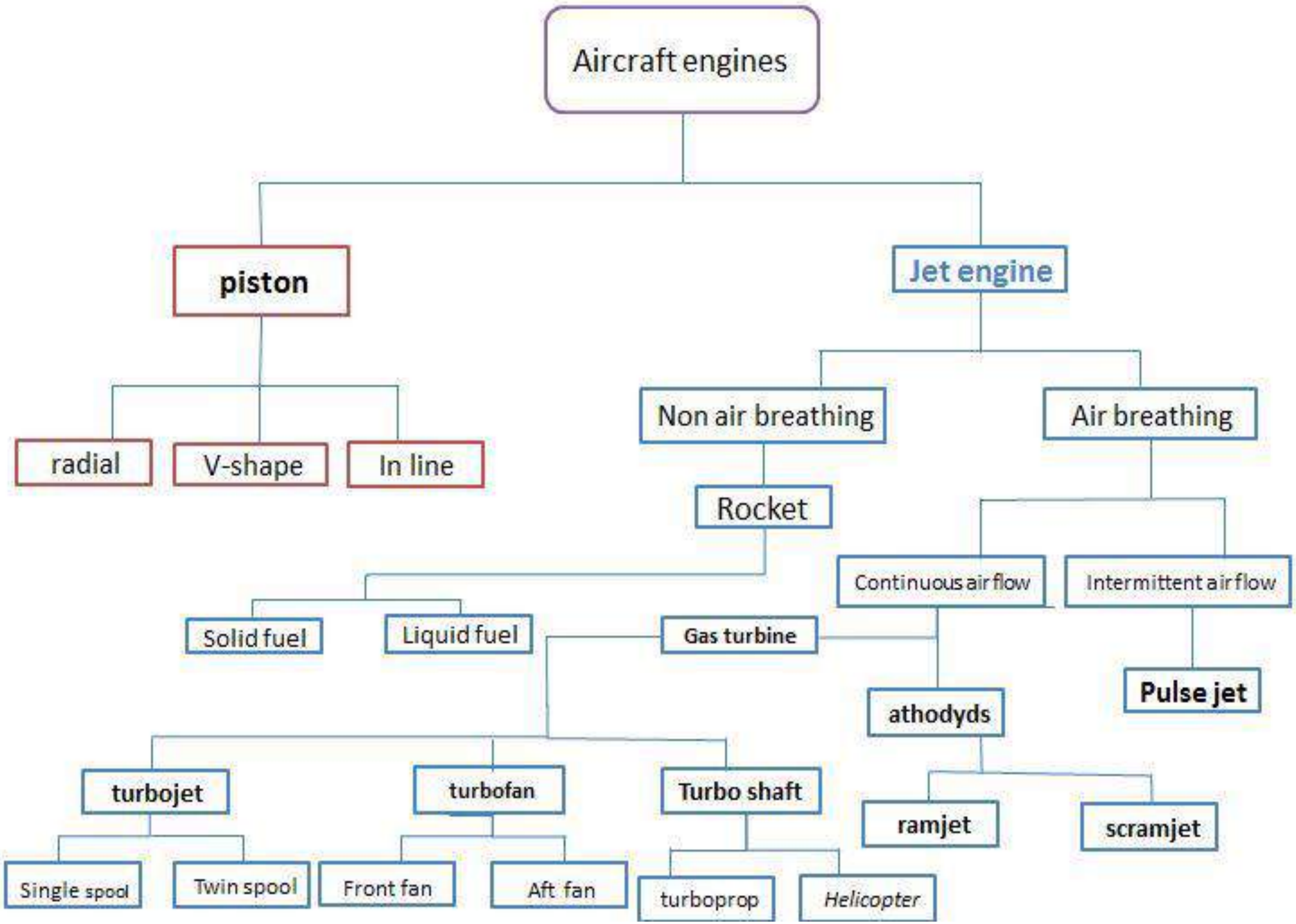
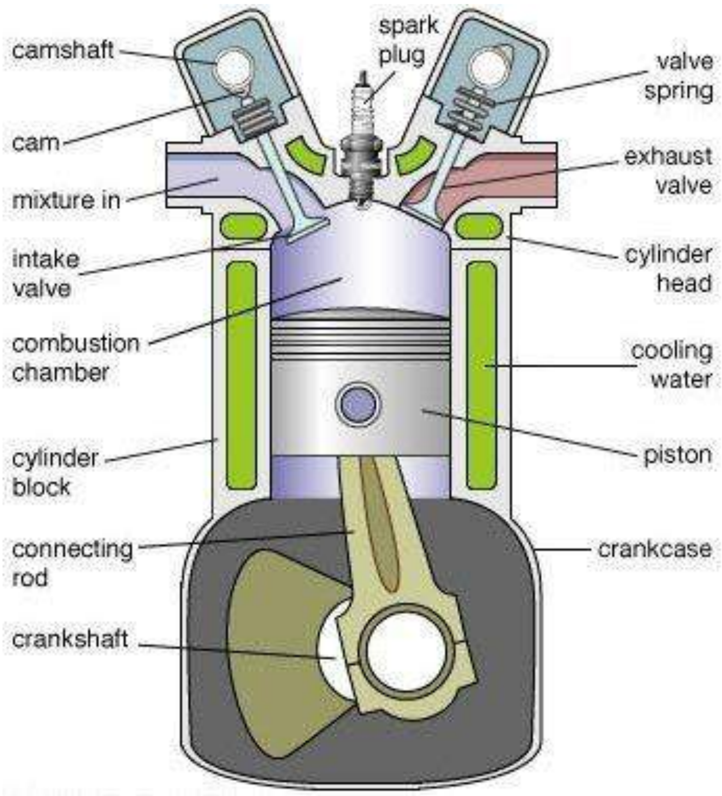

Aircraft engines
piston
Jet engine
radial
V-shape
In line
Non air breathing
Air breathing
Rocket
Continuous air flow
Intermittent air flow
Solid fuel
Liquid fuel
Gas turbine
athodyds
Pulse jet
turbojet
turbofan
Turbo shaft
ramjet
scramjet
Single spool
Twin spool
Front fan
Aft fan
turboprop
Helicopter

Parts of a Reciprocating Engine

Reciprocating engine is a heat engine that utilizes one or more reciprocating pistons to convert pressure into rotating motion. A reciprocating engine is also referred to as an internal combustion engine. The naming criterion derives from the fuel mixture burned within the engine. Major parts of a reciprocating engine include the cylinders, pistons, connecting rods, crankshaft, valves, spark plugs and a valve operating mechanism. These are all used to power conventional vehicles.

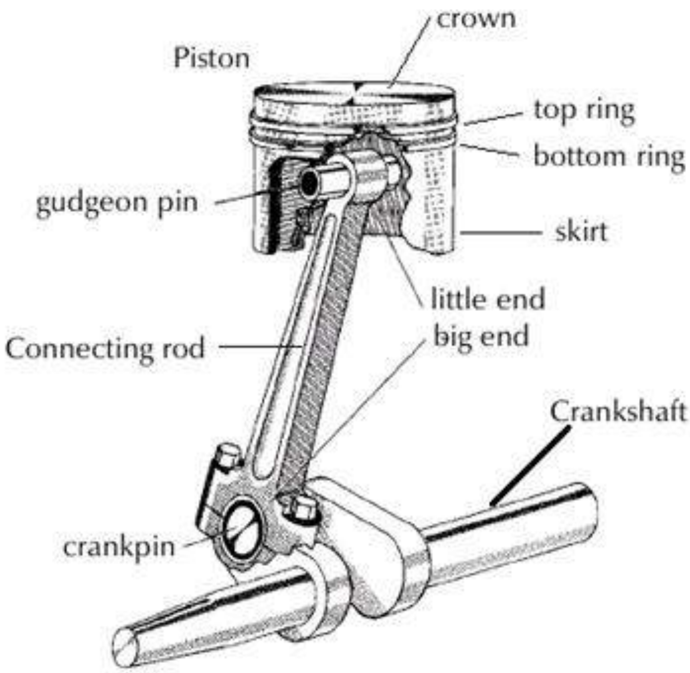
cylinder

A cylinder in a reciprocating engine refers to the confined space in which combustion takes place. Cylinders are arranged in several ways. These include: a single row arrangement, a V-shape arrangement, a W-shape arrangement and a horizontal or flat arrangement.



Piston

Pistons in a reciprocating engine are usually attached to each cylinder. In a reciprocating engine, a piston slides up and down to create a rotary motion. A piston's wall is usually grooved to hold rings that fit tightly against a cylinder wall, preventing gases from escaping the combustion chamber.



Connecting Rod

A connecting rod in a reciprocating engine links a piston and the crankcase held by a crankshaft. The connecting rod in a reciprocating engine, while connected to a rotary motion piston, is used to turn a propeller. This results in the rotary motion of the crankshaft.

1- Crankshaft

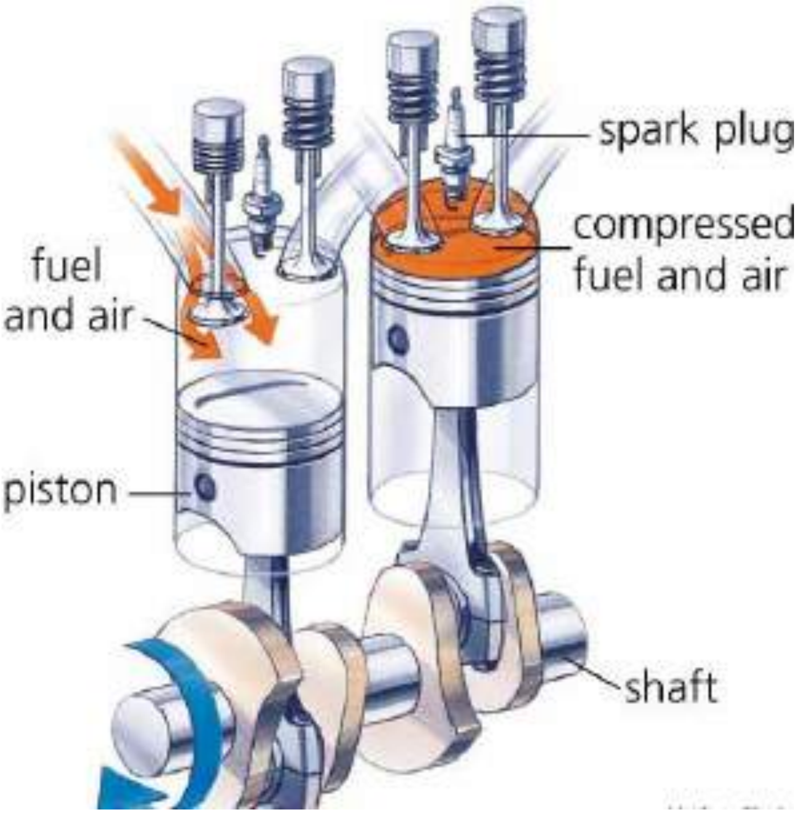
A crankshaft in a reciprocating engine transforms the up and down movement of a piston into rotary motion. While connected to a piston with a connecting rod, a crankshaft yields a rotary motion as the piston moves up and down. During an intake stroke in a piston engine, a piston is pulled downward, creating a vacuum in the cylinder chamber. During a compression stroke in a reciprocating engine, a crankshaft drives a piston upward in the cylinder. This back and forth motion yields a rotary movement in the crankshaft while varying pressure levels in a cylinder.

Valves

A reciprocating engine has an intake and an exhaust valve. These are located adjacent to the fuel-air mixture inlet and exhaust outlet at the top of a cylinder, respectively. An intake valve in a reciprocating engine regulates entry of the air and fuel mixtures while an exhaust valve lets out exhaust and burned gases from the combustion chamber.

Spark Plugs

Spark plugs in a reciprocating engine are usually located on top of a cylinder above the valves. They serve to ignite the compressed air and fuel mixture during the compression and ignition strokes in a reciprocating engine. Ignition takes place just before a piston reaches its top position. This results in very hot gases expanding rapidly to drive a piston down while turning the crankshaft to yield rotary motion.



FUNDAMENTALS OF GAS TURBINE

INTRODUCTION

The jet engine is an internal combustion engine that uses air as the working fluid. The engine extracts chemical energy from fuel and converts it to mechanical energy using the gaseous energy of the working fluid (air) to drive the engine and propeller, which, in turn, propel the airplane.

THE GAS TURBINE CYCLE

The basic principle of the airplane turbine engine is identical to any and all engines that extract energy from chemical fuel. The basic 4 steps for any internal combustion engine are:

1. Intake of air (and possibly fuel).
2. Compression of the air (and possibly fuel).
3. Combustion, where fuel is injected (if it was not drawn in with the intake air) and burned to convert the stored energy.
4. Expansion and exhaust, where the converted energy is put to use.

In the case of a piston engine, such as the engine in a car or reciprocating airplane engine, the intake, compression, combustion, and exhaust steps occur in the same place (cylinder head) at different times as the piston goes up and down.

In the turbine engine, however, these same four steps occur at the same time but in different places. As a result of this fundamental difference, the turbine has engine sections called:

1. The inlet section
2. The compressor section
3. The combustion section (the combustor)
4. The turbine (and exhaust) section.

The turbine section of the gas turbine engine has the task of producing usable output shaft power to drive the propeller. In addition, it must also provide power to drive the compressor and all engine accessories. It does this by expanding the high temperature, pressure, and velocity gas and converting the gaseous energy to mechanical energy in the form of shaft power.

A large mass of air must be supplied to the turbine in order to produce the necessary power. This mass of air is supplied by the compressor, which draws the air into the engine and squeezes it to provide high-pressure air to the turbine. The compressor does this by converting mechanical energy from the turbine to gaseous energy in the

form of pressure and temperature.

If the compressor and the turbine were 100% efficient, the compressor would supply all the air needed by the turbine. At the same time, the turbine would supply the necessary power to drive the compressor. In this case, a perpetual motion machine would exist. However, frictional losses and mechanical system inefficiencies do not allow a perpetual motion machine to operate. Additional energy must be added to the air to accommodate for these losses. Power output is also desired from the engine (beyond simply driving the compressor); thus, even more energy must be added to the air to produce this excess power. Energy addition to the system is accomplished in the combustor. Chemical energy from fuel as it is burned is converted to gaseous energy in the form of high temperatures and high velocity as the air passes through the combustor. The gaseous energy is converted back to mechanical energy in the turbine, providing power to drive the compressor and the output shaft.

SOME BASIC PRINCIPLES

As air passes through a gas turbine engine, aerodynamic and energy requirements

PERFORMANCE AND EFFICIENCY

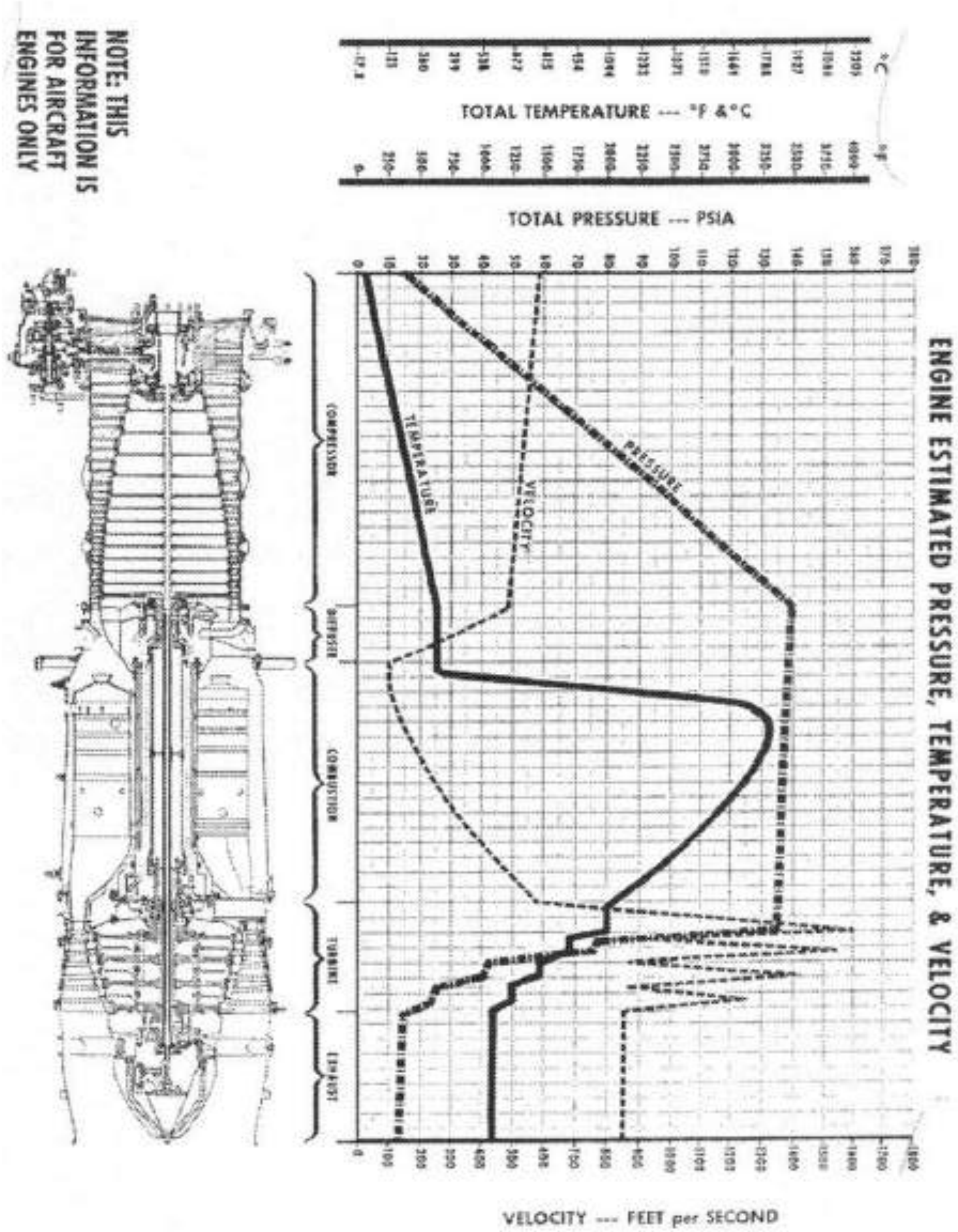
The type of operation for which the engine is designed dictates the performance requirement of a gas turbine engine. The performance requirement is mainly determined by the amount of shaft horse power (s.h.p.) the engine develops for a given set of conditions.

The need for high efficiency in the engine becomes more important as fuels become more costly. Engine efficiency is primarily defined by the specific fuel consumption (s.f.c.) of the engine at a given set of conditions.

Many factors affect both the efficiency and the performance of the engine. The mass flow rate of air through the engine will dictate engine performance. Any restrictions acting against the smooth flow of air through the engine will limit the engine's performance. The pressure ratio of the compressor, the engine operating temperatures (turbine inlet temperature), and the individual component efficiencies will also influence both the performance and the efficiency of the overall engine. All these factors are considered during the design of the engine. An optimum pressure ratio, turbine inlet temperature, and air mass flow rate are selected to obtain the required performance in the most efficient manner. In addition, individual engine components are designed to minimize flow losses to maximize component efficiencies.

the following graphic shows the typical temperature and pressure rise through the gas

flow path.



Engine tempreture and pressure flow

2.4.6 SPECIFIC FUEL CONSUMPTION

This performance parameter of the engine has a direct influence on the costs of aircraft trip and flight economics. Fuel consumption is either defined per unit thrust force (for ramjet, turbojet, and turbofan engines) or per horsepower (e.g., for turboprop and piston-propeller engines).

1. *Ramjet, turbojet, and turbofan engines:*

The TSFC is defined as

$$\text{TSFC} = \frac{\dot{m}_\text{f}}{T}$$

where the thrust force (T) is expressed by Equation 2.3 for a turbojet engine and Equation 2.4 for a turbofan engine; values ofTSFC strongly depend on the flight speed. So its typical values for both turbojet and turbofan are defined for static condition. Typical values [9] for turbojet engines are 0.075–0.11 kg/N · h (0.75–1.1 ib/ib$_f$ · h), while turbofans are more economic and have the following range 0.03–0.05 kg/N · h (0.3–0.5 ib/ib$_f$ · h). However, for ramjet reference values for TSFC are defined at flight Mach number of 2. Typical values are 0.17–0.26 kg/N · h (1.7–2.6 ib/ib$_f$ · h). Some empirical formulae are presented in Reference 11 for TSFC. These formulae have the form:

$$\text{TSFC} = (a + bM_0)\sqrt{\theta}$$

where a and b are constants that vary from one engine to another, M_0 is the flight Mach number, and θ is the dimensionless ratio (T_a/T_{ref}), which is the ratio between the ambient temperature at the flight altitude and the standard temperature at sea level (288.2 K). For a high bypass ratio turbofan, the values of these constants are $a = 0.4$ and $b = 0.45$.

2. *Turboprop engines*

For engines that produce shaft power, fuel consumption is identified by brake-specific fuel consumption (BSFC) or simply SFC, and defined as

$$\text{SFC} \equiv \text{BSFC} = \frac{\dot{m}_\text{f}}{SP}$$

When appreciable thrust is produced by the hot gases, the fuel consumption is identified by the equivalent EBSFC or simply equivalent specific fuel consumption (ESFC) and is defined by

$$\text{ESFC} = \frac{\dot{m}_\text{f}}{\text{ESP}}$$

Typical values [9] for ESFC are 0.45–0.60 ((*ib*/*h*)/*hp*) or 0.27–0.36 (kg/kW · h). The values of the corresponding constants in the empirical relation (2.25b) are $a = 0.2$ and $b = 0.9$ [11].

ENGINE SECTIONS

Inlet

The air inlet duct must provide clean and unrestricted airflow to the engine. Clean and undisturbed inlet airflow extends engine life by preventing erosion, corrosion, and foreign object damage (FOD).

Consideration of atmospheric conditions such as dust, salt, industrial pollution, foreign objects (birds, nuts and bolts), and temperature (icing conditions) must be made when designing the inlet system. Fairings should be installed between the engine air inlet housing and the inlet duct to ensure minimum airflow losses to the engine at all airflow conditions.

The inlet duct assembly is usually designed and produced as a separate system rather than as part of the design and production of the engine.

Dust does damage the blades but the effect is like sand blasting; it takes some time to cause structural damage. The turbine engines of airliners can take some dust (by design) but also get extensive maintenance regularly..

The biggest danger dust-wise is volcanic ash as that will adhere to the turbine blades behind the combustion chamber and potentially shut the engine down due to compressor stall.Lots of helos use separator devices, variously called EAPS (engine air particle separator) or IPS (inertial particle separator) to remove sand, dust etc. from inlet air. It's a bit of a drag on engine performance, but pays off by reducing compressor erosion for longer service life. helicopters which have air filters in their engine intakes, but jet fighters and commercial turbojet airliners don't seem to have any filters in their air intakes.

Compressor

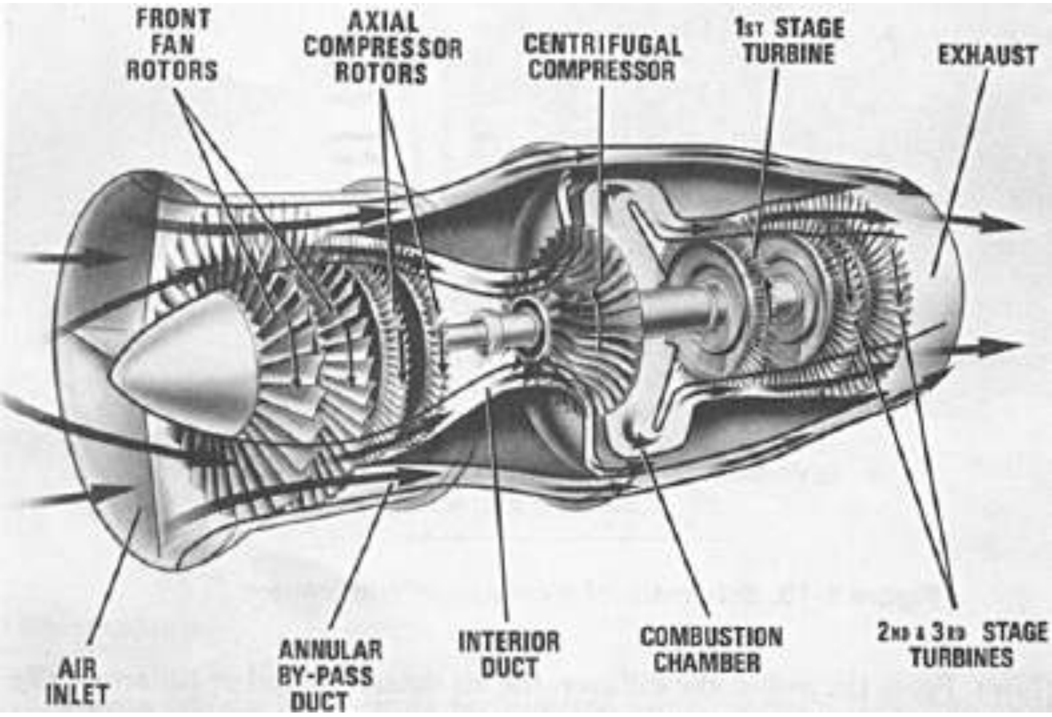
The compressor is responsible for providing the turbine with all the air it needs in an efficient manner. In addition, it must supply this air at high static pressures. The example of a large turboprop axial flow compressor will be used. The compressor is

Engine Temperature and Pressure Flow

assumed to contain fourteen stages of rotor blades and stator vanes. The overall pressure ratio (pressure at the back of the compressor compared to pressure at the front of the compressor) is approximately 9.5:1. At 100% (>13,000) RPM, the engine compresses approximately 433 cubic feet of air per second. At standard day air conditions, this equals approximately 33 pounds of air per second. The compressor also raises the temperature of the air by about 550 F as the air is compressed and moved rearward. The power required to drive a compressor of this size at maximum rated power is approximately 7000 horsepower.

In an axial flow compressor, each stage incrementally boosts the pressure from the previous stage. A single stage of compression consists of a set of rotor blades attached to a rotating disk, followed by stator vanes attached to a stationary ring. The flow area between the compressor blades is slightly divergent. Flow area between compressor vanes is also divergent, but more so than for the blades.

In general terms, the compressor rotor blades convert mechanical energy into gaseous energy. This energy conversion greatly increases total pressure (pt) Most of the increase is in the form of velocity (Pi), with a small increase in static pressure (Ps) due to the divergence of the blade flow paths.



The stator vanes slow the air by means of their divergent duct shape, converting 'the accelerated velocity (Pi) to higher static pressure (Ps). The vanes are positioned at an angle such that the exiting air is directed into the rotor blades of the next stage at the most efficient angle. This process is repeated fourteen times as the air flows from the first stage through the fourteenth stage.

In addition to the fourteen stages of blades and vanes, the compressor also incorporates the inlet guide vanes and the outlet guide vanes. These vanes, located at the inlet and the outlet of the compressor, are neither divergent nor convergent.

The inlet guide vanes direct air to the first stage compressor blades at the "best" angle. The outlet guide vanes "straighten" the air to provide the combustor with the proper airflow direction.

The efficiency of a compressor is primarily determined by the smoothness of the airflow. During design, every effort is made to keep the air flowing smoothly through the compressor to minimize airflow losses due to friction and turbulence. This task is a difficult one, since the air is forced to flow into ever-higher pressure zones.
Air has the natural tendency to flow toward low-pressure zones. If air were allowed to flow "backward" into the lower pressure zones, the efficiency of the compressor would decrease tremendously as the energy used to increase the pressure of the air was wasted. To prevent this from occurring, seals are incorporated at the base of each row of vanes to prevent air leakage. In addition, the tip clearances of the rotating blades are also kept at a minimum by the use of coating on the inner surface of the compressor case.
All components used in the flow path of the compressor are shaped in the form of airfoils to maintain the smoothest airflow possible. Just as is the case for the wings of an airplane, the angle at which the air flows across the airfoils is critical to performance. The blades and vanes of the compressor are positioned at the optimum angles to achieve the most efficient airflow at the compressor’s maximum rated speed.
Any deviation from the maximum rated speed changes the characteristics of the airflow within the compressor. The blades and vanes are no longer positioned at their optimum angles. Many engines use bleed valves to unload the force of excess air in the compressor when it operates at less than optimum speed. The example engine incorporates four bleed valves at each of the fifth and tenth compressor stages. They are open until 13,000 RPM (~94% maximum) is reached, and allow some of the compressed air to flow out to the atmosphere. This results in higher air velocities over the blade and vane airfoils, improving the airfoil angles. The potential for airfoil stalling is reduced, and compressor acceleration can be accomplished without surge.

Diffuser

Air leaves the compressor through exit guide vanes, which convert the radial component of the air flow out of the compressor to straight-line flow. The air then enters the diffuser section of the engine, which is a very divergent duct. The primary function of the diffuser structure is aerodynamic. The divergent duct shape converts most of the air's velocity (Pi) into static pressure (PS). As a result, the highest static pressure and lowest velocity in the entire engine is at the point of diffuser discharge and combustor inlet. Other aerodynamic design considerations that are important in the diffuser section arise from the need for a short flow path, uniform flow distribution, and low drag loss.

