JET PROPULSION

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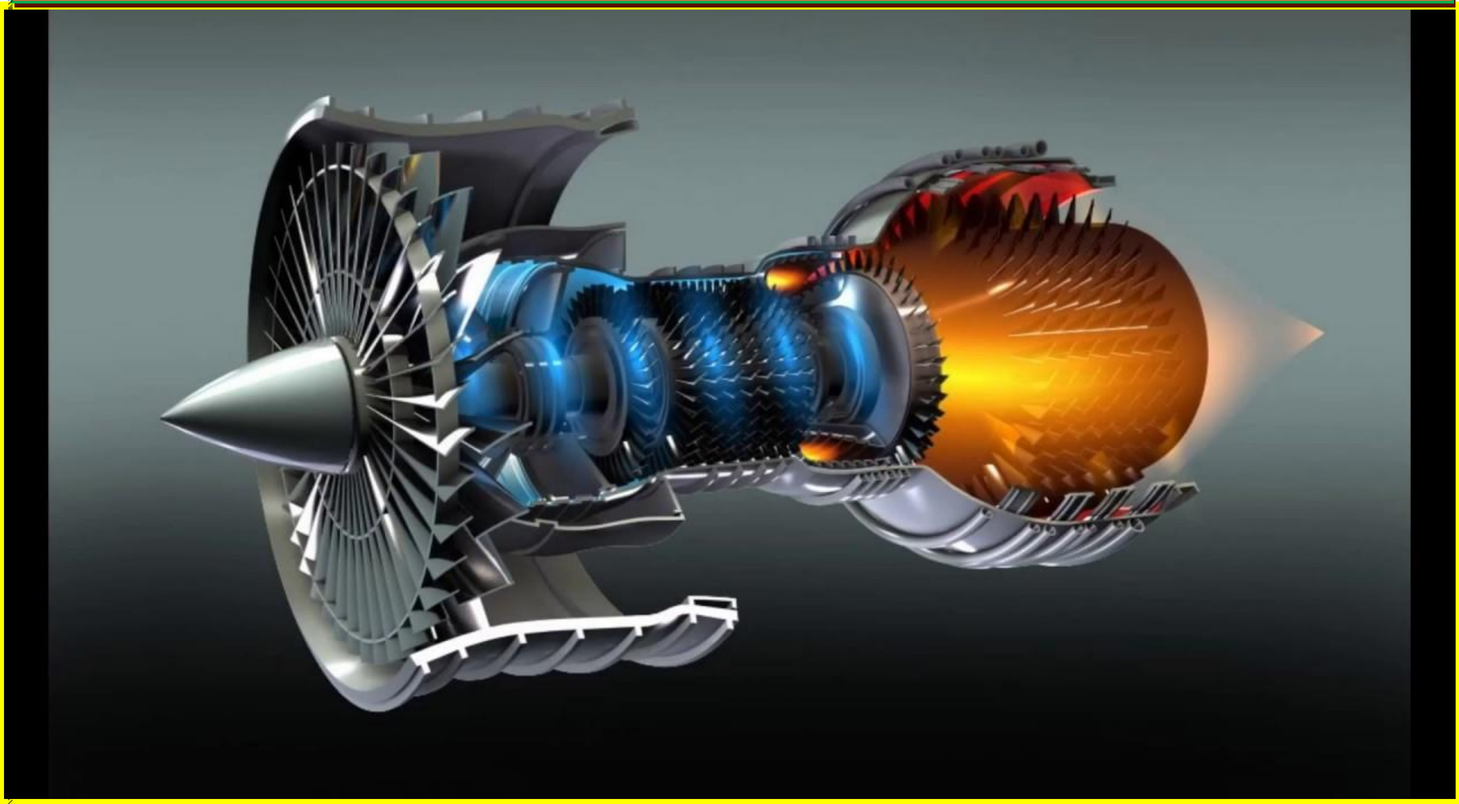

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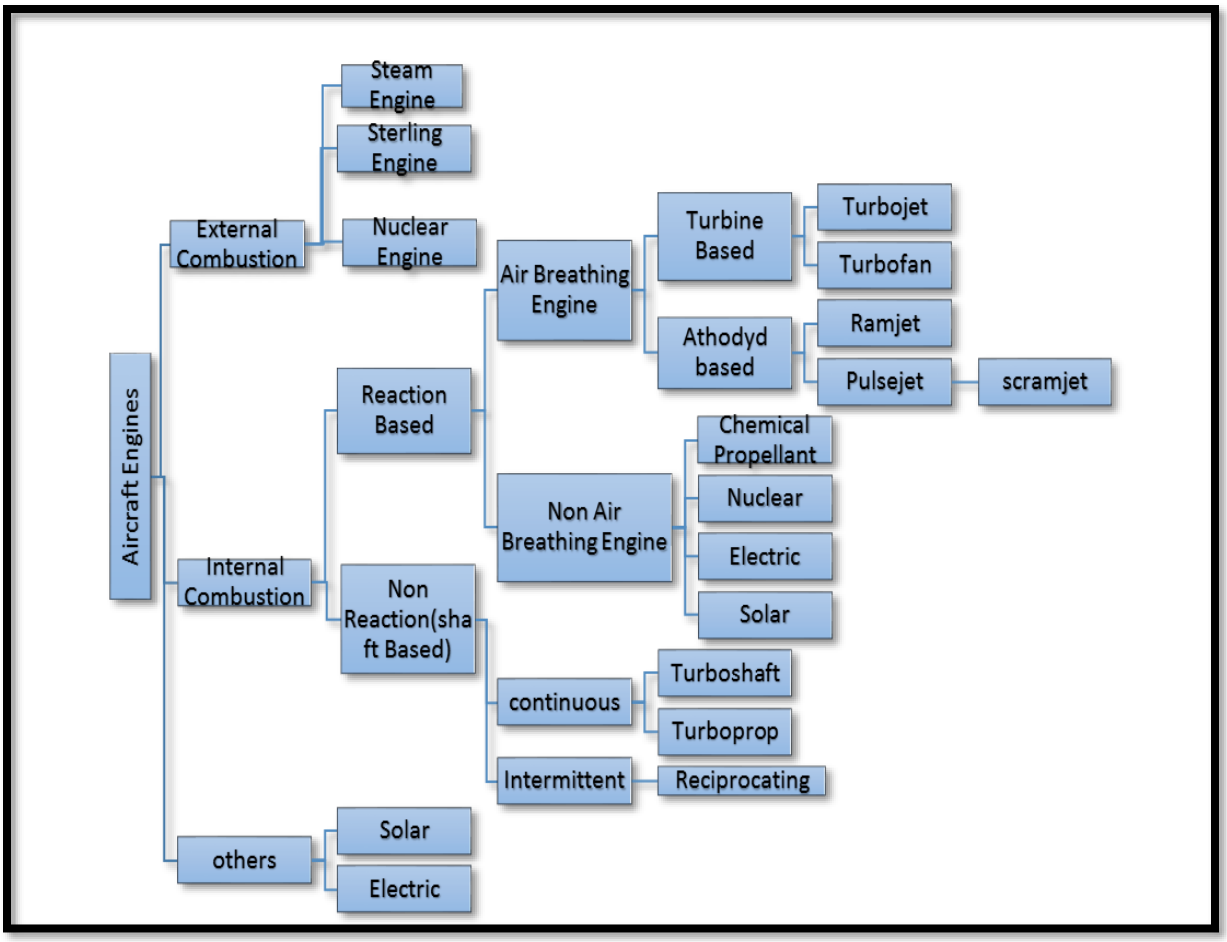
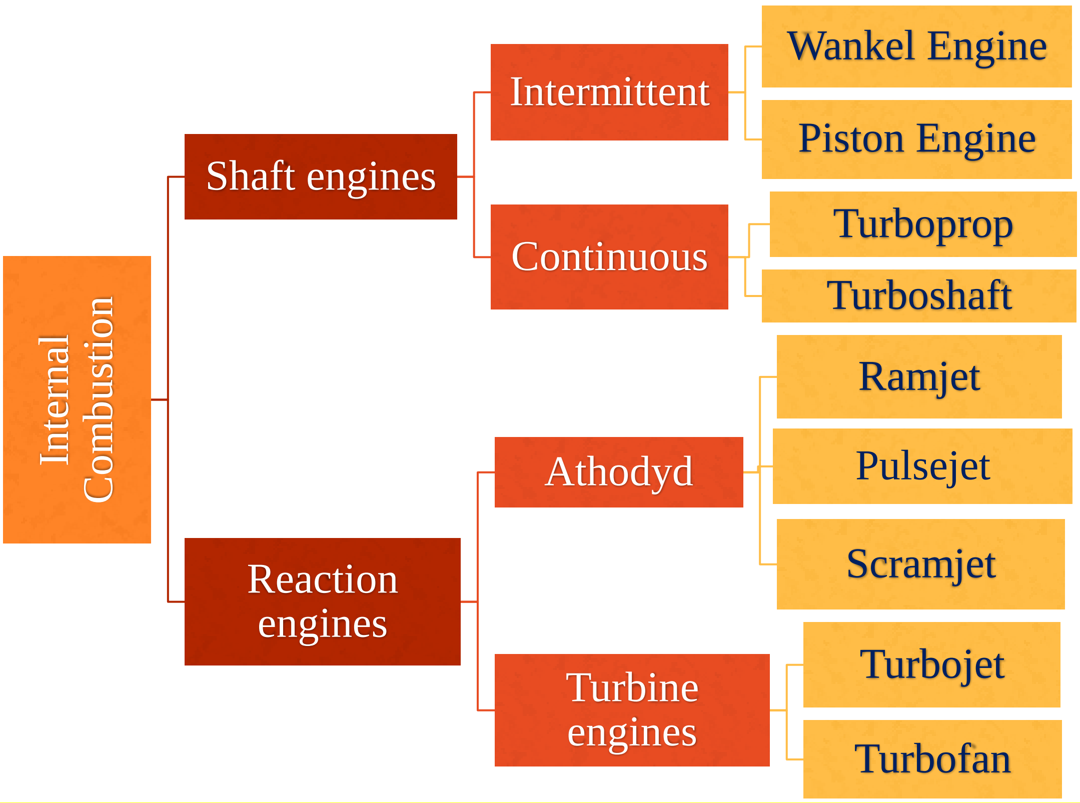
JET PROPULSION

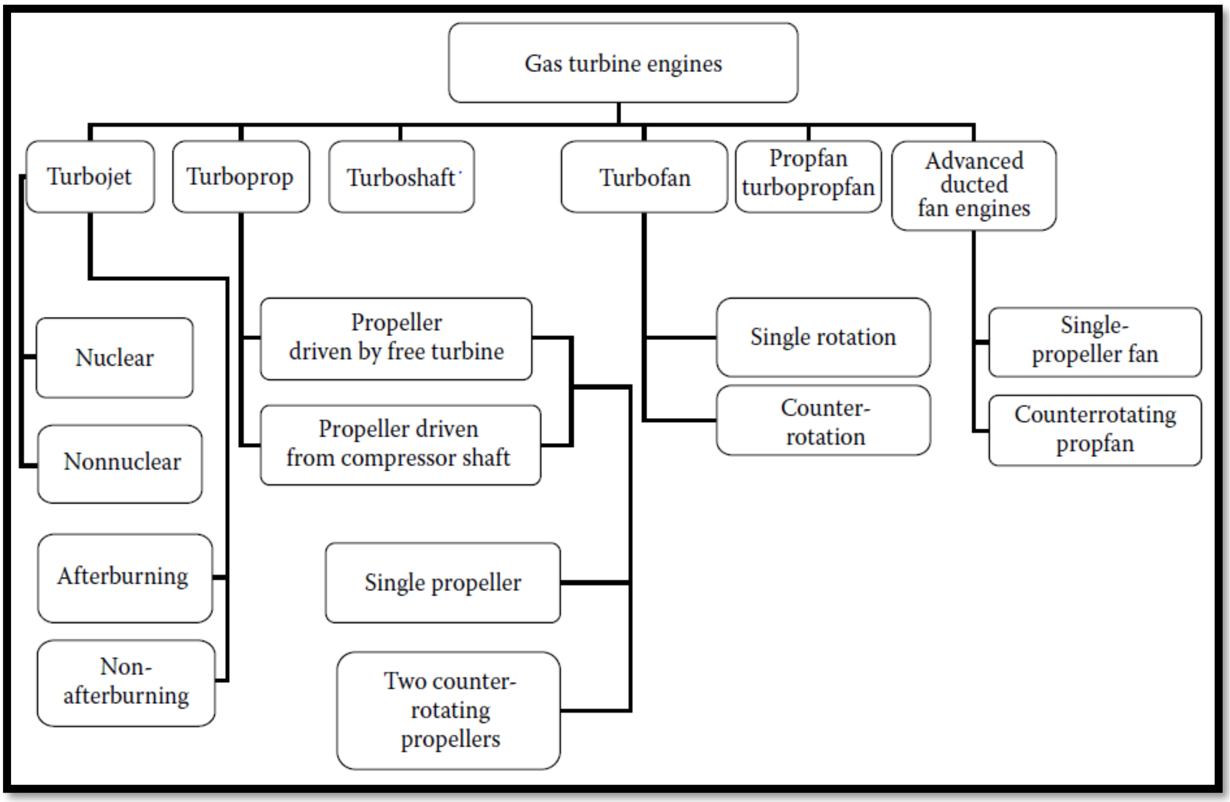
- It is the propulsion of a jet aircraft (or) other missiles by the reaction of jet coming out with high velocity.
- Jet propulsion is based on Newton's second and third law of motion.
- Newton's second law states that 'the rate of change momentum in any direction is proportional to the force acting in that direction'.
- Newton's third law states that for every action there is an equal and opposite reaction.
- The jet propulsion in used when the oxygen is obtained from the surrounding atmosphere.


