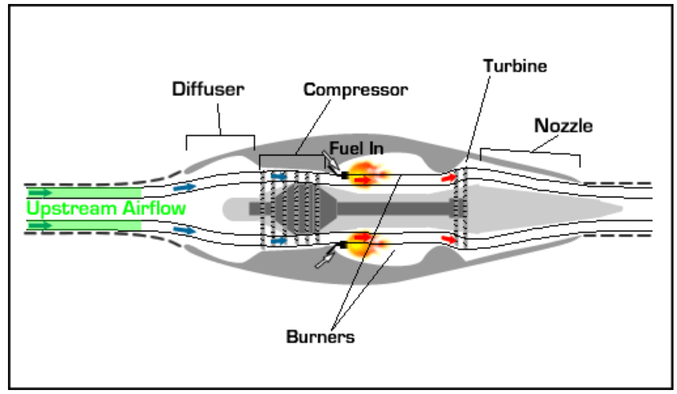


Figure 1.3.2 Working principle of turbojet engine, source: [s2.smu.edu]

for low Mach number, Turbojets have low efficiency, they approach peak efficiency at around Mach 2.

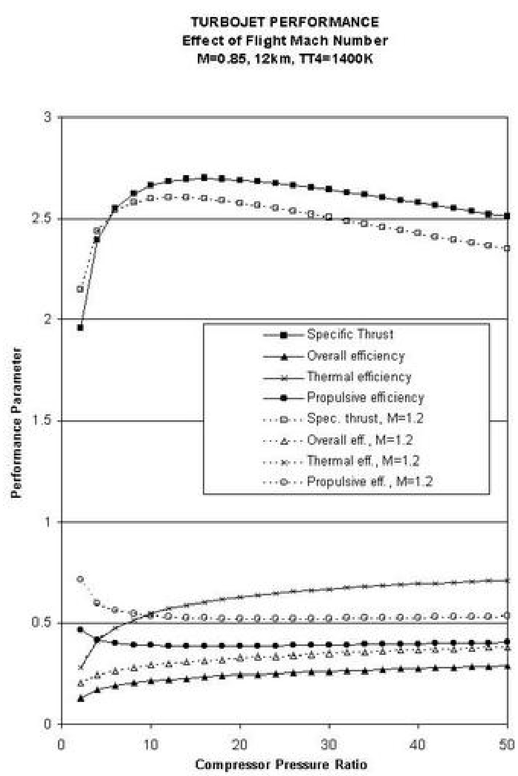


Figure 1.3.3 Performance of an ideal turbojet engine as a function of compressor pressure ratio and flight Mach number. Source: [web.mit.edu]

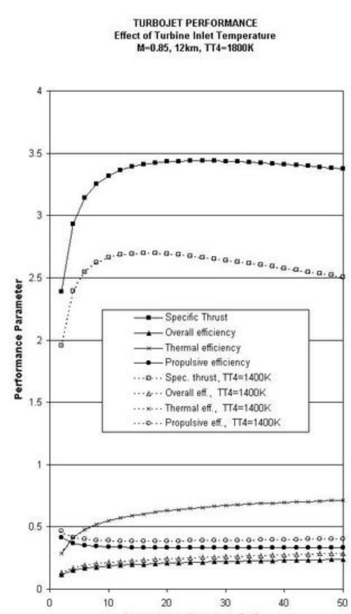


Figure 1.3.4 Performance of an ideal turbojet engine as a function of compressor pressure ratio and turbine inlet temperature. source: [web.mit.edu]

Turboshaft

It operates in the same principle of a turboprop, atmospheric gases are ingested at the inlet, compressed, mixed with fuel and combusted, then expanded through a turbine which powers the compressor, meanwhile it has an additional turbine expansion that extracts energy from the exhaust and converts it into shaft power, it is mainly used in applications that require small size, high reliability, high power output and light weight such as helicopters, also they are used in aircrafts as auxiliary power units.

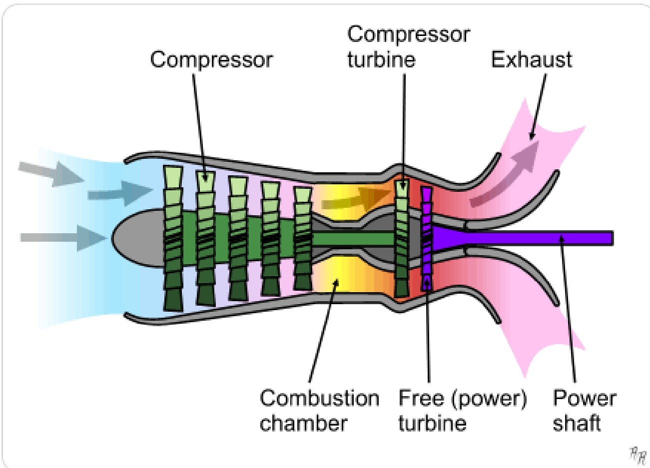


Figure 1.4.1 Turboshaft working principle, source: [andrasmeridian.com]()

Figure 1.4.2 Turboshaft Engine PBS TS100, source: [pbs.cz]()

Ramjet engine

The air enters the inlet with supersonic speed, then the supersonic diffuser slows it down to sonic velocity, which increases air pressure and temperature of air. While the fuel is injected into combustion chamber, it burns with the help of flame igniter. Then it blasts through the nozzle, converting pressure energy into kinetic energy and providing required forward thrust.

Unlike other turbine engines, it has no moving parts, it operates by combustion of fuel in a stream of air compressed by the forward speed of the aircraft itself, it is used primarily in guided-missile systems, and space vehicles

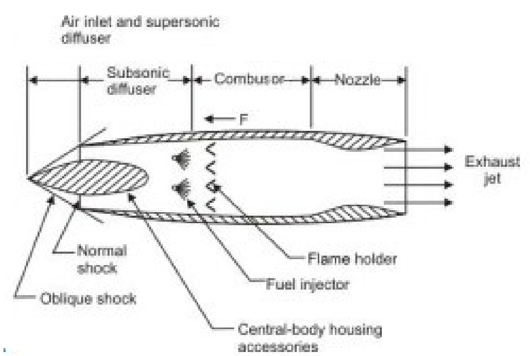


Figure 1.5.1 D sketch or Ramje engine, source:[mechdiploma.com]()

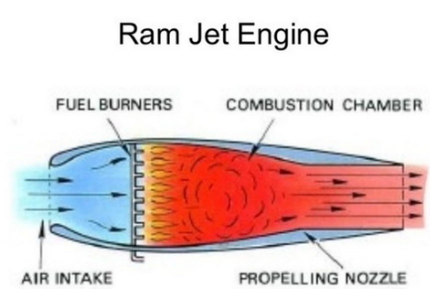


Figure 1.5.2 Ramjet engine working principle, source: [wordpress.com]()

It operates efficiently at around Mach 3, and can operate up to Mach 6 for certain engines