In addition to critical aerodynamic functions, the diffuser also provides:

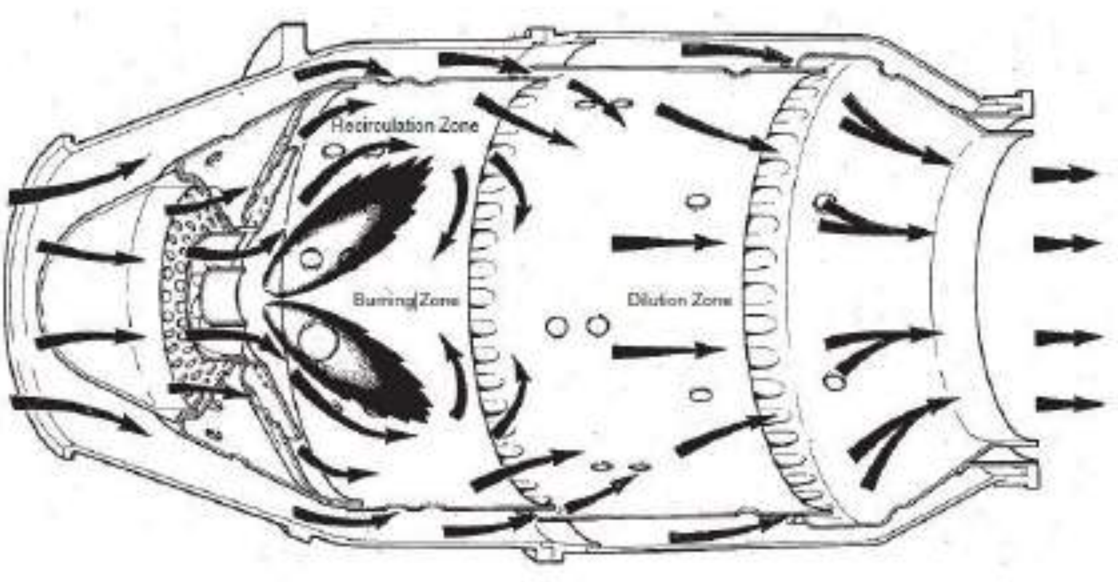
- Engine structural support, including engine mounting to the nacelle
- Support for the rear compressor bearings and seals
- Bleed air ports, which provide pressurized air for:
- airframe "customer" requirements (air conditioning, etc.)
- engine inlet anti-icing
- control of acceleration bleed air valves
- Pressure and scavenge oil passages for the rear compressor and front turbine bearings.

Combustor

Once the air flows through the diffuser, it enters the combustion section, also called the combustor. The combustion section has the difficult task of controlling the burning of large amounts of fuel and air. It must release the heat in a manner that the air is expanded and accelerated to give a smooth and stable stream of uniformly-heated gas at all starting and operating conditions. This task must be accomplished with minimum pressure loss and maximum heat release. In addition, the combustion liners must position and control the fire to prevent flame contact with any metal parts.

The engine in this example uses a can-annular combustion section. Six combustion liners (cans) are positioned within an annulus created by inner and outer combustion cases. Combustion takes place in the forward end or primary zone of the cans.

Primary air (amounting to about one fourth of the total engine's total airflow) is used to support the combustion process. The remaining air, referred to as secondary or dilution air, is admitted into the liners in a controlled manner. The secondary air



controls the flame pattern, cools the liner walls, dilutes the temperature of the core gasses, and provides mass. This cooling air is critical, as the flame temperature is above 1930□C (3500'F), which is higher than the metals in the engine can endure. It is important that the fuel nozzles and combustion liners control the burning and mixing of fuel and air under all conditions to avoid excess temperatures reaching the turbine or combustion cases. Maximum combustion section outlet temperature (turbine inlet temperature) in this engine is about 1070□C (>1950□F).

The rear third of the combustion liners is the transition section. The transition section has a very convergent duct shape, which begins accelerating the gas stream and reducing the static pressure in preparation for entrance to the turbine section.

Turbine

This example engine has a four-stage turbine. The turbine converts the gaseous energy of the air/burned fuel mixture out of the combustor into mechanical energy to drive the compressor, driven accessories, and, through a reduction gear, the propeller. The turbine converts gaseous energy into mechanical energy by expanding the hot, high-pressure gases to a lower temperature and pressure.

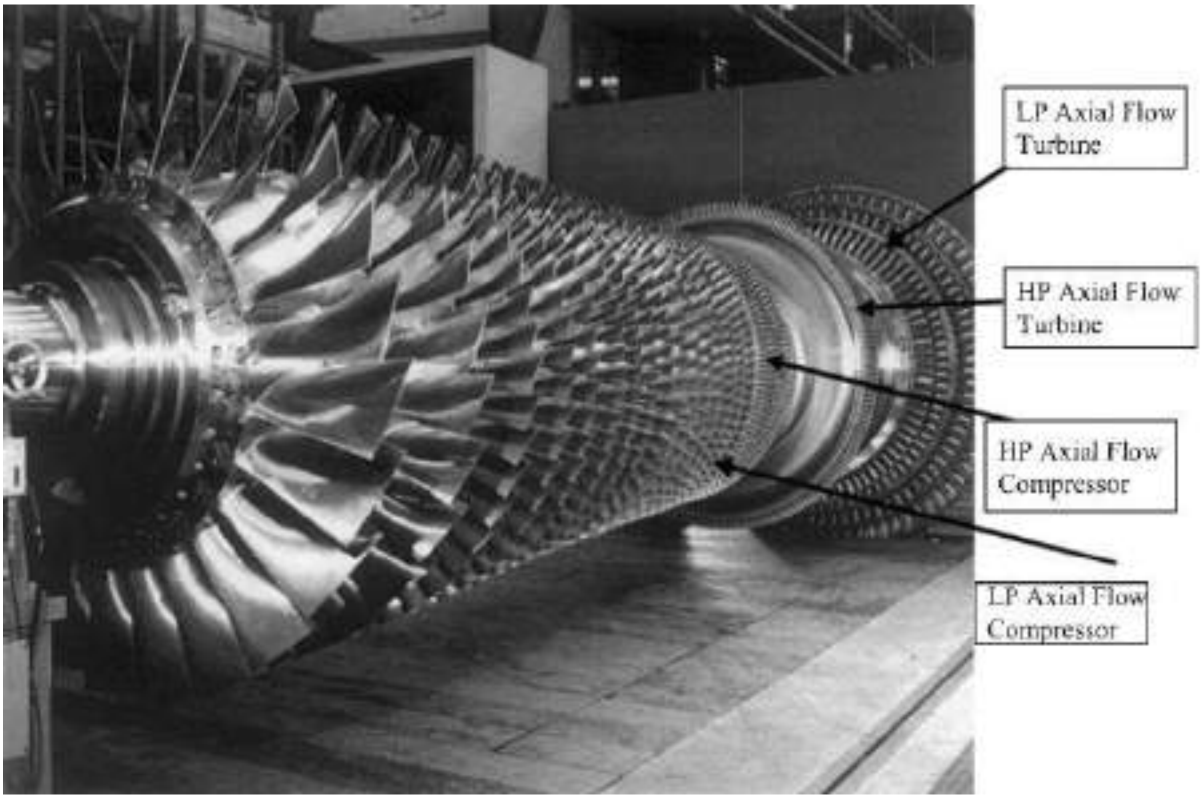
Each stage of the turbine consists of a row of stationary vanes followed by a row of rotating blades. This is the reverse of the order in the compressor. In the compressor, energy is added to the gas by the rotor blades, then converted to static pressure by the stator vanes. In the turbine, the stator vanes increase gas velocity, and then the rotor blades extract energy.

The vanes and blades are airfoils that provide for a smooth flow of the gases. As the airstream enters the turbine section from the combustion section, it is accelerated through the first stage stator vanes. The stator vanes (also called nozzles) form convergent ducts that convert the gaseous heat and pressure energy into higher velocity gas flow (Pi). In addition to accelerating the gas, the vanes "turn" the flow to direct it into the rotor blades at the optimum angle.

As the mass of the high velocity gas flows across the turbine blades, the gaseous energy is converted to mechanical energy. Velocity, temperature, and pressure of the gas are sacrificed in order to rotate the turbine to generate shaft power.

The efficiency of the turbine is determined by how well it extracts mechanical energy from the hot, high-velocity gasses. Since air flows from a high-pressure zone to a low pressure zone, this task is accomplished fairly easily. The use of properly positioned airfoils allows a smooth flow and expansion of gases through the blades and vanes of the turbine.

All the air must flow across the airfoils to achieve maximum efficiency in the turbine. In order to ensure this, seals are used at the base of the vanes to minimize gas flow around the vanes instead of through the intended gas path. In addition, the first three stages of the turbine blades have tip shrouds to minimize gas flow around the blade tips.



Exhaust

After the gas has passed through the turbine, it is discharged through the exhaust. Though most of the gaseous energy is converted to mechanical energy by the turbine, a significant amount of power remains in the exhaust gas. This gas energy is accelerated through the convergent duct shape of the exhaust to make it more useful as jet thrust - the principle of equal and opposite reaction means that the force of the exhausted air drives the airplane forward.

Turbojet Engine

4.2.2 THERMODYNAMIC ANALYSIS

As described earlier, single-spool turbojet may have either one or two compressors as well as a single driving turbine. It may or may not have an afterburner. Figure 4.3 illustrates a single-spool turbojet engine having a single compressor and an afterburner together with designations for each state. The different processes that are encountered within the engine are described here.

(a)–(1): The air flows from far upstream, where the velocity of air relative to engine is the flight velocity up to the intake, usually with some deceleration during cruise and acceleration during takeoff.

(2)–(3): The air flows through the inlet diffuser and ducting system, where the air velocity is decreased as the air is carried to the compressor inlet.

(2)–(3): The air is compressed in a dynamic compressor.

(3)–(4): The air is “heated” by mixing and burning of fuel in the air.

(4)–(5): The air is expanded through a turbine to obtain power to drive the compressor.

(5)–(6): The air may or may not be further heated by the addition and burning of fuel in an afterburner.

(6)–(7): The air is accelerated and exhausted through the exhaust nozzle.

If the engine is not fitted with an afterburner, then states 5 and 6 are coincident. It will be simpler not to discard point 6 in the following analysis. Thus, the flow in the nozzle remains from points 6 to 7. The amount of mass flow is usually set by flow choking in the nozzle throat.

4.2.3 IDEAL CASE

The components except the burners are assumed to be reversible adiabatic or isentropic. Moreover,
the burners are replaced by frictionless heaters; thus, the velocities at stations 2 through 6 are negligible.

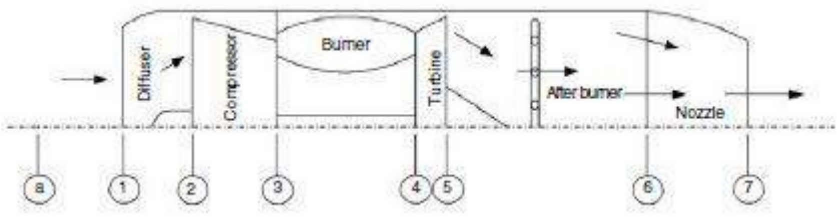


FIGURE 4.3 Single-spool afterburning turbojet.

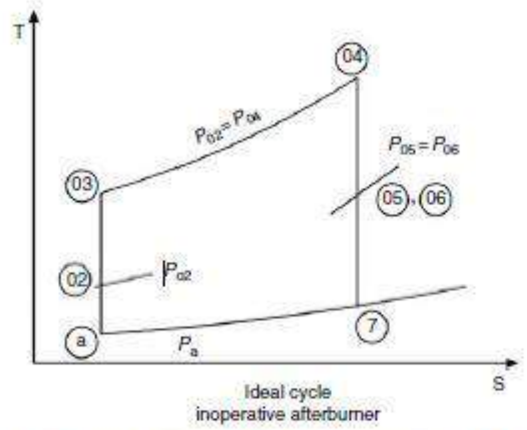


FIGURE 4.4 Temperature–entropy diagram for ideal turbojet, inoperative afterburner.

1 History and Classifications of Aero Engines

Humans have long dreamt of flying. Various ancient and medieval people who fashioned wings in order to fly met with disastrous consequences in leaping from towers or roofs and flapping their wings vigorously. The Greek myth of Daedalus and his son Icarus who were imprisoned on the island of Crete and tried to escape by fastening wings with wax and flying through the air is also well known. The dream of the humans to fly was achieved only in the twentieth century—at about 10:35 a.m. on Tuesday, December 17, 1903—when Orville Wright made the first successful flight in Kitty Hawk, North Carolina. That flying machine, identified as the Wright Flyer I, was the first heavier-than-air flyer designed and flown by the Wright brothers, Wilbur (1867–1912) and Orville (1871–1948). The Wright brothers, who are the inventors of the first practical airplane, are the premier aeronautical engineers in the history of mankind. A comparison of the Wright Flyer I and the aircraft of the twenty-first century, such as the Boeing 787 and Airbus 350, outlines the miracles that have taken place in the aviation industry. The Wright Flyer I did not have any fuselage and the only position the pilot could fly in was probably by lying prone on the bottom wing. Nevertheless, the effort marked the beginning of human-controlled powered flight. Several years lapsed before the design of the conventional aircraft with a closed fuselage installed to wings, a tail unit, and having an undercarriage or landing gears. Tremendous development in aviation industry now allows passengers in civil aircraft travel in air-conditioned compartments of the fuselage, in comfortable seats, and to eat, and watch video movies up in the air. However, it was a long story how people could be convinced to use aircraft as one of the modes of transportation.

Some milestones in such a long journey spanning nearly one century may be mentioned briefly.

For several decades, piston engine coupled with propellers provided the necessary power for early aircraft. The turbojet engines (the first jet engines) invented independently by Sir Frank Whittle in Britain and Dr. von Ohain in Germany powered aircraft from the early 1940s. Such jet engines paved the way to the now, highly sophisticated military and comfortable civil aircraft. During the middle of the twentieth century, airlines relied upon low-speed subsonic aircraft whose flight speeds were less than 250 miles per hour (mph) powered by turbojet and/or turboprop engines. In the late 1960s and early 1970s, the wide-bodied aircraft (Boeing 747, DC-10 and AirbusA300) powered by turbofan engines flew at transonic speeds, that is, at speeds less than 600 mph. Even now civilian aircraft fly at the same transonic speeds. On the other hand, military airplanes, fighter airplanes, for example, fly at supersonic speeds, that is at speeds less than 1500 mph. Such fighter planes are fitted with turbofan engines that have afterburners. X-planes, which are hypersonic, aircraft fitted with scramjet/rocket engines, fly at speeds less than 6000 mph. Space shuttles, which also have rocket engines, fly at hypersonic speeds of less than 17,500 mph.

It is interesting to compare the flight time between popular destinations such as Los Angeles and Tokyo on different airplanes—9.6 h for Boeing 747, 6.2 h for Concorde, and only 2 h for hypersonic aircraft [1]. It may be stated here that given humankind's endless ambitions, it is difficult to anticipate the shape, speed, and the fuel of the flying machines even for the next few decades.

It is a fact that evolution of aero-vehicles and aero-propulsion are closely linked. Unlike the eternal question of the chicken and the egg, there is no doubt as to which came first. The lightweight and powerful engine enabled humans to design the appropriate vehicle structure for both civil and military aircraft. Owing to the interdependence of the performance characteristics of aero-vehicles (including aircraft, missiles, airships, and balloons) and their aero-propulsion system, the evolution review of historical inventions will be divided into two phases. The first of these phases is related to the invention of prejet engines, while the second describes the invention and development of jet engines.

1.1 PREJET ENGINES—HISTORY

This section gives a brief description of the long history of flight events. It starts with some activities around the year 250 BC and ends just before the invention of jet engines in 1930s. Activities related to unpowered flight machines and important patents will be described first. Next, powered flights employing internal combustion gasoline engines will be described

.

1.1.1 Early Activities in Egypt and China

Jet propulsion is based on the reaction principle that governs the motion or flight of both aircraft and missiles. Though such a principle was one of the three famous laws of motion stated by Sir Isaac Newton in 1687, ancient Egyptians and Chinese utilized this principle several hundred years before him. The first known reaction engine was built by a noted Egyptian mathematician and inventor, Hero (sometimes called Heron) of Alexandria sometime around the year 250 BC [2]; some references go back to 150 BC [3]. Hero called his device aeolipile; see Figure 1.1. It consisted of a boiler or bowl, which held a supply of water. Two hollow tubes extended up from the boiler and supported a hollow sphere, which was free to turn on these supports. The steam escaped from two bent tubes mounted opposite one another on the surface of a sphere. The force created by the escaping steam transformed the nozzles into jet nozzles caused the sphere to rotate about its axis. It is said that Hero attached a pulley, ropes, and linkages to the axle on which the sphere rotated to use the aeolipile to pull open the temple doors without the aid of any visible power. Further details can be found in the website [4].

The Chinese discovered gunpowder around AD 1000. Some inventive person probably knew that a cylinder filled with gunpowder and open at one end would dart across a surface when the gun powder was ignited. Such a discovery was automatically employed in the Chinese battles by tying tiny cylinders filled with gunpowder to arrows. Thus these arrows would rocket into the air when ignited (Figure 1.2). The records of a battle that took place in China in AD 1232 provides evidence that solid rockets were used as weapons. These early Chinese scientists were the first people

Pulsejet and Ramjet Engines

3.1 INTRODUCTION

The two types of engines treated in this chapter are air-breathing engines of the athodyd type. This abbreviation stands for aero-thermodynamic duct. Thus, these engines have no major rotating parts (fan/compressor or turbine) and consist of an entry duct, a combustion chamber, and a nozzle [1]. The first engine that will be described and analyzed is the pulsejet engine. Pulsejet engine operates intermittently and has found limited applications [2]. The reasons are the difficulty of its integration into manned aircraft as well as its high fuel consumption, poor efficiency, severe vibration, and high noise. The second engine described here will be the ramjet engine. The ramjet engine is appropriate for supersonic flight speeds [3], where the ram compression of the air becomes sufficient to overcome the need for mechanical compression achieved normally by compressors or fans in other jet engines.
If the flight speed is so high that fuel combustion must occur supersonically, then this ramjet is called a scramjet [4].

3.2 PULSEJET ENGINES

3.2.1 INTRODUCTION

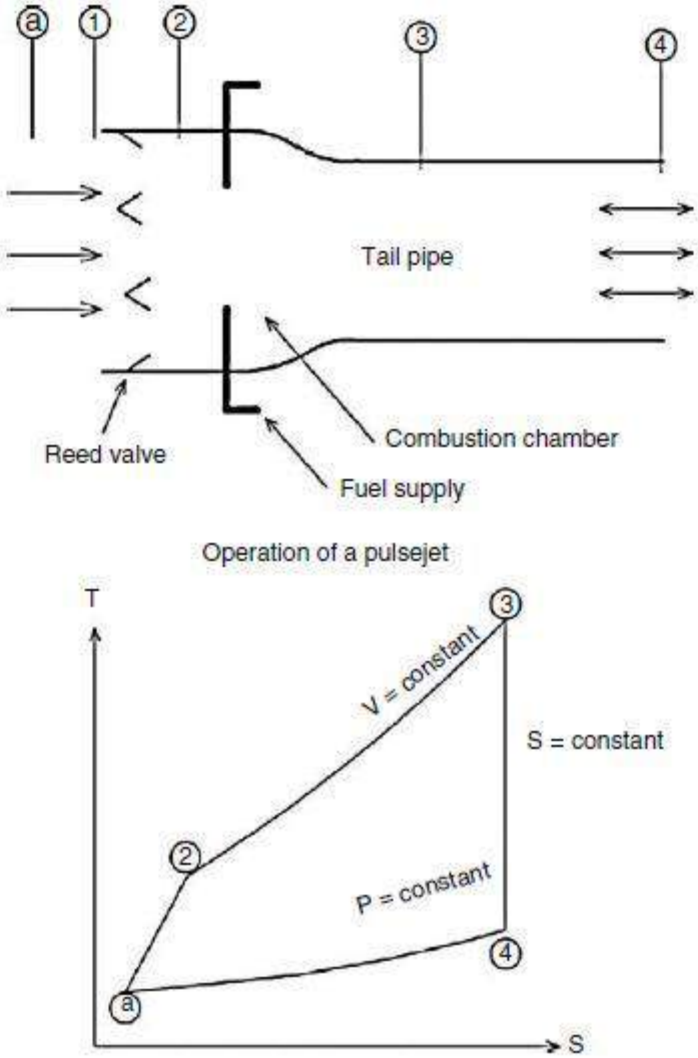
Pulsejet engine is a very simple jet engine, which comprises an air intake, a combustion chamber, and an acoustically resonant exhaust pipe. It is a jet engine in which the intake of air is intermittent; thus, combustion occurs in pulses, resulting in a pulsating thrust rather than a continuous one. Presently, there are two types of pulsejets, namely, *valved and valveless* jet engines.Athird future type known as pulse detonation engine (PDE) is in the research and testing phase of production. The valved type is fitted with a one-way valve while a valveless engine has no mechanical valves at its intake. The PDE is a concept currently in active development to create a jet engine that operates on the supersonic detonation of fuel. Historically, Martin Wiberg (1826–1905) developed the first pulsejet in Sweden. The first type—valved or traditional pulsejet—was used to power a German cruise missile called the Vergeltungswaffe 1 (Vengeance 1), or V1, and normally referred to as the V-1 flying bomb (refer to Figure 1.38). It was extensively used in bombing England and Belgium during World War II. The engines made a distinctive sound, leading the English to call them “buzz bombs.” The pulsejet engine is extremely simple, cheap, and easy to construct. However, it has low reliability, poor fuel economy, and very high noise levels. The high noise levels make them impractical for applications other than the military and similar restricted applications. Pulsejets have been used to power experimental helicopters, the engines being attached to the extreme ends of the rotor blades. They have also been used in both tethered and radio-control model aircraft. The speed record for tethered model aircraft is 186 miles per hour (299 km/h), set in the early 1950s. The valved-type pulsejet engine has a set of one-way valves (or check valves) through which the incoming air passes. The valving is accomplished through the use of reed valves that consist of thin flexible metal or fiberglass strips fixed on one end that open and close upon changing pressures across opposite sides of the valve much like heart valves do. Fuel in the form of a gas or liquid aerosol is either mixed with the air in the intake or injected into the combustion chamber. When the air–fuel mixture is ignited, these valves slam shut, which means that the hot gases can only leave through the engine's tailpipe, thus creating forward thrust. The cycle frequency is primarily dependent onthe length of the engine. For a small model-type engine the frequency may be typically around 250 pulses per second, whereas for a larger engine such as the one used on the German V1 flying bomb, the frequency was closer to 45 pulses per second. Once the engine is running it requires only an input of fuel, but it usually requires forced air and an ignition method for the fuel–air mix. Once running, the engine is self-sustaining. The main drawback of valved type is that the valves require regular replacement and represent a weak link in the engine's reliability. The pulsejet powered V1 "flying bombs" were only good for about 20–30 min of continuous operation. So, by eliminating the valves, it should be possible to make an engine that is as simple as a ramjet having absolutely no moving parts. Thus, a *valveless* type is employed.

The name valveless is really a misnomer. These engines have no mechanical valves, but they do have aerodynamic valves, which, for the most part, restrict the flow of gases to a single direction just as their mechanical counterparts. Thus, this type of pulsejet has no mechanically moving parts at all and in that respect is similar to a ramjet.

3.2.2 VALVED PULSEJET

Figure 3.1 illustrates the simple construction of valved type together with its cycle on the temperature–

entropy (T–S) diagram.



Valved pulsejet engine.

FUNDAMENTALS OF GAS TURBINE

INTRODUCTION

The jet engine is an internal combustion engine that uses air as the working fluid. The engine extracts chemical energy from fuel and converts it to mechanical energy using the gaseous energy of the working fluid (air) to drive the engine and propeller, which, in turn, propel the airplane.

THE GAS TURBINE CYCLE

The basic principle of the airplane turbine engine is identical to any and all engines that extract energy from chemical fuel. The basic 4 steps for any internal combustion engine are:

1. Intake of air (and possibly fuel).
2. Compression of the air (and possibly fuel).
3. Combustion, where fuel is injected (if it was not drawn in with the intake air) and burned to convert the stored energy.
4. Expansion and exhaust, where the converted energy is put to use.

In the case of a piston engine, such as the engine in a car or reciprocating airplane engine, the intake, compression, combustion, and exhaust steps occur in the same place (cylinder head) at different times as the piston goes up and down.

In the turbine engine, however, these same four steps occur at the same time but in different places. As a result of this fundamental difference, the turbine has engine sections called:

1. The inlet section
2. The compressor section
3. The combustion section (the combustor)
4. The turbine (and exhaust) section.

The turbine section of the gas turbine engine has the task of producing usable output shaft power to drive the propeller. In addition, it must also provide power to drive the compressor and all engine accessories. It does this by expanding the high temperature, pressure, and velocity gas and converting the gaseous energy to mechanical energy in the form of shaft power.

A large mass of air must be supplied to the turbine in order to produce the necessary power. This mass of air is supplied by the compressor, which draws the air into the engine and squeezes it to provide high-pressure air to the turbine. The compressor does this by converting mechanical energy from the turbine to gaseous energy in the

form of pressure and temperature.

If the compressor and the turbine were 100% efficient, the compressor would supply all the air needed by the turbine. At the same time, the turbine would supply the necessary power to drive the compressor. In this case, a perpetual motion machine would exist. However, frictional losses and mechanical system inefficiencies do not allow a perpetual motion machine to operate. Additional energy must be added to the air to accommodate for these losses. Power output is also desired from the engine (beyond simply driving the compressor); thus, even more energy must be added to the air to produce this excess power. Energy addition to the system is accomplished in the combustor. Chemical energy from fuel as it is burned is converted to gaseous energy in the form of high temperatures and high velocity as the air passes through the combustor. The gaseous energy is converted back to mechanical energy in the turbine, providing power to drive the compressor and the output shaft.

SOME BASIC PRINCIPLES

As air passes through a gas turbine engine, aerodynamic and energy requirements

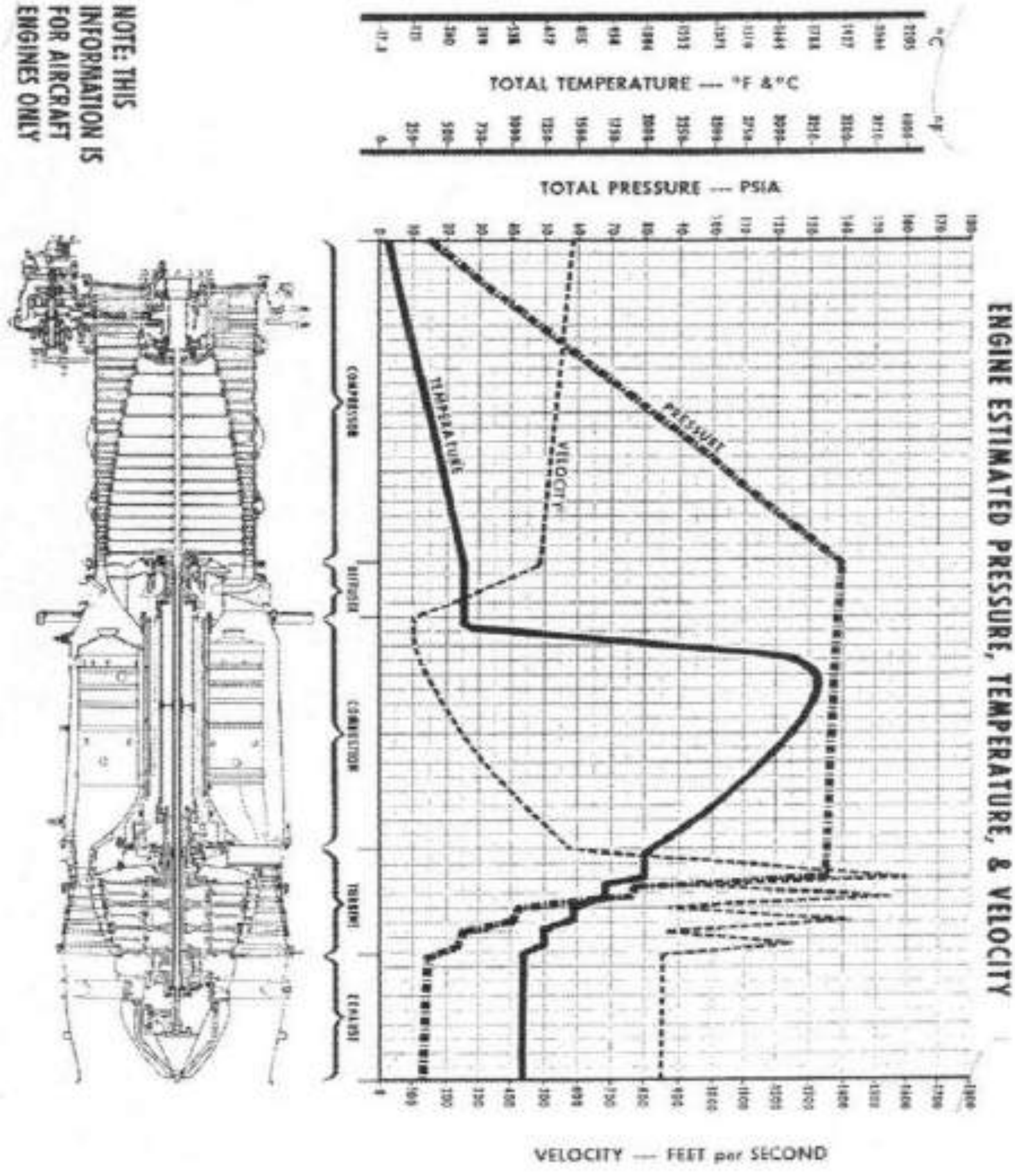
PERFORMANCE AND EFFICIENCY

The type of operation for which the engine is designed dictates the performance requirement of a gas turbine engine. The performance requirement is mainly determined by the amount of shaft horse power (s.h.p.) the engine develops for a given set of conditions.