Aircraft Engines
External Combustion
Steam Engine
Sterling Engine
Nuclear Engine
Internal Combustion
Reaction Based
Air Breathing Engine
Turbine Based
Turbojet
Turbofan
Athodyd based
Ramjet
Pulsejet
scramjet
Non Air Breathing Engine
Chemical Propellant
Nuclear
Electric
Solar
Non Reaction(shaft Based)
continuous
Turboshaft
Turboprop
Intermittent
Reciprocating
others
Solar
Electric

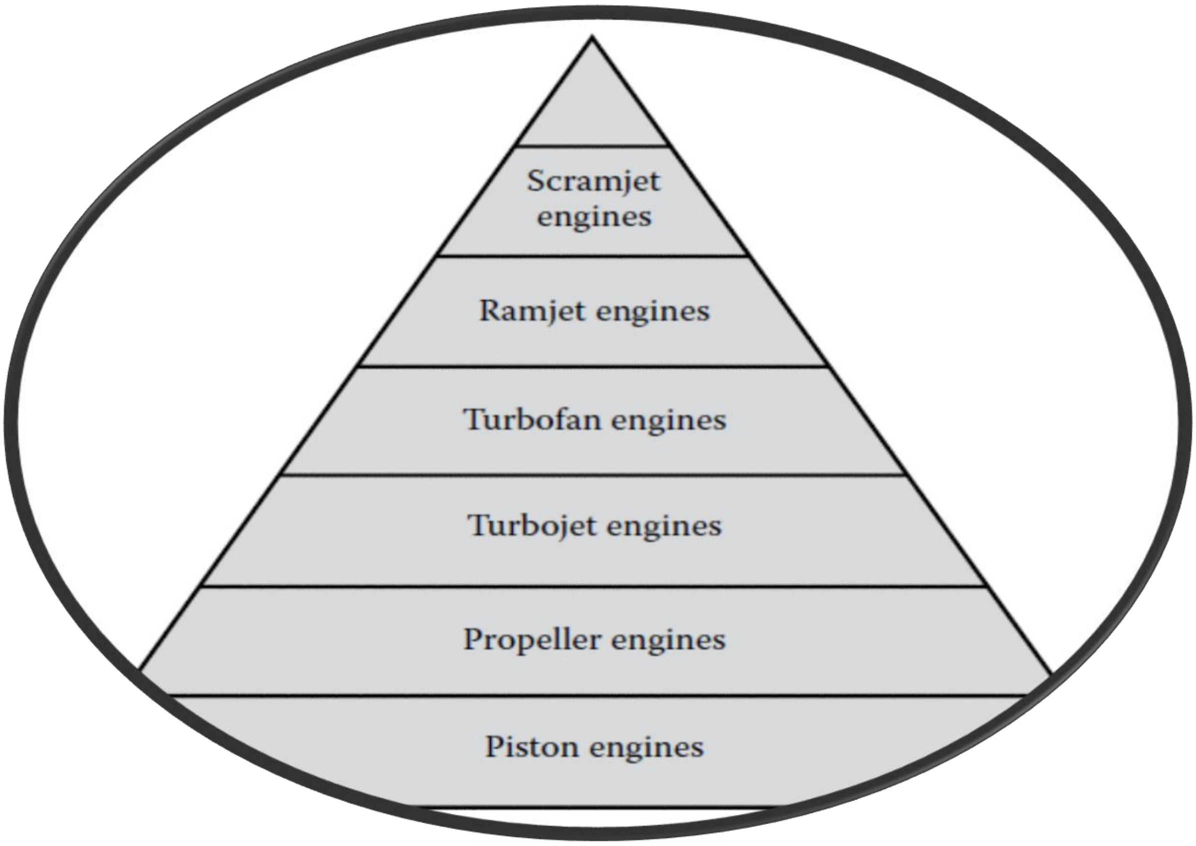
GATE-AE

Internal Combustion Engine



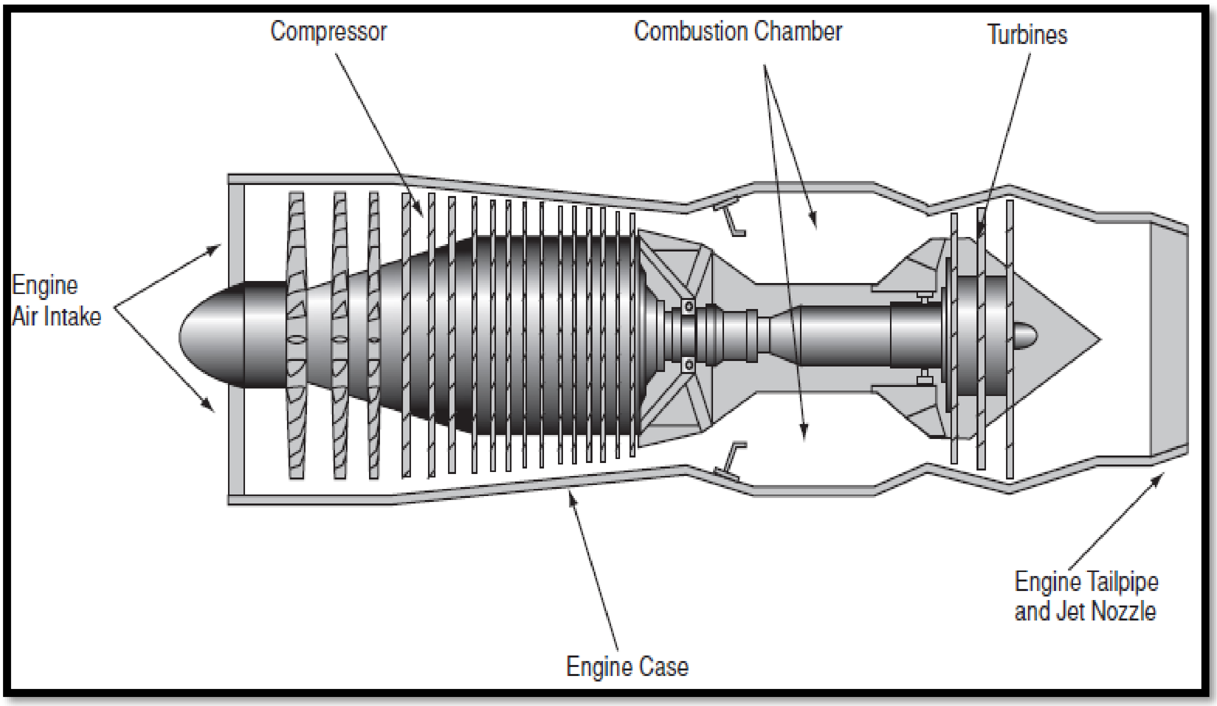

Gas turbine engines
Turbojet
Turboprop
Turboshaft
Turbofan
Propfan turbopropfan
Advanced ducted fan engines
Nuclear
Nonnuclear
Afterburning
Non-afterburning
Propeller driven by free turbine
Propeller driven from compressor shaft
Single propeller
Two counter-rotating propellers
Single rotation
Counter-rotation
Single-propeller fan
Counterrotating propfan

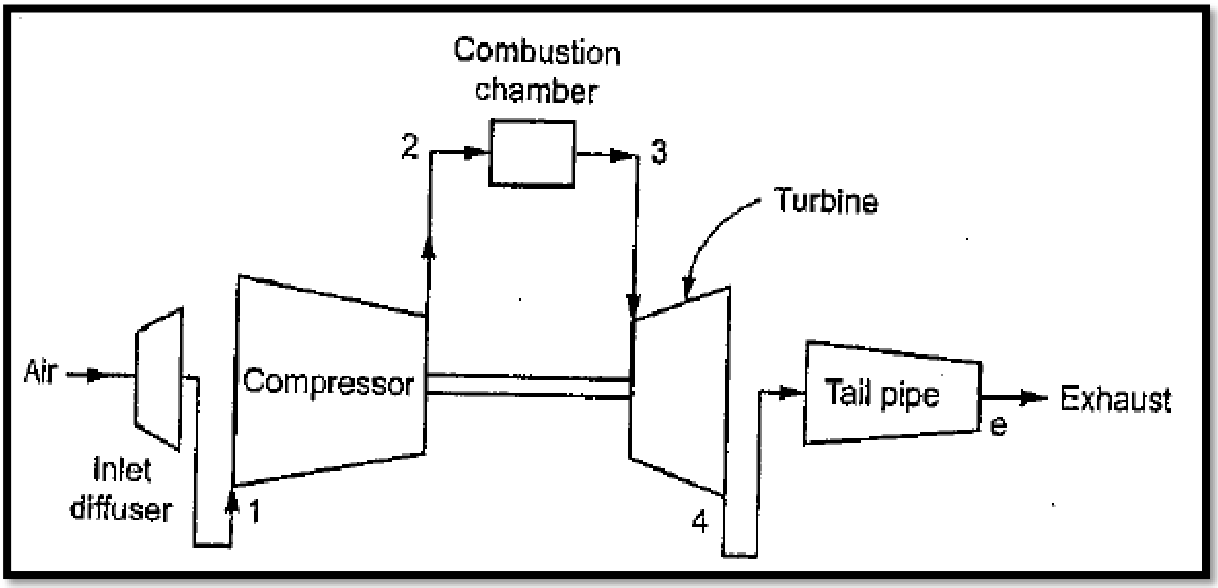
Evolution of aero-engine



TURBOJET ENGINE

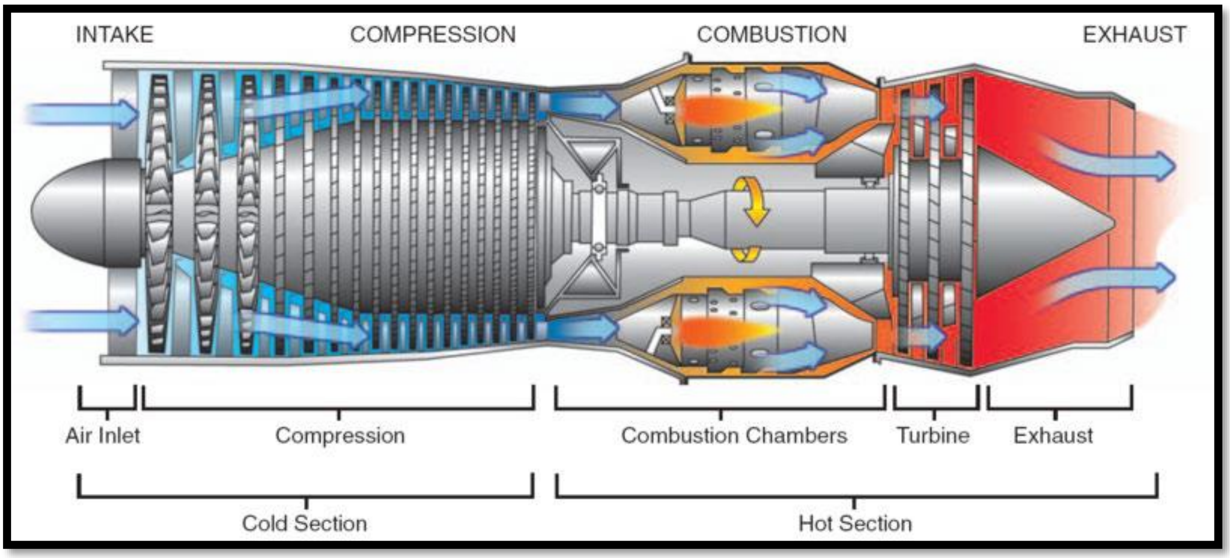
The most common type of air-breathing engine is Turbojet engine.




Combustion chamber
2
3
Turbine
Air
Compressor
Tail pipe
Exhaust
e
Inlet diffuser
1
4

Construction:

The construction of turbojet engine is shown in fig.



It consists of:

- Diffuser
- Rotary compressor
- Combustion chamber
- Turbine
- Exhaust nozzle

- The function of the diffuser is to convert the kinetic energy of the entering air into pressure energy.
- The function of the nozzle is to convert the pressure energy of the combustion gases into kinetic energy

Working:

1. Air from the atmosphere enters into turbojet engine. The air velocity gets reduced and its static pressure is increased by diffuser.
2. Then the air passes through the rotary compressor in which the air is further compressed.
3. Then the high pressure air flows into the combustion chamber. In the combustion chamber, the fuel is injected by suitable injectors and the air-fuel mixture is burnt, heat is supplied at constant pressure.
4. The highly heated products of combustion gases are then entering the turbine and partially expanded.
5. The power produced by the turbine is just sufficient to drive the compressor, fuel pump and other auxiliaries.
6. The hot gases from the turbine are then allowed to expand in the exhaust nozzle section
7. In the nozzle, pressure energy of the gas is converted into kinetic energy. So the gases coming out from the unit with very high velocity.
8. Due to high velocity of gases coming out from the unit, a reaction or thrust is produced in the opposite direction. This thrust propels the aircraft.
9. Like ramjet engine, the turbojet engine is a continuous flow engine.
10. Because of turbine material limitations, only a limited amount of fuel can be burnt in the combustion chamber

Advantages:

1. Construction is simple
2. Less wear and tear
3. Less maintenance cost
4. It runs smoothly because continuous thrust is produced by continuous combustion of fuel.
5. The speed of a turbojet is not limited by the propeller and it can attain higher flight speed than turbo propeller aircrafts.
6. Low grade fuels like kerosene, paraffin, etc., can be used. This reduces the fuel cost.
7. Reheat is possible to increases the thrust
8. Since turbojet engine has a compressor, it can be operated under static conditions.