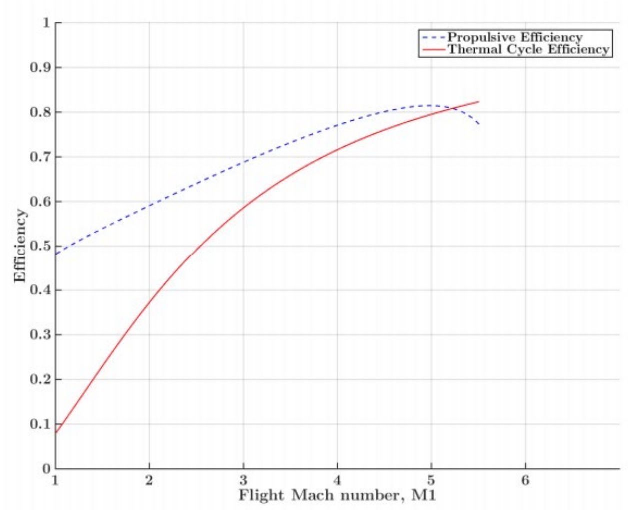


Figure 1.5.3 Propulsive and Thermal cycle efficiency variation with flight Mach, source: ramjet engine calculator

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Chapter 02: thermodynamic – theoretical background

Introduction: in this chapter, we start our analysis by defining some important thermodynamic concepts, then we demonstrate the final derivation of certain important laws that we will eventually use in the analysis and computation of turbojet engine thermal cycle

the first law of thermodynamics

there is no difference between heat and mechanical work exchange with the system , instead their net interaction with the system has to be accounted for the energy balance

For an adiabatic process with no mechanical exchange of work, the system energy remains constant, this principle is mainly used in the study of inlet and exhaust systems of an aircraft engine

Second Law of thermodynamics

It introduces the absolute temperature scale and entropy, and distinguishes between heat and work it states the impossibility of a heat engine to exchange heat with a single reservoir and produce mechanical work continuously, instead it called for a second reservoir at a lower temperature where heat is rejected to by the heat engine

not all heat transfer to a system may be converted into system energy continuously, meanwhile all mechanical work may be converted into system energy

Molecule's degrees of freedom are represented by the sum of the energy states that a molecule possesses. For example, atoms or molecules possess kinetic energy in three spatial directions. If they rotate as well, they will have kinetic energy associated with their rotation. In a molecule, the atoms may vibrate with respect to each other, which then creates kinetic energy of vibration as well as the potential energy of intermolecular forces. Finally, the energy levels (both kinetic energy and potential) that depend on the electron position around the nucleus define the electrons in an atom or a molecule, As the temperature of the gas increases, the successively higher energy states are excited thus the degrees of freedom increases.

A monatomic gas has negligible rotational energy about the axes that pass through the atom because of its negligible moment of inertia. A monatomic gas will not have a vibrational energy, as vibrational mode requires at least two atoms. At higher temperatures, the electronic energy state of the gas is affected, which eventually leads to ionization of the gas. For a diatomic gas, that might be modeled as a dumbbell, there are five degrees of freedom under "normal" temperature conditions, and two of which are in rotational motion, and three of which are in translational motion.

three of which are in translational motion and two of which are in rotational motion. The third rotational motion along the intermolecular axis of the dumbbell is negligibly small. hence a diatomic gas, the ratio of specific heats is $\gamma= 1.4$, At high temperatures, molecular vibrational modes and the excitation of electrons add to the degrees of freedom and that lowers γ.

two energy states corresponding to the kinetic energy of vibration and the potential energy associated with the intermolecular forces represents The vibrational mode

Isentropic process and isentropic flow

In isentropic process, where entropy remains constant, the Gibbs equation relates the pressure and temperature ratios by an isentropic exponent

Conservation principle of systems and control volumes

A system is a collection of matter of fixed identity, hence fixed mass, whereas a control volume is a fixed region in space with fluid that crosses its boundaries. A control volume approach seems to be a more practical method for treating the fluid flow problems in gas turbine engines. meanwhile, all the classical laws of Newtonian physics are written for a matter of fixed mass

In Newtonian mechanics, the mass of a matter is described by a system, does not change with time, in contrast, in control volume, mass is expressed under the continuity equation as the integral of density over volume

Thermodynamic laws

Some important thermodynamic laws:

The medium in gas turbine engines: $p = \rho * R * T$ (2.1.1)

Molecular weight of the gas:$R = \frac{\bar{R}}{MW}$ (2.1.2)

Specific heat at constant volume: $dh = c_p * d * T$; $de = c_v * d * T$ (2.1.3)

Cycle efficiency of a constant pressure: $n_{th} = 1 - \frac{T1}{T2}$ (2.4)

Cycle efficiency of a constant volume: $n_{th} = \frac{\left(1-\gamma\left(\frac{T1}{T2}\right)\left[\left(\frac{T3}{T2}\right)^{\frac{1}{\gamma}}-1\right]\right)}{\left[\frac{T3}{T2}-1\right]}$ (2.1.4)

Ideal Carnot efficiency: $n_{th} = 1 - \frac{T1}{T2}$

Humphrey cycle at a constant-volume combustor instead of a constant pressure

$$n_{th} = \left(1 - \gamma\left(\frac{T1}{T2}\right)\left[\left(\frac{T3}{T2}\right)^{\frac{1}{\gamma}} - 1\right]\right) \quad (2.1.5)$$

For isentropic process and isentropic flow: (valid for calorically perfect gas)

$$\frac{p_2}{p_1} = \left(\frac{T_2}{T_1}\right)^{\frac{c_p}{R}} = \left(\frac{T_2}{T_1}\right)^{\frac{\gamma}{\gamma-1}} \rightarrow \frac{p_2}{P_1} = \left(\frac{\rho_2}{\rho_1}\right)^{\gamma} \quad (2.1.6)$$

Specific heat transfer:

$$\gamma = \frac{5}{3}\ for\ monatomic\ gas\ at\ normal\ temperature$$

$$\gamma = \frac{7}{5} = 1.4\ diatomic\ gas\ at\ normal\ temperature$$

$$\gamma = \frac{8}{6}\ diatomic\ gas\ at\ elevated\ temperature$$

$$\gamma = \frac{9}{7}\ diatomic\ gas\ at\ higher\ temperature$$

Cycle Analysis of the turbojet engine

Turbojet engine is a gas generator fitted with an inlet and exhaust system, that requires us to calculate the exhaust component efficiencies in order to find the performance of the Turbojet engine

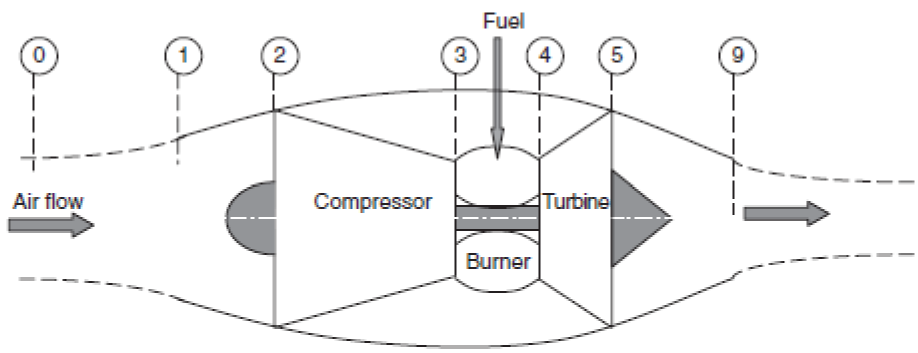


Figure 2.1 schematic Drawing of a gas generator, source: Farokhi, Saeed. Aircraft propulsion

The station numbers in a turbojet engine are defined at the unperturbed flight conditions

00: flight condition

01: inlet lip

02: compressor face

03: compressor exit

04: burner exit

05: turbine exit

09: and the nozzle exit plane

The inlet: it delivers the air to the compressor at the right Mach Number M_2 and the right quality, that is low distortion, for all commercial aircrafts, which include transport and military that has a Mach number of 0.5-0.6, the inlet is required to decelerate or diffuse the air flow , and thus decelerate the air efficiency, flow deceleration is accompanied by the static pressure rise.