The need for high efficiency in the engine becomes more important as fuels become more costly. Engine efficiency is primarily defined by the specific fuel consumption (s.f.c.) of the engine at a given set of conditions.

Many factors affect both the efficiency and the performance of the engine. The mass flow rate of air through the engine will dictate engine performance. Any restrictions acting against the smooth flow of air through the engine will limit the engine's performance. The pressure ratio of the compressor, the engine operating temperatures (turbine inlet temperature), and the individual component efficiencies will also influence both the performance and the efficiency of the overall engine. All these factors are considered during the design of the engine. An optimum pressure ratio, turbine inlet temperature, and air mass flow rate are selected to obtain the required performance in the most efficient manner. In addition, individual engine components are designed to minimize flow losses to maximize component efficiencies.

the following graphic shows the typical temperature and pressure rise through the gas



Engine tempreture and pressure flow

: The cycle can be described briefly as follows

The air is sucked into the combustion chamber through a bank of spring-loaded check .1

reed) valves by the vacuum created by the exhaust of the previous cycle. These valves)

are normally closed, but if a predetermined pressure differential exists, they will open to

.permit high-pressure air from the diffusing section to pass into the combustion chamber

.They never permit flow from the combustion chamber back into the diffuser

A spark plug initiates the combustion process, which occurs at something approaching a .2

constant volume process [5]. Such combustion occurs in the form of an explosion, which

raises the pressure in the combustion chamber to a high level and closes the spring valve

.at the intake

The resultant high pressure and temperature force the gases to flow out of the tail pipe at .3

.high velocity

At the end of the discharge, the inertia of the gas creates a vacuum in the combustion chamber

that together with the ram pressure developed in the diffuser causes sufficient pressure differential to

open the check valves.Anew charge of air enters the chamber and a new cycle starts. The frequency

of the cycle depends on the size of the engine and the dynamic characteristics of the valves must be

matched carefully to this frequency. Small-size engines operate at frequencies as high as 300–400

.cycles per second, and large engines operate at frequencies as low as 40 cycles per second

To start the engine, a carefully adjusted amount of fuel is sprayed into the cold combustion chamber in order to create a mixture and the resultant strong explosion starts the cycle. The fuel

flow, injected directly into the combustion chamber, is continuous throughout the cycle with some

variation in fuelflowresulting from the pressure in the combustion chamber. This fuelflowfluctuation

.can be neglected

Intake or diffuser .1

This process occurs from state (a) to state (2) as shown in Figure 3.1. State (a) is far

upstream, while state (1) is at the pulsejet inlet and state (2) is just aft of the check

valve upstream of the combustion chamber. Owing to ram effect, both the pressure and

temperature rise from the ambient conditions to the values to be calculated hereafter. The

total conditions at state (1) are given by the relations

$$P_{01} = P_{0a} = P_a \left(1 + \frac{\gamma_c - 1}{2} M^2\right)^{\frac{\gamma_c}{\gamma_c - 1}} \tag{3.1}$$

Here M is the flight Mach number and γ_c is the specific heat ratio for cold air. The stagnation temperatures for states (a), (1), (2) are also equal

$$T_{01} = T_{0a} = T_a \left(1 + \frac{\gamma_c - 1}{2} M^2\right) \tag{3.2}$$

If it is assumed that the diffuser is an ideal one, then the total pressure and temperature at state (2) is equal to those at state (1), or

$$P_{02} = P_{01} \tag{3.3}$$

and

$$T_{02} = T_{01} \tag{3.4}$$

:

VALVELESS PULSEJET

Valveless pulsejet engine is sometimes identified as acoustic jet engine [6]. This idea was the brainchild of the French propulsion research group SNECMA. They developed these valveless pulsejets in the late forties for use in drones. One application was the DutchAT-21 target drone built byAviolanda Aircraft from 1954 to 1958. The main design difficulties encountered for valved pulsejets are difficult-to-resolve wear issues. Thus, a valveless pulsejet (or pulsejet) is designed to replace valved type.Valveless pulsejets are low in cost, lightweight, powerful, and easy to operate. They have all the advantages and most of the disadvantages of conventional valved pulsejets. Fuel consumption is excessively high and the noise level is unacceptable as per recent standards. However, they do not have the troublesome reed valves that need frequent replacement. They can operate for their entire useful life with practically zero maintenance. They have been used to power model aircraft, experimental go-karts, and even some unmanned military aircraft such as cruise missiles and target drones. These engines have no mechanical valves, but they do have aerodynamic valves that, for the most part, restrict the flow of gases to a single direction just as their mechanical counterparts. Thus, the intake and exhaust pipes usually face the same direction. This necessitates bending the engine into a “U” shape as shown in Figure 3.2 or placing a 180 ◦ bend in the intake tube. The Lockwood–Hille is an example for the “U” shape design.

When the air–fuel mixture inside the engine ignites, hot gases will rush out through both the intake tube and the exhaust tube, since the aerodynamic valve “leak.” If both tubes were not facing in the same direction, less thrust would be generated because the reactions from the intake and exhaust gas flows would partially cancel each other

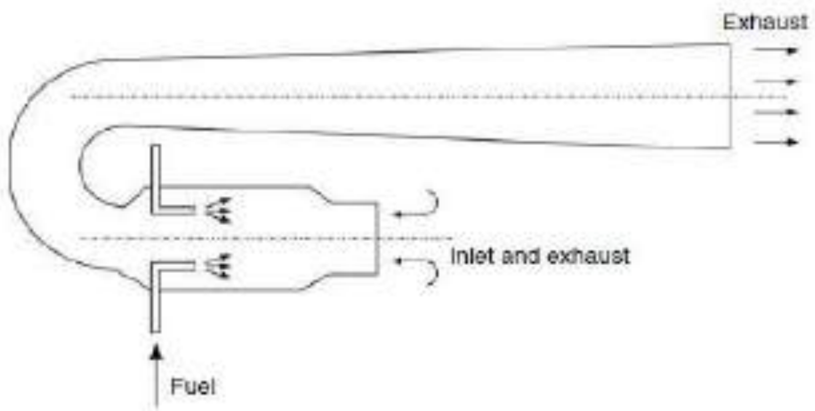


FIGURE 3.2 Valveless pulsejet engine.

In this type of pulsejet, combustion process generates two shock wave fronts, and one travels down each tube. By properly “tuning” the system, a stable, resonating combustion process can be
achieved, which yields a considerable thrust.
Successful valveless pulsejets have been built from a few centimeters in length to huge sizes, though the largest and smallest have not been used for propulsion. The smallest ones are only successful
when extremely fast-burning fuels such as acetylene or hydrogen, for example, are employed. Medium- and larger-sized engines can be made to burn almost any flammable material that can be
delivered uniformly to the combustion zone, though of course volatile flammable liquids (gasoline,
kerosene, various alcohols) and standard fuel gases (propane, butane, MAPP gas) are easiest to use. Because of the deflagrating nature of pulsejet combustion, these engines are extremely efficient combustors, producing practically no hazardous pollutants, even when using hydrocarbon fuels. Wit modern high temperature metals for the main structure, engine weight can be kept extremely low. Untill now, the physical size of successful valveless designs has always been somewhat larger than valved engines for the same thrust value, though this is theoretically not a requirement. An ignition system of some sort is required for engine startup. In the smallest sizes, forced air at the intake is also typically needed for startup. There is still much room for improvement in the development of really efficient, fully practical designs for propulsion uses.

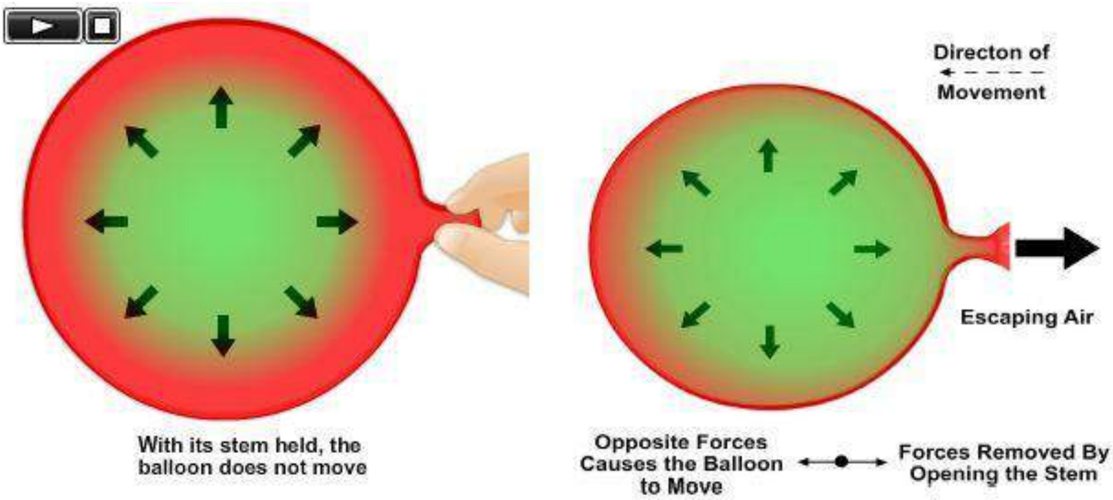
PULSE DETONATION ENGINE

The pulse detonation engine (PDE) marks a new approach toward noncontinuous combustion in jet engines and promises higher fuel efficiency compared even to turbofan jet engines. With the aid of the latest design techniques and high pulse frequencies the drawbacks of the early designs can be overcome. To date no practical PDE engine has been put into production, but several test bed engines have been built by Pratt & Whitney and General Electric, which have proven the basic concept. Extensive research work is also carried out in different NASAcenters. In theory, the design can produce an engine with the efficiency far
These systems should be put to use in .surpassing gas turbine with almost no moving parts
All regular jet engines operate on the deflagration of fuel, that is, the rapid but .the near futu
subsonic combustion

of fuel. The PDE is a concept currently in active development to create a jet engine that PDEs are an extension of pulsejet engines. .operates on the supersonic detonation of fuel They share many similarities. However, there is one important difference between them: PDEs detonate, rather than deflagrate, their fuel. Detonation of fuel is a supersonic combustion of fuel that results in immense pressure, which in turn is used as thrust. These combustors have an advantage over traditional near-constant pressure combustors in being more thermodynamically efficient by approximating constant volume combustion. However,

BASIC THEORY OF JET PROPULSION

Jet propulsion is the propelling force generated in the direction opposite to the flow of a mass of gas or liquid under pressure. The mass escapes through a hole or opening called a jet nozzle. A familiar example is the nozzle at the end of a fire hose. The nozzle forms a smaller passageway through which the water must flow. The nozzle increases the velocity of the water, giving the term, “a jet of water.” Another example of the theory of jet propulsion is an inflated balloon. With the opening in the balloon closed (*Figure1-1*), there is no action because the pressure of the gas inside the balloon is equal in all directions. When you allow the opening to release the air *(Figure 1-2*), the balloon moves. Its movements appear to be in all directions. However, it is always moving in the opposite direction from the open end where the air is exiting



Look at the balloon example from the mechanic’s point of view. Igniting a hydrocarbon fuel (a compound containing only hydrogen and carbon) and oxygen in a closed container *(Figure 1-3)* releases heat. The burning fuel causes the trapped gases to expand rapidly. The expansion occurs equally in all directions, the force of the pressure is balanced, and the container does not move.

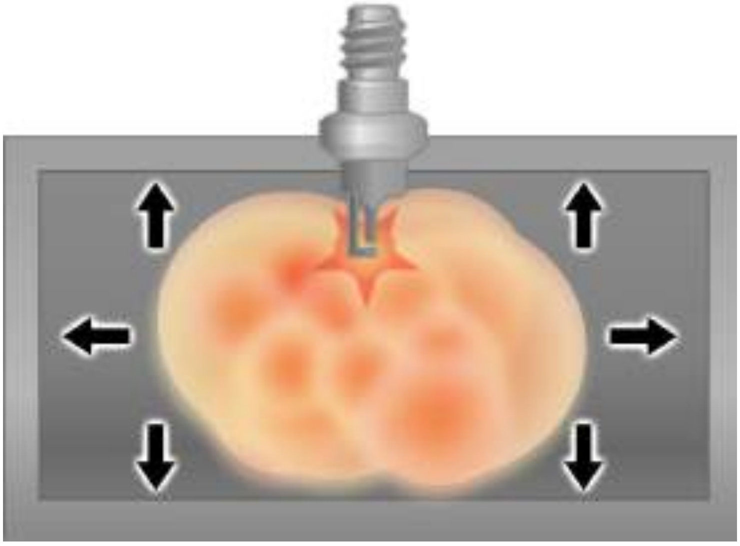
Rocket

When combustion takes place in an open container, the expanding gases rush out the opening at a high velocity (*Figure 1-4).* The release of internal pressure at the nozzle end of the container leaves an unbalanced pressure at the other end. The released pressure propels the container (a rocket) in the direction opposite of the exhaust gases. Obviously, propulsion depends solely upon internal conditions. The container does not “push against” external air. In fact, a complete vacuum would produce greater force, the basic operating principle for all jets. The rocket (propulsion unit) is one of the four main classes of jet engines. Before we continue on to the physical principles of jet engines, we will review the three other types of jet engines.

The Ramjet

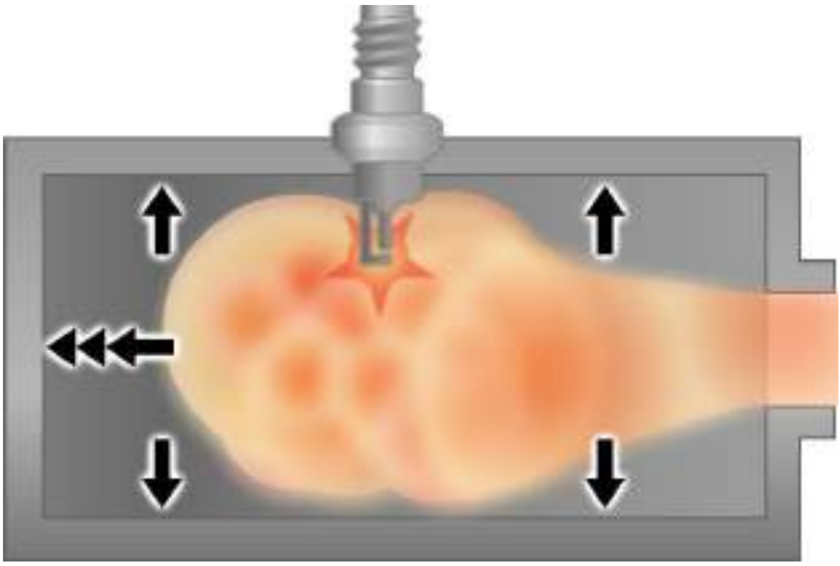
Suppose you attach a plain cylinder with open ends under the wing of an aircraft flying at high speed. Air enters the front of the duct and leaves at the rear. Nothing increases the force of flow through the duct. There is a loss of energy because of skin friction and airflow disturbances at the entrance and exit. If you add heat energy to the air as it passes through the duct, the air would expand and increase the jet velocity. *(Figure 1-5).* The amount of heat you can add is largely dependent upon the pressure of the air treated. A simple method of raising the pressure is to pass the air through a DIVERGENT entry nozzle. A divergent entry nozzle converts gaseous energy from velocity to pressure and temperature. This also provides a forward pressure wall for the jet to react. A CONVERGENT exit nozzle converts gaseous energy from pressure and temperature to velocity. The simple gas unit *(Figure 1-6)* created has little practical use because of the following

:



- Air compression depends solely on "ram effect."
- A limited amount of heat is added.
- Considerable heat is lost by radiation.

The next step is to improve the method of adding heat, through internal combustion*. Figure1-7* shows a divergent-convergent duct. Fuel is injected and burned, releasing heat directly into the airstream. This simple "Aero Thermo Dynamic Duct" (ATHODYD) or RAMJET is used in remotely piloted vehicle (RPV), cruise missiles, and National Aeronautics and Space Administration (NASA) on their X-15 and X-34 projects



Performance Parameters of

Jet Engines

2.1 INTRODUCTION

The designer of an aircraft engine must recognize the differing requirements for takeoff, climb, cruise and maneuvering, the relative importance of these being different for civil and military applications and for long- and short-haul aircraft. In the early aircraft, it was common practice to focus on the take-off thrust. This is no longer adequate for later and present day aircraft. For long-range civil aircraft such as Boeing 747, 777, and Airbus A340 as well as the future Boeing 787, Airbus A380 (the world's truly double-deck airliner), andA350XWB(extra wide body expected in 2012), the fuel consumption through some ten or more flight hours is the dominant parameter. Military aircraft have numerous criteria such as the rate of climb, maneuverability for fighters, short takeoff distance for aircraft operating from air carriers and maximum ceilings for high-altitude reconnaissance aircraft such as SR-71 Blackbird aircraft. For civil and military freighter airplanes the maximum payload is the main requirement.

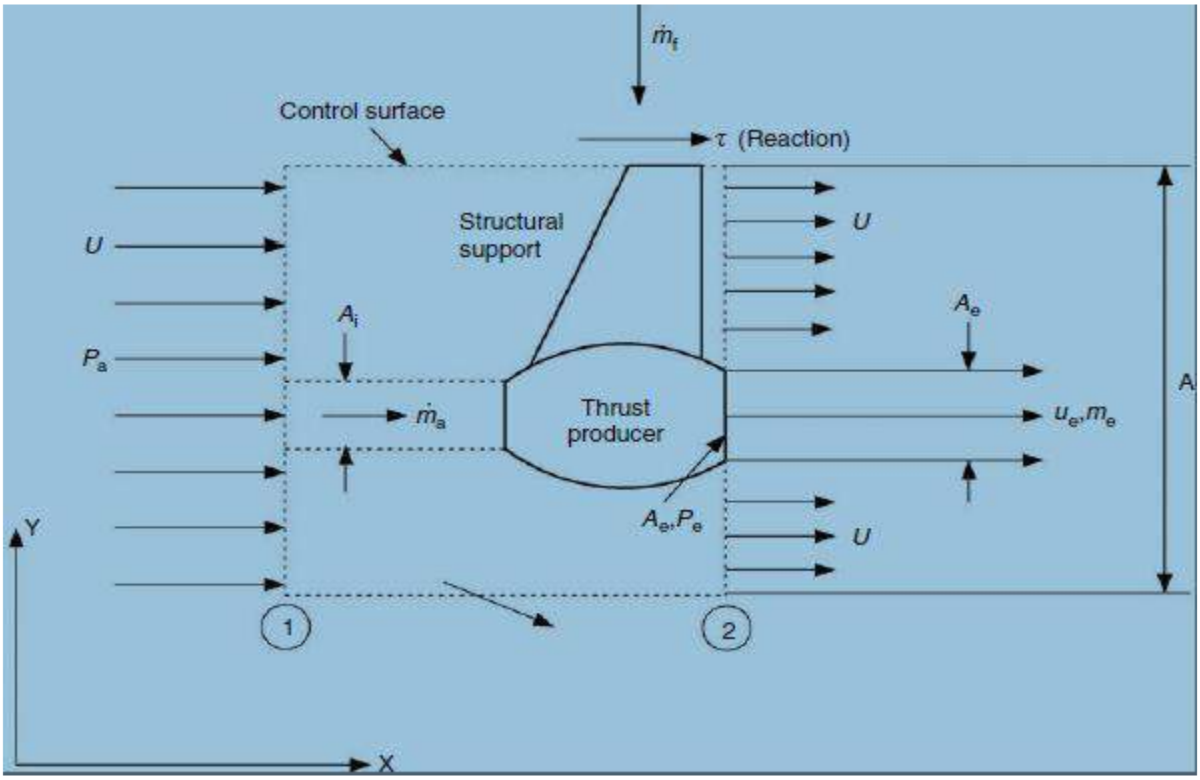
In all types of aircraft, the engines are expected to provide efficiently the thrust force necessary for their propelling during different flight phases and at different operating conditions, including hottest or coldest ambient temperatures and rainy, windy, or snowing weather.

This chapter resembles a first window for air-breathing engines. It starts by a derivation for the thrust force or the propelling force generated in the direction opposite to the flow of air entering the engine in accordance with Sir Isaac Newton's laws of motion. Consequently, all jet engines including rocket motors belong to the class of power plants called *reaction engines*. It is the internal imbalance of forces within the gas turbine engines that gives all reaction engines their names. The propulsive force developed by a jet engine is the result of a complex series of actions and reactions that occur within the engine. The thrust constituents and the different factors affecting the thrust are next explained. Some of these factors are related to the engine; others are related to the medium in which the engine operates.

The performance of jet engines is evaluated through the propulsive, thermal, and overall efficiencies. The propeller efficiency of turboprop engines is also evaluated. Fuel consumption is properly evaluated through a parameter identified as the thrust-specific fuel consumption (TSFC), which is the ratio of fuel flow rate into the engine to the generated thrust. Thus, different jet engines may be compared. The range of aircraft is a combined engine/aircraft parameter where the fuel consumption through the engine is coupled to the aircraft's lift and drag forces.

2.2 THRUST FORCE

Thrust force is the force responsible for propelling the aircraft in its different flight regimes.
It is in addition to the lift, drag, and weight that together represent the four forces that govern the aircraft motion. During the cruise phase of flight, where the aircraft is flying steadily at a constant speed and altitude, the four forces are in equilibrium in pairs—lift and weight as well as thrust and drag. During landing, thrust force is either fully or partially used in braking the aircraft through a thrust-reversing mechanism. The basic conservation laws of mass and momentum are used in their integral forms to derive an expression for thrust force.



Consider a schematic diagram (Figure 2.1) for an engine with a part of its pod installation (i.e., a structural support for hanging the engine to the wing). Next define a control volume which control surface passes through the engine outlet (exhaust) plane (2) and extends far upstream at (1). The two side faces of the control volume are parallel to the flight velocity u. The upper surface side cuts the structural support while the lower one is far from the engine. The surface area at planes (1) and (2) are equal and denoted by A. The stream tube of air entering the engine has an area A_i at plane (1), while the exhaust area for gases leaving the engine is A_e. The velocity and pressure over plane (1) are u (which is the flight speed) and P_a (ambient pressure at this altitude). Over plane (2) the velocity and pressure are still u and P_a except over the exhaust area of the engine A_e where the values will be u_e and P_e. The x and y directions employed here are chosen parallel and normal to the centerline of the engine.

The following assumptions are made:

1. The flow is steady within the control volume; thus all the properties within the control do not change with time.
2. The external flow is reversible; thus the pressures and velocities are constant over the control surface except over the exhaust area P_e of the engine.

Conservation of mass across the engine gives

The Pulsejet Engine
The “intermittent impulse” jet engine *(Figure1-8)*, known as the aero-pulse or pulsejet, improves compression by sacrificing the principle of continual power generation. The pulsejet is like the ramjet, but with a series of non-return shutter valves. Fuel injection nozzles located just aft of the shutter valves provide fuel. As the engine travels through the air, pressure on the nose opens the valve and rams air into the duct, mixing air with fuel. Igniting the combustible mixture creates a high pressure (from the expanding gases), closing the valves. The violent ejection of the gases forms a relatively low pressure area inside the duct, admitting a fresh charge of air through the flat spring valves. Because of the temperature of the duct and the return of part of the flaming exhaust gases, the rest of the charges burn without an igniter plug. This operating cycle or pulsation creates a loud buzzing

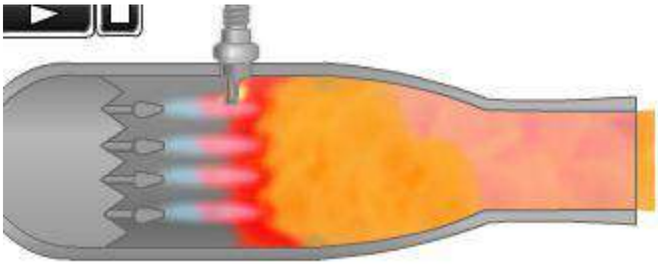


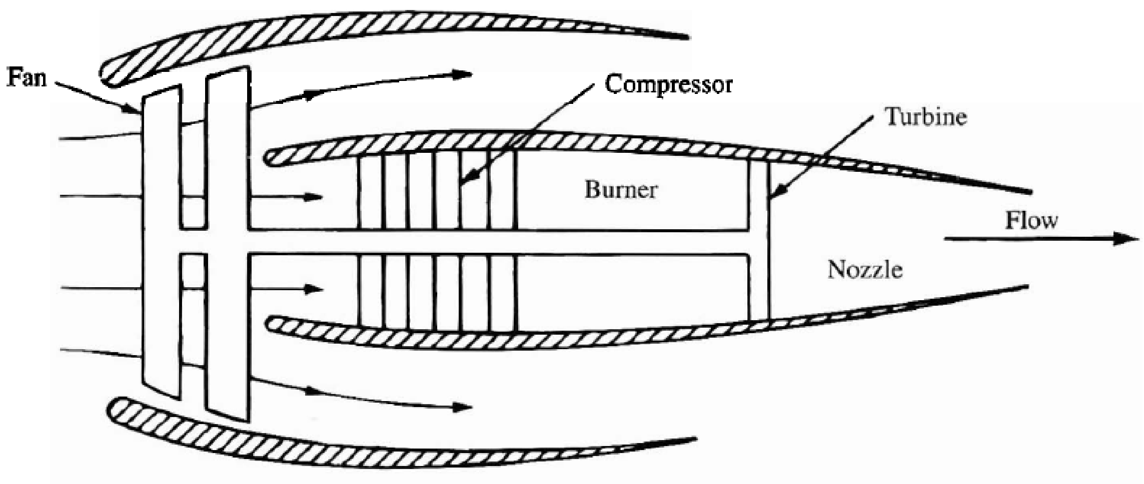
Figure 1-8 — The Aeropulse or pulsejet.

sound. “Buzz bomb” described an
early application of this unit, the German V-1 flying bomb designed for the German Air .Ministry in 1933, and more recently used for ultra-light type aircraft

We learned the basic principle of jet propulsion with the rocket. The ram jet taught us that adding heat would expand the gases and increase velocity. It also showed the amount of heat that is possible to add is dependent upon the amount of air available. The pulsejet proved that the more air an engine could compress the greater the power (thrust) it produced

THE TURBOFAN ENGINE

The ***turbofan*** engine is a propulsive mechanism to combine the high thrust of a turbojet with the high efficiency of a propeller. Basically, a turbojet engine forms the ***core*** of the turbofan; the core contains the diffuser, compressor, burner, turbine, and nozzle. However, in the turbofan engine, the turbine drives not only the compressor, but also a large fan ***external*** to the core. The fan itself is contained in a shroud that is wrapped around the core. The flow through a turbofan engine is split into two paths. One passes through the fan and flows externally over the core; this air is processed only by the fan, which is acting in the manner of a sophisticated, shrouded propeller. The propulsive thrust obtained from this flow through the fan is generated with an efficiency approaching that of a propeller. The second air path is through the core itself. The propulsive thrust is obtained from the flow through the core is generated with an efficiency associated with a turbojet. The overall propulsive efficiency of a turbofan is therefore a compromise between that of a propeller and that of a turbojet.



This compromise has been found to be quite successful- the vast majority of jet-propelled airplanes today are powered by turbofan engines.