Disadvantages:

1. It has low take-off thrust and hence poor starting characteristics
2. Fuel consumption is high
3. Costly materials are used
4. The fuel economy at low operational speed is extremely poor
5. Sudden decreases of speed is difficult
6. Propulsive efficiency and thrust are lower at lower speeds

APPLICATIONS:

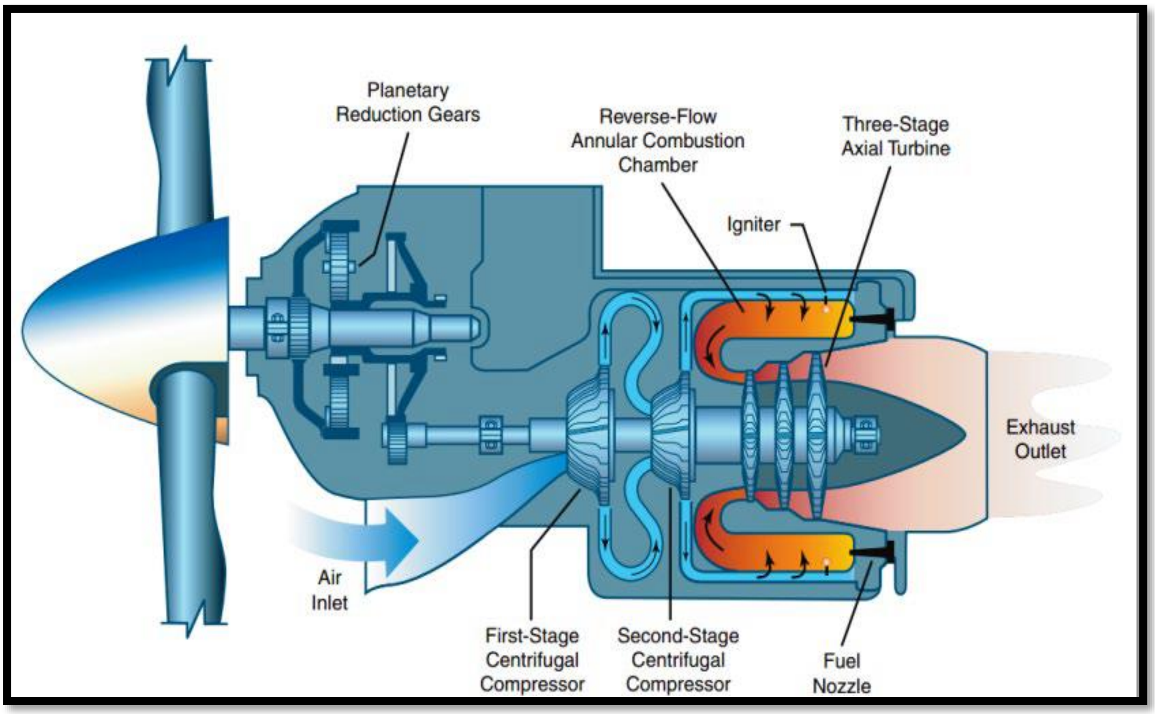
It is best suited for piloted air-crafts, Military aircrafts, etc

TURBO-PROP ENGINE (OR) TURBO-PROPELLER ENGINE

It is very similar to turbojet engine. In this type, the turbine drives the compressor and propeller.

CONSTRUCTION:

The construction of Turbo-Prop Engine is shown in fig



It consists of:

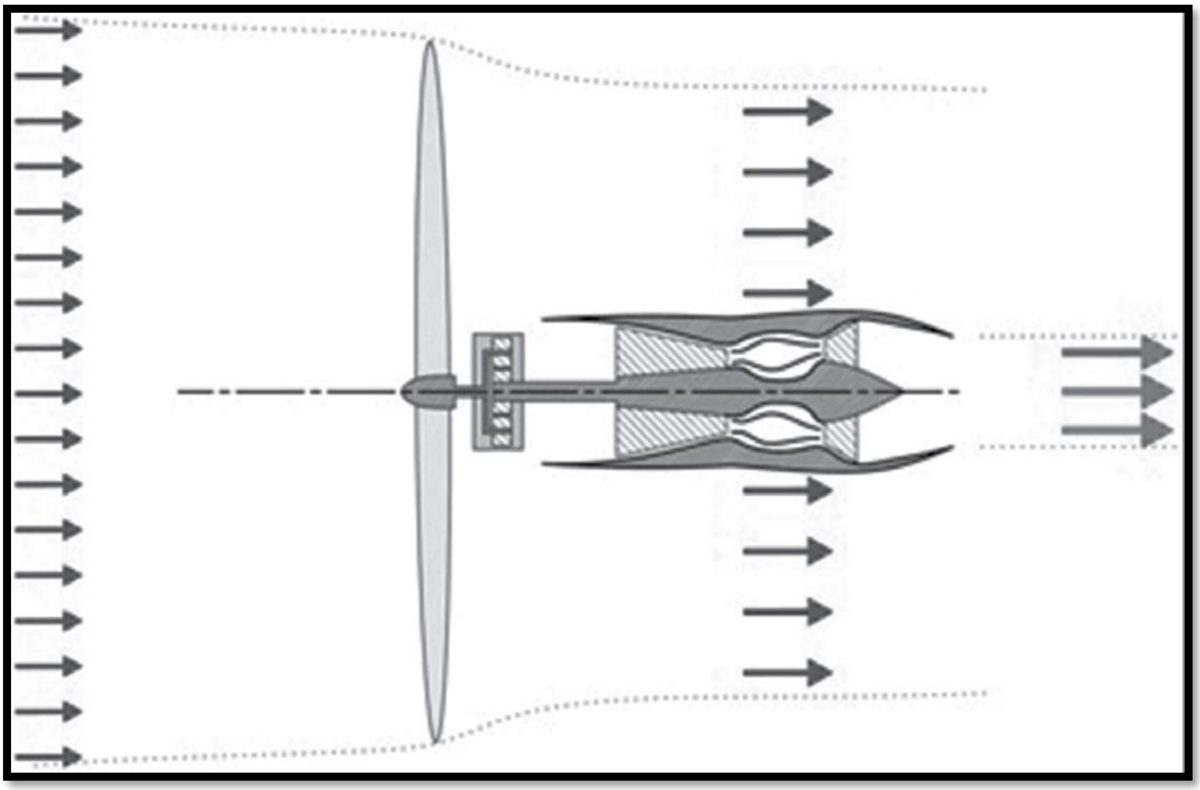
- Diffuser
- Compressor
- Combustion chamber
- Turbine
- Exhaust nozzle
- Reduction gear
- Propeller

- The function of diffuser is to convert the kinetic energy of the entering air into pressure energy.
- The function of nozzle is to convert the pressure energy of the combustion gases into kinetic energy.
- The angular velocity of the shaft is very high. But the propeller cannot run at higher angular velocity. So reduction gear box is provided before the power is transmitted to the propeller.

WORKING:

1. Air from the atmosphere enters into turbo prop engine. The air velocity gets reduced and its static pressure is increased by diffuser.
2. Then the air passes through the rotary compressor in which the air is further compressed. So, the static pressure of the air is further increased.
3. Then the high pressure air flows into the combustion chamber. In the combustion chamber, the fuel is injected by suitable injectors and the air-fuel mixture is burnt. Heat is supplied at constant pressure.
4. The highly heated products of combustion gases are then enters the turbine and partially (about 80 to 90%) expanded.

5. The power produced by the turbine is used to drive the compressor and propeller.
6. Propeller is used to increase the flow rate of air which results in better fuel economy.
7. The hot gases from the turbine are then allowed to expand in the exhaust nozzle section.
8. In the nozzle, pressure energy of the gas is converted into kinetic energy. So the gases coming out form the unit with very high velocity.
9. Due to high velocity of gases coming out from the unit, a reaction (or) thrust is produced in the opposite direction.
10. The total thrust produced in this engine is the sum of the thrust produced by the propeller and the thrust produced by the nozzle. This total thrust propels the air craft.



ADVANTAGES:

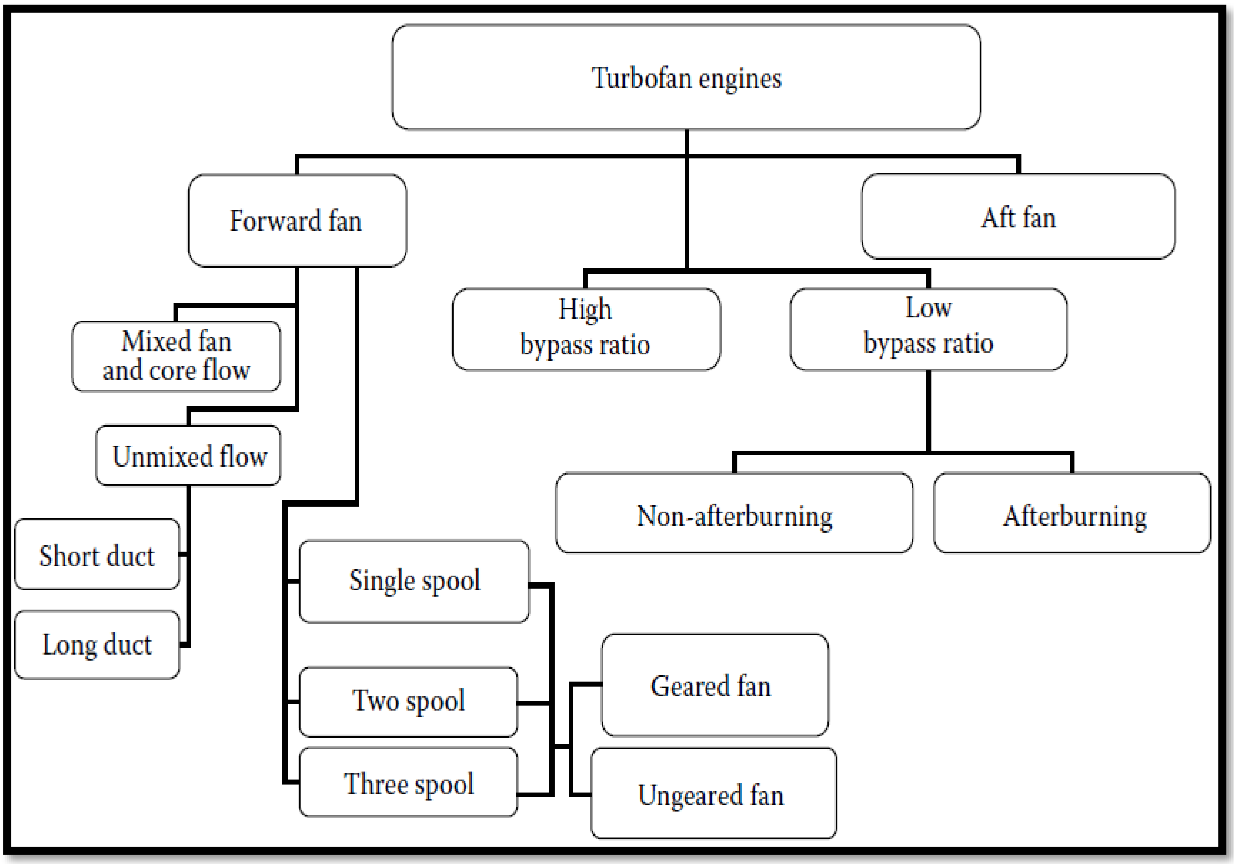
1. HIGH TAKE-OFF THRUST
2. GOOD PROPELLER EFFICIENCY AT A SPEED BELOW 800KM/HR
3. REDUCED VIBRATION AND NOISE
4. BETTER FUEL ECONOMY
5. EASY MAINTENANCE
6. IT OPERATES OVER A WIDE RANGE OF SPEEDS DUE TO MULTI SHAFT ARRANGEMENT.
7. THE POWER OUTPUT IS NOT LIMITED.
8. SUDDEN DECREASE OF SPEED IS POSSIBLE BY THRUST REVERSAL.

DISADVANTAGES:

1. The main disadvantage is, the propeller efficiency is rapidly decreasing at high speeds due to shocks and flow separation.
2. It requires a reduction gear which increases the cost of the engine.
3. More space needed than turbojet engine
4. Engine construction is more complicated

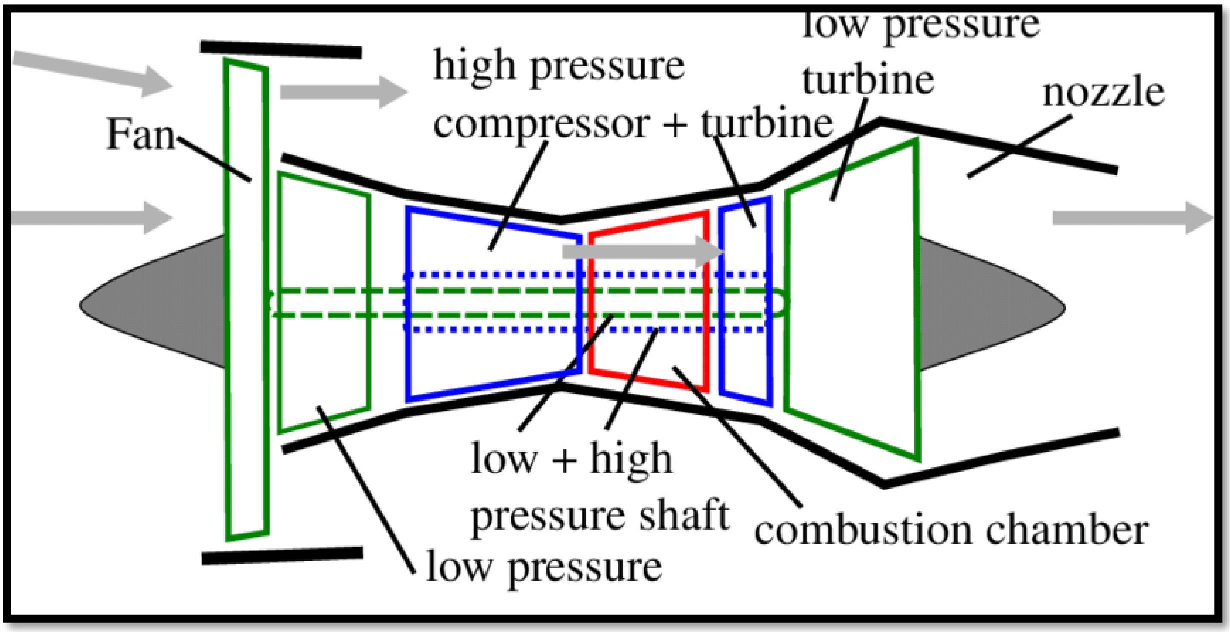
TURBOFAN ENGINE

The turbofan engine is a combination of the turbo prop and the turbojet engines combining the advantages of both.