the boundary layers being of a low energy and momentum deficit zone, facing an adverse pressure gradient environment tend to separate, therefore the designer must prevent it by tailoring the geometry of the inlet to avoid rapid diffusion or possibly through variable geometry inlet design

an ideal inlet would provide a reversible adiabatic Process, which is an isentropic compression of the captured flow to the engine.

there for from Fourier law of heat transfer through the inlet wall

$$\mathbf{q_n} = \frac{\dot{Q}_n}{A} = -k\frac{\delta T}{\delta n} \quad (2.2.1.1)$$

Where k: is the thermal conductivity of the wall

q_n: heat flux, which is defined as the heat transfer rate $\dot{Q}_n$per unit Area

Following the combined first and second law of thermodynamics, Gibbs equation is introduced

$$\frac{p_{t_2}}{p_{t_0}} = \exp\left(\frac{-s_2 - s_0}{R}\right) = \exp\left(-\frac{\Delta s}{R}\right) \quad (2.2.1.2)$$

The gap between the states T and t_0 represents the amount of dissipated kinetic energy

Inlet adiabatic efficiency: $\eta = \frac{h_{t_{2s}} - h_0}{h_{t_2} - h_0} = \frac{\left(\frac{V^2}{2}\right)_{ideal}}{\frac{V_0^2}{2}}$ (2.2.) (2.2.1.3)

Dividing the nominator and denominator by h_0:

$$\eta_d = \frac{\left(\frac{h_{t2s}}{h_0} - 1\right)}{\left(\frac{h_{t_2}}{h_0} - 1\right)} = \frac{\frac{T_{t_{2s}}}{T_0} - 1}{\frac{h_{t_0}}{h_0} - 1} = \frac{\left(\left(\frac{P_{t_2}}{P_0}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} - 1\right)}{\frac{\gamma_{cold}-1}{2}M_0^2} = \frac{\left(\frac{P_{t_2}}{P_0}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} - 1}{\tau_r - 1} \quad (2.2.1.4)$$

we can separate the unknown term $p_{t2}$and write the following expression:

$$\frac{P_{t_2}}{P_0} = \left\{1 + \frac{\eta_d(\gamma_{cold}-1)}{2}M_0^2\right\}^{\frac{\gamma-1}{\gamma}} \quad (2.2.1.5)$$

Another parameter is the inlet total pressure recovery:

$$\pi_d = \frac{p_{t_2}}{p_{t_0}} \quad (2.2.1.6)$$

And thus, we derive the relations ship between inlet efficiency and compressor pressure ratio:

$$\frac{p_{t_2}}{p_0} = \frac{p_{t_2} * p_{t_0}}{p_{t_0} p_0} = \left\{1 + \frac{\eta_d(\gamma_{cold}-1)}{2}M_0^2\right\}^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} \quad (2.2.1.7)$$

$$\pi_d = \frac{\left\{1 + \frac{\eta_d(\gamma_{cold}-1)}{2}M_0^2\right\}^{\frac{\gamma_{cold}-1}{\gamma_{cold}}}}{\frac{p_{t0}}{p_0}} = \left\{\frac{1 + \frac{\eta_d(\gamma_{cold}-1)}{2}M_0^2}{1 + \frac{\gamma-1}{2}M_0^2}\right\}^{\left(\frac{\gamma_{cold}}{\gamma_{cold}-1}\right)} = \frac{\left\{1 + \frac{\eta_d(\gamma_{cold}-1)}{2}M_0^2\right\}^{\frac{\gamma_{cold}-1}{\gamma_{cold}}}}{\pi_r} \quad (2.2.1.8)$$

The Compressor

it requires external power to operate, this power comes from the turbine via a shaft, the process is considered to be adiabatic in both ideal and real compressor, which states that only a negligible amount of heat transfer occurs between the air flowing through the compressor and the ambient air.

the source of the power delivered to the medium of a compressor is one or more rows of rotating rotors attached to one or more spinning shafts, each rotor blade causes a spin change, and a counter torque reaction, the order of magnitude of the power source, causes the adiabatic property of a real compressor. whereas the sources of irreversibility in a compressor is the relative supersonic flow through compressor blades which causes shock waves and the presence of wall friction acting on the medium through the boundary layers

an ideal compressor consumes less power than an actual compressor, due to the absence of dissipative mechanisms, which leads to lost work, meanwhile the actual temperature is higher than isentropic temperature to achieve the same compressor pressure ratio, because the lost work (friction and shock waves) is converted into heat

examining the static states of the gas in a compressor will help us with the fluid dynamics of the compressor

the power delivered to the medium by the spinning of the rotor at angular speed w follow Newtonian mechanics:

$$\wp = \tau\, \mathrm{w} \quad (2.2.2.1)$$

The rate of mechanical energy transfer is several orders of magnitude larger than heat transfer

$$\wp_c \gg \dot{Q}_w \quad (2.2.2.2)$$

There for the real compressor is considered to be adiabatic

$$\wp_c = \dot{m}\left(h_{t_3} - h_{t_2}\right) \quad (2.2.2.3)$$

$$\pi_c = \frac{p_{t_3}}{p_{t_2}} \quad (2.2.2.4)$$

since $t_3$is the actual state of the gas and the exit of the compressor: $\tau_c = \frac{T_{t_3}}{T_{t_2}}$ (2.2.2.4)

therefor we cannot expect the isentropic relationship between $\tau_c$and $\pi_c$to hold

The compressor adiabatic efficiency: $\eta_c = \frac{h_{t_{3s}} - h_{t_2}}{h_{t_3} - h_{t_2}} = \frac{\Delta h_{t,isentropic}}{\Delta h_{t,actual}}$ (2.2.2.5)

We divide the numerator and denominator by h_{t2}, $\eta_c = \frac{\frac{T_{t_{3s}}}{T_{t2}} - 1}{\frac{T_{t_3}}{T_{t_2}} - 1}$ (2.2.2.6)

Temperature and pressure are related via isentropic formula: $\frac{T_{t_{3s}}}{T_{t_2}} = \left(\frac{p_{t_{3s}}}{p_{t_2}}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} = \left(\frac{p_{t_3}}{p_{t_2}}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} = \pi_c^{\frac{\gamma_{cold}-1}{\gamma_{cold}}}$ (2.2.2.7)

Compression adiabatic efficiency: $\eta_c = \frac{\pi_c^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} - 1}{\tau_c - 1}$; $e_c = \frac{dh_{ts}}{dh_t}$; $T_t ds = dh_t - \frac{dp_t}{\rho_t}$ (2.2.2.8)

$\mathrm{dh_{ts}} = \frac{\mathrm{dp_t}}{\rho_t}$; if we substitute in $\mathrm{e_c} = \frac{\mathrm{dh_{ts}}}{\mathrm{dh_t}}$ and replace density with pressure and temperature we get:

$$\mathrm{e_c} = \frac{\frac{\mathrm{dp_t}}{\mathrm{p_t}}}{\frac{\mathrm{dh_t}}{\mathrm{RT_t}}} = \frac{\frac{\mathrm{dp_t}}{\mathrm{p_t}}}{\frac{\mathrm{C_p dT_t}}{\mathrm{RT_t}}} = \frac{\frac{\mathrm{dp_t}}{\mathrm{p_t}}}{\frac{\gamma_{\mathrm{cold}}\mathrm{dT_t}}{(\gamma_{\mathrm{cold}}-1)\,\mathrm{T_t}}} \text{ (2.2.2.9)}$$

$$\frac{\mathrm{dp_t}}{\mathrm{p_t}} = \left(\frac{\gamma_{\mathrm{cold}}\mathrm{e_c dT_t}}{(\gamma_{\mathrm{cold}}-1)\mathrm{T_t}}\right) \text{ (2.2.2.10)}$$