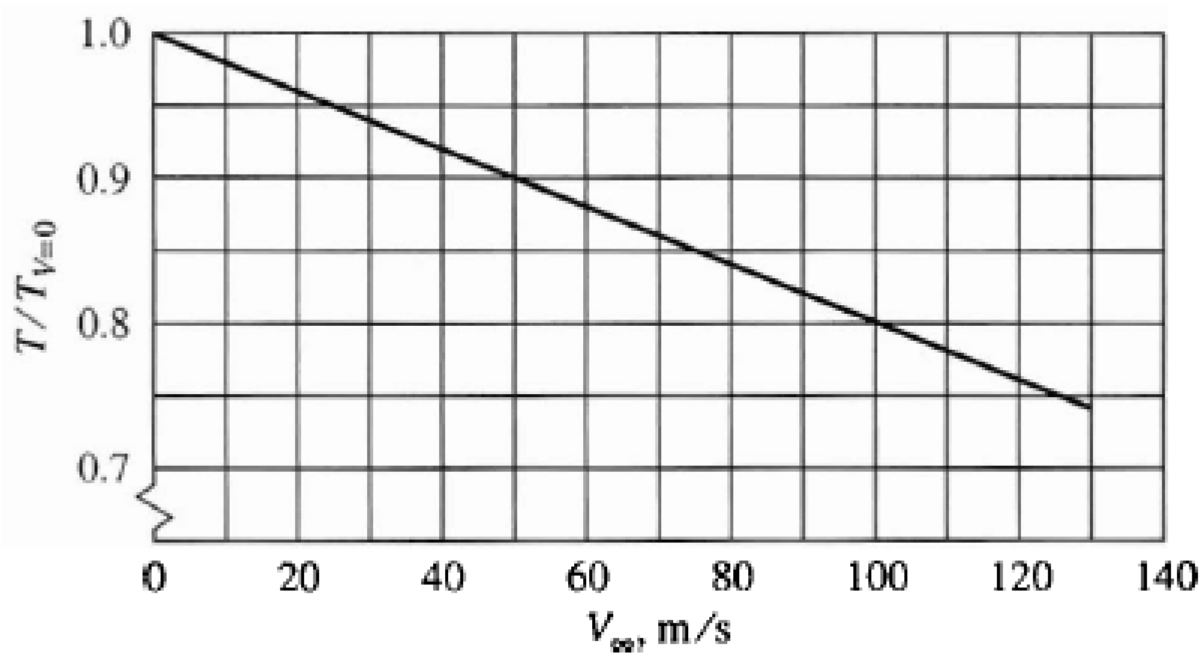
An important parameter of a turbofan engine is the bypass ***ratio,*** defined as the mass flow passing through the fan, externally to the core divided by the mass flow through the core itself. Everything else being equal, the higher the bypass ratio, the higher the propulsive efficiency. For the large turbofan engines that power airplanes such as the Boeing 747, for example, the Rolls-Royce RB211 and the Pratt & Whitney JTBD, the bypass ratios are on the order of 5. Typical values of the thrust specific fuel consumption for these turbofan engines are 0.6 lb/(lb h) almost half that of a conventional turbojet engine.

Variations of Thrust and Specific Fuel Consumption with Velocity and Altitude

For high-bypass-ratio turbofans-those with bypass ratios on the order of 5 (these are the class of turbofans that power civil transports) the performance seems to be closer to that of a propeller than that of a turbojet in some respects. The thrust of a civil turbofan engine has a strong variation with velocity; thrust decreases as V_∞ increases:

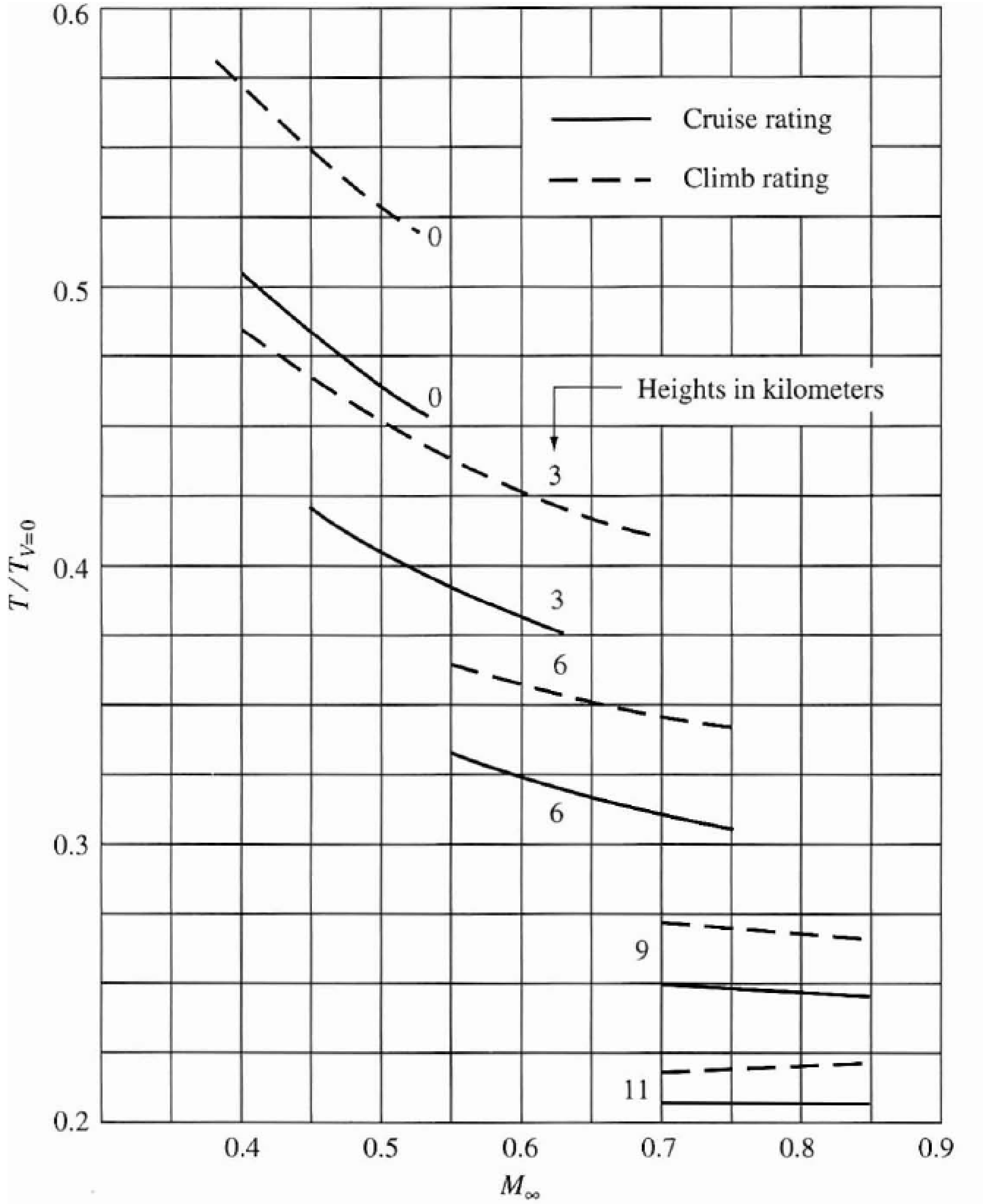
$$\frac{T}{T_{V=0}} = 1 - 2.52 \times 10^{-3} V_\infty + 4.34 \times 10^{-6} V_\infty^2$$

The equation holds for V_∞ < 130 m/s.



At higher subsonic velocities for a given, constant altitude, the decrease in thrust with Mach number can be correlated by

$$\frac{T}{T_{V=0}} = A M_\infty^{-n}$$



Although the variation of T for a civil turbofan is a strong function of V, (or Ma) at lower altitudes, at the relatively high altitude of 11 **km,** T is relatively constant for the narrow Mach number range from 0.7 to 0.85. This corresponds to normal cruise Mach numbers for civil transports such as the Boeing 747. Hence, for the analysis of airplane performance in the cruise range, it appears reasonable to assume T= constant.

The variation of T with altitude is approximated by

$$\frac{T}{T_0} = \left(\frac{\rho}{\rho_0}\right)^m$$

The variation of thrust specific fuel consumption with both altitude and Mach number is shown in Fig. The ratio of the thrust specific fuel consumption at the specified altitude and Mach number, to the value at zero velocity and at sea level, is shown. The variation with velocity at a given altitude follows the relation:

$$c_t = B(1 + kM_\infty)$$

where B and k are empirical constants

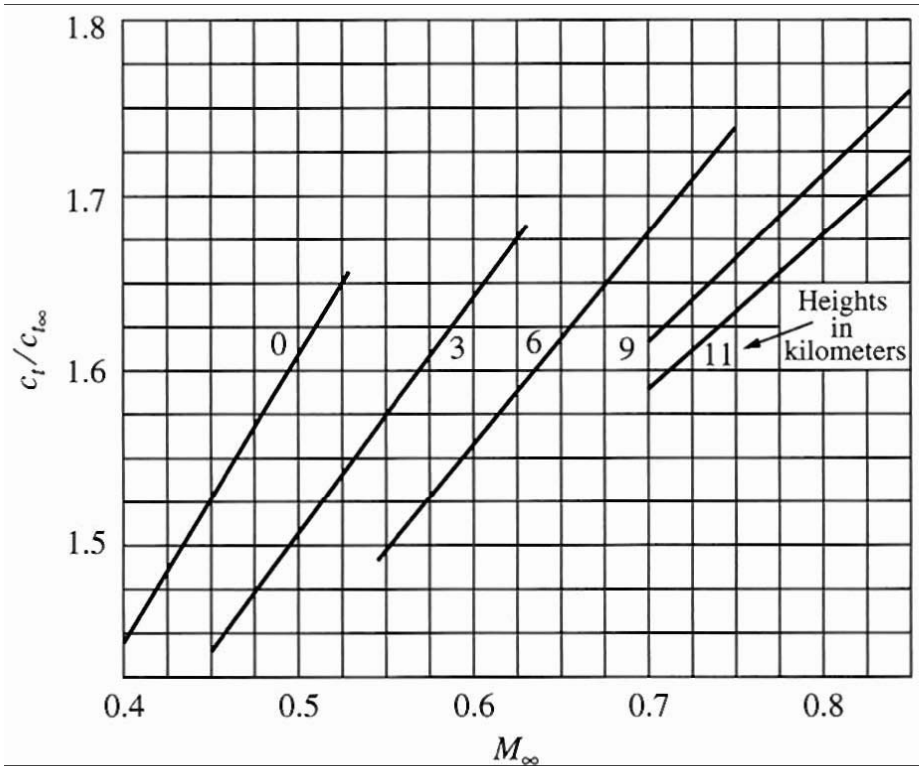
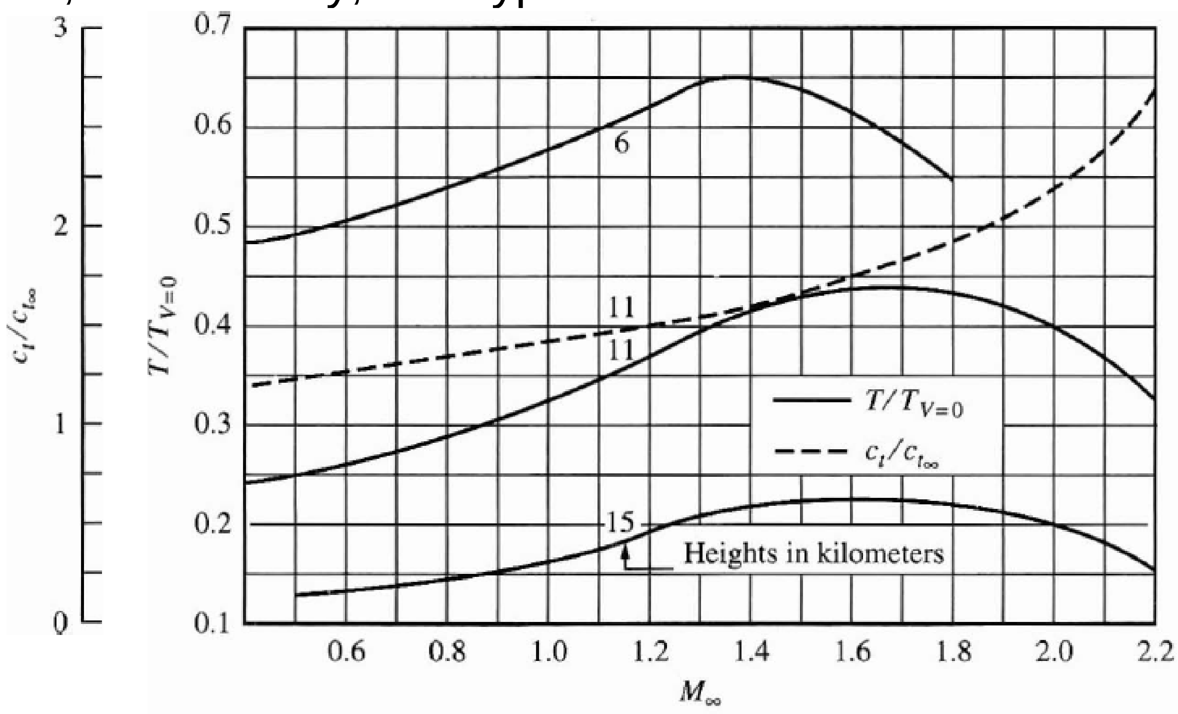


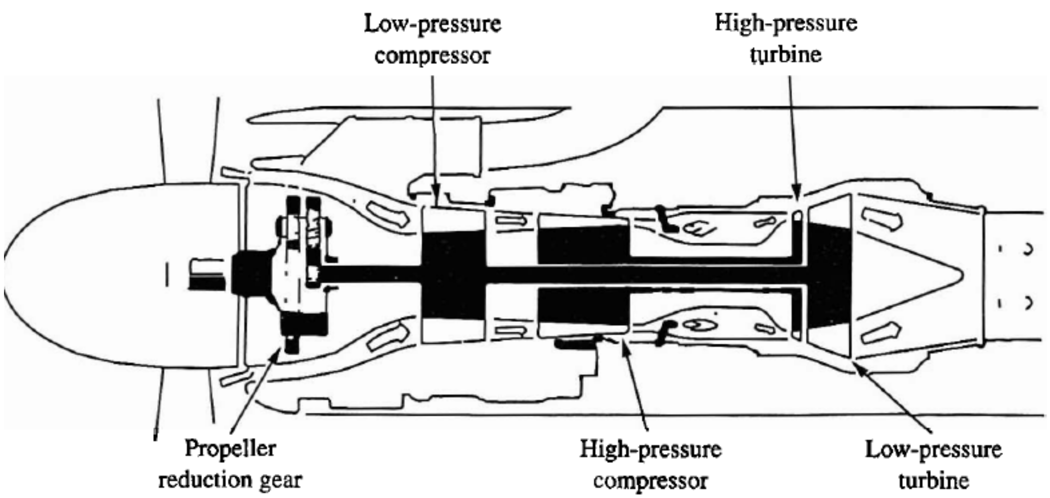
Figure shows why turbofans were not used on the Concorde supersonic transport, with its cruising Mach number of 2.2. The thrust specific fuel consumption of a turbojet engine is almost constant with speed **in** the supersonic regime. However, for a turbofan, ***ct*** increases markedly with an increase in Ma. For this reason, a turbojet is more fuel-efficient than a turbofan if the design Mach number is 2.2. The ordinate in Figure is expanded. Hence, the altitude effect on ***ct,*** looks larger than it really is. To first order, is constant with altitude.

For low-bypass-ratio turbofans-those with bypass ratios between 0 and 1-the performance is somewhat different from that for the high-bypass-ratio case discussed above. The performance of low-bypass-ratio turbofans is much closer to that of a turbojet than that of a propeller. Typical generic variations of $T/T_{v=0}$ and $c_t/c_{t\infty}$ versus Ma, for a military, low-bypass-ratio turbofan are



after a small initial decrease at low subsonic Mach numbers, the thrust increases for increasing Mach number well above Mach 1. The dashed line gives the variation of thrust specific fuel consumption versus Mach number for a military turbofan. Note that **ct,** for the low-bypass-ratio turbofan gradually increases as M, increases for subsonic and transonic speeds, and begins to rapidly increase at Mach **2** and beyond. This is unlike the variation of **ct,** for a pure turbojet engine, which is relatively constant in the low supersonic regime.

THE TURBOPROP



The turboprop is essentially a propeller driven by a gas-turbine engine, it is the closest to the reciprocating engine/propeller combination**. T**he inlet air is compressed by an axial-flow compressor, mixed with fuel and burned in the combustor, expanded through a turbine, and then exhausted through a nozzle. Unlike the turbojet, the turbine powers not only the compressor but also the propeller. By design, most of the available work in the flow is extracted by the turbines, leaving little available for jet thrust. For most turboprops, only about 5% of the total thrust is associated with the jet exhaust, and remaining 95% comes from the propeller.

the turboprop falls in between the reciprocating engine/propeller combination and the turbofan or turbojet. The turboprop generates more thrust than a reciprocating engine/propeller device, but less than a turbofan or turbojet. On the other hand, the turboprop has a specific fuel consumption higher than that of the reciprocating engine/propeller combination, but lower than that of **a** turbofan or turbojet. Also, the maximum speed of a turboprop-powered airplane is limited to that at which the propeller efficiency becomes seriously degraded by shock wave formation on the propeller usually around Ma=0.6 to 0.7.

the thrust generated by the turboprop is the sum of the propeller thrust *Tp,* and the jet thrust *Tj.* For the engine in flight at velocity V, the power available from the turboprop is

$$\boxed{P_A = \left(T_p + T_j\right) V_\infty}$$

The main business end of a turboprop is the shaft coming from the engine to which the propeller is attached via some type of gearbox mechanism. Hence the *shaft power Ps,* coming from the engine is a meaningful quantity. Because of losses associated with the propeller the power obtained from the propeller/shaft combination is η_{pr} *Ps.* Hence, the net power available, which includes the jet thrust, is

$$\boxed{P_A = \eta_{pr} P_s + T_j V_\infty}$$

Sometimes manufacturers rate their turboprops in terms of the *equivalent shaftpower Pes* which is an overall power rating that *includes* the effect of the jet thrust:

$$\boxed{P_A = \eta_{pr} P_{es}}$$

Combining the two:

$$\eta_{pr} P_{es} = \eta_{pr} P_s + T_j V_\infty$$

$$\boxed{P_{es} = P_s + \frac{T_j V_\infty}{\eta_{pr}}}$$

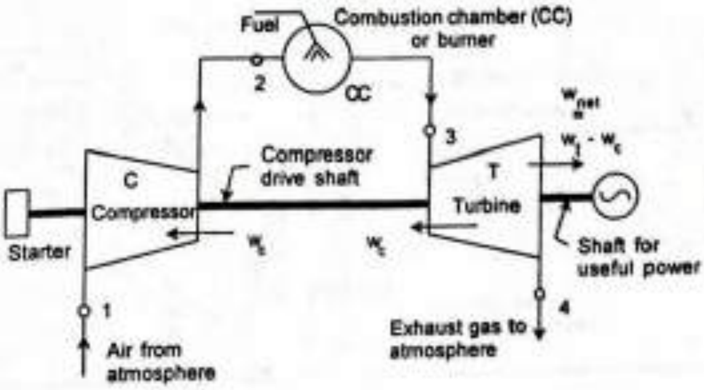


Fig. 17.2(a). Simple Open Cycle Gas Turbine.

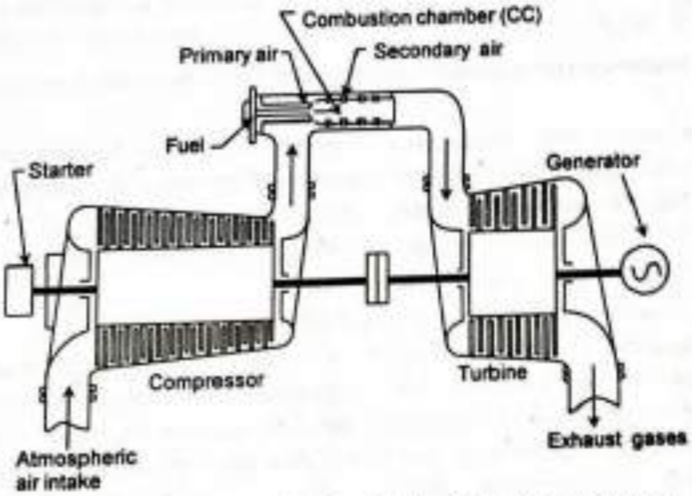


Fig. 17.2(b). Simple Open Cycle Gas Turbine Showing Blade Rows.

system will be free from combustion products The proto type hybrid unit producing 40 MW will be completed by 2004.

17.2. Simple Open Cycle Gas Turbine (Constant Pressure Heat Addition) or Air Standard Brayton (or Joule) Cycle.

The Joule or Brayton cycle is the most idealized cycle for the simple gas turbine power plant shown in Fig. 17.2(a) & (b). Atmospheric air is compressed from p_1 to a high pressure p_2 in the compressor and delivered to the burner or combustion chamber (CC) where fuel is injected and burned. The combustion process occurs nearly at constant pressure. Due to combustion heat is added to the working fluid in the burner (combustor) from T_2 to T_3. The products of combustion from the combustion chamber are expanded in the turbine from p_2 to atmosphere pressure, p_1 and then discharged into the atmosphere. The turbine and compressor are mechanically coupled, so the net work is equal to the difference between the work done by the turbine and work consumed by the compressor.

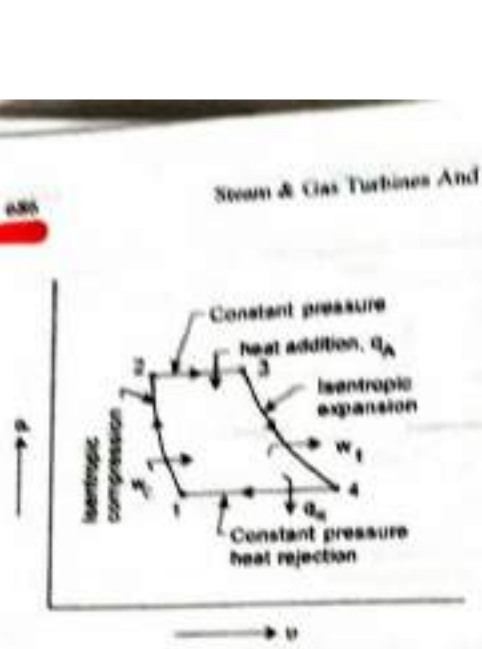


Fig. 17.3(a). Representation of Joule Cycle on p-v Diagram.

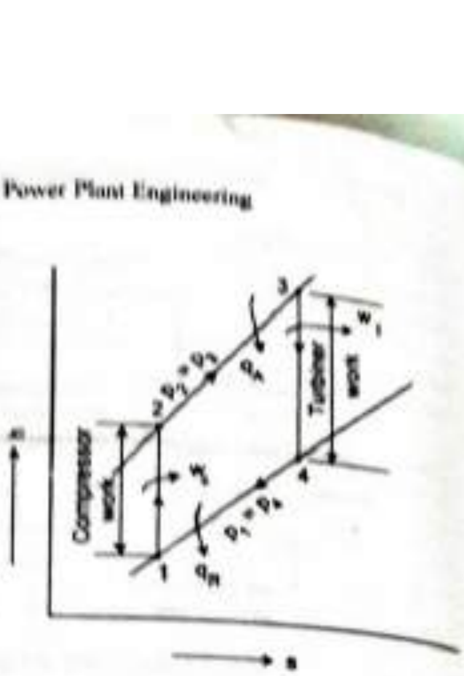


Fig.17.3 (b). Representation of Joule Cycle on *h-s or (T-s)* Diagram.

To first run the compressor, a starter is needed. When the turbine starts running, the starter is cut-off.

Representation of ideal Brayton cycle on *p-v* and *h-s* or *(T-s)* diagrams are shown in Fig. 17.3 (a) and 17.3 (b). All the assumptions are valid here for air standard cycle.

Process 1-2 is the isentropic compression in the compressor.

Process 2-3 is constant pressure heat additiod in the combustion chamber.

Process 3-4 is isentropic expansion in the turbine.

Process 4.1 is constant pressure heat rejection (cycle is not closed).

Referring to Fig. 17.3, we may find out the thermal efficiency on the basis of 1 kg of working fluid flow. Use the steady flow energy equation for analysis. Since the velocity of gas is high in gas turbine so the analysis should be based on *stagnation values*, but for simplicity, the analysis is based on static values. *However, with the same analysis stagnation values may be put by placing suffix 0 in enthalpy, temperature and pressure terms.*

Heat supplied $= q_A = h_3 - h_2 = c_p (T_3 - T_2)$ if c_p is assumed constant in process 2-3.

Heat rejected $= q_R = h_4 - h_1 = c_p (T_4 - T_1)$ if c_p is assumed constant in process 4-1.

Net work $= w_{net} = q_A - q_R = c_p \{(T_3 - T_2) - (T_4 - T_1)\}$

This net work may also be found from turbine and compressor work.

Work done by turbine $= w_t = h_3 - h_4 = c_p (T_3 - T_4)$

Work consumed by compressor $= w_c = h_2 - h_1 = c_p (T_2 - T_1)$

$$w_{net} = w_t - w_c = c_p\{(T_3 - T_4) - (T_2 - T_1)\} = c_p\{(T_3 - T_2) - (T_4 - T_1)\}$$

The thermal efficiency $= \eta_{th} = \dfrac{w_{net}}{q_A} = \dfrac{c_p\{(T_3 - T_2) - (T_4 - T_1)\}}{c_p (T_3 - T_2)} = 1 - \dfrac{T_4 - T_1}{T_3 - T_2}$ (17.1)

We know that $\dfrac{T_2}{T_1} = \left(\dfrac{p_2}{p_1}\right)^{(\gamma-1)/\gamma}$ and $\dfrac{T_3}{T_4} = \left(\dfrac{p_3}{p_4}\right)^{(\gamma-1)/\gamma}$

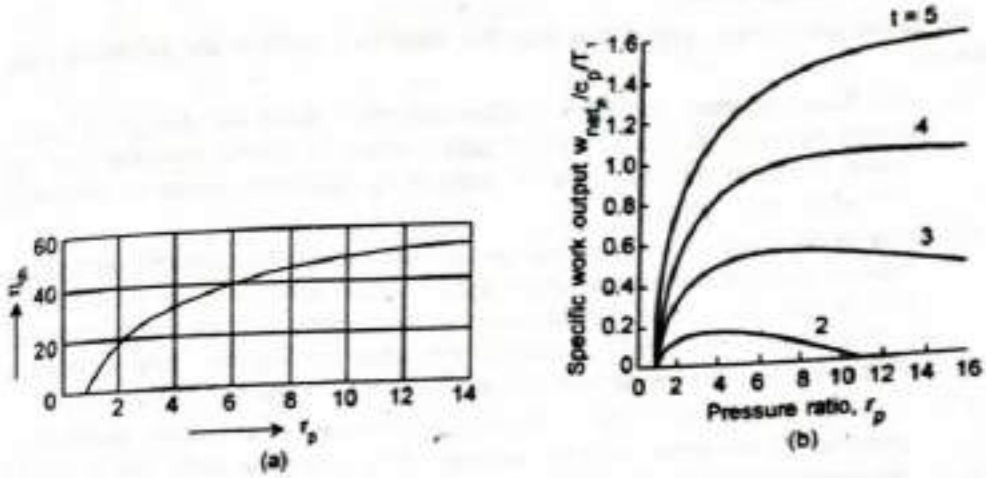


Fig. 17.4. Thermal Efficiency and Specific Work Output for Simple Cycle.

Since $p_1 = p_4$ and $p_2 = p_3$, so $\frac{T_2}{T_1} = \frac{T_3}{T_4} = \left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma}$ or $\frac{T_1}{T_2} = \frac{T_4}{T_3} = \frac{T_4 - T_1}{T_3 - T_2}$

Putting the value of $\frac{T_4 - T_1}{T_3 - T_2}$ in equation (17.1), we get

$$\eta_{th} = 1 - \frac{T_1}{T_2} = 1 - \frac{T_4}{T_3}$$

Let $\quad r_p = \text{pressure ratio} = \frac{p_2}{p_1}$

So, $\quad \frac{T_2}{T_1} = \left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} = (r_p)^{(\gamma-1)/\gamma}$

Therefore, the thermal efficiency $= \eta_{th} = 1 - \frac{1}{r_p^{(\gamma-1)/\gamma}}$ (17.2)

From above equation we see that the thermal efficiency of the Brayton cycle is the same as that of the Otto cycle. *Note that for turbine and compressor work we have neglected the change in K.E and P.E. energies.*

A plot of η_{th} as a function of r_p is shown in Fig. 17.4.(a). It is obvious from the Fig. 17.4(a) that the efficiency progressively increases with increasing value of pressure ratio. The turbine specific work is found to be a function of pressure ratio and turbine inlet temperature as shown in Fig. 17.4(b) in which $T_3/T_1 = T_{max}/T_{min} = t$ and expressed as

$$\frac{w_{net}}{c_pT_1} = t\left[1 - \frac{1}{r_p(\gamma-1)/\gamma}\right] - \left[r_p^{(\gamma-1)/\gamma} - 1\right] \tag{17.3}$$

It is obvious that specific work increases with t at any pressure ratio but there exists an optimum value of r_p for every value of t i.e. T_3/T_1

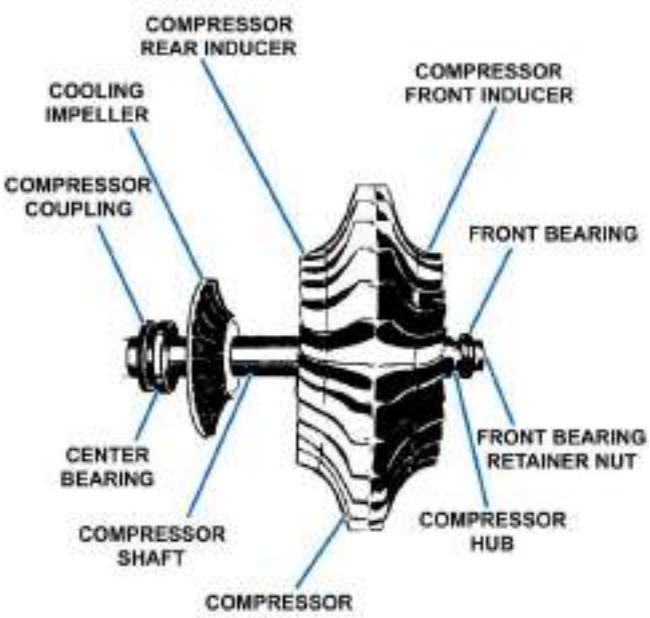
COMPRESSOR SECTION

The primary function of the compressor is to supply air in enough quantity to satisfy the requirements of the combustion burners. Specifically, the compressor increases the air mass received from the air inlet duct and directs it to the burners in the quantity and at the pressures required. A secondary function is to supply compressor bleed air for various purposes in the engine and aircraft. The compressor provides space for mounting accessories and engine parts. There are two basic types of compressors. The compressor type is also the engine type, so a centrifugal-flow compressor is in a centrifugal engine. Centrifugal-flow compressors have a compression ratio of 5:1. Present-day axial flow compressors have compression ratios approaching 15:1 and airflows up to 350 lb. The addition of a fan raises these values to 25:1 and 1,000 lb. /sec.