CONSTRUCTION:

The construction of turbofan engine is shown in fig

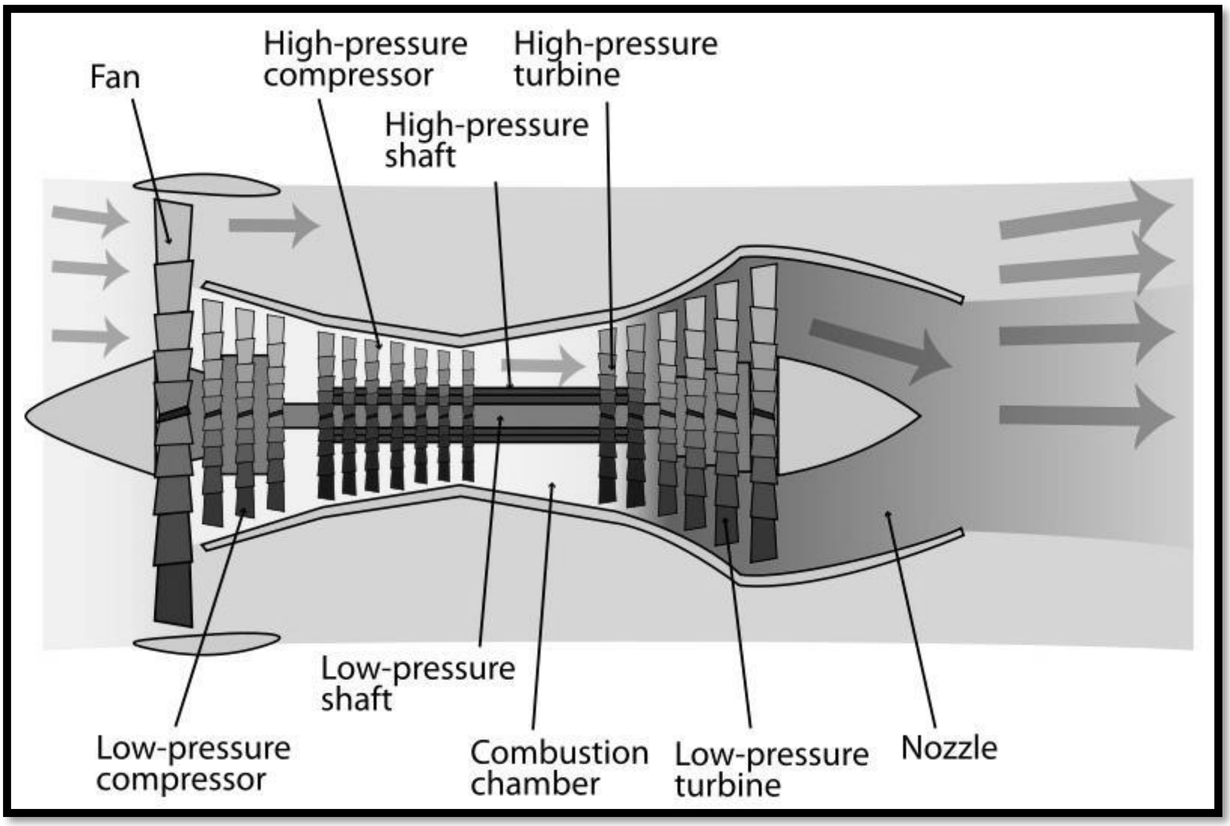


WORKING:

1. Air from the atmosphere enters into turbofan engine, employing a low pressure ducted fan.
2. The air after passing through the fan is divided into two streams, namely primary air and secondary air.
3. The primary air flow through the turbofan engine consisting of compressor, combustion chamber, turbine and exhaust nozzle.

Combustion takes place in the combustion chamber and the thrust is produced in the opposite direction

4. The secondary air (or) by pass air (or) cold air at relatively lower pressure flows around the turbofan engine and expands in the fan nozzle. Hence thrust is produced.



5. The thrust developed by the secondary air is at lower velocity and the thrust developed by the primary air is at much higher velocity.

6. The total thrust produced in this engine is the sum of thrust produced by the primary air $(\dot{m_h})$ and the secondary air $(\dot{m_c})$. This total thrust propels the air craft.
7. The ratio of the mass flow rates of cold air $(\dot{m_c})$ and the hot air $(\dot{m_h})$ is known as Bypass Ratio.

$$\beta = \frac{\dot{m}_{COLD}}{\dot{m}_{HOT}}$$

Total mass flow rate to the engine is:

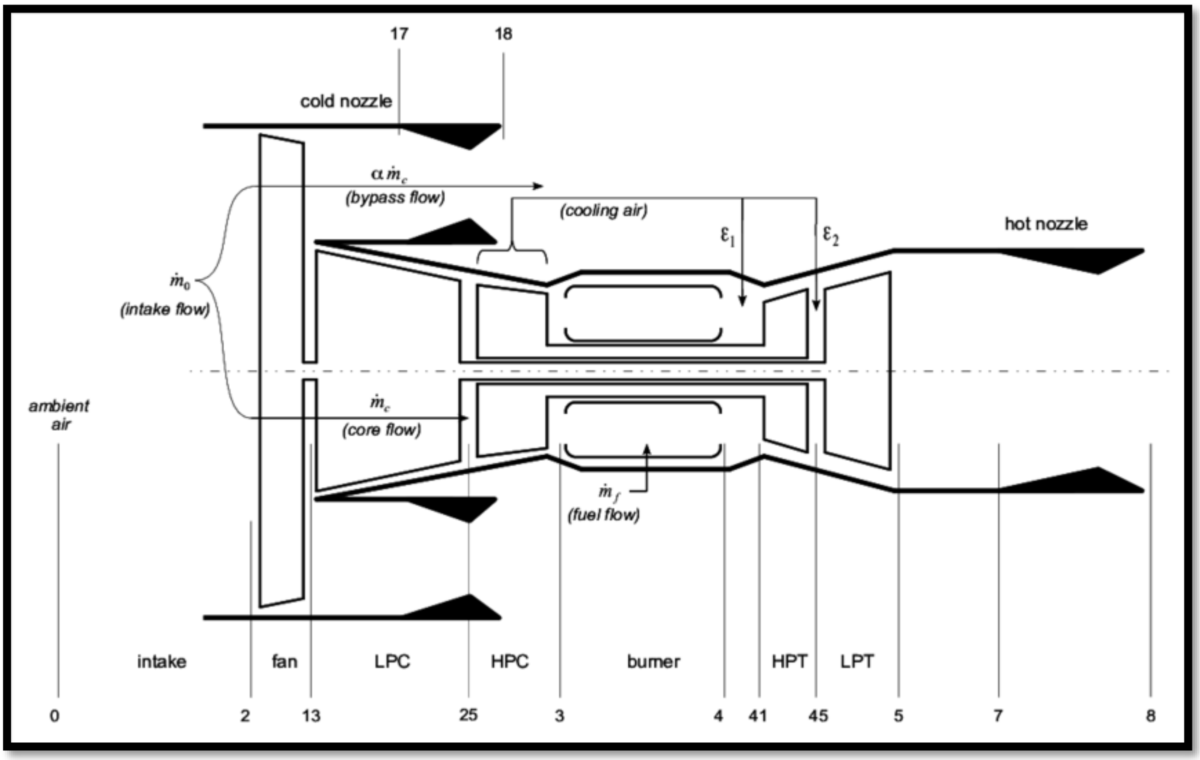
$$\dot{m}_{total} = \dot{m}_{COLD} + \dot{m}_{HOT}$$

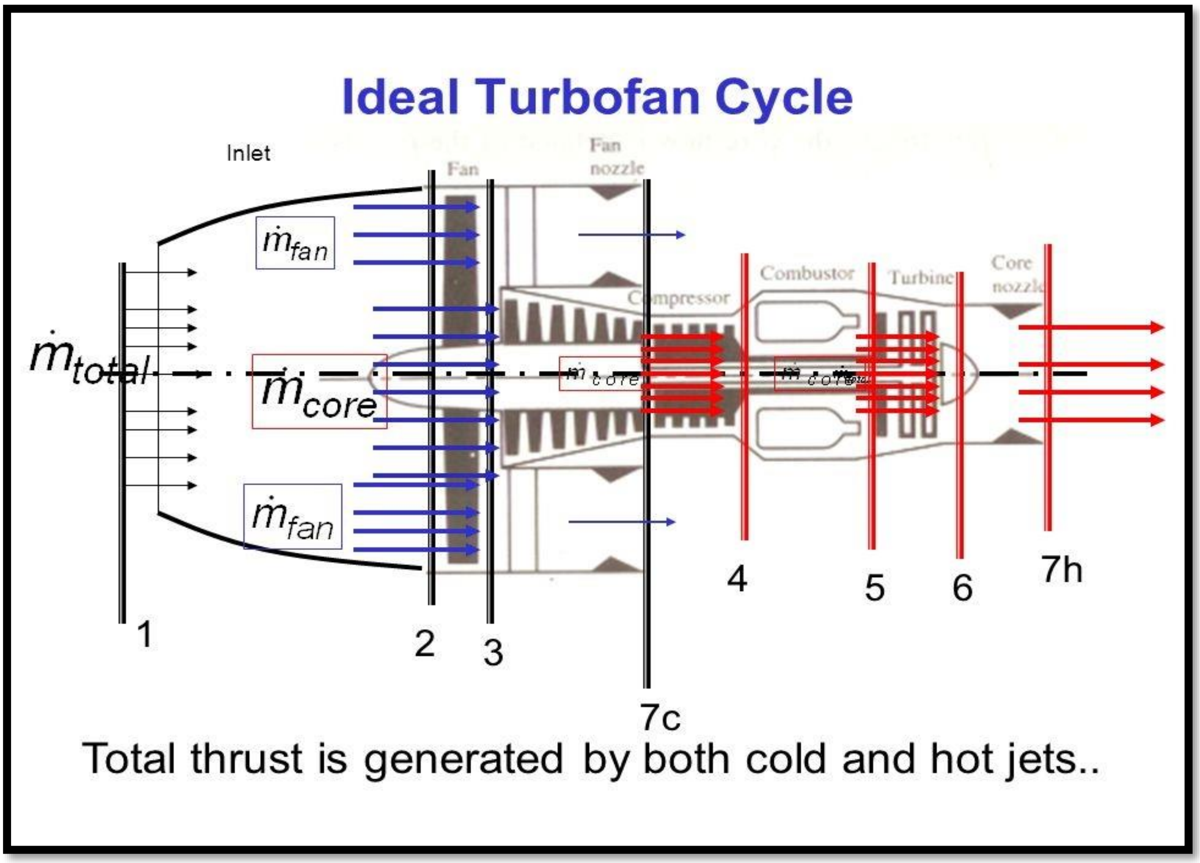
Hence primary mass flow rate through core or hot section

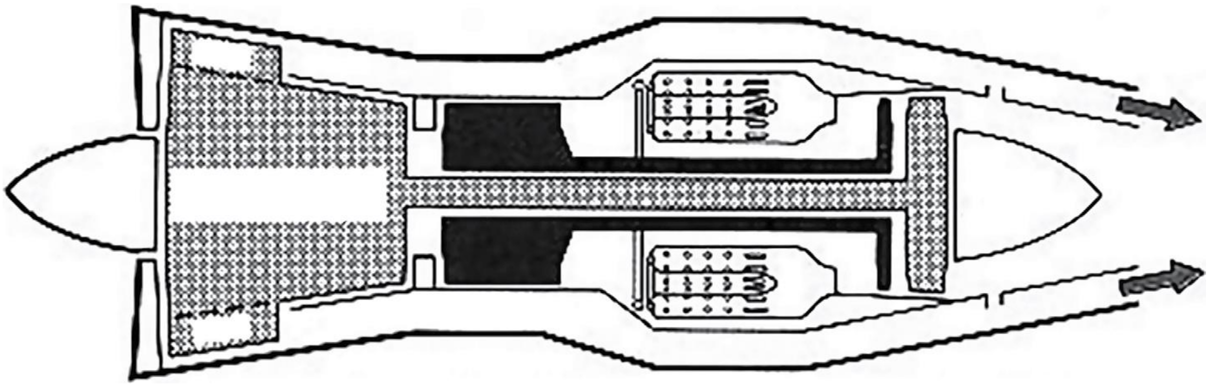
$$\dot{m}_{HOT} = \frac{\dot{m}_{total}}{1+\beta}$$

Secondary mass flow rate through fan or cold section

$$\dot{m}_{COLD} = \left(\frac{\beta}{1+\beta}\right)\dot{m}_{total}$$


17
18
cold nozzle
$\alpha \dot{m}_c$
(bypass flow)
(cooling air)
ε_1
ε_2
hot nozzle
$\dot{m}_0$
(intake flow)
ambient air
$\dot{m}_c$
(core flow)
$\dot{m}_f$
(fuel flow)
intake
fan
LPC
HPC
burner
HPT
LPT
0
2
13
25
3
4
41
45
5
7
8


Ideal Turbofan Cycle
Inlet
Fan
Fan nozzle
Compressor
Combustor
Turbine
Core nozzle
$\dot{m}_{total}$
$\dot{m}_{fan}$
$\dot{m}_{core}$
$\dot{m}_{fan}$
1
2
3
7c
4
5
6
7h
Total thrust is generated by both cold and hot jets..