Integrating gives: $\frac{\mathrm{p_{t_3}}}{\mathrm{p_{t_2}}} = \pi_c = \left(\frac{\mathrm{T_{t_3}}}{\mathrm{T_{t_2}}}\right)^{\frac{\gamma_{\mathrm{cold}}\mathrm{e_c}}{\gamma_{\mathrm{cold}}-1}} = \tau_c^{\frac{\gamma_{\mathrm{cold}}\mathrm{e_c}}{(\gamma_{\mathrm{cold}}-1)}}$ (2.2.2.11)

Polytropic efficiency: $\tau_c = \pi^{\frac{\gamma_{\mathrm{cold}}-1}{\gamma_{\mathrm{cold}}\mathrm{e_c}}}$ (2.2.2.12); $\tau_{c,\mathrm{real}} > \tau_{c,\mathrm{isentropic}}$ or $\mathrm{T_{t_3}} > \mathrm{T_{t_{3s}}}$

The relations ship between adiabatic and polytropic efficiency:

$$\eta_c = \frac{\pi_c^{\frac{\gamma_{\mathrm{cold}}-1}{\gamma_{\mathrm{cold}}}} - 1}{\tau_c - 1} = \frac{\pi_c^{\frac{\gamma_{\mathrm{cold}}-1}{\gamma_{\mathrm{cold}}}} - 1}{\pi_c^{\frac{\gamma_{\mathrm{cold}}-1}{\gamma_{\mathrm{cold}}\mathrm{e_c}}} - 1} \text{ ; (2.2.2.13)}$$

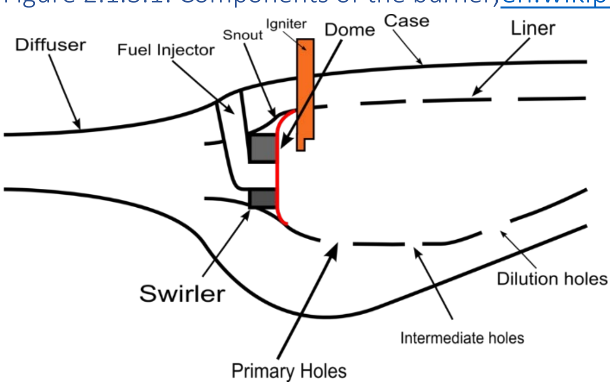
$$V_2^2 \approx V_3^2 \Rightarrow V_2 \approx V_3 \text{ ; (2.2.2.14)}$$

The burner

A combustion must contain and maintain stable combustion despite very high air flow rates, therefore the air and the fuel must first mix then ignite, and then mix in more air to complete the combustion process

Components:

Figure 2.1.3.1: Components of the burner,en.wikipedia.com



Case: the outer shell of the combustor, it requires little maintenance and protected from thermal loads by the flowing of air through it, it also serves as a pressure vessel that have to withstand the pressure difference outside and inside the combustor

The diffuser: it slows the high compressed fast air from the compressor to a velocity optimal for the combustor, which causes unavoidable total pressure, it is mostly designed to be light and short

Liner: it has to withstand extended high temperature cycles, therefore it 's made of superalloy ,furthermore it is cooled with air flow or in addition to thermal barrier coating

Snout: it separates the primary air from the secondary air flows

Dome\swirler: it generate turbulence in the flow to rapidly mix the air with fuel. Old combustors use dome which create wake turbulence to mix, while new combustors use swirlers to establish a local low-pressure zone that forces some of the combustion products to recirculate, creating a high turbulence

Fuel injector: it mixes the fuel with air and introduces fuel to the combustion zone

Igniter: it is located far enough upstream in order not to be damaged by the combustion, the oxygen is injected to the ignition area, helping the fuel easily combust

The ideal burner burns slowly because burner Mach number and wall friction are zero, which cause the pressure to conserve, meanwhile the real burner, contain wall friction, turbulent mixing, and chemical reaction at finite Mach number

Real combustion in chamber: $\pi_b = \frac{p_{t_4}}{p_{t_3}} < 1$ (2.2.3.1)

Ideal combustion in chamber: $\pi_b = 1$ (2.2.3.2)

Kerrebrock approximate expression: $\pi_b = 1 - \frac{\epsilon\gamma}{2} M_b^2 \,;\ (2.2.3.3) \text{where } 1 < \epsilon < 2$

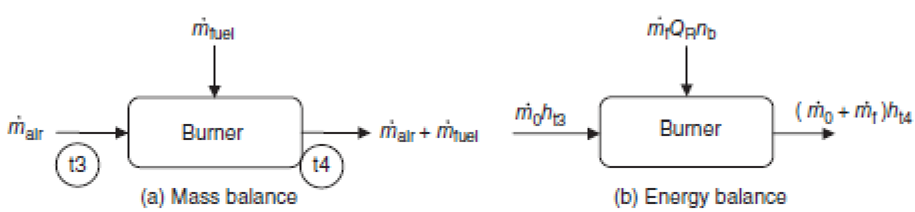


Figure 2.1.3.2: equilibrium representation of the TJ burner,source: Farokhi, Saeed. Aircraft propulsion

Equilibrium equation yields:

$$\dot{m}_4 = \dot{m}_0 + \dot{m}_f = \dot{m}_0(1 + f) \text{ ; (2.2.3.4) where f= fuel to air ratio}$$

$$\dot{m}_0 h_{t_3} + \dot{m}_f Q_R \eta_b = (\dot{m}_0 + \dot{m}_f) h_{t_4} = \dot{m}_0 (1 + f) h_{t_4} \text{ (2.2.3.5)}$$

Dividing by m_0,

$$f = \frac{h_{t_4} - h_{t_3}}{Q_R \eta_b - h_{t_4}} = \frac{\frac{h_{t_4}}{h_0} - \frac{h_{t_3}}{h_0}}{\frac{Q_R \eta_b}{h_0} - \frac{h_{t_4}}{h_0}} = \frac{\tau_\lambda - \tau_r \tau_c}{Q_R \frac{\eta_b}{h_0} - \tau_\lambda} \text{ (2.2.3.6)}$$

Burner efficiency: $\eta = \frac{Q_{R,Actual}}{Q_{R,ideal}}$; (2.2.3.7)

The ideal heat of reactions: $Q_R = 42000 \left[\frac{kJ}{Kg}\right]$

$$h_{t_3} + f\, Q_R\, \eta_b = (1 + f) h_{t_4}; \text{ (2.2.3.8)}$$

$$\tau_r\, \tau_c = \frac{h_{t_3}}{h_0} \text{ ; (2.2.3.9)}$$

Cyclical thermal limit parameter: $\tau_\lambda = \frac{h_{t_4}}{h_0}$; (2.2.3.10)

The turbine

it is a rotary mechanical device that that extracts energy from a fluid flow and converts it into useful work