Centrifugal-Flow Compressors

The single entry centrifugal-flow compressor *(Figure 1-16)* consists of an impeller (rotor element), a diffuser (stator element), and a manifold. The impeller picks up and accelerates air outward to the diffuser. The diffuser directs air into the manifold. The manifold distributes air into the combustion section.

Double entry centrifugal-flow compressors *(Figure 1-17)* handle the same airflow with a smaller diameter. Small multi-stage centrifugal-flow engines used in aircraft *(Figure 1-18),* or as Auxiliary Power Units (APUs) that take advantage of this feature.



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axial-Flow Compressors

The term *axial flow* applies to the axial (straight-line) flow of air through the compressor section of the engine. The axial-flow compressor has two main elements—a rotor and a stator. Each consecutive pair of rotor and stator blades makes a pressure stage. The rotor is a shaft with blades attached to it. These blades impel air rearward in the same manner as a propeller, by reason of their angle and airfoil contour. The rotor, turning at high speed, takes in air at the compressor inlet and impels it through a series of stages. The action of the rotor increases the compression of the air. At each stage it accelerates rearward. The stator blades act as diffusers, partially converting high velocity to pressure. Maintaining high efficiency requires small changes in the rate of diffusion at each stage. The number of stages depends on the amount of air and total pressure rise required. A greater number of stages means a higher compression ratio. Most present day engines use from 10 to 16 stages.

An axial-flow compressor follows the same rules and has the same limitations as an aircraft wing. The concept is more complicated than a single airfoil, because the blades are close together. Each trailing edge blade affects the next leading edge. This cascade effect is of prime importance in determining blade design and placement. The axial-flow compressor has

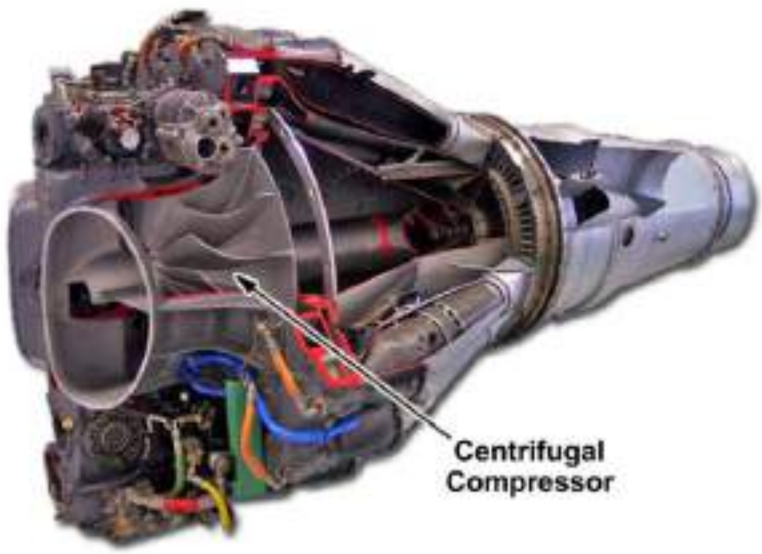


Figure 1-18 — Centrifugal-flow engine

its disadvantages, the most important of which is the stall problem. If, for some reason, the angle of attack—the angle at which the airflow strikes the rotor blades—becomes too low, the pressure zones, shown in *Figure 1-19*, will be of low value, and the airflow and compression will be low. If the angle of attack is high, the pressure zones will be high, and the airflow and compression ratio will be high. If the angle of attack is too high, the compressor will stall.

The airflow over the upper foil surface will become turbulent and destroy the pressure zones. This will decrease the compression airflow. The angle of attack will vary with engine rpm, compressor-inlet temperature, and compressor discharge or burner pressure. Any action that decreases airflow relative to engine speed will increase the angle of attack and increase the tendency to stall.

The decrease in airflow may result from a too-high compressor-discharge pressure. During ground operation of the engine, the prime action that causes a stall is choking. If there is a decrease in the engine speed, the compression ratio will decrease with the lower rotor velocities. With a decrease in compression, the volume of air in the rear of the compressor will be greater. This excess volume of air causes a choking action in the rear of the compressor with a decrease in airflow. This, in turn, decreases the air velocity in the front of the compressor and increases the tendency to stall. If no corrective action is taken, the front of the compressor will stall at low engine speeds.

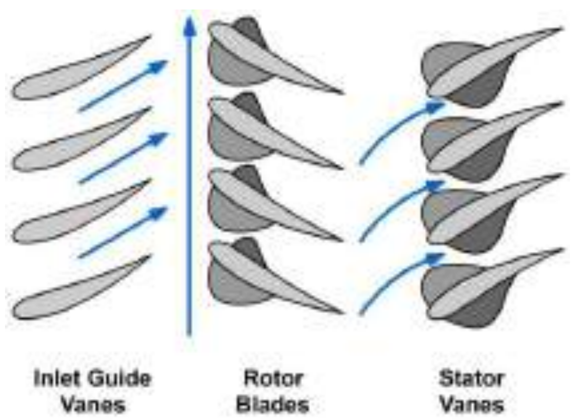
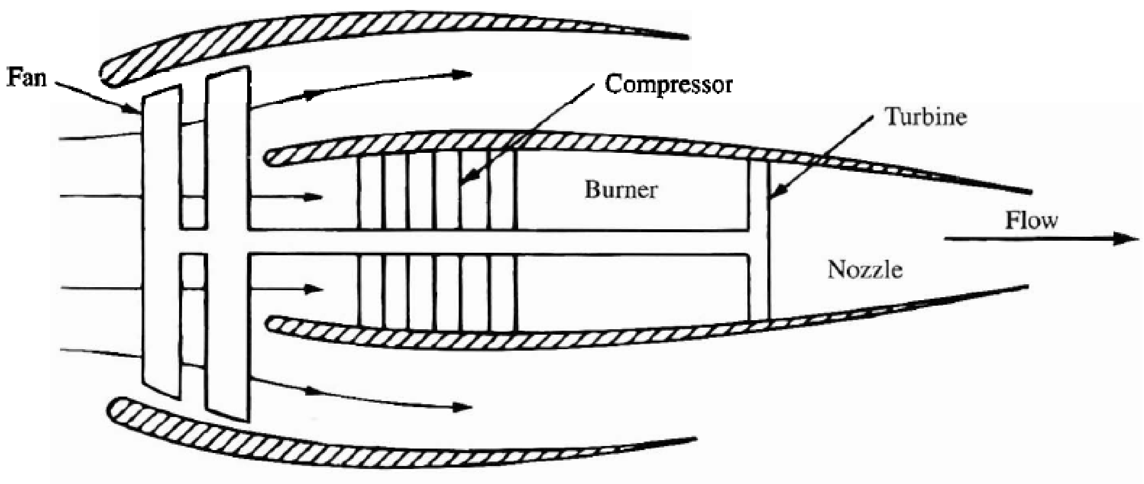


Figure 1-19 — The cascade effect.

THE TURBOFAN ENGINE

The ***turbofan*** engine is a propulsive mechanism to combine the high thrust of a turbojet with the high efficiency of a propeller. Basically, a turbojet engine forms the ***core*** of the turbofan; the core contains the diffuser, compressor, burner, turbine, and nozzle. However, in the turbofan engine, the turbine drives not only the compressor, but also a large fan ***external*** to the core. The fan itself is contained in a shroud that is wrapped around the core. The flow through a turbofan engine is split into two paths. One passes through the fan and flows externally over the core; this air is processed only by the fan, which is acting in the manner of a sophisticated, shrouded propeller. The propulsive thrust obtained from this flow through the fan is generated with an efficiency approaching that of a propeller. The second air path is through the core itself. The propulsive thrust is obtained from the flow through the core is generated with an efficiency associated with a turbojet. The overall propulsive efficiency of a turbofan is therefore a compromise between that of a propeller and that of a turbojet.



This compromise has been found to be quite successful- the vast majority of jet-propelled airplanes today are powered by turbofan engines.

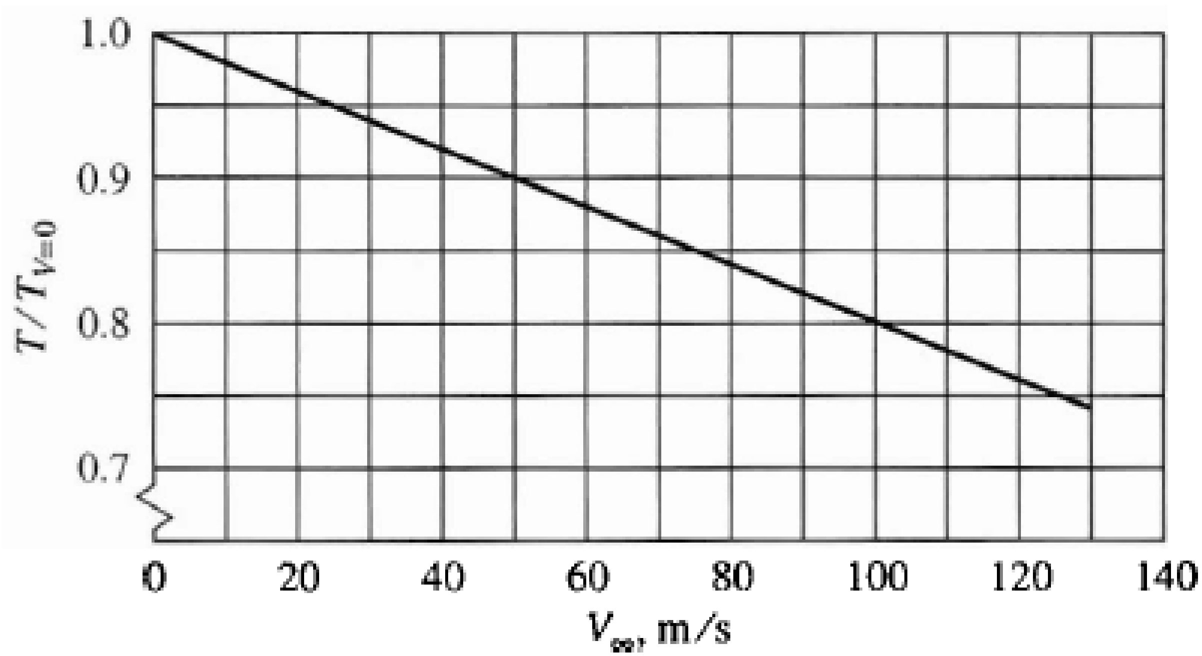
An important parameter of a turbofan engine is the bypass ***ratio,*** defined as the mass flow passing through the fan, externally to the core divided by the mass flow through the core itself. Everything else being equal, the higher the bypass ratio, the higher the propulsive efficiency. For the large turbofan engines that power airplanes such as the Boeing 747, for example, the Rolls-Royce RB211 and the Pratt & Whitney JTBD, the bypass ratios are on the order of 5. Typical values of the thrust specific fuel consumption for these turbofan engines are 0.6 lb/(lb h) almost half that of a conventional turbojet engine.

Variations of Thrust and Specific Fuel Consumption with Velocity and Altitude

For high-bypass-ratio turbofans-those with bypass ratios on the order of 5 (these are the class of turbofans that power civil transports) the performance seems to be closer to that of a propeller than that of a turbojet in some respects. The thrust of a civil turbofan engine has a strong variation with velocity; thrust decreases as V_∞ increases:

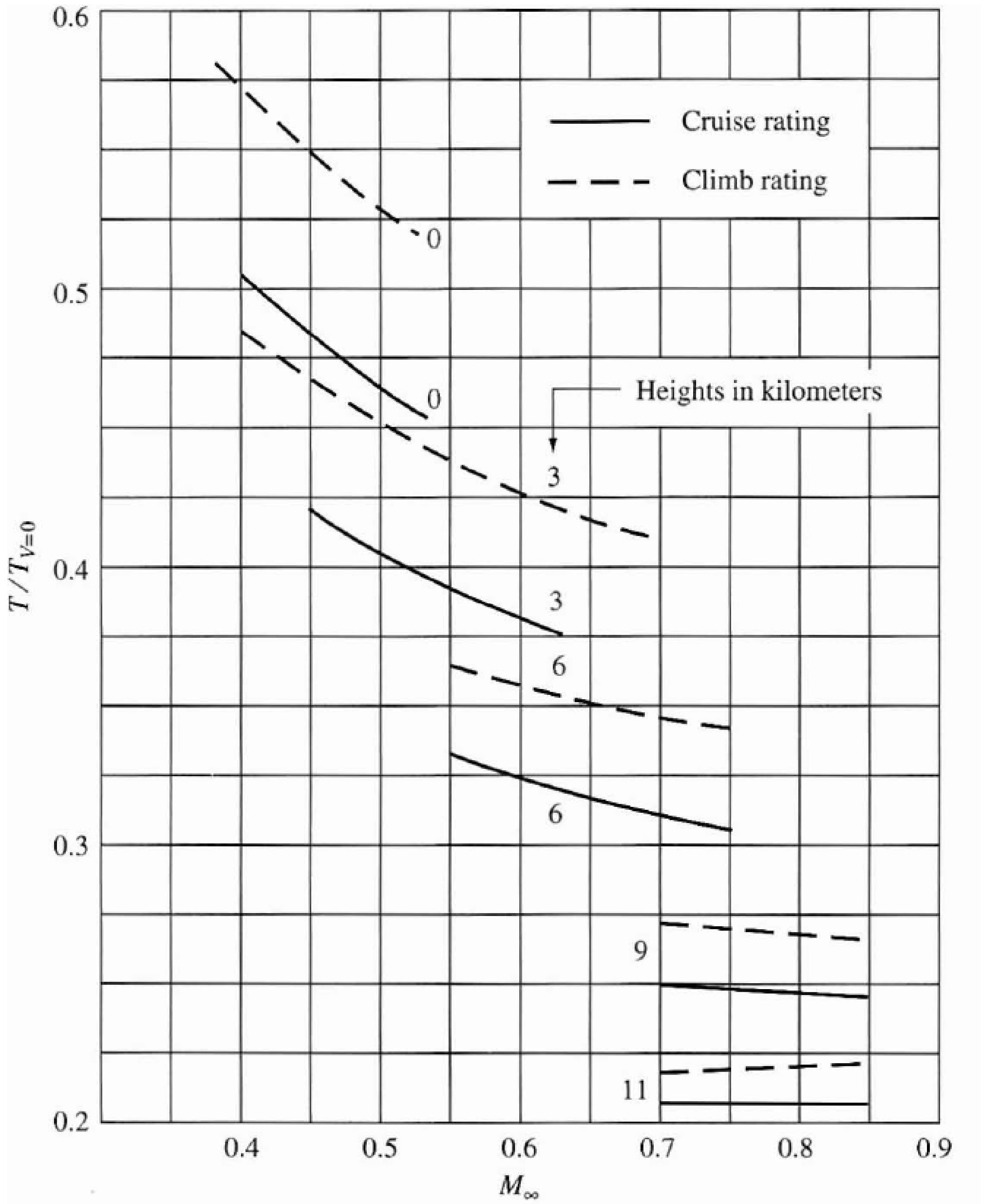
$$\frac{T}{T_{V=0}} = 1 - 2.52 \times 10^{-3} V_\infty + 4.34 \times 10^{-6} V_\infty^2$$

The equation holds for V_∞ < 130 m/s.



At higher subsonic velocities for a given, constant altitude, the decrease in thrust with Mach number can be correlated by

$$\frac{T}{T_{V=0}} = AM_\infty^{-n}$$



Although the variation of T for a civil turbofan is a strong function of V, (or Ma) at lower altitudes, at the relatively high altitude of 11 **km,** T is relatively constant for the narrow Mach number range from 0.7 to 0.85. This corresponds to normal cruise Mach numbers for civil transports such as the Boeing 747. Hence, for the analysis of airplane performance in the cruise range, it appears reasonable to assume T= constant.

The variation of T with altitude is approximated by

$$\frac{T}{T_0} = \left(\frac{\rho}{\rho_0}\right)^m$$

The variation of thrust specific fuel consumption with both altitude and Mach number is shown in Fig. The ratio of the thrust specific fuel consumption at the specified altitude and Mach number, to the value at zero velocity and at sea level, is shown*.* The variation with velocity at a given altitude follows the relation:

$$c_t = B(1 + kM_\infty)$$

where B and k are empirical constants

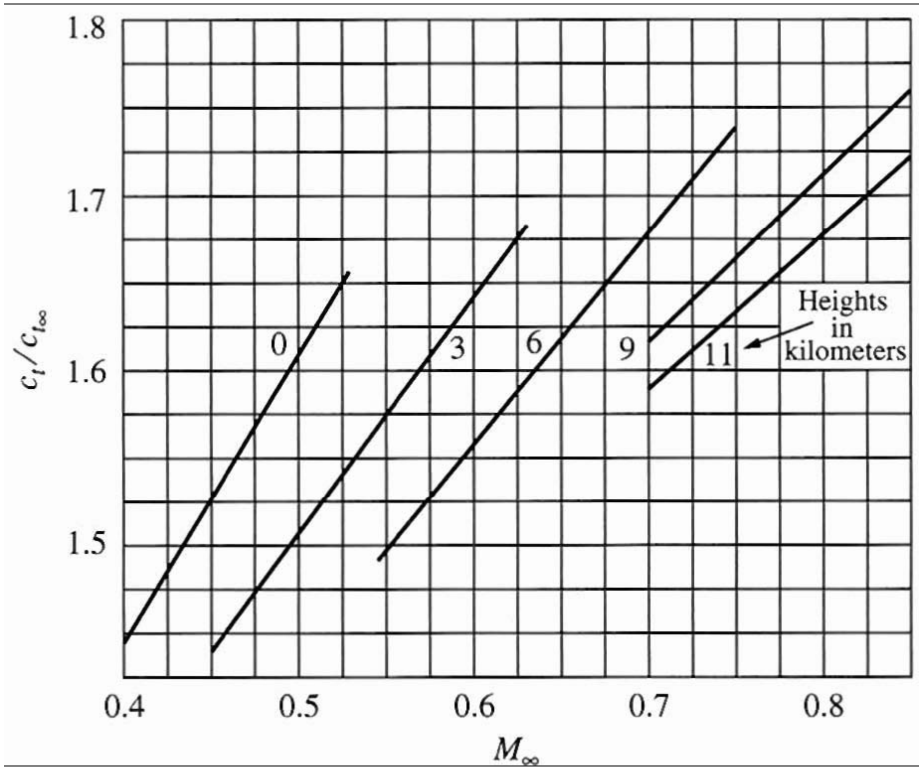


Figure shows why turbofans were not used on the Concorde supersonic transport, with its cruising Mach number of 2.2. The thrust specific fuel consumption of a turbojet engine is almost constant with speed **in** the supersonic regime. However, for a turbofan, ***ct*** increases markedly with an increase in Ma. For this reason, a turbojet is more fuel-efficient than a turbofan if the design Mach number is 2.2. The ordinate in Figure is expanded. Hence, the altitude effect on ***ct,*** looks larger than it really is. To first order, is constant with altitude.

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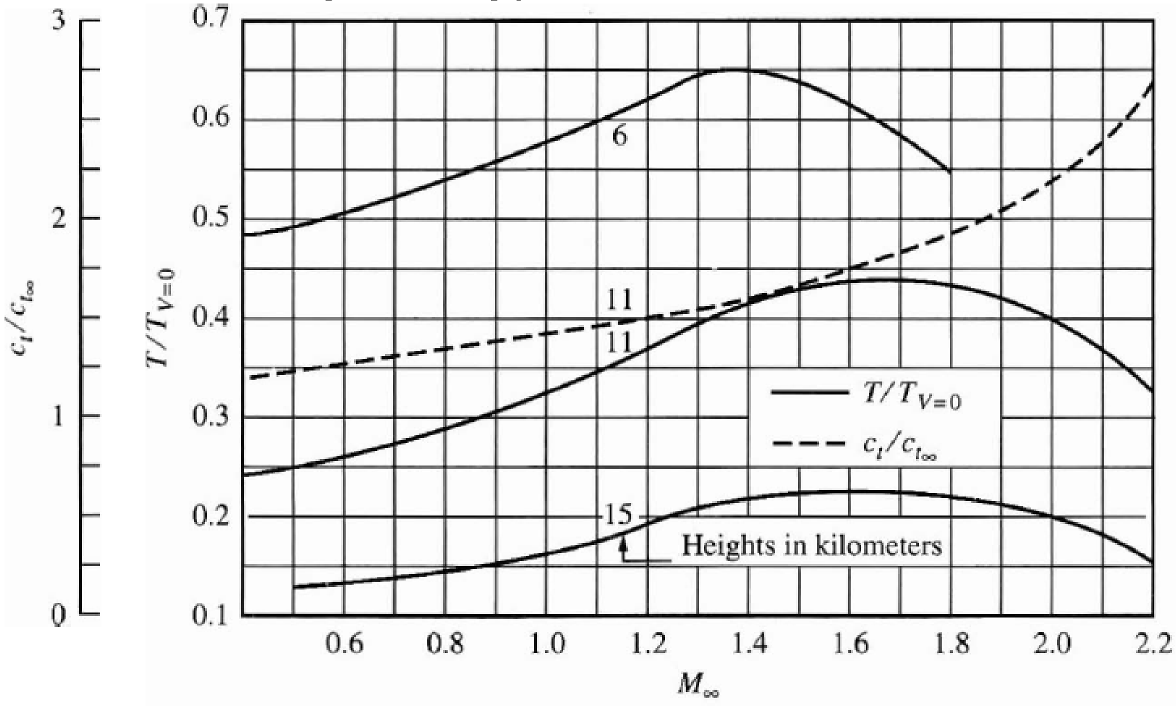

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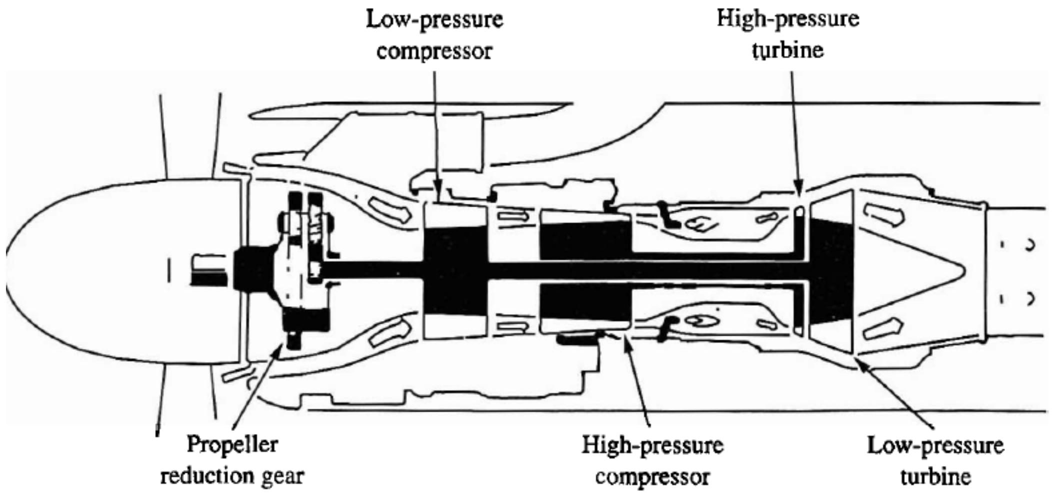
THE TURBOPROP



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The main business end of a turboprop is the shaft coming from the engine to which the propeller is attached via some type of gearbox mechanism. Hence the *shaft power Ps,* coming from the engine is a meaningful quantity. Because of losses associated with the propeller the power obtained from the propeller/shaft combination is η_{pr} *Ps.* Hence, the net power available, which includes the jet thrust, is

$$\boxed{P_A = \eta_{pr} P_s + T_j V_\infty}$$

Sometimes manufacturers rate their turboprops in terms of the *equivalent shaftpower Pes* which is an overall power rating that *includes* the effect of the jet thrust:

$$\boxed{P_A = \eta_{pr} P_{es}}$$

Combining the two:

$$\eta_{pr} P_{es} = \eta_{pr} P_s + T_j V_\infty$$

$$\boxed{P_{es} = P_s + \frac{T_j V_\infty}{\eta_{pr}}}$$

17.3. Actual Brayton Cycle.

The actual gas turbine cycle differs from the theoretical cycle in the following main respects:–

(1) The fluid velocities are high in turbo-machinery, hence the change in kinetic energy between inlet and outlet of each component cannot necessarily be ignored. This may be taken care by considering stagnation values of properties instead of static values.

(2) Due to *frictional losses* in the compressor and turbine the compression and expansion processes are not frictionless and take place with some increase in entropy (*i.e.* the processes are irreversible adiabatic). In the ideal case, the compressor and turbine efficiencies were 100 percent but actual turbine and compressor have efficiency less than 100 percent. (Generally, $\eta_c = 87\%$ and $\eta_t = 88\%$)

(3) A *small pressure loss (about 2% of inlet pressure) occurs* in the combustion chamber. Combustion is never complete. This loss is so little that it can be neglected for the sake of simplification of the problem wherever necessary. Thus, $p_3 = p_2 - (\Delta p)_{loss, comb}$

(4) The mass of the gas flowing through the turbine is $(1 + F/A)$ times the mass of air flowing through the compressor where F/A represents the fuel-air ratio.

(5) The *specific heat of combustion gas is slightly higher than that of air* ($c_{pg} = 1.20$ kJ/kgK). This increase is so little that the specific heat of combustion gas may be taken as that of air for simplicity wherever necessary.

(6) There is a pressure loss in the exhaust hood of turbine and as a result the expansion of gas will take place upto a pressure higher than atmospheric so that after pressure loss the exhaust pressure will be equal to the atmospheric pressure. Thus $p_4 = p_1 + \Delta p_{loss}$. If $p_4 < p_1$, the exhaust will not go to atmosphere and there will be a back flow.

(7) The gas turbine bladings are subjected to high pressure and high temperature and thus they are vulnerable to oxidation and creep. For the safe working of gas turbine blading, it is cooled. Generally, the blade are cooled by air bleeding, from compressor. The amount of the cooling air required varies from 7 to 20 percent of inlet air flow depending upon turbine inlet temperature. In the present analysis, it is neglected

h–s diagram for an actual Brayton Cycle is shown in Fig. 17.5. The pressure loss in the combustion chamber is represented by $p_2 - p_3$ and the exhausthood, $p_4 - p_1$. In this cycle.

1 – 2 is isentropic compression.	1 – 2′ is actual compression.
3 – 4 is isentropic expansion.	3 – 4′ is actual expansion.

The compressor efficiency also known as isentropic compressor efficiency, η_c is—

$$\eta_c = \frac{\text{Isentropic compression work}}{\text{Actual compression work}} = \frac{w_c}{w_{ca}}$$

Based on static values, $$\eta_c = \frac{h_2 - h_1}{h_2' - h_1} = \frac{c_p(T_2 - T_1)}{c_p(T_2' - T_1)} = \frac{T_2 - T_1}{T_2' - T_1} \quad (17.4)$$

Based on total or stagnation values, $$\eta_{oc} = \frac{h_{02} - h_{01}}{h_{02}' - h_1} = \frac{T_{02} - T_{01}}{T_{02}' - T_{01}} \quad (17.4a)$$

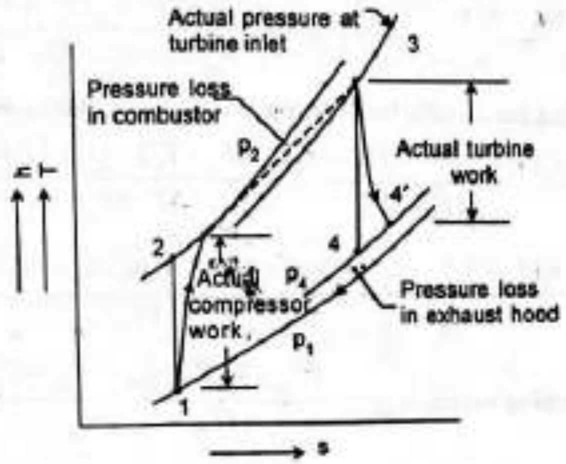


Fig. 17.5. Actual Gas Turbine Cycle Representation on h-s Chart

Here suffix 0 denotes the stagnation values.