ADVANTAGES:

1. Thrust developed is higher than turbojet engine
2. Weight per unit thrust is lower than turbo prop engine
3. Less noise
4. High take-off thrust

Disadvantages

- Increased frontal area
- Fuel consumption is high compared to turbo prop engine.
- Construction is complicated compared to turbojet engine

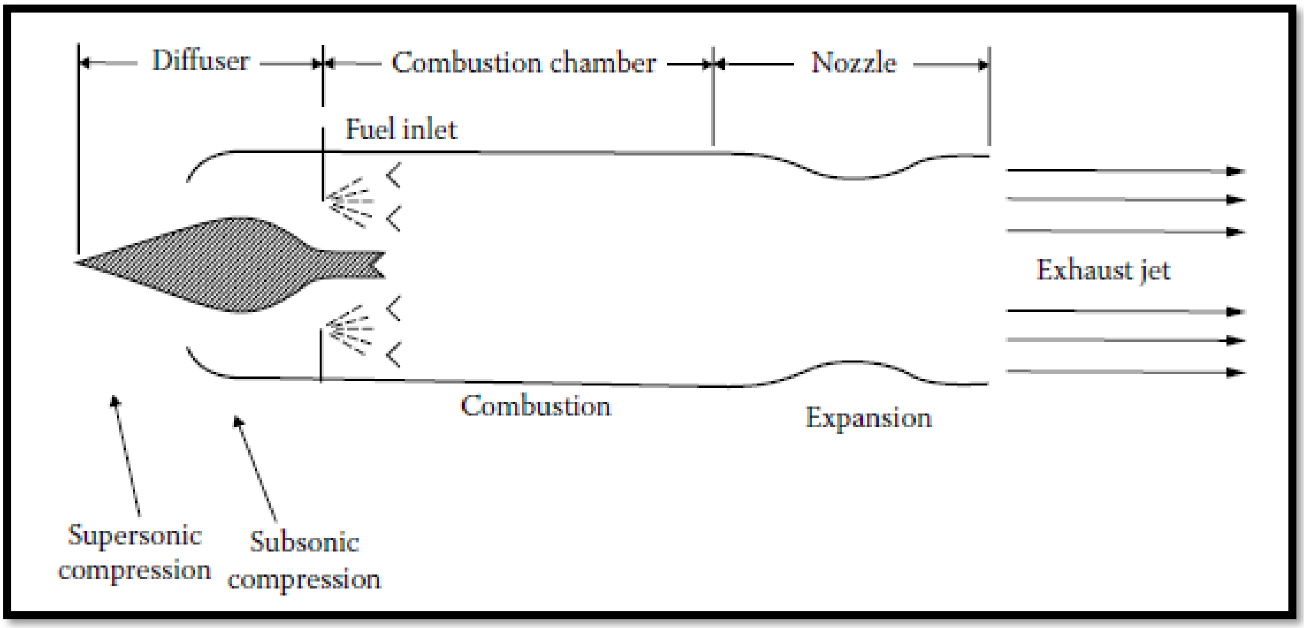
APPLICATION:

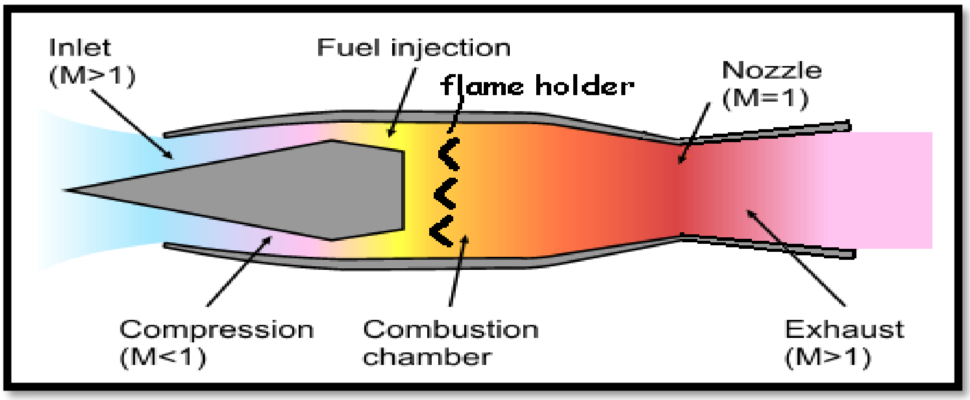
THE TURBOFAN ENGINE IS BEST SUITED FOR COMMERCIAL AND MILITARY AIR-CRAFT OPERATION DUE TO ITS HIGH FLEXIBILITY OF OPERATION AND GOOD FUEL ECONOMY.

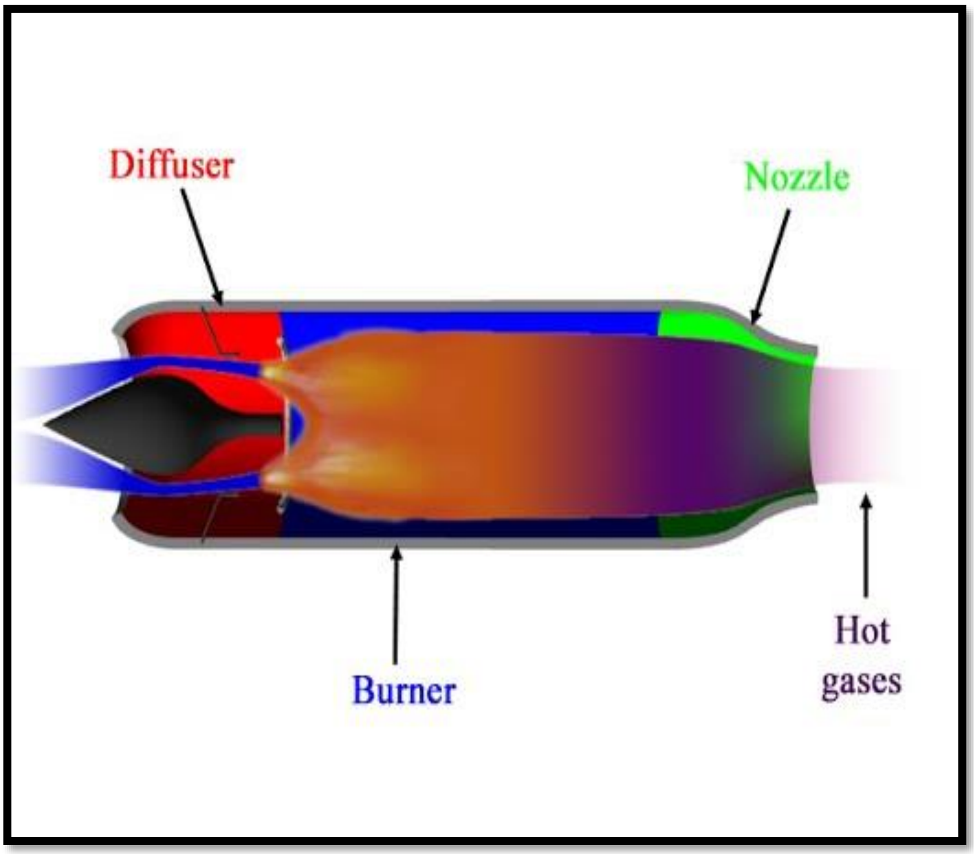
RAM JET ENGINE

Construction:

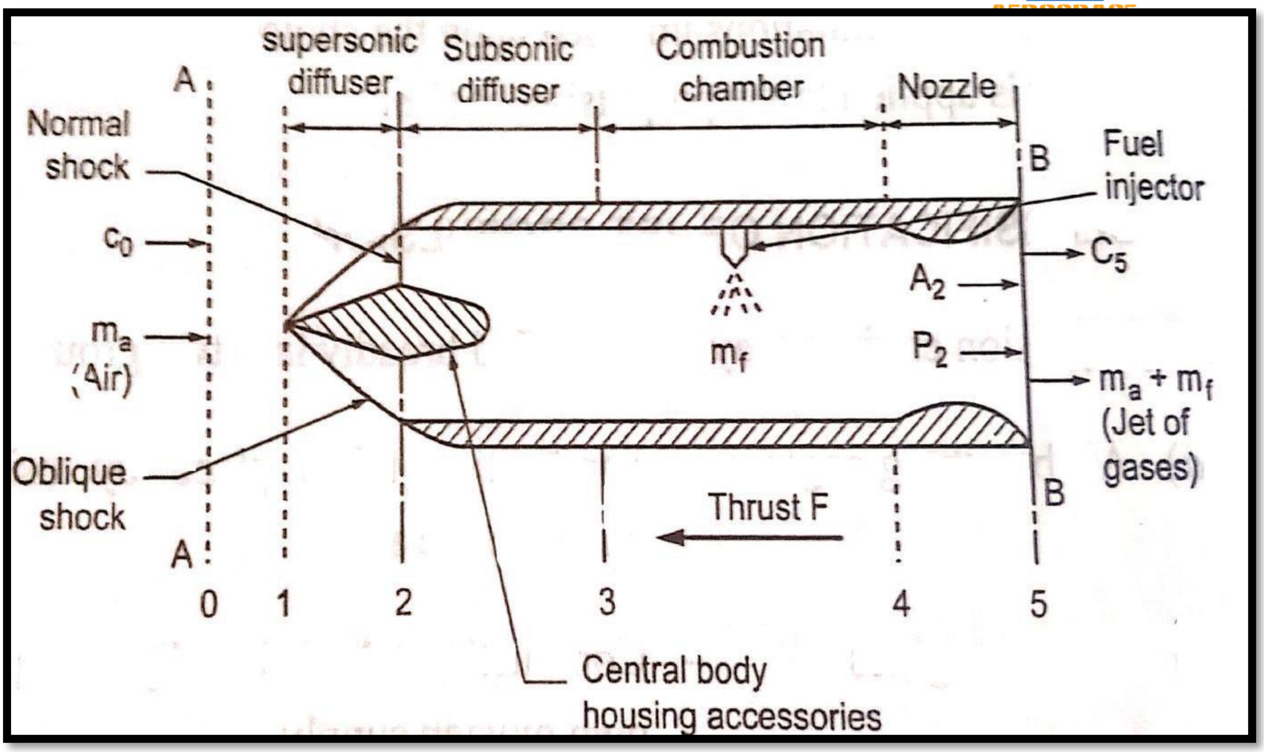
THE CONSTRUCTION OF RAMJET ENGINE IS SHOWN IN FIG. WHICH IS SIMPLEST TYPES OF AIR-BREATHING ENGINE.




Inlet
(M>1)
Fuel injection
flame holder
Nozzle
(M=1)
Compression
(M<1)
Combustion
chamber
Exhaust
(M>1)


Diffuser
Nozzle
Burner
Hot
gases

GATE-AE



IT CONSISTS OF:

- Supersonic diffuser (1-2)
- Subsonic diffuser (2-3)
- Combustion chamber (3-4)
- Discharge nozzle section (4-5)
- The function of supersonic and subsonic diffusers is to convert the kinetic energy of the entering air into pressure energy.
- This energy transformation is called **ram effect** and the pressure rise is called the **ram pressure.** The function of nozzle is to convert pressure energy of gas into kinetic energy.

Working:

1. Air from the atmosphere enters the engine at a very high speed and its velocity gets reduced and its static pressure is increased by supersonic diffuser.
2. Then the air passes through the subsonic diffuser and its velocity further reduces to subsonic value. Due to this, the pressure of air increases to ignition pressure.
3. Then the high pressure air flows into the combustion chamber. In the combustion chamber, the fuel is injected by suitable injectors and the air fuel mixture is burnt.
4. The highly heated products of combustion gases are then allowed to expand in the exhaust nozzle section.
5. In the nozzle pressure energy of the gas is converted into kinetic energy. So the gases coming out form the unit with very high velocity.
6. Due to high velocity of gases coming out from the unit, a reaction or thrust is produced in the opposite direction. This thrust propels the air craft.
7. Ramjet produces very high thrust with high efficiency at supersonic speeds. So, it is best suitable for high speed aircraft
8. The air enters the engine with a supersonic speed must be reduced to subsonic speed. This is necessary to prevent the blow out of the flame in the combustion chamber. The velocity must be small enough to make it possible to add the required quantity of fuel for stable combustion.
9. Both theory and experiment indicate that the speed of the air entering the combustion chamber should not be higher than that corresponding to a local mach number of 0.2 approximately.

Advantages:

1. Ramjet engine is very simple and does not have any moving part.
2. Cost is low
3. Less maintenance
4. The specific fuel consumption is better than other gas turbine power plants at high speed.
5. There is no upper limit to the flight speed.
6. Light weight when compared with turbojet engine.