A hot combustion gas expands through the turbine, it spins the rotating blades. The rotating blades perform a dual function: they drive the compressor to bring more pressurized air into the combustion chamber, and they spin a generator to produce electricity, Gas turbine have very high-power densities, since they run at very high speeds

In a real gas turbine, mechanical energy is changed irreversibly into pressure and thermal energy (due to internal friction and turbulence)

Turbine function is limited by temperature, this can be improved by a combination of cooling technologies and advanced materials, or install a recuperator or heat recovery steam generator (HRSF), a recuperator captures heat loss to preheat the compressor discharge before it enters the combustion chamber while HRSG generates steam by capturing heat from turbine exhaust, a single cycle gas turbine can achieve efficiency between (20-35)%

Turbine adiabatic efficiency:

$$\mathcal{P}_{t,actual} = \dot{m}_t \left(h_{t_4} - h_{t_5}\right) = \dot{m}_t \Delta h_{t,actual} \text{ ; (2.2.4.1)}$$

$$\mathcal{P}_{i,ideal} = \dot{m} \left(h_{t_4} - h_{t_{5s}}\right) = \dot{m}_t \Delta h_{t,isentropic}; \text{ (2.2.4.2)}$$

Where m_t: turbine mass flow rate

Equilibrium equation: $\dot{m}_t = \dot{m}_0 + \dot{m}_f = (1 + f)\dot{m}_0$; (2.2.4.3)

Turbine adiabatic efficiency: $\eta_t = \frac{h_{t_4} - h_{t_5}}{h_{t_4} - h_{t_{5s}}} = \frac{\Delta h_{t,actual}}{\Delta h_{t,isentropic}}$; (2.2.4.4)

Diving by h_{t_4}

$$\eta_t = \frac{1-\frac{T_{t_5}}{T_{t_4}}}{1-\frac{T_{t_{5s}}}{T_{t_4}}} \text{ ; (2.2.4.5)}$$

Isentropic formula: $\frac{T_{t_{5s}}}{T_{t4}} = \left(\frac{p_{T_{5s}}}{p_{T_4}}\right)^{\frac{\gamma_{hot}-1}{\gamma_{hot}}} = \left(\frac{p_{T_5}}{p_{T_4}}\right)^{\frac{\gamma_{hot}-1}{\gamma_{hot}}} = \pi_t^{\frac{\gamma_{hot}-1}{\gamma_{hot}}}$; (2.2.4.6)

$turbine\ adiabatic\ efficiency\ and\ total\ pressure\ temperature\ ratio$ $\eta_t = \frac{1-\tau_t}{1-\pi_t^{\frac{\gamma-1}{\gamma}}}$; (2.2.4.7)

Turbine Polytropic efficiency: $e_t = \frac{dh_t}{dh_{ts}} = \frac{\frac{dh_t}{dp_t}}{\rho_t} = \frac{1}{1-\frac{C_n}{C_{p_{hot}}}}$; (2.2.4.8)

The nozzle

It is a device used to control fluid flow characteristics as it enters an enclosed chamber, it increases the kinetic energy of the flowing medium at the expense of its pressure and internal energy.

propelling nozzle: it converts the internal energy of a working gas into propulsive force, it can accelerate the gas to subsonic, transonic, or supersonic velocities, it might have a convergent shape which can accelerate the jet beyond sonic speed or convergent-divergent shape which take the flow past it's choke point and accelerate the jet to supersonic velocities, the nozzle has to withstand high heat and pressure in which it can expand and contract without damage or distortion, therefor it has to be insulated by isolating the jet pipe from the aircraft or by a short section of insulation

propelling nozzle generate pressure thrust which is different from thrust obtained from the momentum change of gas stream

When the gas enters the convergent section of the nozzle, the gas velocity increases with a corresponding fall in static pressure. As the gas leaves the restriction of the throat and flows into the divergent section, it progressively increases in velocity towards the exit. The reaction to this further increase in momentum is a pressure force acting on the inner wall of the nozzle. A component of this force acting parallel to the longitudinal axis of the nozzle produces the further increase in thrust.

The geometry of the propelling nozzle is important in shaping the correct balance of pressure, temperature and thrust, while a fixed area propelling nozzle is only efficient over a narrow range if engine operating conditions, a variable area nozzle which is automatically controlled can function in all operating conditions, the hot and cold nozzle are coaxial, and the area of each nozzle is designed to obtain maximum efficiency

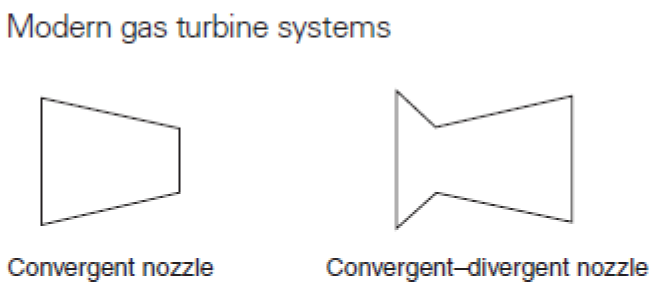
Construction: the main material used for nozzle construction are as follows:

Inconel: a nickel chromium alloy classified as superalloy, it is resistant to high or low temperatures, and can withstand long exposure to sea water

Stainless steel alloys: which has at least 10% chromium that is austenitic, ferritic, and martensitic

Titanium: it has high tensile strength to density ratio, it can withstand a high temperature since it is high corrosion-resistant, due to the high demand, the purchase is regulated in the united states

Figure 2.1.5.1: Schematic representation of convergent and convergent-divergent propelling nozzles Farokhi, Saeed. Aircraft propulsion



Gross thrust: $\mathrm{F_g} = \dot{\mathrm{m}}_9\mathrm{V}_9 + (\mathrm{p}_9 - \mathrm{p}_0)\mathrm{A}_9$; (2.2.5.1)

Rule 01: if $M_{jet} < 1$, then $P_{jet}, P_{ambient}$ (2.2.5.2)

Rule 02: if $P_{jet} = P_{ambient}$: we have perfectly expanded nozzle which results in $F_{g,max}$ (2.2.5.3)

Rule 03: $if\ NPR \geq (NPR)_{critical}$: the nozzle throat velocity is sonic, $M_8 = 1.0$ (2.2.5.4)

Rule 04: $\frac{P_{t_7}}{P_0} > \approx 2: nozzle\ throat\ can\ be\ choked\ M_8 = 1.0$ (2.2.5.5)

Rule 05: across the slip stream of a jet exhaust plume, static pressure must be continuous (2.2.5.6)

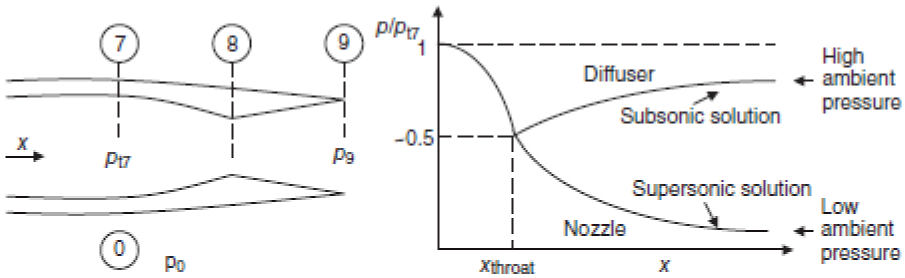


Figure 2.1.5.2: a choked convergent divergent nozzle and the static pressure along nozzle axis, Farokhi, Saeed. Aircraft propulsion