The turbine efficiency, $\eta_t = \dfrac{\text{Actual turbine work}}{\text{Isentropic turbine work}} = \dfrac{w_{ta}}{w_t}$

Based on static values, we have :–

Actual turbine work $= w_{ta} = h_3 - h_4' = c_{pg}\left(T_3 - T_4'\right)$ kJ/kg

If specific heat of combustion gas c_{pg} may be taken as that of air, then

Actual turbine work $= w_{ta} = c_p(T_3 - T_4')$ kJ/kg

$$\text{So, turbine efficiency} = \eta_t = \frac{h_3 - h_4'}{h_3 - h_4} = \frac{c_p(T_3 - T_4')}{c_p(T_3 - T_4)} = \frac{T_3 - T_4'}{T_3 - T_4} \qquad (17.5)$$

Based on total or stagnation values, the turbine isentropic efficiency is

$$\eta_{ot} = \frac{h_{03} - h_{04}'}{h_{03} - h_{04}} = \frac{T_{03} - T_4'}{T_{03} - T_{04}} \qquad \mathbf{(17.5a)}$$

The thermal efficiency of the actual cycle is computed as follows for static values :–

Actual net work $= w_{neta} = w_{ta} - w_{ca} = [(1 + F/A)(h_3 - h_4')] - (h_2' - h_1)$ kJ/kg

where $F/A = \dot{m}_f / \dot{m}_a$ = fuel-air ratio

Actual heat supplied $= q_{Aa} = [(1 + F/A)\, h_3 - h_2']$ kJ/kg of air.

$$\text{So, thermal efficiency} = \eta_{tha} = \frac{(1 + F/A)(h_3 - h_4') - (h_2' - h_1)}{(1 + F/A)h_3 - h_2'}$$

$$= \frac{(1 + F/A) \cdot c_{pg}(T_3 - T_4') - c_{pa}(T_2' - T_1)}{(1 + F/A) \cdot c_{pg} \cdot T_3 - c_{pa} \cdot T_2'} \qquad \mathbf{(17.6)}$$

If we neglect the mass of fuel in comparison to the mass of air used, then

17.3. Actual Brayton Cycle.

The actual gas turbine cycle differs from the theoretical cycle in the following main respects:–

(1) The fluid velocities are high in turbo-machinery, hence the change in kinetic energy between inlet and outlet of each component cannot necessarily be ignored. This may be taken care by considering stagnation values of properties instead of static values.

(2) Due to *frictional losses* in the compressor and turbine the compression and expansion processes are not frictionless and take place with some increase in entropy (*i.e.* the processes are irreversible adiabatic). In the ideal case, the compressor and turbine efficiencies were 100 percent but actual turbine and compressor have efficiency less than 100 percent. (Generally, $\eta_c = 87\%$ and $\eta_t = 88\%$)

(3) A *small pressure loss (about 2% of inlet pressure) occurs* in the combustion chamber. Combustion is never complete. This loss is so little that it can be neglected for the sake of simplification of the problem wherever necessary. Thus, $p_3 = p_2 - (\Delta p)_{loss,\ comb}$

(4) The mass of the gas flowing through the turbine is $(1 + F/A)$ times the mass of air flowing through the compressor where F/A represents the fuel-air ratio.

(5) The *specific heat of combustion gas is slightly higher than that of air* ($c_{pg} = 1.20$ kJ/kgK). This increase is so little that the specific heat of combustion gas may be taken as that of air for simplicity wherever necessary.

(6) There is a pressure loss in the exhaust hood of turbine and as a result the expansion of gas will take place upto a pressure higher than atmospheric so that after pressure loss the exhaust pressure will be equal to the atmospheric pressure. Thus $p_4 = p_1 + \Delta p_{loss}$. If $p_4 < p_1$, the exhaust will not go to atmosphere and there will be a back flow.

(7) The gas turbine bladings are subjected to high pressure and high temperature and thus they are vulnerable to oxidation and creep. For the safe working of gas turbine blading, it is cooled. Generally, the blade are cooled by air bleeding, from compressor. The amount of the cooling air required varies from 7 to 20 percent of inlet air flow depending upon turbine inlet temperature. In the present analysis, it is neglected

h–s diagram for an actual Brayton Cycle is shown in Fig. 17.5. The pressure loss in the combustion chamber is represented by $p_2 - p_3$ and the exhausthood, $p_4 - p_1$. In this cycle.

1 – 2 is isentropic compression. 1 – 2' is actual compression.

3 – 4 is isentropic expansion. 3 – 4' is actual expansion.

The compressor efficiency also known as isentropic compressor efficiency, η_c is—

$$\eta_c = \frac{\text{Isentropic compression work}}{\text{Actual compression work}} = \frac{w_c}{w_{ca}}$$

Based on static values, $$\eta_c = \frac{h_2 - h_1}{h_2' - h_1} = \frac{c_p(T_2 - T_1)}{c_p(T_2' - T_1)} = \frac{T_2 - T_1}{T_2' - T_1} \qquad (17.4)$$

Based on total or stagnation values, $$\eta_{oc} = \frac{h_{02} - h_{01}}{h_{02}' - h_1} = \frac{T_{02} - T_{01}}{T_{02}' - T_{01}} \qquad (17.4a)$$

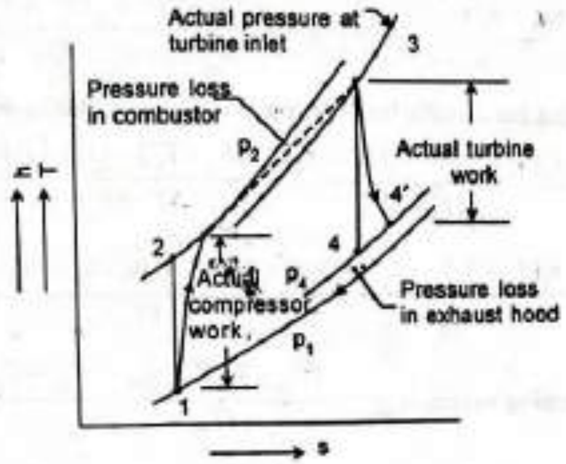


Fig. 17.5. Actual Gas Turbine Cycle Representation on *h-s* Chart

Here suffix 0 denotes the stagnation values.

The turbine efficiency, $\eta_t = \dfrac{\text{Actual turbine work}}{\text{Isentropic turbine work}} = \dfrac{w_{ta}}{w_t}$

Based on static values, we have :–

Actual turbine work $= w_{ta} = h_3 - h_4' = c_{pg}\left(T_3 - T_4'\right)$ kJ/kg

If specific heat of combustion gas c_{pg} may be taken as that of air, then

Actual turbine work $= w_{ta} = c_p(T_3 - T_4')$ kJ/kg

So, turbine efficiency $= \eta_t = \dfrac{h_3 - h_4'}{h_3 - h_4} = \dfrac{c_p(T_3 - T_4')}{c_p(T_3 - T_4)} = \dfrac{T_3 - T_4'}{T_3 - T_4}$ (17.5)

Based on total or stagnation values, the turbine isentropic efficiency is

$$\eta_{ot} = \frac{h_{03} - h_{04}'}{h_{03} - h_{04}} = \frac{T_{03} - T_4'}{T_{03} - T_{04}} \qquad \textbf{(17.5a)}$$

The thermal efficiency of the actual cycle is computed as follows for static values :–

Actual net work $= w_{neta} = w_{ta} - w_{ca} = [(1 + F/A)(h_3 - h_4')] - (h_2' - h_1)$ kJ/kg

where $F/A = \dot{m}_f / \dot{m}_a$ = fuel-air ratio

Actual heat supplied $= q_{Aa} = [(1 + F/A)\,h_3 - h_2']$ kJ/kg of air.

So, thermal efficiency $= \eta_{tha} = \dfrac{(1 + F/A)(h_3 - h_4') - (h_2' - h_1)}{(1 + F/A)h_3 - h_2'}$

$$= \frac{(1 + F/A)\,.\,c_{pg}(T_3 - T_4') - c_{pa}\,(T_2' - T_1)}{(1 + F/A)\,.\,c_{pg}\,.\,T_3 - c_{pa}\,.\,T_2'} \qquad \textbf{(17.6)}$$

If we neglect the mass of fuel in comparison to the mass of air used, then

Illustrative Example 5.1: In an axial flow compressor air enters the compressor at stagnation pressure and temperature of 1 bar and 292K, respectively. The pressure ratio of the compressor is 9.5. If isentropic efficiency of the compressor is 0.85, find the work of compression and the final temperature at the outlet. Assume g ¼ 1.4, and Cp ¼ 1.005 kJ/kg K.

Solution:

$$T_{01} = 292\text{K}, \quad P_{01} = 1\,\text{bar}, \quad \eta_c = 0.85.$$

Using the isentropic P–T relation for compression processes,

$$\frac{P_{02}}{P_{01}} = \left[\frac{T'_{02}}{T_{01}}\right]^{\frac{\gamma}{\gamma-1}}$$

where $T_{02'}$ is the isentropic temperature at the outlet.

Therefore,

$$T'_{02} = T_{01}\left[\frac{P_{02}}{P_{01}}\right]^{\frac{\gamma-1}{\gamma}} = 292(9.5)^{0.286} = 555.92\,\text{K}$$

Now, using isentropic efficiency of the compressor in order to find the actual temperature at the outlet,

$$T_{02} = T_{01} + \frac{\left(T'_{02} - T_{01}\right)}{\eta_c} = 292 + \frac{(555.92 - 292)}{0.85} = 602.49\,\text{K}$$

Work of compression:

$$W_c = C_p(T_{02} - T_{01}) = 1.005(602.49 - 292) = 312 \text{ kJ/kg}$$

Illustrative Example 5.2: In one stage of an axial flow compressor, the pressure ratio is to be 1.22 and the air inlet stagnation temperature is 288K. If the stagnation temperature rise of the stages is 21K, the rotor tip speed is 200 m/s, and the rotor rotates at 4500rpm, calculate the stage efficiency and diameter of the rotor.

Solution:
The stage pressure ratio is given by:

$$R_s = \left[1 + \frac{\eta_s \Delta T_{0s}}{T_{01}}\right]^{\frac{\gamma}{\gamma-1}}$$

or

$$1.22 = \left[1 + \frac{\eta_s(21)}{288}\right]^{3.5}$$

that is,

$$\eta_s = 0.8026 \quad \text{or} \quad 80.26\%$$

The rotor speed is given by:

$$U = \frac{\pi DN}{60}, \quad \text{or} \quad D = \frac{(60)(200)}{\pi(4500)} = 0.85 \text{ m}$$

Illustrative Example 5.3: An axial flow compressor has a tip diameter of 0.95 m and a hub diameter of 0.85 m. The absolute velocity of air makes an angle of 28° measured from the axial direction and relative velocity angle is 56°. The absolute velocity outlet angle is 56° and the relative velocity outlet angle is 28°. The rotor rotates at 5000rpm and the density of air is 1.2 kg/m^3. Determine:

1. The axial velocity.
2. The mass flow rate.
3. The power required.
4. The flow angles at the hub.
5. The degree of reaction at the hub.

Solution:
1. Rotor speed is given by:

$$U = \frac{\pi DN}{60} = \frac{\pi(0.95)(5000)}{60} = 249 \text{ m/s}$$

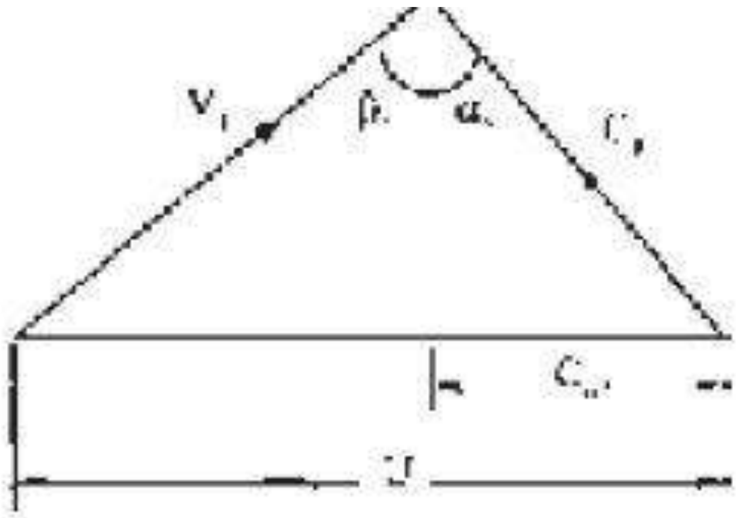


Figure 5.13 Inlet velocity triangle.

Blade speed at the hub:

$$U_h = \frac{\pi D_h N}{60} = \frac{\pi(0.85)(5000)}{60} = 223\,\text{m/s}$$

From the inlet velocity triangle (Fig. 5.13),

$$\tan\alpha_1 = \frac{C_{w1}}{C_a} \qquad \text{and} \qquad \tan\beta_1 = \frac{(U - C_{w1})}{C_a}$$

Adding the above two equations:

$$\frac{U}{C_a} = \tan\alpha_1 + \tan\beta_1$$

or:

$$U = C_a(\tan 28^\circ + \tan 56^\circ) = C_a(2.0146)$$

Therefore, $C_a = 123.6$ m/s (constant at all radii)

2. The mass flow rate:

$$\begin{aligned} \dot{m} &= \pi(r_t^2 - r_h^2)\rho C_a \\ &= \pi(0.475^2 - 0.425^2)(1.2)(123.6) = 20.98\,\text{kg/s} \end{aligned}$$

3. The power required per unit kg for compression is:

$$\begin{aligned} W_c &= \lambda U C_a(\tan\beta_1 - \tan\beta_2) \\ &= (1)(249)(123.6)(\tan 56^\circ - \tan 28^\circ)10^{-3} \\ &= (249)(123.6)(1.483 - 0.53) \\ &= 29.268\,\text{kJ/kg} \end{aligned}$$

The total power required to drive the compressor is:

$$W_c = \dot{m}(29.268) = (20.98)(29.268) = 614\,\text{kW}$$

4. At the inlet to the rotor tip:

$$C_{w1} = C_a \tan\alpha_1 = 123.6 \tan 28^{\circ} = 65.72\,\text{m/s}$$

Using free vortex condition, i.e., $C_w r$ = constant, and using h as the subscript for the hub,

$$C_{w1h} = C_{w1t}\frac{r_t}{r_h} = (65.72)\frac{0.475}{0.425} = 73.452\,\text{m/s}$$

At the outlet to the rotor tip,

$$C_{w2t} = C_a \tan\alpha_2 = 123.6 \tan 56^{\circ} = 183.24\,\text{m/s}$$

Therefore,

$$C_{w2h} = C_{w2t}\frac{r_t}{r_h} = (183.24)\frac{0.475}{0.425} = 204.8\,\text{m/s}$$

Hence the flow angles at the hub:

$$\tan\alpha_1 = \frac{C_{w1h}}{C_a} = \frac{73.452}{123.6} = 0.594 \text{ or, } \alpha_1 = 30.72^{\circ}$$

$$\tan\beta_1 = \frac{(U_h)}{C_a} - \tan\alpha_1 = \frac{223}{123.6} - 0.5942 = 1.21$$

i.e., $\beta_1 = 50.43^{\circ}$

$$\tan\alpha_2 = \frac{C_{w2h}}{C_a} = \frac{204.8}{123.6} = 1.657$$

i.e., $\alpha_2 = 58.89^{\circ}$

$$\tan\beta_2 = \frac{(U_h)}{C_a} - \tan\alpha_2 = \frac{223}{123.6} - \tan 58.59^{\circ} = 0.1472$$

i.e., $\beta_2 = 8.37^{\circ}$

5. The degree of reaction at the hub is given by:

$$\begin{aligned}\Lambda_h &= \frac{C_a}{2U_h}(\tan\beta_1 + \tan\beta_2) = \frac{123.6}{(2)(223)}(\tan 50.43^{\circ} + \tan 8.37^{\circ}) \\ &= \frac{123.6}{(2)(223)}(1.21 + 0.147) = 37.61\%\end{aligned}$$

Illustrative Example 5.4: An axial flow compressor has the following data:

Blade velocity at root:	140 m/s
Blade velocity at mean radius:	185 m/s
Blade velocity at tip:	240 m/s
Stagnation temperature rise in this stage:	15K
Axial velocity (constant from root to tip):	140 m/s
Work done factor:	0.85
Degree of reaction at mean radius:	50%

Calculate the stage air angles at the root, mean, and tip for a free vortex design.

Solution:

Calculation at mean radius:

From Eq. (5.1), $W_c = U(C_{w2} - C_{w1}) = U\Delta C_w$

or:

$$C_p(T_{02} - T_{01}) = C_p\Delta T_{0s} = \lambda U \Delta C_w$$

So:

$$\Delta C_w = \frac{C_p \Delta T_{0s}}{\lambda U} = \frac{(1005)(15)}{(0.85)(185)} = 95.87\,\text{m/s}$$

Since the degree of reaction (Fig. 5.14) at the mean radius is 50%, $\alpha_1 = \beta_2$ and $\alpha_2 = \beta_1$.

From the velocity triangle at the mean,

$$U = \Delta C_w + 2C_{w1}$$

or:

$$C_{w1} = \frac{U - \Delta C_w}{2} = \frac{185 - 95.87}{2} = 44.57\text{ m/s}$$

Hence,

$$\tan \alpha_1 = \frac{C_{w1}}{C_a} = \frac{44.57}{140} = 0.3184$$

that is,

$$\alpha_1 = 17.66° = \beta_2$$

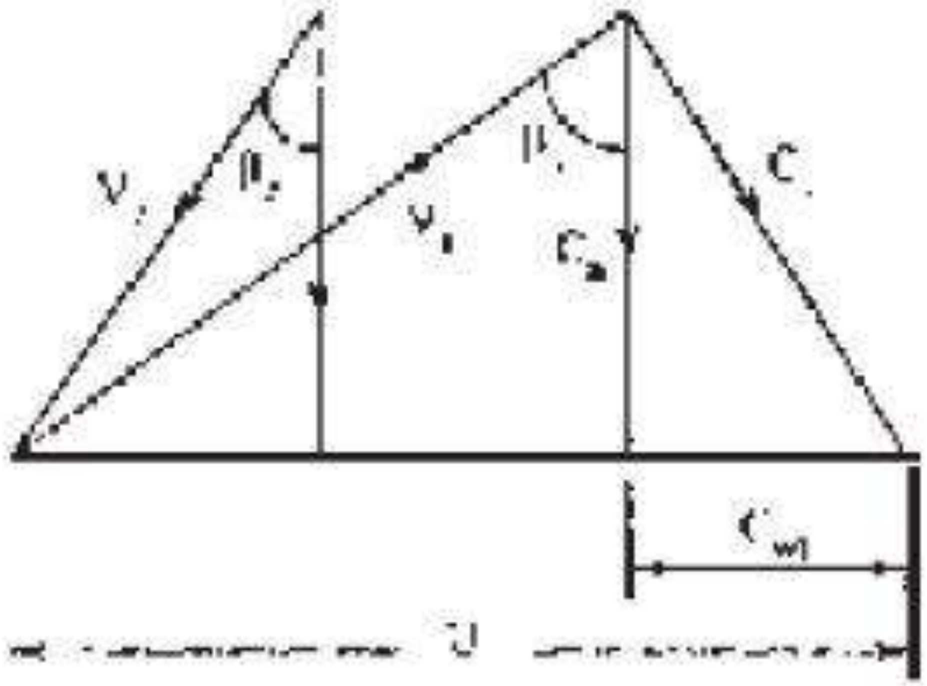


Figure 5.14 Velocity triangle at the mean radius.

and

$$\tan \beta_1 = \frac{(\Delta C_w + C_{w1})}{C_a} = \frac{(95.87 + 44.57)}{140} = 1.003$$

i.e., $\beta_1 = 45.09° = \alpha_2$

Calculation at the blade tip:
Using the free vortex diagram (Fig. 5.15),

$$(\Delta C_w \times U)_t = (\Delta C_w \times U)_m$$

Therefore,

$$\Delta C_w = \frac{(95.87)(185)}{240} = 73.9 \text{ m/s}$$

Whirl velocity component at the tip:

$$C_{w1} \times 240 = (44.57)(185)$$

Therefore:

$$C_{w1} = \frac{(44.57)(185)}{240} = 34.36 \text{ m/s}$$

$$\tan \alpha_1 = \frac{C_{w1}}{C_a} = \frac{34.36}{140} = 0.245$$

or:

$$\Delta C_w \times 140 = (95.87)(185) \quad \text{and} \quad \Delta C_w = 126.69\,\text{m/s}$$

Also:

$$(C_{w1} \times U)_r = (C_{w1} \times U)_m$$

or:

$$C_{w1} \times 140 = (44.57)(185) \quad \text{and} \quad C_{w1} = 58.9\,\text{m/s}$$

and

$$(C_{w2} \times U)_t = (C_{w2} \times U)_r$$

so:

$$C_{w2,tip} = C_a \tan\alpha_2 = 140 \tan 37.71^\circ = 108.24\,\text{m/s}$$

Therefore:

$$C_{w2,root} = \frac{(108.24)(240)}{140} = 185.55\,\text{m/s}$$

$$\tan\alpha_1 = \frac{58.9}{140} = 0.421$$

i.e., $\alpha_1 = 22.82^\circ$

From the velocity triangle at the blade root, (Fig. 5.16)

or:

$$x_2 = C_{w2} - U = 185.55 - 140 = 45.55$$

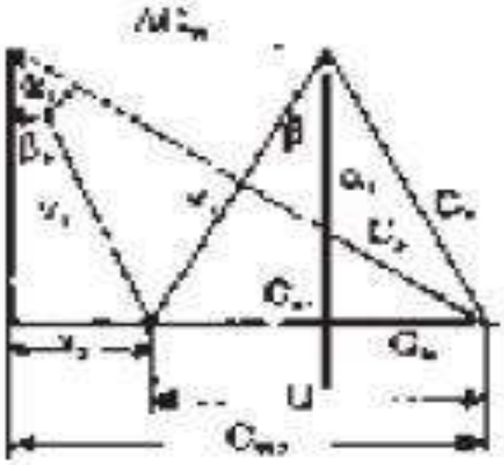


Figure 5.16 Velocity triangles at root.

Therefore:

$$\tan \beta_1 = \frac{U - C_{w1}}{C_a} = \frac{140 - 58.9}{140} = 0.579$$

i.e., $\beta_1 = 30.08°$

$$\tan \alpha_2 = \frac{C_{w2}}{C_a} = \frac{185.55}{140} = 1.325$$

i.e., $\alpha_2 = 52.96°$

$$\tan \beta_2 = -\frac{x_2}{C_a} = -\frac{45.55}{140} = -0.325$$

i.e., $\beta_2 = -18°$

Design Example 5.5: From the data given in the previous problem, calculate the degree of reaction at the blade root and tip.

Solution:

Reaction at the blade root:

$$\Lambda_{root} = \frac{C_a}{2U_r}(\tan \beta_{1r} + \tan \beta_{2r}) = \frac{140}{(2)(140)}(\tan 30.08^{\circ} + \tan(-18^{\circ}))$$

$$= \frac{140}{(2)(140)}(0.579 - 0.325) = 0.127, \text{ or } 12.7\%$$

Reaction at the blade tip:

$$\Lambda_{tip} = \frac{C_a}{2U_t}(\tan \beta_{1t} + \tan \beta_{2t}) = \frac{140}{(2)(240)}(\tan 55.75^{\circ} + \tan 43.26^{\circ})$$

$$= \frac{140}{(2)(240)}(1.469 + 0.941) = 0.7029, \text{ or } 70.29\%$$

Illustrative Example 5.6: An axial flow compressor stage has the following data:

Air inlet stagnation temperature:	295K
Blade angle at outlet measured from the axial direction:	32°
Flow coefficient:	0.56
Relative inlet Mach number:	0.78
Degree of reaction:	0.5

Find the stagnation temperature rise in the first stage of the compressor.

Solution:

Since the degree of reaction is 50%, the velocity triangle is symmetric as shown in Fig. 5.17. Using the degree of reaction equation [Eq. (5.12)]:

$$\Lambda = \frac{C_a}{2U}(\tan\beta_1 + \tan\beta_2) \quad \text{and} \quad \varphi = \frac{C_a}{U} = 0.56$$

Therefore:

$$\tan\beta_1 = \frac{2\Lambda}{0.56} - \tan 32^\circ = 1.16$$

i.e., $\beta_1 = 49.24^\circ$

Now, for the relative Mach number at the inlet:

$$M_{r1} = \frac{V_1}{(\gamma R T_1)^{\frac{1}{2}}}$$

or:

$$V_1^2 = \gamma R M_{r1}^2 \left(T_{01} - \frac{C_1^2}{2C_p}\right)$$

From the velocity triangle,

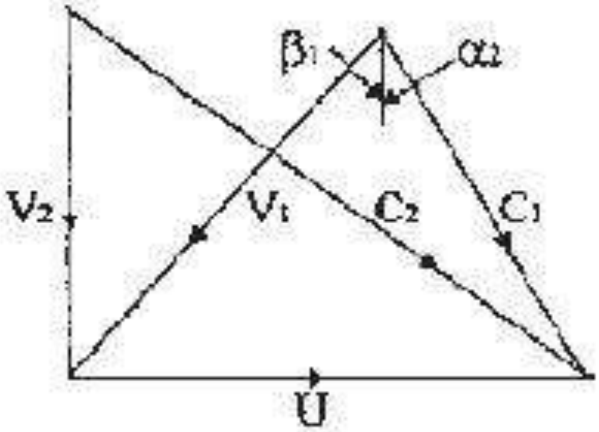
$$V_1 = \frac{C_a}{\cos\beta_1}, \quad \text{and} \quad C_1 = \frac{C_a}{\cos\alpha_1}$$

and:

$$\alpha_1 = \beta_2 (\text{since } \Lambda = 0.5)$$

Therefore:

$$C_1 = \frac{C_a}{\cos 32^\circ} = \frac{C_a}{0.848}$$



$= 1.128[(1523 - 753.2) - (1.005 - 285)] = 480.42$ kJ/kg of air

Heat supplied $= (c_{pg}T_3 - c_{pa}T_2') = 1.128 \times 1523 - 1.005 \times 673.98 = 1040.59$ **kJ/kg of air**

(a) Thermal efficiency $= \dfrac{\text{Net work}}{\text{Heat supplied}} = \dfrac{480.42}{1040.59} = 46.16\%$ ***Ans.***

(b) Specific output $= 480.42$ kW/kg of air ***Ans.***

(c) Fuel-air ratio $= \dfrac{\text{External heat supplied in kJ/kg of air}}{\text{Calorific value of fuel in kJ/kg of fuel}} = \dfrac{1040.59}{42000} = 0.0247$ ***Ans.***

(d) Specific air consumption $= \dfrac{1}{\text{Specific output}} = \dfrac{1}{480.42}$ kg/s kW

Specific fuel consumption = Specific air consumption × (F/A)

$= \dfrac{3600}{480.42} \times 0.0247 = 0.185$ kg/kWh ***Ans.***

Problem 17.4. An open cycle gas turbine plant operates with a pressure ratio of 4.5 while using 82 kg/min. of air and 1.4 kg/min. of fuel. The net output of the plant is 200 kW when 230 kW is needed to drive the compressor. Air enters the compressor at 1 bar and 15°C and combustion gases enters the turbine at 765°C. Assuming specific heats of air and combustion gases as 1.005 and 1.128 respectively, the index of compression 1.4, the index of expansion 1.34 and mechanical efficiency for both the compressor and the turbine 0.98 each, estimate ; (a) the isentropic compressor efficiency, (b) the isentropic turbine efficiency (c) overall thermal efficiency of the plant.