Disadvantages:

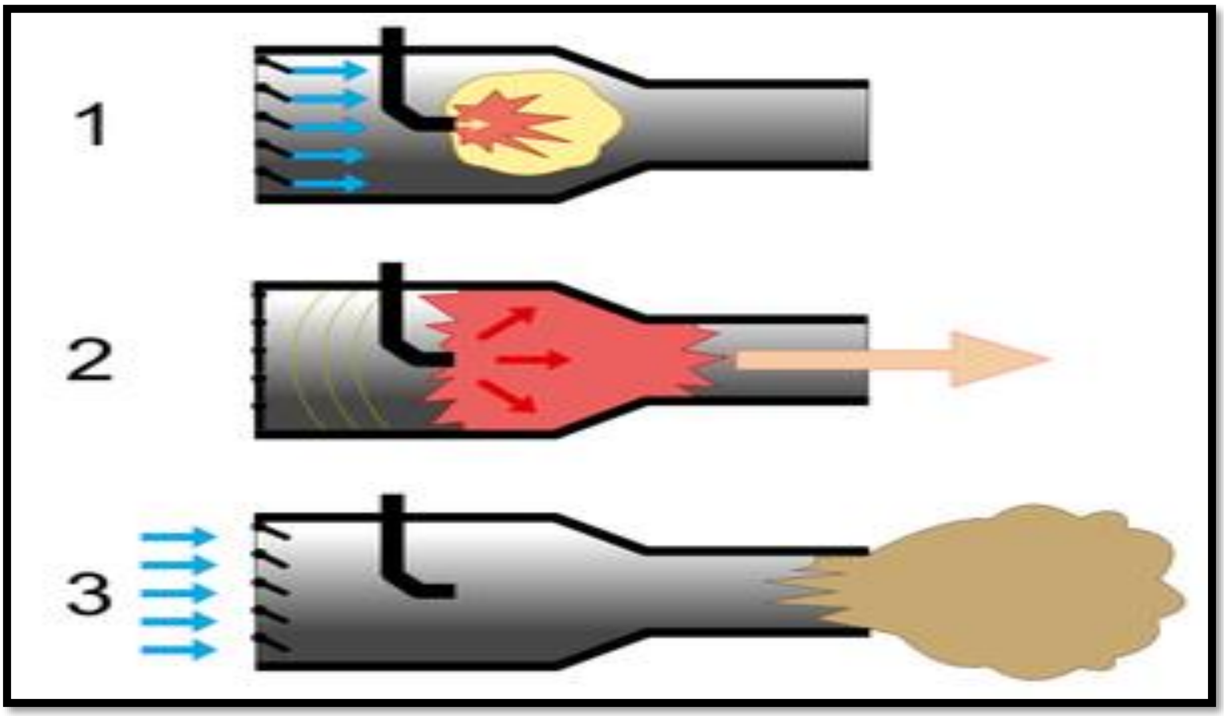
1. Since the take-off thrust is zero, it is not possible to start a ramjet engine without an external launching device.
2. The combustion chamber required flame holder to stabilize the combustion due to high speed of air.
3. It is very difficult to design a diffuser which will give good pressure recovery over a wide range of speeds.
4. It has low thermal efficiency.

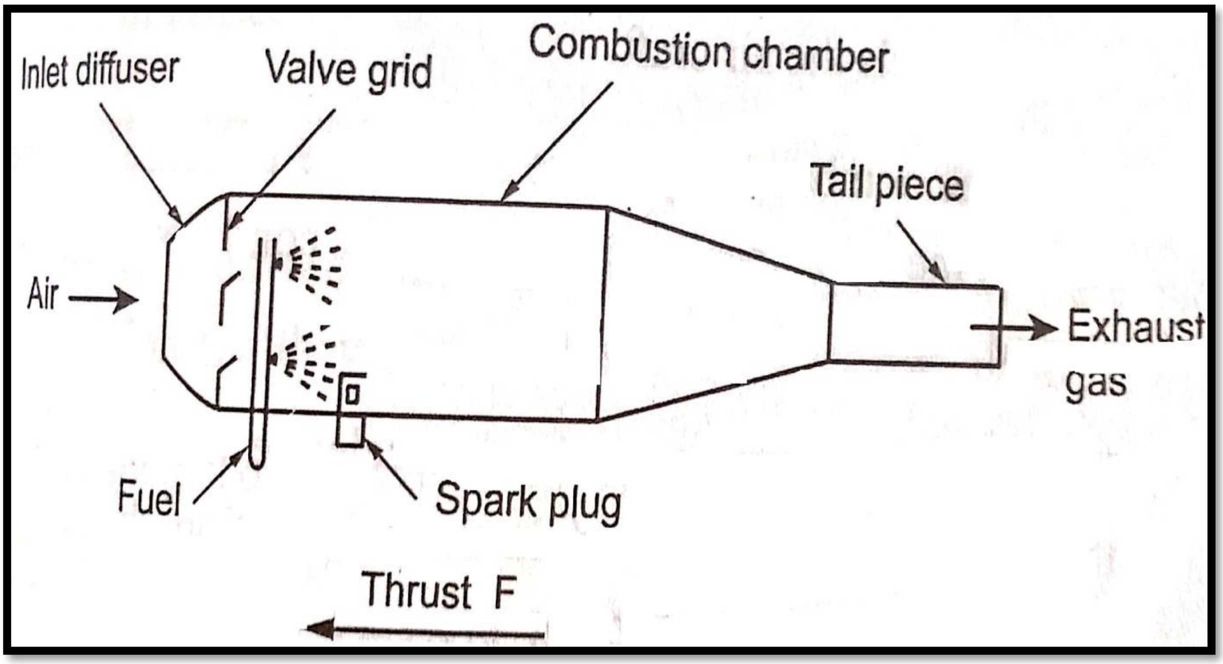
Applications:

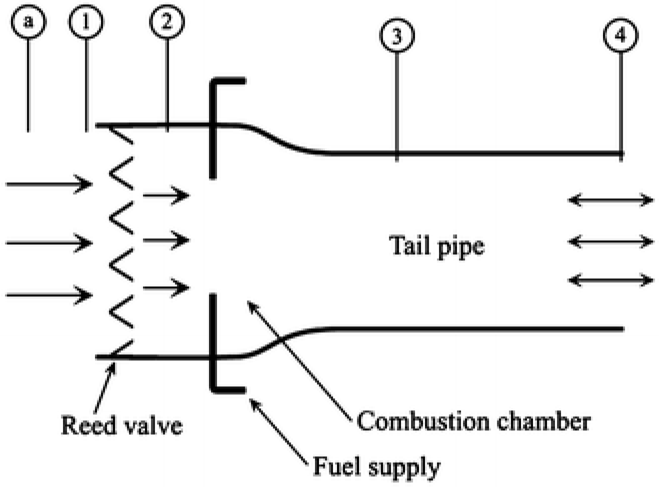
1. It is widely used in high speed aircrafts and missiles due to its high thrust and high operational speed.
2. Subsonic ramjets are used in target weapons.

PULSE JET ENGINE (OR) FLYING BOMB

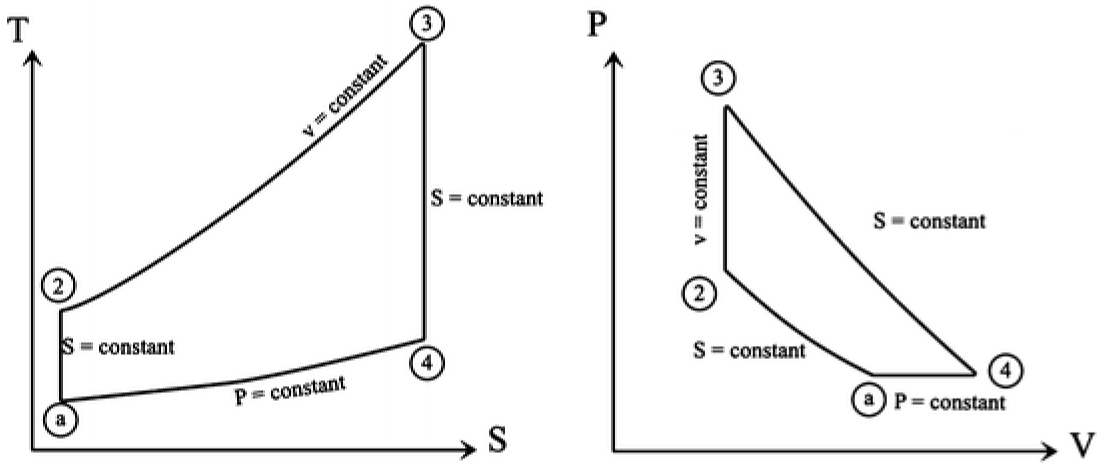
The construction of pulse jet engine is shown in fig. which is similar to ramjet engine.




Inlet diffuser
Valve grid
Combustion chamber
Tail piece
Air
Exhaust gas
Fuel
Spark plug
Thrust F



Operation of a pulsejet



It consists of:

- A valve grid which contains springs that close on their own spring pressure
- A diffuser.

- Combustion chamber
- Spark plug.
- A tail pipe (or discharge nozzle)

- The function of diffuser is to convert the kinetic energy of the entering air into pressure energy.
- The function of nozzle is to convert pressure energy of gas into kinetic

  energy.

WORKING:

1. Air from the atmosphere enters into pulse jet engine. The air velocity gets reduced and its static pressure is increased by diffuser.
2. When a certain pressure difference exists across the valve grid, the valve will open and allow the air to enter into the combustion chamber.
3. In the combustion chamber, fuel is mixed with air and combustion starts by the use of spark plug.
4. Once the combustion starts it proceeds at constant volume. So there is a rapid increase in pressure, which causes the valve to close rapidly.
5. The highly heated products of combustion gases are then allowed to expand in the exhaust nozzle (Tail pipe) section.
6. In the nozzle pressure energy of the gas is converted into kinetic energy. So the gases coming out from the unit with very high velocity
7. Due to high velocity of gases coming out form the unit, a reaction (or) thrust is produced in the opposite direction. This thrust propels the air craft

8. Since the combustion process causes the pressure to increases, the engine can operate even at static condition once it gets started
9. When the combustion products accelerate from the chamber, they have left a slight vacuum in the combustion chamber. This, in turn, produces sufficient pressure drop across the valve grid, allowing the valves to open again and new charge of air enters the combustion chamber.

ADVANTAGES:

1. Pulse jet engine is very simple device next to ramjet engine
2. Less maintenance.
3. Cost is low
4. Light weight when compared with turbojet engine.
5. Unlike the ramjet engine, the pulse jet engine develops thrust at zero speed.

DISADVANTAGES:

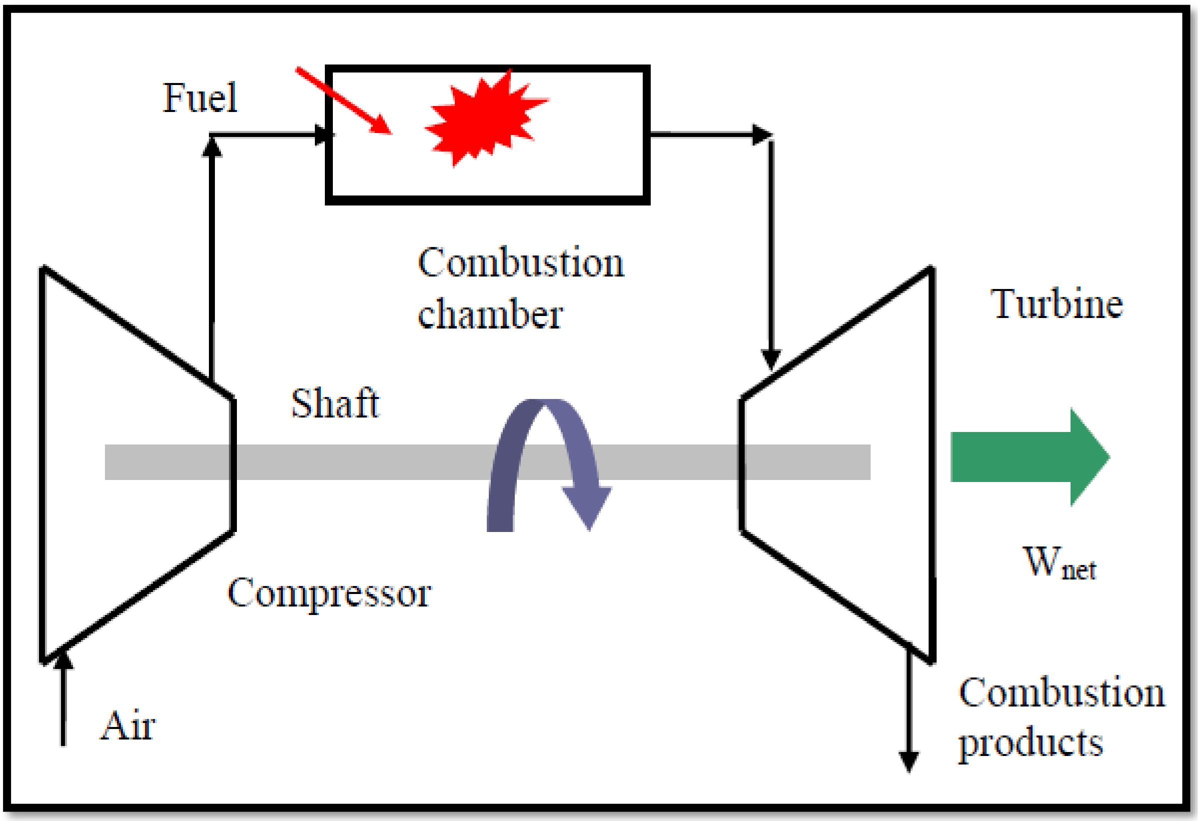
1. High rates of fuel consumption
2. The biggest disadvantage is very short life of flapper valve and high rates of fuel consumption. The SFC is as high as that of ramjet.
3. The maximum flight speed of the pulse jet engine is limited to 750km/h, because of the limitations in the aerodynamic design of an efficient diffuser
4. suitable for a wide range