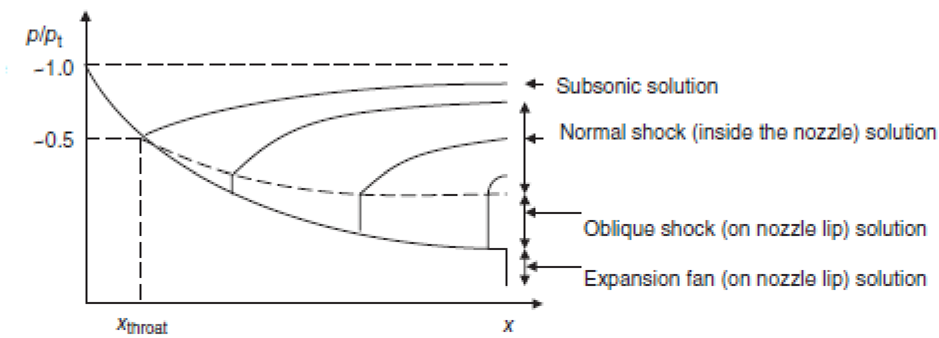


Figure 2.1.5.3: possible static pressure distribution inside a choked supersonic nozzle, Farokhi, Saeed. Aircraft propulsion

Nozzle adiabatic efficiency: $\eta_n = \frac{h_{t_7}-h_9}{h_{t_7}-h_{9s}} = \frac{\frac{V_9^2}{2}}{\frac{V_{9s}^2}{2}}$; (2.2.5.7)

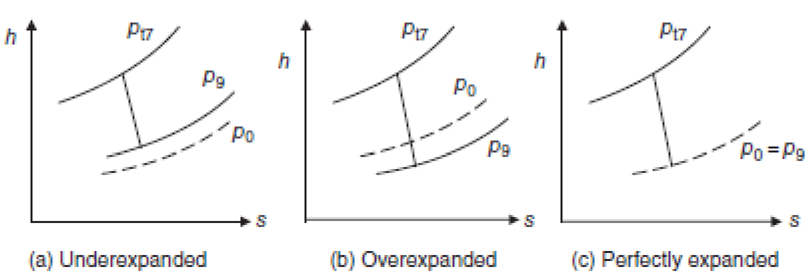


Figure 2.1.5.4: h-s diagram of three possible nozzle expansion, Farokhi, Saeed. Aircraft propulsion

Thermal efficiency of a turbojet engine:

An ideal turbo jet engine is the same as Brayton cycle

Brayton cycle efficiency: $\eta_{th} = 1 - \frac{T_0}{T_3}$; (2.2.6.1)

Carnot cycle efficiency: $\eta_{th-Carnot} = 1 - \frac{T_0}{T_4}$;(2.2.6.2)

Cycle specific work: $w_c = (h_{t4} - h_9) - (h_{t3} - h_0)$; (2.2.6.3)

Engine thermal limit: $\tau_\lambda = \frac{h_{t4}}{h_{t0}}$; (2.2.6.4)

Ram temperature ratio: $\tau_r = 1 + \gamma_{cold} - \frac{1}{2} M_0^2$; (2.2.6.5)

Compressor temperature ratio $\tau_c = \frac{T_{t_3}}{T_{t_2}}$; (2.2.6.6)

References:

Farokhi, Saeed. Aircraft propulsion. John Wiley & Sons, 2014.

Chapter 03: Experiment description

Introduction: in this chapter we are going to explain the procedure of the total pressure and temperature measurements on CM14 axial gas turbine engine

Description

We have integrated CM14 axial gas turbine into a robust metal frame that fixes it firmly, while enabling accurate engine thrust measurement, the engine inlet has been replaced with a custom fabricated frontal duct, that enables an accurate measurement of the air mass flow rate. The engine supervision was conducted through a preprogrammed controller to ensure safe operation condition all time

Figure 3.1 CM 14 axial gas turbine setting, source: armfield.co.uk

We have controlled the engine speed using a high precision fuel gear pump through a graphical interface, in which optimal start-up and power-down sequences are already set to assure minimum mechanical stresses.

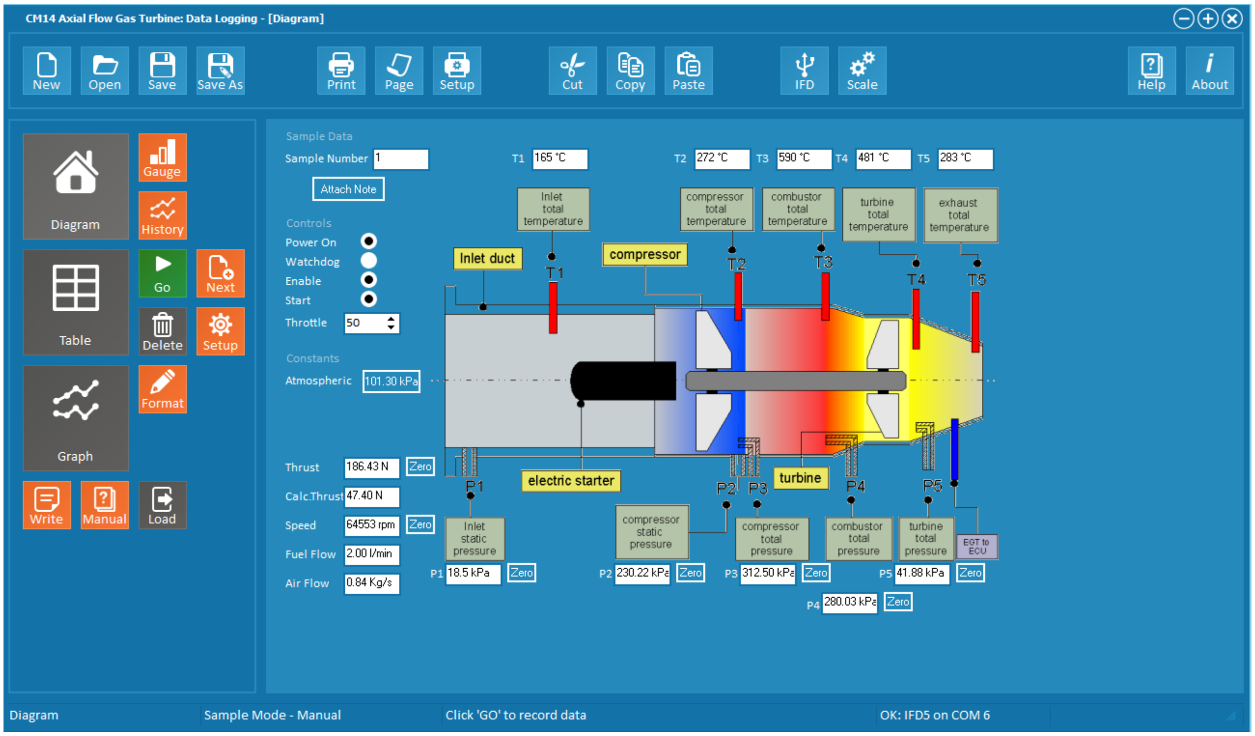


Figure 3.2 graphical interface of CM14 Axial flow gas turbine, source: armfield.co.uk

We have measured 9 inter-related input variables in which 101 measurement were recorded for each one and saved in an excel sheet. input variables together with their symbolic representations are shown in table below

Input variable	Symbols
Inlet static pressure	P_{t2}
Compressor total pressure	P_{t3}
Combustor total pressure	P_{t4}
Turbine total pressure	P_{t5}
Inlet total temperature	T_{t2}
Compressor total temperature	T_{t3}
Combustor total temperature	T_{t4}
Turbine total temperature	T_{t5}
Exhaust total temperature	T_9

Chapter 04: Choice of parameters for identification:

Introduction: in this chapter we are going to explain the reason why we have chosen those parameters as outputs during the analysis of CM14 axial gas turbine engine

Choice arguments

Compressor efficiency η_c: It is a function of compressor pressure ratio. It is also the ratio of work output for an ideal isentropic compression process to the work input to develop the required heat.