Solution. Refer to Fig. 17.8. Given : $p_1 = 1$ bar, $p_2 = 4.5$ bar, $T_1 = 288$K, $T_3 = 1038$ K, $w_{net} = 200$ kW, $w_{cw} = 230$ kW, $c_{pa} = 1.005$ kJ/kgK, $\dot{m}_a = 82$ kg/min, $c_{pg} = 1.128$ kJ/kgK, $\dot{m}_f = 1.4$ kg/min, $\dot{m}_g = 83.4$ kg/min, $\gamma_a = 1.4$, $\gamma_g = 1.34$, $\eta_{mt} = \eta_{mc} = 0.98$.

$$T_2 = T_1\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} = 288(4.5)^{(0.4)/1.4} = 442.61\text{K}$$

$$\text{Now,}\quad T_2' - T_1 = \frac{T_2 - T_1}{\eta_c} = \frac{442.61 - 288}{\eta_c} = \frac{154.61}{\eta_c}$$

$$\text{Power consumed by the compressor} = w_{cw} = \frac{\dot{m}_a c_{pa}(T_2 - T_1)}{\eta_{mc} \times \eta_c} = 230$$

$$\text{(a) or}\quad \eta_c = \frac{82 \times 1.005 \times 154.61}{0.98 \times 230 \times 60} = 94.21\%$$

Ans.

$$\text{Now,}\quad T_4 = \frac{T_3}{(4.5)^{.34/1.34}} = \frac{1038}{1.4646} = 708.69\text{K}$$

$$T_3 - T_4' = \eta_t(T_3 - T_4) = \eta_t(1038 - 708.69) = \eta_t \times 329.31$$

$$\text{Actual power developed by the turbine} = 230 + 200 = \frac{\eta_{mt} \times \dot{m}_g \times c_{pg} \times (T_3 - T_4)\eta_t}{60}\ \text{kW}$$

$$\text{or}\quad \eta_t = \frac{430 \times 60}{0.98 \times 83.9 \times 1.128 \times 329.31} = 84.979\%$$

Ans.

Now, $T_2' = 288 + \dfrac{154.61}{0.9421} = 452.46$ K

Heat supplied $= (\dot{m}_g \cdot c_{pg} T_3 - \dot{m}_a c_{pa} T_2')$ kg/min

$$= \frac{83.4 \times 1.128 \times 1038 - 82 \times 1.005 \times 452.46}{60} = 1019.75 \text{ kW}$$

Overall thermal efficiency $= \dfrac{\text{Net work}}{\text{Heat supplied}} = \dfrac{200}{1019.75} = 19.61\%$ ***Ans.***

Problem 17.5. A simple open cycle gas turbine takes in air at atmospheric pressure and 15°C and compresses air in the compressor upto 12 bar. Then air enters the combustion chamber and is heated to a maximum temperature of 1350°C, then it enters the turbine and expands to atmospheric pressure. If the isentropic efficiency of compressor and turbine is 0.86, combustion efficiency, is 0.97, fall of pressure through the combustion system is 0.3 bar, c_p for both air and gas 1.005, $\gamma = 1.4$. Determine the flow of air and gas for net power of 200MW developed. Calculate also the heat supplied per kg of air, work ratio, thermal efficiency and specific fuel consumption if C.V. of fuel is 42000 kJ/kg

Solution. Refer to Fig. 17.9. Given : $p_1 = 1.013$ bar, $T_1 = 288$K, $T_3 = 1623$K, $p_4 = 1.013$ bar; $p_2 = 12$ bar, $p_3 = 11.7$ bar, $\eta_c = \eta_t = 0.86$, $\eta_{comb} = 0.97$, Power = 200 MW

$$T_2 = T_1\left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} = 288 \times \left(\frac{12}{1.013}\right)^{(0.4)/1.4} = 288 \times 2.0264 = 583.61\text{K}$$

$$T_2' = T_1 + \frac{T_2 - T_1}{\eta_c} = 288 + \frac{583.61 - 288}{0.86} = 288 + 343.73 = 613.73 \text{ K}$$

$$T_4 = \frac{T_3}{(p_3/p_4)^{(\gamma-1)/\gamma}} = \frac{1623}{(11.7/1.013)^{(0.4)/1.4}} = 810.59\text{K}$$

and $\quad T_3 - T_4' = \eta_t(T_3 - T_4) = 0.86(1623 - 810.59) = 698.67$

Net work = Turbine work – Compressor work = $c_p(T_3 - T_4') - c_p(T_2' - T_1)$

$= 1.005\,(698.67 - 343.73) = 356.71$ kJ/kg of air

Power developed $= 200 \times 10^3 = \dot{m}_a \times 356.71$ where $\dot{m}_a$ is the rate of flow of air

or $\quad \dot{m}_a = 560.67$ kg/s ***Ans.***

Effective heat supplied $= c_p(T_3 - T_2')$ kJ/kg $= 1.005(1623 - 613.73) = 991.27$ kJ/kg of air

Actual heat supplied $= q_{Aa} = \dfrac{991.27}{0.97} = 1021.92$ kJ/kg of air ***Ans.***

Work ratio $= \dfrac{\text{Actual work}}{\text{Turbine work}} = \dfrac{356.71}{698.67} = 51.05\%$ ***Ans.***

Actual thermal efficiency $= \dfrac{w_{net}}{q_{Aa}} = \dfrac{356.71}{1021.92} = 34.9\%$ ***Ans.***

compressor and alternator respectively.

$$\frac{T_3}{T_4} = \left(\frac{p_2}{p_1}\right)^{(0.4)/1.4} = (14)^{0.286} = 2.127 \text{ or } T_4 = \frac{1523}{2.127} = 716.03\text{K}$$

For process $3-4'$, $T_3 - T_4' = \eta_t(T_3 - T_4) = 0.88(1523 - 716.03) = 710.13$ K

$$\therefore \quad T_4' = 1523 - 710.13 = 812.87\text{K}$$

For process $3-4''$, $T_3 - T_4'' = 0.86(1523 - 716.03) = 693.99$K

$$\therefore \quad T_4'' = 1523 - 693.99 = 829\text{K}$$

Let $\dot{m}_{a1}$ of air be required by the turbine which drives the compressor for 1 kg of air compressed, hence, $1 \times c_{pa}(T_2' - T_1) = \dot{m}_{1a} \times c_{pg}(T_3 - T_4')$

$$\therefore \quad \dot{m}_{a1} = \frac{c_{pa}}{c_{pg}} \cdot \frac{(T_2' - T_1)}{(T_3 - T_4')} = \frac{1.005 \times 376.88}{1.11 \times 710.13} = 0.4805 \text{ kg/kg of air}$$

Hence, the amount of air which passes through the power turbine

$= 1 - 0.4805 = 0.5195$ kg

(*a*) Ratio of air which passes through the power turbine to total air compressed

$$= \frac{0.5195}{1} = 0.5195 \qquad \textbf{Ans.}$$

Output per kg of air compressed

$= \dot{m}_{a1}\, c_{pg}(T_3 - T_4'') = 0.5195 \times 1.11 \times 693.99 = 400.18$ kW

Now, $\dot{m}_a$ kg of air per sec, is compressed and the alternator output is 120 MW with 0.98 efficiency, hence $\dot{m}_a \times 400.18 = \dfrac{120}{0.98}$

$$(b) \therefore \; \dot{m}_a = \frac{120 \times 10^3}{400.18 \times 0.98} = 305.98 \text{ kg/s} \qquad \textbf{Ans.}$$

(*c*) Heat supplied $= c_{pg}\, T_3 - c_{pa}\, T_2' = 1.11 \times 1523 - 1.005 \times 664.88 = 1022.23$ kJ/kg.

$$\text{Thermal efficiency} = \frac{\text{Work output}}{\text{Heat supplied}} = \frac{400.18}{1022.23} = 39.14\% \qquad \textbf{Ans.}$$

17.6. Optimum Pressure Ratio for Maximum Specific Output in Actual Simple Gas Turbine Cycle

Referring to Fig. 17.5 (without pressure loss), we have

$$\frac{T_2}{T_1} = \frac{T_3}{T_4} = \left(\frac{p_2}{p_1}\right)^{(\gamma-1)/\gamma} = \left(\frac{p_3}{p_4}\right)^{(\gamma-1)/\gamma} = (r_p)^{(\gamma-1)/\gamma} = x$$

$$\text{Actual compressor work} = w_{ca} = (h_2' - h_1) = \frac{(h_2 - h_1)}{\eta_c} = \frac{c_p(T_2 - T_1)}{\eta_c} \text{ kJ/kg}$$

$$\text{Actual turbine work} = w_{ta} = (h_3 - h_4') = (h_3 - h_4)\eta_t = c_p(T_3 - T_4)\eta_t \text{ kJ/kg}$$

$$w_{neto} = c_p\left(T_3 - T_4\right)\eta_t - \frac{c_p\,(T_2 - T_1)}{\eta_c} = c_p\,\eta_t\;T_3(1 - \frac{T_4}{T_3}) - \frac{c_pT_1}{\eta_c}\left(\frac{T_2}{T_1} - 1\right)$$

$$= c_p\,\eta_t\,T_3\,(1 - \frac{1}{x}) - \frac{c_pT_1}{\eta_c}(x - 1) \qquad (17.16)$$

For maximum output, differentiating w_{net} w.r.t. to x and equating to zero

$$\frac{dw_{neto}}{dx} = 0 = c_p.\,\eta_t.\;T_3\left(\frac{1}{x^2}\right) - \frac{c_p.T_1}{\eta_c}$$

So, $\quad x = \sqrt{\left(\eta_t.\;\eta_c.\frac{T_3}{T_1}\right)} \quad$ or $\quad (r_p)^{(\gamma-1)/\gamma} = \left(\eta_t.\;\eta_c.\frac{T_3}{T_1}\right)^{1/2}$

or $\quad (r_p)_{opt} = \left(\eta_t.\;\eta_c.\frac{T_3}{T_1}\right)^{\gamma/2(\gamma-1)} = \left(\eta_t.\eta_c.\frac{T_{max}}{T_{min}}\right)^{\gamma/2(\gamma-1)} \qquad (17.17)$

For ideal simple cycle, $(r_p)_{opt} = \left(\frac{\mathrm{T}_{max}}{\mathrm{T}_{min}}\right)^{\gamma/2(\gamma-1)}$

17.7. Optimum Pressure Ratio for Maximum Cycle Thermal Efficiency.

From the above article the net output = $w_{neto} = c_p\eta_t\,T_3(1 - \frac{1}{x}) - \frac{c_pT_1}{\eta_c}(x - 1)$

Heat supplied $= c_p(T_3 - T_2') \approx c_p(T_3 - T_2)$. This approximation is only for simplification of the problem without any appreciable error.

Let $\quad \frac{T_3}{T_1} = y \;$ and $\; \frac{T_3}{T_4} = \frac{T_2}{T_1} = (r_p)^{(\gamma-1)/\gamma} = x$

So, the heat supplied $= c_p(T_3 - T_1\,.\,x) = c_p\,T_1(y - x)$

$$\eta_{tho} = \frac{\eta_tT_3(1 - 1/x) - T_1(x - 1)/\eta_c}{T_1(y - x)} = \frac{\eta_t(1 - 1/x)\,y - (x - 1)/\eta_c}{(y - x)}$$

For maximum thermal efficiency,

$$\frac{d\eta_{tho}}{dx} = 0 = \left\{\eta_t y\left(\frac{1}{x^2}\right)(y - x) - \frac{1}{\eta_c}(y - x)\right\}$$

$$- \left(\eta_t y\left(1 - \frac{1}{x}\right) - \frac{1}{\eta_c}(x - 1)\right)(-1)$$

or $\quad \left[\frac{\eta_c\,\eta_t\,.\,y - x^2}{x^2\,\eta_c}\right](y - x) + (x - 1)\left[\frac{\eta_c\,.\,\eta_t\,.\,y - x}{x\,.\,\eta_c}\right] = 0$

or $\quad \eta_c\,.\,\eta_t y^2 - yx^2 - \eta_c\eta_t yx - x^3 = x^3 - x^2\eta_c\eta_t\,.\,y - x^3 + x\,.\,y\,.\,\eta_c\eta_t$

or $\quad x^3[y(\eta_c\eta_t - 1) + 1] - 2\eta_c\eta_t yx + \eta_c\eta_t y^2 = 0$

The **turbofan** or **fanjet** is a type of airbreathing jet engine that is widely used in aircraft propulsion. The word "turbofan" is a portmanteau of "turbine" and "fan": the *turbo* portion refers to a gas turbine engine which achieves mechanical energy from combustion,[1] and the *fan*, a ducted fan that uses the mechanical energy from the gas turbine to accelerate air rearwards. Thus, whereas all the air taken in by a turbojetpasses through the turbine (through the combustion chamber), in a turbofan some of that air bypasses the turbine. A turbofan thus can be thought of as a turbojet being used to drive a ducted fan, with both of those contributing to the thrust. The ratio of the mass-flow of air bypassing the engine core compared to the mass-flow of air passing through the core is referred to as the bypass ratio. The engine produces thrust through a combination of these two portions working together; engines that use more jet thrust relative to fan thrust are known as *low-bypass turbofans*, conversely those that have considerably more fan thrust than jet thrust are known as *high-bypass*. Most commercial aviation jet engines in use today are of the high-bypass type,[2][3] and most modern military fighter engines are low-bypass.[4][5] Afterburnersare not used on high-bypass turbofan engines but may be used on either low-bypass turbofan or turbojet engines.

Most of the air flow through a high-bypass turbofan is low-velocity bypass flow: even when combined with the much higher velocity engine exhaust, the average exhaust velocity is considerably lower than in a pure turbojet. Turbojet engine noise is predominately jet noise from the high exhaust velocity, therefore turbofan engines are significantly quieter than a pure-jet of the same thrust with jet noise no longer the predominant source. Other noise sources are the fan, compressor and turbine.[6] Jet noise is reduced by using chevrons - sawtooth patterns on the exhaust nozzles - on the Rolls-Royce Trent 1000 and General Electric GEnx engines, which are used on the Boeing 787. [7]

Since the efficiency of propulsion is a function of the relative airspeed of the exhaust to the surrounding air, propellers are most efficient for low speed, pure jets for high speeds, and ducted fans in the middle. Turbofans are thus the most efficient engines in the range of speeds from about 500 to 1,000 km/h (310 to 620 mph), the speed at which most commercial aircraft operate.[8][9] Turbofans retain an efficiency edge over pure jets at low supersonic speeds up to roughly Mach 1.6 (1,960.1 km/h; 1,217.9 mph).

Modern turbofans have either a large single-stage fan or a smaller fan with several stages. An early configuration combined a low-pressure turbine and fan in a single rear-mounted unit.

Low-bypass turbofan

A high-specific-thrust/low-bypass-ratio turbofan normally has a multi-stage fan, developing a relatively high pressure ratio and, thus, yielding a high (mixed or cold) exhaust velocity. The core airflow needs to be large enough to give sufficient core power to drive the fan. A smaller core flow/higher bypass ratio cycle can be achieved by raising the (HP) turbine rotor inlet temperature.

To illustrate one aspect of how a turbofan differs from a turbojet, they may be compared, as in a re-engining assessment, at the same airflow (to keep a common intake for example) and the same net thrust (i.e. same specific thrust). A bypass flow can be added only if the turbine inlet temperature is not too high to compensate for the smaller core flow. Future improvements in turbine cooling/material technology can allow higher turbine inlet temperature, which is necessary because of increased cooling air temperature, resulting from an overall pressure ratio increase.

The resulting turbofan, with reasonable efficiencies and duct loss for the added components, would probably operate at a higher nozzle pressure ratio than the turbojet, but with a lower exhaust temperature to retain net thrust. Since the temperature rise across the whole engine (intake to nozzle) would be lower, the (dry power) fuel flow would also be reduced, resulting in a better specific fuel consumption (SFC).

Some low-bypass ratio military turbofans (e.g. F404) have variable inlet guide vanes to direct air onto the first fan rotor stage. This improves the fan surge margin (see compressor map).

High-bypass turbofan

The low-specific-thrust/high-bypass-ratio turbofans used in today's civil jetliners (and some military transport aircraft) evolved from the high-specific-thrust/low-bypass-ratio turbofans used in such aircraft back in the 1960s

Low specific thrust is achieved by replacing the multi-stage fan with a single-stage unit. Unlike some military engines, modern civil turbofans do not have any stationary inlet guide vanes in front of the fan rotor. The fan is scaled to achieve the desired net thrust.

The core (or gas generator) of the engine must generate sufficient core power to at least drive the fan at its design flow and pressure ratio. Through improvements in turbine cooling/material technology, a higher (HP) turbine rotor inlet temperature can be used, thus facilitating a smaller (and lighter) core and (potentially) improving the core thermal efficiency. Reducing the core mass flow tends to increase the load on the LP turbine, so this unit may require additional stages to reduce the average stage loading and to maintain LP turbine efficiency. Reducing core flow also increases bypass ratio. Bypass ratios greater than 5:1 are increasingly common with the Pratt & Whitney PW1000G attaining 12.5:1.

Further improvements in core thermal efficiency can be achieved by raising the overall pressure ratio of the core. Improved blade aerodynamics reduces the number of extra compressor stages required. With multiple compressors (i.e., LPC, IPC, and HPC) dramatic increases in overall pressure ratio have become possible. Variable geometry (i.e., stators) enable high-pressure-ratio compressors to work surge-free at all throttle settings.

Cutaway diagram of the General Electric CF6-6 engine

The first (experimental) high-bypass turbofan engine was built and run on February 13, 1964 by AVCO-Lycoming.[13][14] Shortly after, the General Electric TF39 became the first production model, designed to power the Lockheed C-5 Galaxy military transport aircraft.[9] The civil General Electric CF6 engine used a derived design. Other high-bypass turbofans are the Pratt & Whitney JT9D, the three-shaft Rolls-Royce RB211 and the CFM International CFM56; also the smaller TF34. More recent large high-bypass turbofans include the Pratt & Whitney PW4000, the three-shaft Rolls-Royce Trent, the General Electric GE90/GEnxand the GP7000, produced jointly by GE and P&W.

For reasons of fuel economy, and also of reduced noise, almost all of today's jet airliners are powered by high-bypass turbofans. Although modern combat aircraft tend to use low-bypass ratio turbofans, military transport aircraft (e.g., C-17) mainly use high-bypass ratio turbofans (or turboprops) for fuel efficiency.

The lower the specific thrust of a turbofan, the lower the mean jet outlet velocity, which in turn translates into a high thrust lapse rate (i.e. decreasing thrust with increasing flight speed). See technical discussion below, item 2. Consequently, an engine sized to propel an aircraft at high subsonic flight speed (e.g., Mach 0.83) generate a relatively high thrust at low flight speed, thus enhancing runway performance. Low specific thrust engines tend to have a high bypass ratio, but this is also a function of the temperature of the turbine system.

The turbofans on twin engined airliners are further more powerful to cope with losing one engine during take-off, which reduces the aircraft's net thrust by half. Modern twin engined airliners normally climb very steeply immediately after take-off. If one engine is lost, the climb-out is much shallower, but sufficient to clear obstacles in the flightpath.

The Soviet Union's engine technology was less advanced than the West's and its first wide-body aircraft, the Ilyushin Il-86, was powered by low-bypass engines. The Yakovlev Yak-42, a medium-range, rear-engined aircraft seating up to 120 passengers introduced in 1980 was the first Soviet aircraft to use high-bypass engines.

Afterburning turbofan

Since the 1970s, most jet fighter engines have been low/medium bypass turbofans with a mixed exhaust, afterburner and variable area final nozzle. An afterburner is a combustor located downstream of the turbine blades and directly upstream of the nozzle, which burns fuel from afterburner-specific fuel injectors. When lit, prodigious amounts of fuel are burnt in the afterburner, raising the temperature of exhaust gases by a significant degree, resulting in a higher exhaust velocity/engine specific thrust. The variable geometry nozzle must open to a larger throat area to accommodate the extra volume flow when the afterburner is lit. Afterburning is often designed to give a significant thrust boost for take off, transonic acceleration and combat maneuvers, but is very fuel intensive. Consequently, afterburning can be used only for short portions of a mission.

Unlike the main combustor, where the downstream turbine blades must not be damaged by high temperatures, an afterburner can operate at the ideal maximum (stoichiometric) temperature (i.e., about 2100K/3780Ra/3320F/1826C). At a fixed total applied fuel:air ratio, the total fuel flow for a given fan airflow will be the same, regardless of the dry specific thrust of the engine. However, a high specific thrust turbofan will, by definition, have a higher nozzle pressure ratio, resulting in a higher afterburning net thrust and, therefore, a lower afterburning specific fuel consumption (SFC). However, high specific thrust engines have a high dry SFC. The situation is reversed for a medium specific thrust afterburning turbofan: i.e., poor afterburning SFC/good dry SFC. The former engine is suitable for a combat aircraft which must remain in afterburning combat for a fairly long period, but has to fight only fairly close to the airfield (e.g. cross border skirmishes) The latter engine is better for an aircraft that has to fly some distance, or loiter for a long time, before going into combat. However, the pilot can afford to stay in afterburning only for a short period, before aircraft fuel reserves become dangerously low.

The first production afterburning turbofan engine was the Pratt & Whitney TF30, which initially powered the F-111 Aardvark and F-14 Tomcat. Current low-bypass military turbofans include the Pratt & Whitney F119, the Eurojet EJ200, the General Electric F110, the Klimov RD-33, and the Saturn AL-31, all of which feature a mixed exhaust, afterburner and variable area propelling nozzle.

Cycle improvements

Consider a mixed turbofan with a fixed bypass ratio and airflow. Increasing the overall pressure ratio of the compression system raises the combustor entry temperature. Therefore, at a fixed fuel flow there is an increase in (HP) turbine rotor inlet temperature. Although the higher temperature rise across the compression system implies a larger temperature drop over the turbine system, the mixed nozzle temperature is unaffected, because the same amount of heat is being added to the system. There is, however, a rise in nozzle pressure, because overall pressure ratio increases faster than the turbine expansion ratio, causing an increase in the hot mixer entry pressure. Consequently, net thrust increases, whilst specific fuel consumption (fuel flow/net thrust) decreases. A similar trend occurs with unmixed turbofans.

So turbofans can be made more fuel efficient by raising overall pressure ratio and turbine rotor inlet temperature in unison. However, better turbine materials and/or improved vane/blade cooling are required to cope with increases in both turbine rotor inlet temperature and compressor delivery temperature. Increasing the latter may require better compressor materials.

Overall pressure ratio can be increased by improving fan (or) LP compressor pressure ratio and/or HP compressor pressure ratio. If the latter is held constant, the increase in (HP) compressor delivery temperature (from raising overall pressure ratio) implies an increase in HP mechanical speed. However, stressing considerations might limit this parameter, implying, despite an increase in overall pressure ratio, a reduction in HP compressor pressure ratio.

According to simple theory, if the ratio of turbine rotor inlet temperature/(HP) compressor delivery temperature is maintained, the HP turbine throat area can be retained. However, this assumes that cycle improvements are obtained, while retaining the datum (HP) compressor exit flow function (non-dimensional flow). In practice, changes to the non-dimensional speed of the (HP) compressor and cooling bleed extraction would probably make this assumption invalid, making some adjustment to HP turbine throat area unavoidable. This means the HP turbine nozzle guide vanes would have to be different from the original. In all probability, the downstream LP turbine nozzle guide vanes would have to be changed anyway.

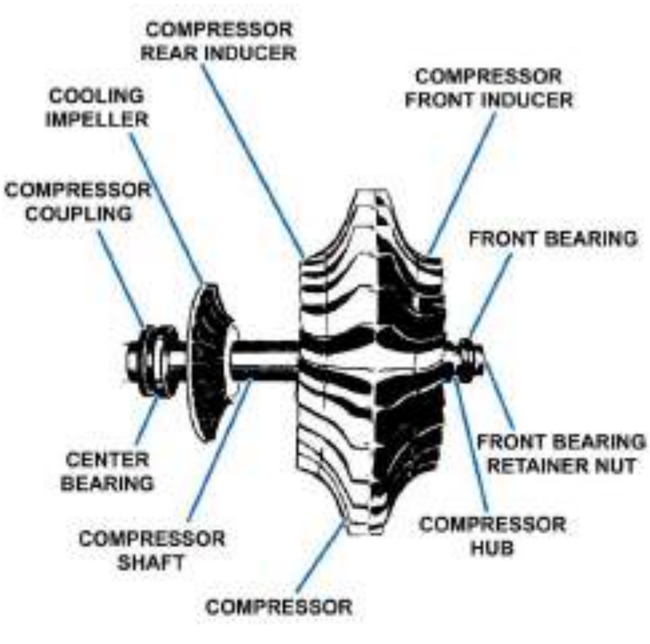
COMPRESSOR SECTION

The primary function of the compressor is to supply air in enough quantity to satisfy the requirements of the combustion burners. Specifically, the compressor increases the air mass received from the air inlet duct and directs it to the burners in the quantity and at the pressures required. A secondary function is to supply compressor bleed air for various purposes in the engine and aircraft. The compressor provides space for mounting accessories and engine parts. There are two basic types of compressors. The compressor type is also the engine type, so a centrifugal-flow compressor is in a centrifugal engine. Centrifugal-flow compressors have a compression ratio of 5:1. Present-day axial flow compressors have compression ratios approaching 15:1 and airflows up to 350 lb. The addition of a fan raises these values to 25:1 and 1,000 lb. /sec.

Centrifugal-Flow Compressors

The single entry centrifugal-flow compressor *(Figure 1-16)* consists of an impeller (rotor element), a diffuser (stator element), and a manifold. The impeller picks up and accelerates air outward to the diffuser. The diffuser directs air into the manifold. The manifold distributes air into the combustion section.

Double entry centrifugal-flow compressors *(Figure 1-17)* handle the same airflow with a smaller diameter. Small multi-stage centrifugal-flow engines used in aircraft *(Figure 1-18),* or as Auxiliary Power Units (APUs) that take advantage of this feature.



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axial-Flow Compressors

The term *axial flow* applies to the axial (straight-line) flow of air through the compressor section of the engine. The axial-flow compressor has two main elements—a rotor and a stator. Each consecutive pair of rotor and stator blades makes a pressure stage. The rotor is a shaft with blades attached to it. These blades impel air rearward in the same manner as a propeller, by reason of their angle and airfoil contour. The rotor, turning at high speed, takes in air at the compressor inlet and impels it through a series of stages. The action of the rotor increases the compression of the air. At each stage it accelerates rearward. The stator blades act as diffusers, partially converting high velocity to pressure. Maintaining high efficiency requires small changes in the rate of diffusion at each stage. The number of stages depends on the amount of air and total pressure rise required. A greater number of stages means a higher compression ratio. Most present day engines use from 10 to 16 stages.

An axial-flow compressor follows the same rules and has the same limitations as an aircraft wing. The concept is more complicated than a single airfoil, because the blades are close together. Each trailing edge blade affects the next leading edge. This cascade effect is of prime importance in determining blade design and placement. The axial-flow compressor has

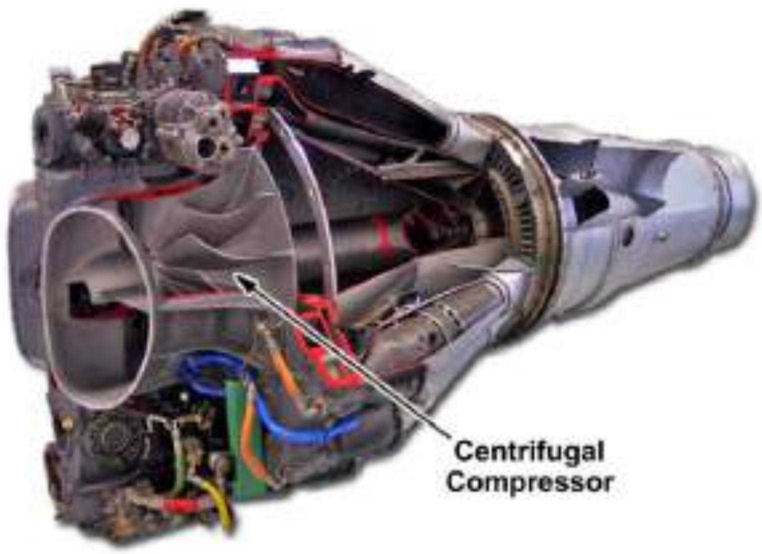


Figure 1-18 — Centrifugal-flow engine

its disadvantages, the most important of which is the stall problem. If, for some reason, the angle of attack—the angle at which the airflow strikes the rotor blades—becomes too low, the pressure zones, shown in *Figure 1-19*, will be of low value, and the airflow and compression will be low. If the angle of attack is high, the pressure zones will be high, and the airflow and compression ratio will be high. If the angle of attack is too high, the compressor will stall.

The airflow over the upper foil surface will become turbulent and destroy the pressure zones. This will decrease the compression airflow. The angle of attack will vary with engine rpm, compressor-inlet temperature, and compressor discharge or burner pressure. Any action that decreases airflow relative to engine speed will increase the angle of attack and increase the tendency to stall.

The decrease in airflow may result from a too-high compressor-discharge pressure. During ground operation of the engine, the prime action that causes a stall is choking. If there is a decrease in the engine speed, the compression ratio will decrease with the lower rotor velocities. With a decrease in compression, the volume of air in the rear of the compressor will be greater. This excess volume of air causes a choking action in the rear of the compressor with a decrease in airflow. This, in turn, decreases the air velocity in the front of the compressor and increases the tendency to stall. If no corrective action is taken, the front of the compressor will stall at low engine speeds.