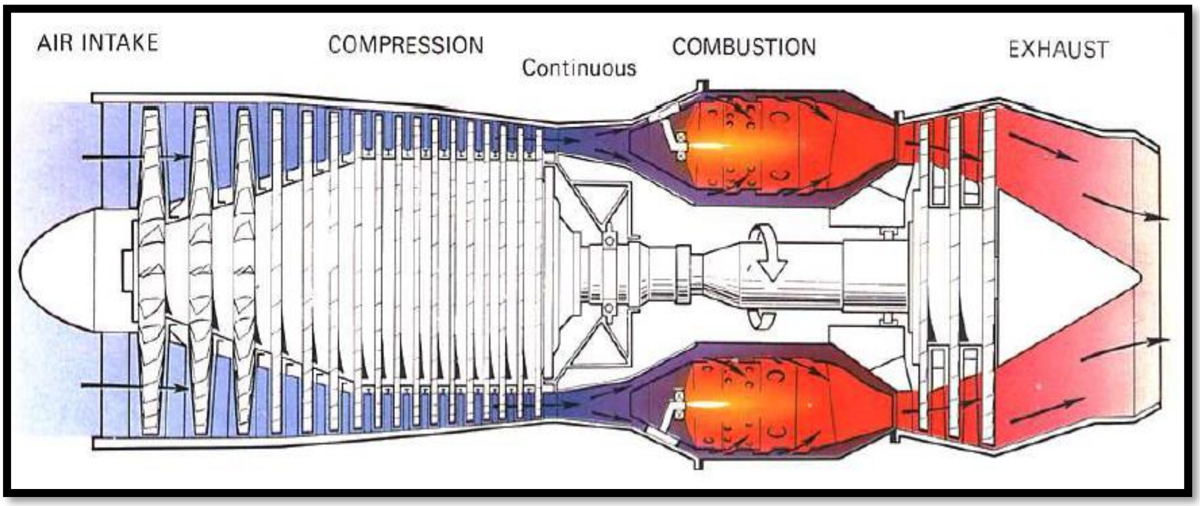
5. Low propulsive efficiency than turbojet engines.
6. High degree of vibration leads to noise pollution

Applications:

It is used in subsonic flights, German V-I missiles, Target aircraft missiles, etc.

Thermodynamic Cycle Analysis of a Single Spool Turbojet Engine





Brayton Cycle

- Brayton cycle is the ideal cycle for gas-turbine engines in which the working fluid undergoes a closed loop. For gas turbine cycle where, working fluid is air/gas and it will perform open cycle of Brayton cycle.
- That is the combustion and exhaust processes are modelled by constant-pressure heat addition and rejection, respectively.
- The Brayton ideal cycle is made up of four internally reversible processes

1-2	Isentropic compression (in compressor)
2-3	Const. pressure heat-addition (in combustion chamber)
3-4	Isentropic expansion (in turbine)
4-1	Const. pressure heat rejection (exhaust)

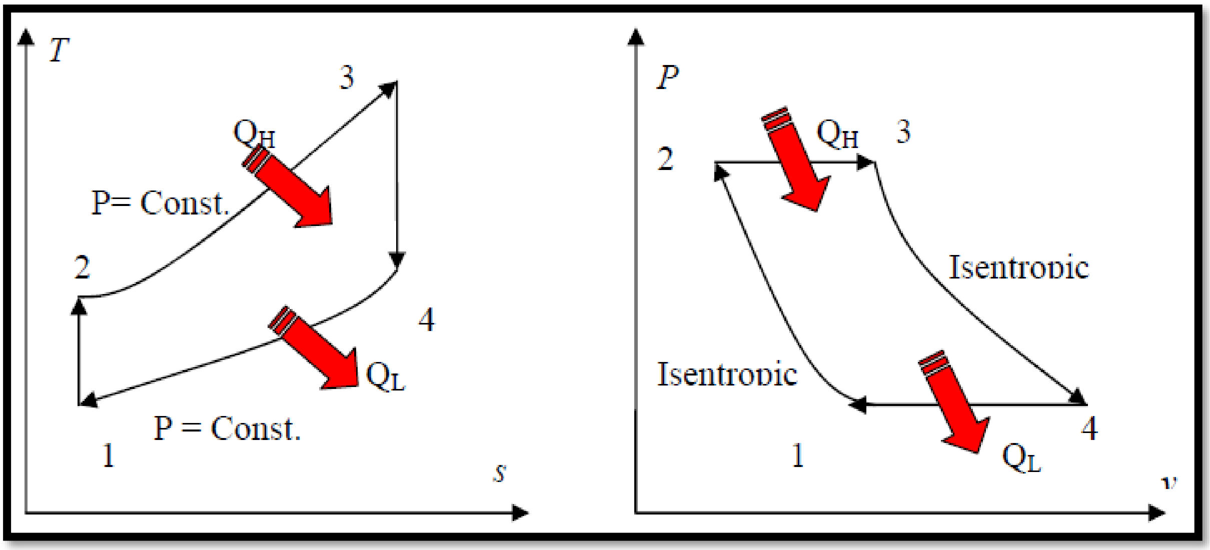


Figure: T-s and P-v diagrams for ideal Brayton cycle.

Thermal efficiency for the Brayton cycle is:

$$\eta_{th,Brayton} = 1 - \frac{q_{out}}{q_{in}} = 1 - \frac{c_p\left(T_4 - T_1\right)}{c_p\left(T_3 - T_2\right)} = 1 - \frac{T_1\left(\dfrac{T_4}{T_1} - 1\right)}{T_2\left(\dfrac{T_3}{T_2} - 1\right)}$$

As we know for isentropic process:

$$\frac{T_2}{T_1} = \left(\frac{P_2}{P_1}\right)^{\frac{(\gamma-1)}{\gamma}}$$

Or

$$\frac{T_2}{T_1} = \left(\frac{P_2}{P_1}\right)^{\frac{(\gamma-1)}{\gamma}} = \left(\frac{P_3}{P_4}\right)^{\frac{(\gamma-1)}{\gamma}} = \frac{T_3}{T_4}$$

$$OR$$

$$\frac{T_3}{T_2} = \frac{T_4}{T_1}$$

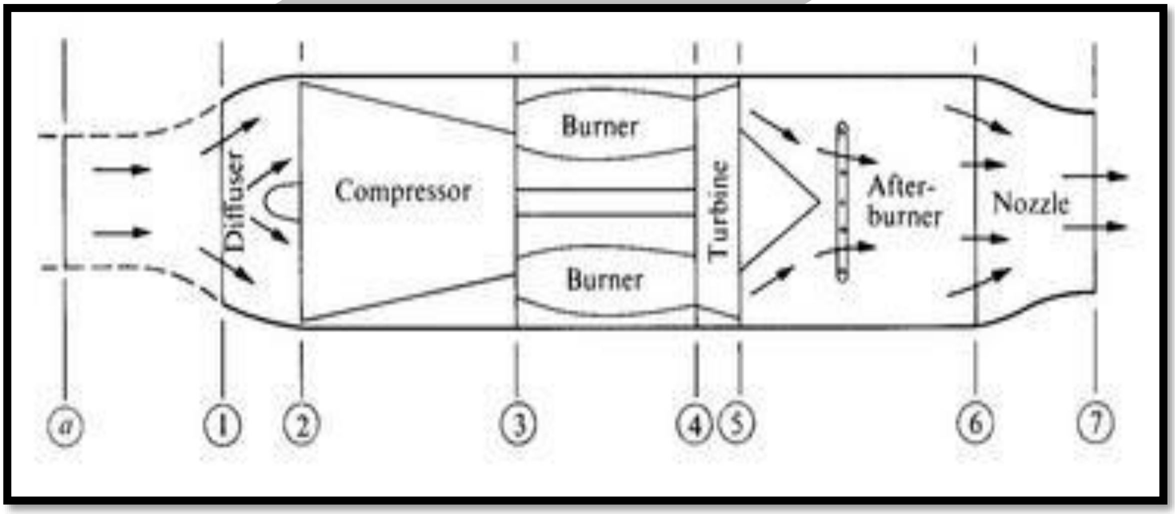
Thus

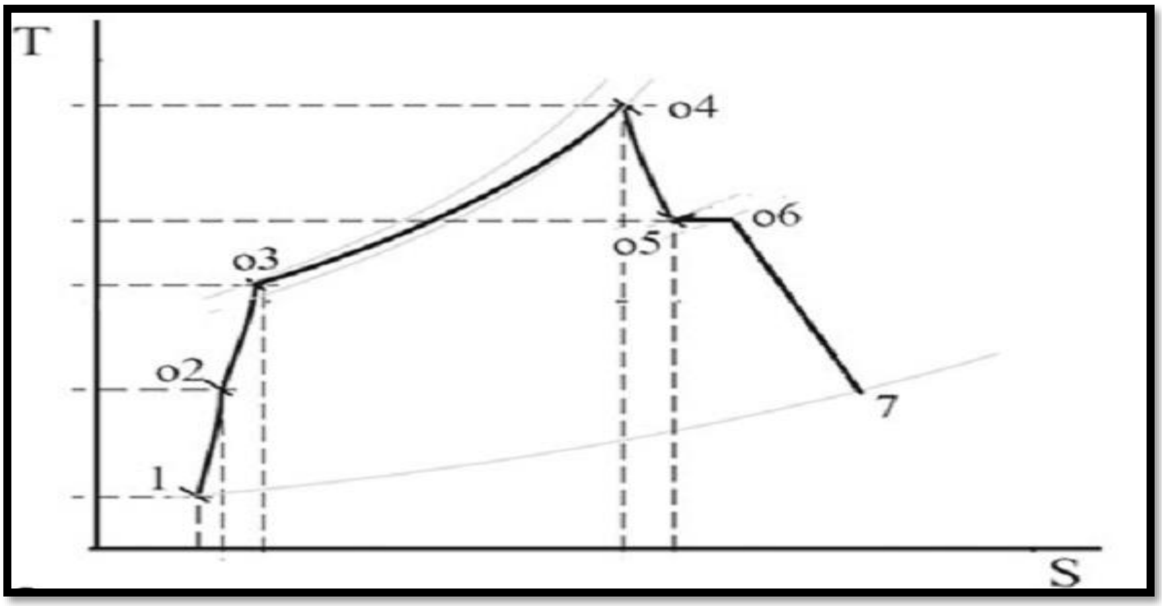
$$\boxed{\eta_{th,Brayton} = 1 - \frac{1}{r_p^{\frac{(\gamma-1)}{\gamma}}}}$$

Where r_p is the pressure ratio which is given by

$$r_p = \frac{P_2}{P_1} = \frac{P_3}{P_4}$$

Turbojet real cycle





(1): Air flows from far upstream up to the intake, usually with some deceleration during cruise and acceleration during takeoff

(1)– (2): Air flows through the inlet (or intake) and ducting system up to the compressor inlet.

(2)– (3): Air is compressed in a dynamic rotating compressor (AFC/CFC).

(3)– (4): Air is heated by mixing and burring of fuel within the combustion chamber.

(4)– (5): Products of combustion are expanded through a turbine to obtain power to
drive the compressor.

(5)– (6): Gases may or may not be further heated if the afterburner is operative or inoperative.

(6)– (7): Gases are accelerated and exhausted through the exhaust nozzle.

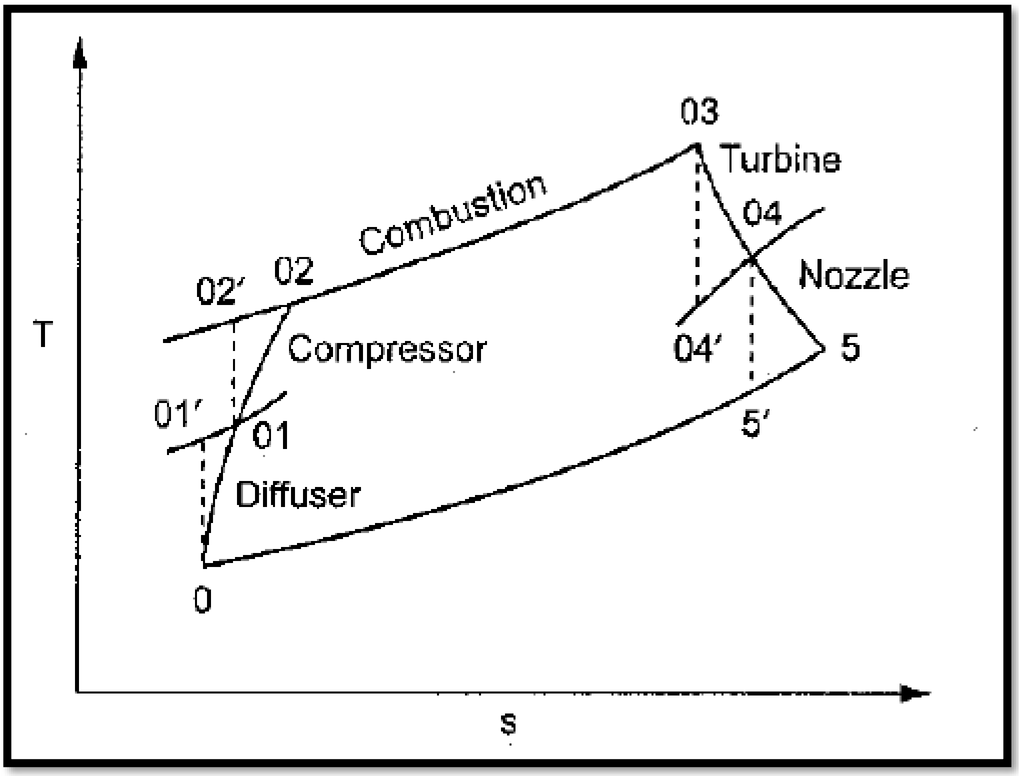


Fig-Real cycle of Turbojet engine

Thrust Equation for a Turbojet Engine

- Let jet engine takes in air at essentially the free stream velocity V_∞, heats it by combustion of fuel inside the duct, and then blasts the hot mixture of air and combustion products out the back end at a much higher velocity V_e