Compressor polytropic efficiency e_t: Also called small stage efficiency, it involves finite jumps Δh_t and infinitesimal stepsdh_t, and Its actually the adiabatic compressor efficiency with small pressure ratios

Total pressure drops in the burner π_b: Because it represents turbulent mixings and chemical reactions at the burner, in a function of specific heat constant and Mach number

Burner efficiency η_b: It relates the actual heat reaction and ideal heat reaction, also it gives the designer a brief knowledge of the burner, since it is related to fuel to air ratio, burner total enthalpy, and compressor total enthalpy. It defines the worthiness of the burner because it represents how much fuel is needed to meet our heating needs

f: It represents how much fuel is burned in respect to real air gas

Turbine adiabatic efficiency η_t: It represents the ratio between actual work output of the turbine and the net input energy supplied in the form of fuel, and since state T_{5s} and T_4 lies on the same isentrope, the turbine adiabatic efficiency is a function of total pressures and total temperatures

Turbine polytropic efficiency e_t: It defines the ratio between real turbine work and isentropic turbine work, therefore it is a function of enthalpies

Nozzle adiabatic efficiency η_n: It is a fraction of ideal exit kinetic energy and actual exit kinetic energy, it is also related to nozzle total pressure π_n

V9: Nozzle exit velocity, it required in the determination of nozzle adiabatic efficiency

Thermal efficiency η_{th}: It shows the engine performance in dependance of temperature, it give the designer a brief knowledge of engine thermal limits

Chapter 05: Parameter identification based on engine thermal cycle

Introduction: in this chapter we are going to identify the parameters of CM 14 gas turbine engine based on the lab measured data. First we start by calculating specific parameters that are going to be used by both methods, the first method in which we calculate the output(mean(input)) , and in the second method we calculate mean(outut(input))

$$C_{p_{cold}} = \frac{\gamma * r}{\gamma - 1} = 0.2857$$

$$C_{p_{hot}} = \frac{\gamma * r}{\gamma - 1} = 0.2481$$

$$C_{v_{cold}} = \left(\frac{r}{\gamma - 1}\right) = 717.5$$

$$C_{v_{hot}} = \left(\frac{r}{\gamma - 1}\right) = 869.697$$

First method

A total of 101 measurement was given for each set, In this part: we will calculate the mean value of the inputs, then we proceed to get the required output, the procedure is shown below:

Representation	**Symbols**	Input mean values	**Units**
Inlet static pressure	P_{t2}	99818	[Pa]
Compressor total pressure	P_{t3}	3.9461e+05	[Pa]
Combustor total pressure	P_{t4}	3.9311e+05	[Pa]
Turbine total pressure	P_{t5}	2.0367e+05	[Pa]
Inlet total temperature	T_{t2}	295.9	[K]
Compressor total temperature	T_{t3}	479.02	[K]
Combustor total temperature	T_{t4}	1158.9	[K]
Turbine total temperature	T_{t5}	998.45	[K]
Exhaust total temperature	T_9	837.85	[K]
Specific heat ratio of the air (cold part of the engine)	γ_{hot}	1.4	
Specific heat ratio of combustion gas (hot part of the engine)	γ_{hot}	1.33	
Specific gas constant	r	287	$[J \cdot kg^{-1} \cdot K^{-1}]$

Table 01: inputs mean values

Inlet

Calculating the total enthalpies:

$$h_{t1} = C_{p_{cold}} * T_{t2} = 2.9723 * 10^5 \left[\frac{J}{kg}\right]$$

$$h_{t2} = h_{t1}$$

Total Pressure recovery: $\pi_D = \frac{P_{t2}}{P_{t2}} = 1$

$$T_{t_{1s}} = T_{t2} * \left(\frac{P_{t2}}{P_{t2}}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} = 295.9\ [K]$$

$$h_{t1s} = C_{p_{cold}} * T_{1s} = 2.9723 * 10^5 [K]$$

Compressor

$$T_{t2_s} = T_{t2} * \left(\frac{P_{t3}}{P_{t2}}\right)^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} = 438.24\ [K]$$

Compressor efficiency: $\eta_c = \frac{\frac{T_{t2_s}}{T_{t2}}-1}{\frac{T_{t3}}{T_{t2}}-1} = 0.77727$

Compressor polytropic efficiency: $e_c = \frac{\left(\frac{P_{t3}-P_{t2}}{P_{t2}}\right)}{\left(\frac{\gamma_{cold}}{\gamma_{cold}-1}\right) * \left(\frac{T_{t4}-T_{t3}}{T_{t3}}\right)} = 0.59448$

Compressor total temperature ratio: $\tau_c = \frac{P_{t3}}{P_{t2}} = 3.9533$

Compressor Pressure Ratio: $\pi_c = \tau_c = 3.9533$

Burner

$$M_b = 0$$

$$pi_b = 1$$

$$Q_R = 42000 * 10^3$$

Total Pressure Drop: $\pi_b = \frac{P_{t4}}{P_{t3}} = 0.9962$

Burner efficiency was chosen as: $\eta_b = 0.995$

$$h_{t3} = C_{p_{hot}} * T_{t3} = 5.5409 * 10^5 \left[\frac{J}{kg}\right]$$

$$h_{t4} = C_{p_{hot}} * T_{t4} = 1.3405 * 10^6 \left[\frac{J}{kg}\right]$$

$$f = \frac{h_{t4} - h_{t3}}{Q_R * \eta_b - h_{t4}} = 0.019443$$

Turbine

$$T_{t4_s} = T_{t4} \; * \left(\frac{P_{t5}}{P_{t4}}\right)^{\frac{\gamma_{hot}-1}{\gamma_{hot}}} = 984.47\ [K]$$

$$h_{t4_s} = C_{p_{hot}} * T_{t4_s} = 1.1387 * 10^6 \left[\frac{J}{kg}\right]$$

Turbine adiabatic efficiency: $\eta_t = \frac{1-\frac{T_{t5}}{T_{t4}}}{1-\frac{T_{t4_s}}{T_{t4}}} = 0.91987$

Polytropic exponent: $n = -\frac{1}{\left(\frac{\log\left(\frac{T_{t5}}{T_{t4}}\right)}{\log\left(\frac{P_{t5}}{P_{t4}}\right)}\right)-1} = 1.2931$

Polytropic specific heat capacity: $C_n = C_{v_{hot}} * \frac{n-\gamma_{hot}}{n-1} = -109.42 \; \left[\frac{J}{K}\right]$

Turbine polytropic efficiency: $e_t = \frac{1}{1-\frac{C_n}{C_{p_{hot}}}} = 0.91357$