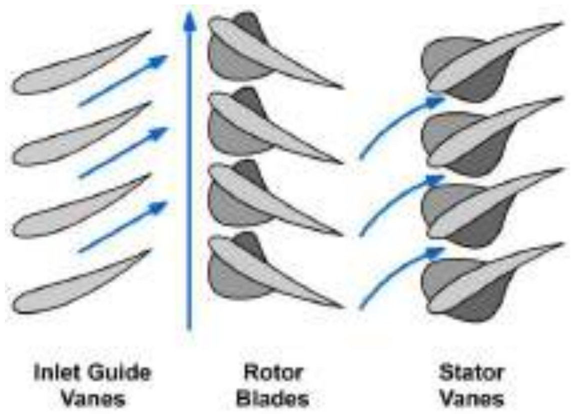
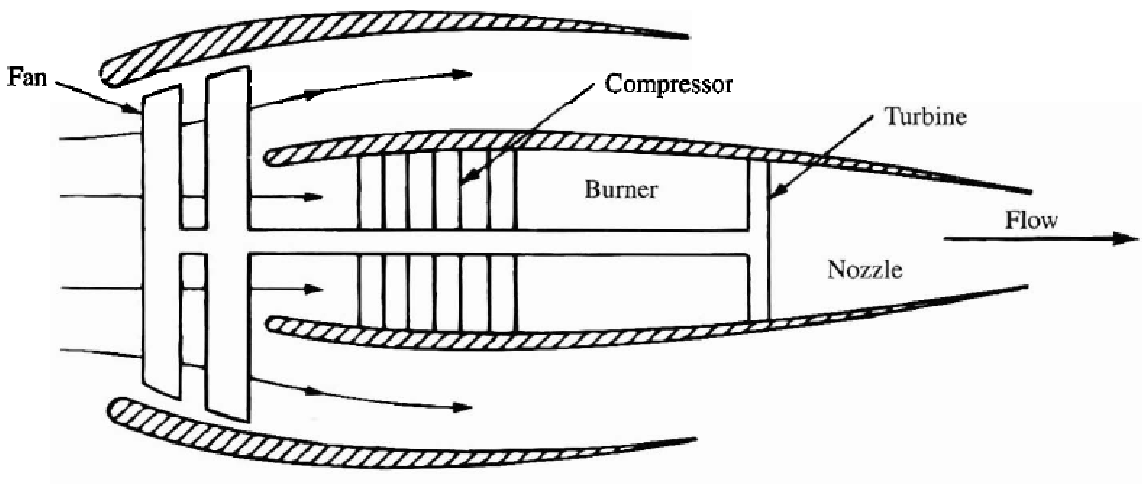


Figure 1-19 — The cascade effect.

THE TURBOFAN ENGINE

The ***turbofan*** engine is a propulsive mechanism to combine the high thrust of a turbojet with the high efficiency of a propeller. Basically, a turbojet engine forms the ***core*** of the turbofan; the core contains the diffuser, compressor, burner, turbine, and nozzle. However, in the turbofan engine, the turbine drives not only the compressor, but also a large fan ***external*** to the core. The fan itself is contained in a shroud that is wrapped around the core. The flow through a turbofan engine is split into two paths. One passes through the fan and flows externally over the core; this air is processed only by the fan, which is acting in the manner of a sophisticated, shrouded propeller. The propulsive thrust obtained from this flow through the fan is generated with an efficiency approaching that of a propeller. The second air path is through the core itself. The propulsive thrust is obtained from the flow through the core is generated with an efficiency associated with a turbojet. The overall propulsive efficiency of a turbofan is therefore a compromise between that of a propeller and that of a turbojet.



This compromise has been found to be quite successful- the vast majority of jet-propelled airplanes today are powered by turbofan engines.

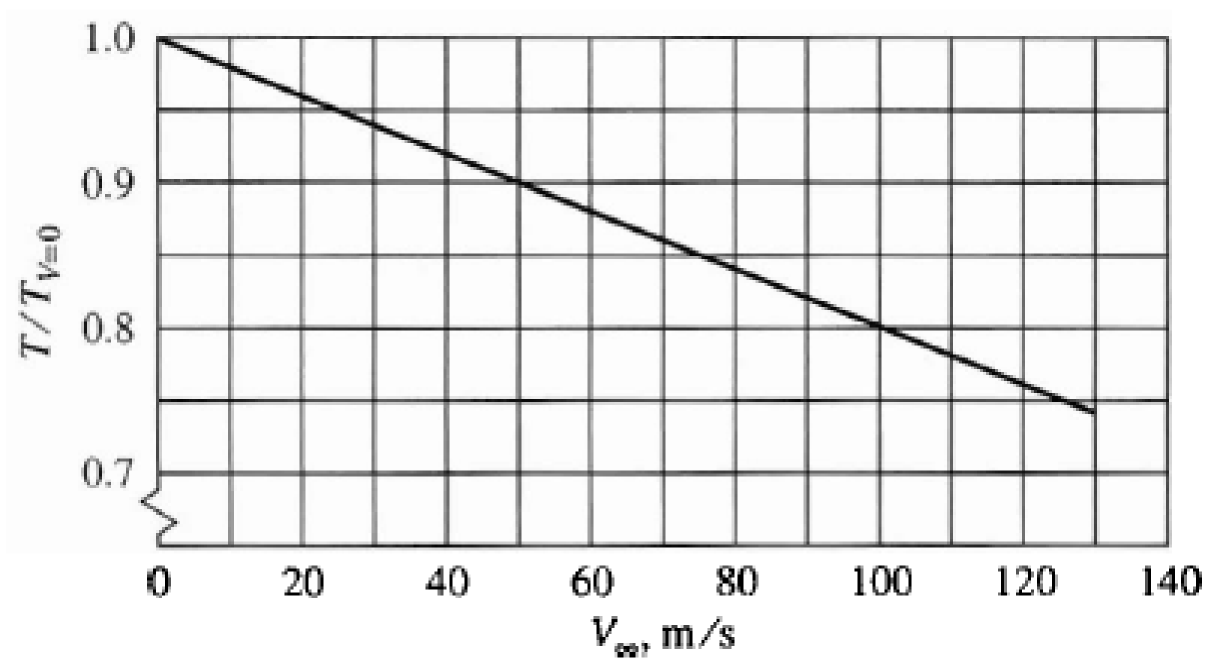
An important parameter of a turbofan engine is the bypass ***ratio,*** defined as the mass flow passing through the fan, externally to the core divided by the mass flow through the core itself. Everything else being equal, the higher the bypass ratio, the higher the propulsive efficiency. For the large turbofan engines that power airplanes such as the Boeing 747, for example, the Rolls-Royce RB211 and the Pratt & Whitney JTBD, the bypass ratios are on the order of 5. Typical values of the thrust specific fuel consumption for these turbofan engines are 0.6 lb/(lb h) almost half that of a conventional turbojet engine.

Variations of Thrust and Specific Fuel Consumption with Velocity and Altitude

For high-bypass-ratio turbofans-those with bypass ratios on the order of 5 (these are the class of turbofans that power civil transports) the performance seems to be closer to that of a propeller than that of a turbojet in some respects. The thrust of a civil turbofan engine has a strong variation with velocity; thrust decreases as V_∞ increases:

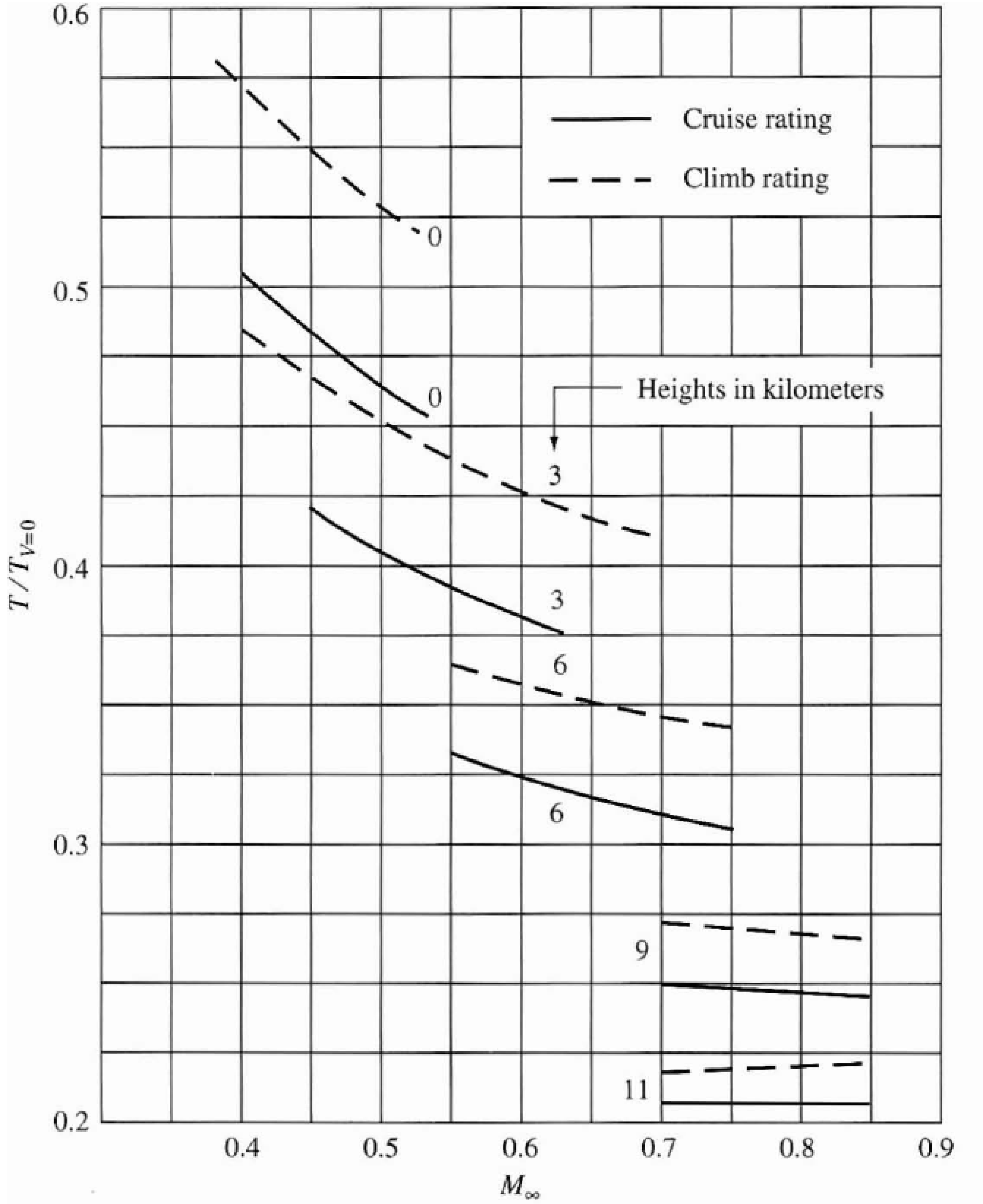
$$\frac{T}{T_{V=0}} = 1 - 2.52 \times 10^{-3} V_\infty + 4.34 \times 10^{-6} V_\infty^2$$

The equation holds for V_∞ < 130 m/s.



At higher subsonic velocities for a given, constant altitude, the decrease in thrust with Mach number can be correlated by

$$\frac{T}{T_{V=0}} = A M_\infty^{-n}$$



Although the variation of T for a civil turbofan is a strong function of V, (or Ma) at lower altitudes, at the relatively high altitude of 11 **km,** T is relatively constant for the narrow Mach number range from 0.7 to 0.85. This corresponds to normal cruise Mach numbers for civil transports such as the Boeing 747. Hence, for the analysis of airplane performance in the cruise range, it appears reasonable to assume T= constant.

The variation of T with altitude is approximated by

$$\frac{T}{T_0} = \left(\frac{\rho}{\rho_0}\right)^m$$

The variation of thrust specific fuel consumption with both altitude and Mach number is shown in Fig. The ratio of the thrust specific fuel consumption at the specified altitude and Mach number, to the value at zero velocity and at sea level, is shown. The variation with velocity at a given altitude follows the relation:

$$c_t = B(1 + kM_\infty)$$

where B and k are empirical constants

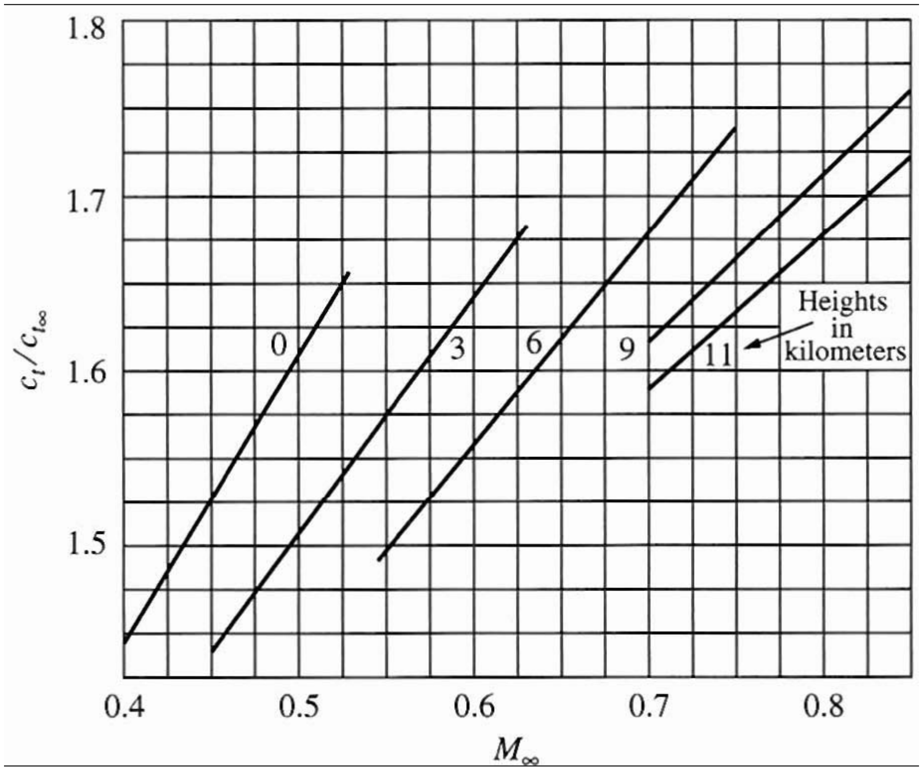
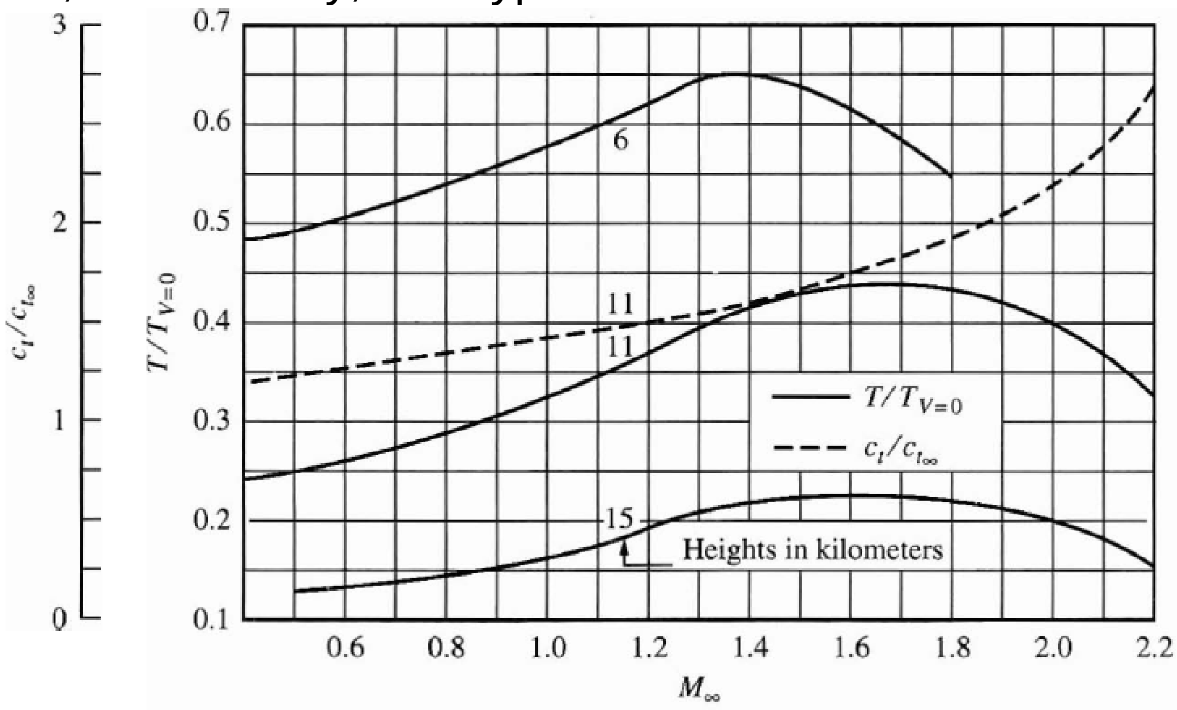


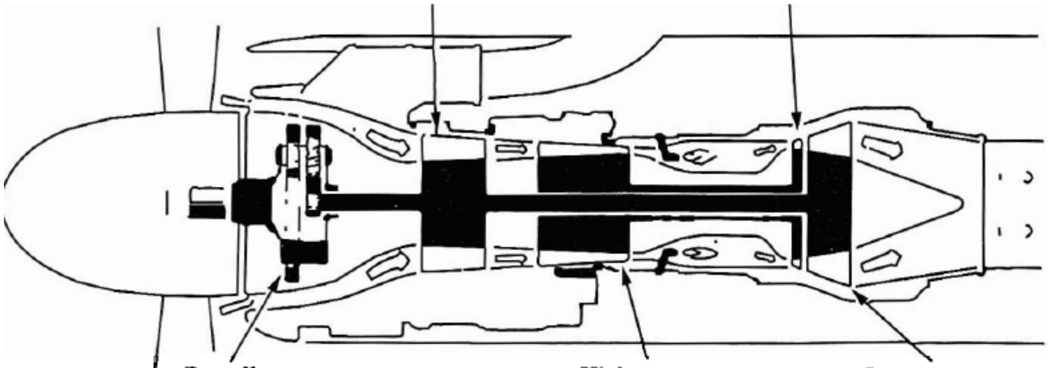
Figure shows why turbofans were not used on the Concorde supersonic transport, with its cruising Mach number of 2.2. The thrust specific fuel consumption of a turbojet engine is almost constant with speed **in** the supersonic regime. However, for a turbofan, ***ct*** increases markedly with an increase in Ma. For this reason, a turbojet is more fuel-efficient than a turbofan if the design Mach number is 2.2. The ordinate in Figure is expanded. Hence, the altitude effect on ***ct,*** looks larger than it really is. To first order, is constant with altitude.

For low-bypass-ratio turbofans-those with bypass ratios between 0 and 1-the performance is somewhat different from that for the high-bypass-ratio case discussed above. The performance of low-bypass-ratio turbofans is much closer to that of a turbojet than that of a propeller. Typical generic variations of $T/T_{v=0}$ and $c_t/c_{t\infty}$ versus Ma, for a military, low-bypass-ratio turbofan are



after a small initial decrease at low subsonic Mach numbers, the thrust increases for increasing Mach number well above Mach 1. The dashed line gives the variation of thrust specific fuel consumption versus Mach number for a military turbofan. Note that **ct,** for the low-bypass-ratio turbofan gradually increases as M, increases for subsonic and transonic speeds, and begins to rapidly increase at Mach **2** and beyond. This is unlike the variation of **ct,** for a pure turbojet engine, which is relatively constant in the low supersonic regime.

THE TURBOPROP



The turboprop is essentially a propeller driven by a gas-turbine engine, it is the closest to the reciprocating engine/propeller combination**. T**he inlet air is compressed by an axial-flow compressor, mixed with fuel and burned in the combustor, expanded through a turbine, and then exhausted through a nozzle. Unlike the turbojet, the turbine powers not only the compressor but also the propeller. By design, most of the available work in the flow is extracted by the turbines, leaving little available for jet thrust. For most turboprops, only about 5% of the total thrust is associated with the jet exhaust, and remaining 95% comes from the propeller.

the turboprop falls in between the reciprocating engine/propeller combination and the turbofan or turbojet. The turboprop generates more thrust than a reciprocating engine/propeller device, but less than a turbofan or turbojet. On the other hand, the turboprop has a specific fuel consumption higher than that of the reciprocating engine/propeller combination, but lower than that of **a** turbofan or turbojet. Also, the maximum speed of a turboprop-powered airplane is limited to that at which the propeller efficiency becomes seriously degraded by shock wave formation on the propeller usually around Ma=0.6 to 0.7.

the thrust generated by the turboprop is the sum of the propeller thrust *Tp,* and the jet thrust *Tj.* For the engine in flight at velocity V, the power available from the turboprop is

$$\boxed{P_A = \left(T_p + T_j\right) V_\infty}$$

The main business end of a turboprop is the shaft coming from the engine to which the propeller is attached via some type of gearbox mechanism. Hence the *shaft power Ps,* coming from the engine is a meaningful quantity. Because of losses associated with the propeller the power obtained from the propeller/shaft combination is η_{pr} *Ps.* Hence, the net power available, which includes the jet thrust, is

$$\boxed{P_A = \eta_{\text{pr}} P_s + T_j V_\infty}$$

Sometimes manufacturers rate their turboprops in terms of the *equivalent shaftpower Pes* which is an overall power rating that *includes* the effect of the jet thrust:

$$\boxed{P_A = \eta_{\text{pr}} P_{\text{es}}}$$

Combining the two:

$$\eta_{\text{pr}} P_{\text{es}} = \eta_{\text{pr}} P_s + T_j V_\infty$$

$$\boxed{P_{\text{es}} = P_s + \frac{T_j V_\infty}{\eta_{\text{pr}}}}$$

FUNDAMENTALS OF GAS TURBINE

INTRODUCTION

The jet engine is an internal combustion engine that uses air as the working fluid. The engine extracts chemical energy from fuel and converts it to mechanical energy using the gaseous energy of the working fluid (air) to drive the engine and propeller, which, in turn, propel the airplane.

THE GAS TURBINE CYCLE

The basic principle of the airplane turbine engine is identical to any and all engines that extract energy from chemical fuel. The basic 4 steps for any internal combustion engine are:

1. Intake of air (and possibly fuel).
2. Compression of the air (and possibly fuel).
3. Combustion, where fuel is injected (if it was not drawn in with the intake air) and burned to convert the stored energy.
4. Expansion and exhaust, where the converted energy is put to use.

In the case of a piston engine, such as the engine in a car or reciprocating airplane engine, the intake, compression, combustion, and exhaust steps occur in the same place (cylinder head) at different times as the piston goes up and down.

In the turbine engine, however, these same four steps occur at the same time but in different places. As a result of this fundamental difference, the turbine has engine sections called:

1. The inlet section
2. The compressor section
3. The combustion section (the combustor)
4. The turbine (and exhaust) section.

The turbine section of the gas turbine engine has the task of producing usable output shaft power to drive the propeller. In addition, it must also provide power to drive the compressor and all engine accessories. It does this by expanding the high temperature, pressure, and velocity gas and converting the gaseous energy to mechanical energy in the form of shaft power.

A large mass of air must be supplied to the turbine in order to produce the necessary power. This mass of air is supplied by the compressor, which draws the air into the engine and squeezes it to provide high-pressure air to the turbine. The compressor does this by converting mechanical energy from the turbine to gaseous energy in the

form of pressure and temperature.

If the compressor and the turbine were 100% efficient, the compressor would supply all the air needed by the turbine. At the same time, the turbine would supply the necessary power to drive the compressor. In this case, a perpetual motion machine would exist. However, frictional losses and mechanical system inefficiencies do not allow a perpetual motion machine to operate. Additional energy must be added to the air to accommodate for these losses. Power output is also desired from the engine (beyond simply driving the compressor); thus, even more energy must be added to the air to produce this excess power. Energy addition to the system is accomplished in the combustor. Chemical energy from fuel as it is burned is converted to gaseous energy in the form of high temperatures and high velocity as the air passes through the combustor. The gaseous energy is converted back to mechanical energy in the turbine, providing power to drive the compressor and the output shaft.

SOME BASIC PRINCIPLES

As air passes through a gas turbine engine, aerodynamic and energy requirements

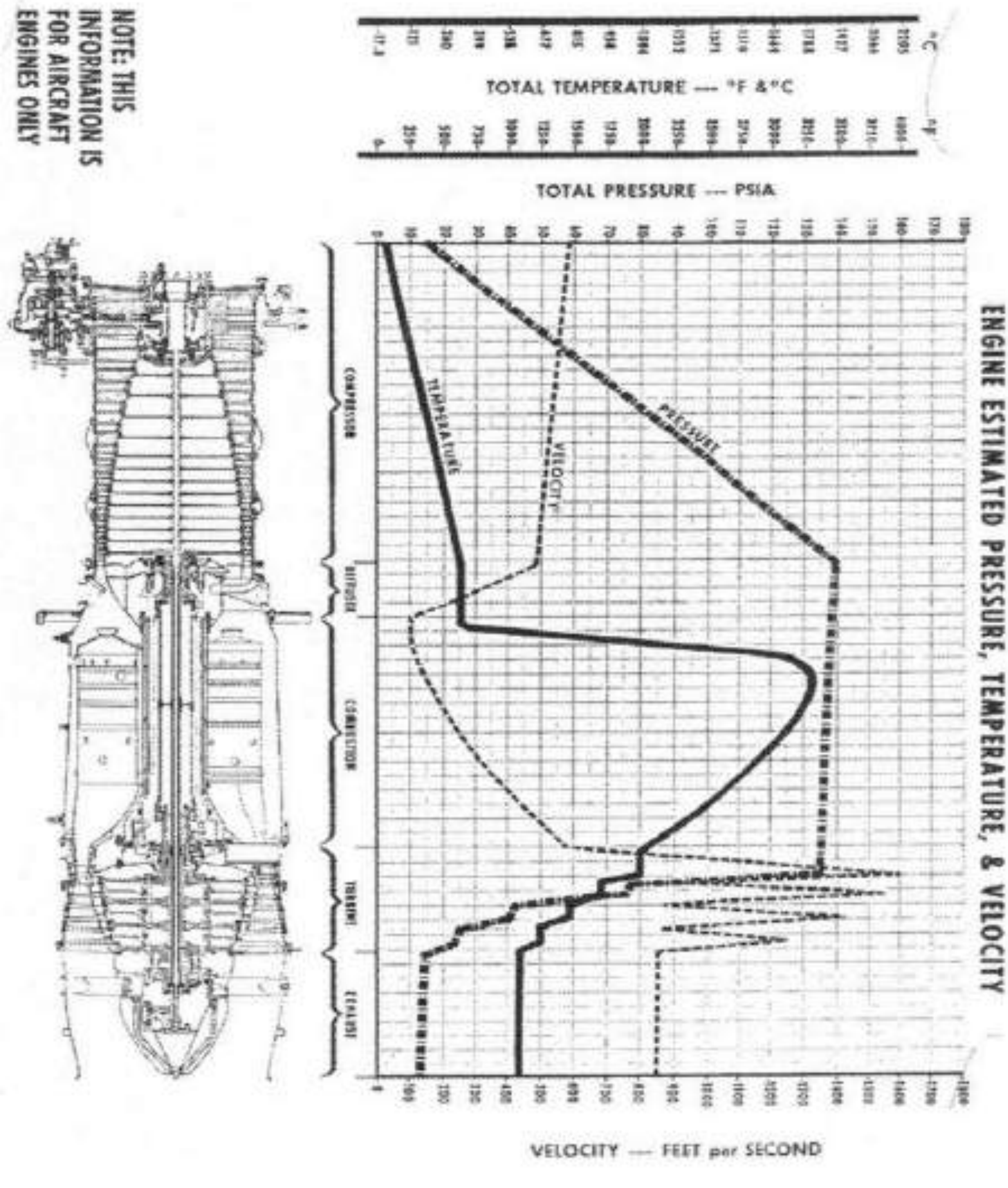
PERFORMANCE AND EFFICIENCY

The type of operation for which the engine is designed dictates the performance requirement of a gas turbine engine. The performance requirement is mainly determined by the amount of shaft horse power (s.h.p.) the engine develops for a given set of conditions.

The need for high efficiency in the engine becomes more important as fuels become more costly. Engine efficiency is primarily defined by the specific fuel consumption (s.f.c.) of the engine at a given set of conditions.

Many factors affect both the efficiency and the performance of the engine. The mass flow rate of air through the engine will dictate engine performance. Any restrictions acting against the smooth flow of air through the engine will limit the engine's performance. The pressure ratio of the compressor, the engine operating temperatures (turbine inlet temperature), and the individual component efficiencies will also influence both the performance and the efficiency of the overall engine. All these factors are considered during the design of the engine. An optimum pressure ratio, turbine inlet temperature, and air mass flow rate are selected to obtain the required performance in the most efficient manner. In addition, individual engine components are designed to minimize flow losses to maximize component efficiencies.

the following graphic shows the typical temperature and pressure rise through the gas



Engine tempreture and pressure flow