Nozzle

$$P9 = P_{t5} * \exp\left(\left(\frac{C_{p_{hot}}}{r}\right) * \log\left(\frac{T9}{T_{t5}}\right)\right) = 99818\ [Pa]$$

$$T_{t5_s} = T_{t5} * \left(\frac{P9}{P_{t5}}\right)^{\frac{\gamma_{hot}-1}{\gamma_{hot}}} = 814.39\ [K]$$

$$h_{t5} = C_{p_{hot}} * T_{t5} = 1.1549 * 10^6 \left[\frac{J}{kg}\right]$$

$$h9 = C_{p_{hot}} * T9 = 9.6914 * 10^5 \left[\frac{J}{Kg}\right]$$

$$h_{t5_s} = C_{p_{hot}} * T_{t5_s} = 9.4201 * 10^5 \left[\frac{J}{Kg}\right]$$

Nozzle total pressure ratio: $\pi_n = \frac{P9}{P_{t5}} = 0.4901$

$$V_9 = \sqrt{\left(2 * \left(h_{t5} - h9\right)\right)} = 609.53 \left[\frac{m}{s}\right]$$

$$V5_s = \sqrt{2 * \left(h_{t5} - h_{t5_s}\right)} = 652.53 \left[\frac{m}{s}\right]$$

Nozzle adiabatic efficiency: $\eta_n = \frac{\frac{V_9^2}{2}}{\frac{V_{5s}^2}{2}} = 0.87256$

Thermal efficiency of turbojet engine

$h_0 = h_{t2}$

$h_{t9} = h_{t5}$

$h_{t0} = h_0$

Thermal efficiency: $\eta_{th} = \frac{(h_{t4}-h_{t3})*\left(1-\frac{T_{t2}}{T_{t3}}\right)}{h_{t4}-h_{t3}} = 0.2362$

Engine thermal limit: $\tau_\lambda = \frac{h_{t4}}{h_{t1}} = 4.5101$

Ram temperature ratio: $\tau_r = 1 + \gamma_{hot} = 2.4$

Compressor temperature ratio: $\tau_c = \frac{T_{t3}}{T_{t2}} = 1.6189$

optimum compressor pressure ratio: $\pi_{c\ Optimum} = \tau_c^{\frac{\gamma_{cold}-1}{\gamma_{cold}}} = 1.1476$

Table 02: Results of the first method

Outputs	First Method $output(mean(input))$
Compressor efficiency η_c	0.77727
Compressor polytropic efficiency e_t	0.59448
Total pressure drops in the burner π_b	0.9962
Burner efficiency η_b	0.995
f	0.019443
Turbine adiabatic efficiency η_t	0.91987
Turbine polytropic efficiency e_t	0.91357
Nozzle adiabatic efficiency η_n	0.87256
V9	609.53
Thermal efficiency η_{th}	0.2362

Second method

In this part we will calculate the output in similar way like the first method, the only difference is that, this time, we calculate the output of each input, then we calculate the mean of every set of outputs

Table 03: results of the second method

Outputs	Second Method $mean(output(input))$
Compressor efficiency η_c	0.76549

Compressor polytropic efficiency e_t	0.59967
Total pressure drops in the burner π_b	0.99619
Burner efficiency η_b	0.995
f	0.019369
Turbine adiabatic efficiency η_t	0.91552
Turbine polytropic efficiency e_t	0.90897
Nozzle adiabatic efficiency η_n	0.8822
V9	612.25
Thermal efficiency η_{th}	0.23933

Comparison

Table 04: comparison between the first and second method

Outputs	First Method $\mathrm{output}(\mathrm{mean}(\mathrm{input}))$	Second Method $\mathrm{mean}(\mathrm{output}(\mathrm{input}))$	$\frac{\mathrm{Output1}}{\mathrm{output2}} - 1$	$\frac{\mathrm{Output1}}{\mathrm{output2}} - 1$
Compressor efficiency η_c	0.77727	0.76549	0.015389653	1.54% ▲
Compressor polytropic efficiency e_t	0.59448	0.59967	-0.008650742	-0.87% ▼
Total pressure drops in the burner π_b	0.9962	0.99619	5.35086E-06	0.00% ▲
Burner efficiency η_b	0.995	0.995	0	
f	0.019443	0.019369	0.003838822	0.38% ▲
Turbine adiabatic efficiency η_t	0.91987	0.91552	0.004886734	0.49% ▲
Turbine polytropic efficiency e_t	0.91357	0.90897	0.005068345	0.51% ▲
Nozzle adiabatic efficiency η_n	0.87256	0.8822	-0.01093527	-1.09% ▼
V9	609.53	612.25	-0.004439756	-0.44% ▼
Thermal efficiency η_{th}	0.2362	0.23933	-0.013082743	-1.31% ▼

From Table 06, we can see that the ratio between the outputs of first method and the second method is in the interval [-1.31%; 1.54%] , the first method output is smaller proximately by [0-1] % for compressor polytropic efficiency e_t, nozzle adiabatic efficiency η_n ,V_9 and Thermal efficiency η_{th}, meanwhile it higher by [0% ; 1.54%] for compressor efficiency η_c, total pressure drop in the burner π_b,Burner efficiency η_b, Turbine adiabatic efficiency η_t, and Turbine polytropic efficiency e_t

To understand the previous results, we perform some Analysis to both input and output sets, Table : shows the respective Data

Std: standard deviation

Table 05: statistical analysis of inputs

Input Sets	Mean(input)	Std(input)
Inlet static pressure P_{t2}	99818	688.81
Compressor total pressure P_{t3}	3.9461e+05	1249.3
Combustor total pressure P_{t4}	3.9311e+05	1251.6
Turbine total pressure P_{t5}	2.0367e+05	1169.8
Inlet total temperature T_{t2}	295.9	4.155
Compressor total temperature T_{t3}	479.02	7.0869
Combustor total temperature t_{t4}	1158.9	2.9917
Turbine total temperatureT_{t5}	998.45	4.0128
Exhaust total temperature T_9	837.85	4.3439

Table 06: statistical analysis of outputs

Output Sets	Mean(output)	Std(output)
Compressor efficiency η_c	0.76549	0.038914
Compressor polytropic efficiency e_t	0.59967	0.01679
Total pressure drops in the burner π_b	0.99619	0.00073978
Burner efficiency η_b	0.995	0
f	0.019369	0.00021542
Turbine adiabatic efficiency η_t	0.91552	0.026488
Turbine polytropic efficiency e_t	0.90897	0.028379
Nozzle adiabatic efficiency η_n	0.8822	0.030654
V9	612.25	11.493
Thermal efficiency η_{th}	0.23933	0.0092138

The highest error is in the compressor efficiency: this is mainly because the pressure and temperature increase in the compressor, in which the random fast motion of the gas in the compressor, causes more disturbance in the pressure and temperature sensors, we can see this in table 05, where the standard deviation has increased significantly between P_{t2} and P_{t3}, $T_{t2} and$ T_{t3} thus the measurement becomes less accurate, but still in an accepted range

The standard deviation for burner efficiency is 0 because is was estimated and not calculated

Conclusion

The purpose of this study was to identify CM 14 axial gas turbine engine parameters through thermal cycle analysis. We started by giving a briefing of different types of gas turbine engines, this will motivate the designers to further learn about this topic.

we discussed some general thermodynamic concepts and the cycle analysis of the turbojet engine, in addition we explained the experiment and further more the reason for choosing our outputs

eventually we have calculated the required parameters of CM 14 axial gas turbine engine using matlab Simulink in both methods, and compared and performed statistical analysis

we can see that the maximum calculation error rate between the two methods is 1.54%, which confirms the efficiency of the proposal model

The presented computational thermodynamic Analysis might be of good use for an aircraft designer, or a TJ engine Tester

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