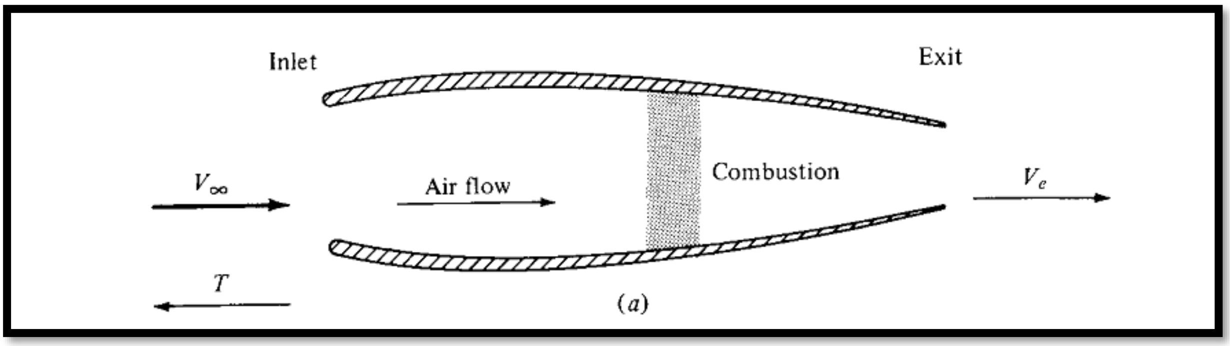


Figure-a

- By Newton's third law, the equal and opposite reaction produces a thrust. The true fundamental source of the thrust of a jet engine is the net force produced by the pressure and shear stress distributions exerted over the surface of the engine. This is sketched in Figure b

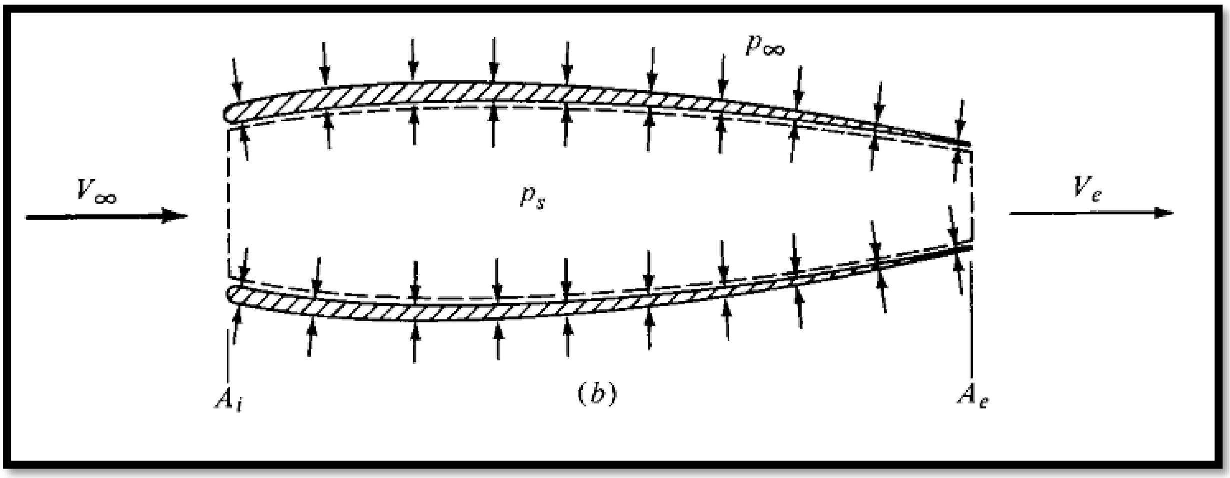


Figure-b

- Which illustrates the distribution of pressure Ps over the internal surface of the engine duct, and the ambient pressure, essentially P_∞ over the external engine surface. Shear stress, which is generally secondary in comparison to the magnitude of the pressures, is ignored here.

- Let x denote the flight direction. The thrust of the engine in this direction is equal to the x component of integrated over the complete internal surface, plus that of pressure integrated over the complete external surface.

- In mathematical symbols,

$T = \int(P_s ds) + \int P_\infty ds$ .... (1)

- Since P_∞ is constant, the last term becomes

$\int P_\infty ds = P_\infty \int ds = p_\infty (A_i - A_e)$......... (2)

➢ Where A_i and A_e are the inlet and exit areas, respectively, of the duct, we obtain for the thrust T of the jet engine

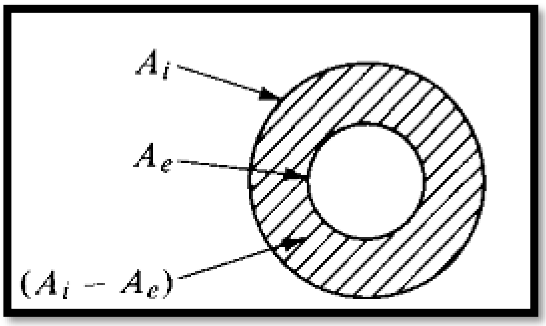


Figure-c

$T = \int P_s ds + P_\infty(A_i - A_e)$............. (3)

➢ Consider the volume of gas bounded by the dashed lines in Figure b. This is called a control volume in aerodynamics. This control volume is sketched again in Figure d.

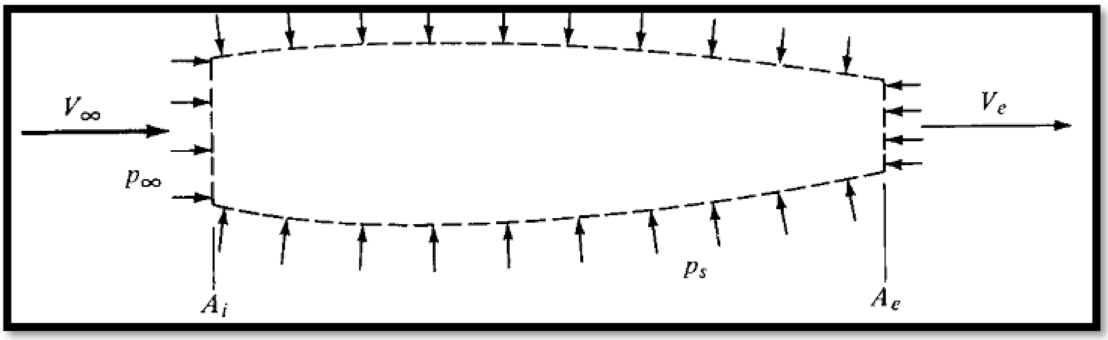


Figure d

➢ The frontal area of the control volume is A_i, on which P_∞ is exerted. The side of the control volume is the same as the internal area of the engine duct.

Since the gas is exerting a pressure P_s in the duct as shown in Figure b, then by Newton's third law, the duct exerts an equal and opposite pressure P_s on the gas in the control volume as shown in Figure d.

- Finally, the rear area of the control volume is A_e on which Pe is exerted. The pressure Pe is the gas static pressure at the exit of the duct, the x component of the force on the gas inside the control volume is

$$F = \int P_s ds + P_\infty A_i - P_e A_e \text{......... (4)}$$

- Newton's second law, namely, F = ma. This can also be written as F = d(mV)/dt. The mass flow of air entering the duct is $(\dot{m}_{air})$ its momentum is $(\dot{m}_{air})V_\infty$. The mass flow of gas leaving the duct is $\dot{m}_{fuel} + \dot{m}_{air}$ and its momentum is $(\dot{m}_{fuel} + \dot{m}_{air})V_e$ .Thus, the time rate of change of momentum of the airflow through the control volume is the difference between what comes out and what comes in:
- From Newton's second law, this is equal to the force on the control volume,

$$F = (\dot{m}_{fuel} + \dot{m}_{air})V_e - (\dot{m}_{air})V_\infty \text{......... (5)}$$

Equating (4) and (5)

$$\int P_s ds = -P_\infty A_i + P_e A_e + (\dot{m}_{fuel} + \dot{m}_{air})V_e - (\dot{m}_{air})V_\infty \text{............... (6)}$$

$$\xrightarrow{6 \text{ in } 3}$$

$$T = (\dot{m}_{fuel} + \dot{m}_{air})V_e - (\dot{m}_{air})V_\infty - P_\infty A_i + P_e A_e + P_\infty(A_i - A_e)$$

$$T = (\dot{m}_{fuel} + \dot{m}_{air})V_e - (\dot{m}_{air})V_\infty + P_e A_e - P_\infty A_e \text{ .... (7)}$$

$$T = (\dot{m}_{fuel} + \dot{m}_{air})V_e - (\dot{m}_{air})V_\infty + (P_e - P_\infty)A_e \text{ .... (8)}$$

- The above equation (8) is the general equation for Thrust generated by the turbojet engine.

- The mass of the fuel is so small compared to mass flow of the air.

$$T = \dot{m}_{air}\left[(1+f)V_e - V_\infty\right] + A_e\left(P_e - P_\infty\right)$$

Where $f = \frac{\dot{m}_f}{\dot{m}_a} = fuel\ air\ ratio$

- Hence
- When mass flow of the fuel is neglected, The above thrust equation can be written as
- $\dot{m}_{fuel} <<< \dot{m}_{air}$

$\boldsymbol{T = (V_e - V_\infty)(\dot{m}_{air}) + (P_e - P_\infty)A_e}$ .... **(8)**

$$\boxed{\begin{gathered}\boldsymbol{T = (V_e - V_\infty)(\dot{m}_{air}) + (P_e - P_\infty)A_e} \\ \text{Or} \\ \boldsymbol{T = (C_j - C_i)(\dot{m}_a) + (P_e - P_\infty)A_e}\end{gathered}}$$

Note

- Thrust Equation of turbojet engine is consisting 2 parts

- ✓ **Part-1**: - Momentum thrust$=\dot{m}_{air}\left(V_e - V_\infty\right) = \dot{m}_a\left(C_j - C_i\right)$
- ✓ **Part-2**: - Pressure thrust$=\left(P_e - P_\infty\right)A_e$
- ✓ **Product of $\dot{m}_{air}$ and V_e is called as gross momentum thrust**

Hence

$$\text{Gross momentum thrust}=\dot{m}_{air}V_e=\dot{m}_{air}C_j$$

- ✓
- ✓
- ✓ Product of $\dot{m}_{air}$ and V_∞ called as **Intake drag**

Similarly

$$\text{Intake drag}=\dot{m}_{air}V_\infty=\dot{m}_{air}C_i$$

- ➢ **In case of optimum expansion in nozzle , thrust will be maximum in this case :-**

Hence

For maximum thrust i.e. optimum expansion at nozzle end $(P_e=P_\infty)$

$$F=Thrust=T=\dot{m}_{air}\left(V_e-V_\infty\right)=\dot{m}_a\left(V_e-V_\infty\right)$$

Thrust Equation for a Turbofan Engine

$$T = \dot{m}_H\left[(1+f)V_{eHOT} - V_\infty\right] + \dot{m}_C\left(V_{eCOLD} - V_\infty\right) + A_{eHOT}\left(P_{eHOT} - P_\infty\right) + A_{eCOLD}\left(P_{eCOLD} - P_\infty\right)$$

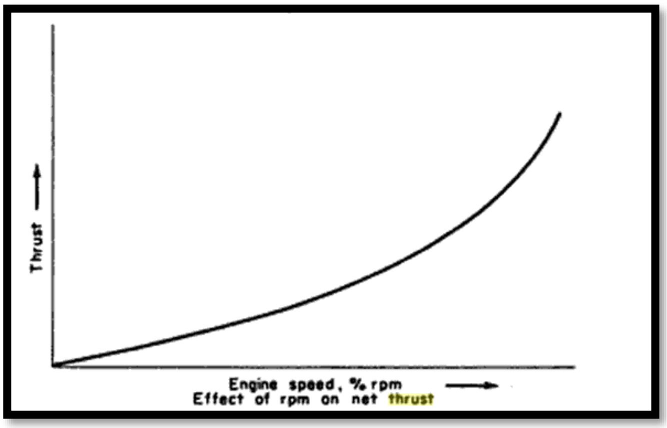
Factors affecting Thrust

- Based on Engine Characteristics
- Based on operating medium

Based on Engine Characteristics

- Engine RPM
- Size of Nozzle Area
- Fuel Flow rate
- Amount of Air Bleed from the compressor
- Turbine Inlet Temperature
- Use of Water Injection

- **Engine rpm effect:** The engine rpm greatly influences the thrust produced by the engine as it determines the air mass flowrate being pumped by the compressor and, from thrust equation it can be seen that the thrust is directly proportional to the air mass flow rate. Since the compressor has fixed blade shapes, it will be more efficient in pumping air at a given rpm range, usually close to the maximum rpm of the engine. An engine rpm is expressed in terms of the maximum rpm for the engine: idle thrust being around 30%, whereas cruise thrust could be at 90% and maximum continuous thrust is at 100% rpm. In some cases, an engine may be able to provide more than 100% continuous thrust for a very short period; this is called the maximum takeoff thrust, and it may damage the engine if used for a prolonged period.



- **Nozzle area:** The nozzle area determines the maximum exhaust mass flow rate, the limiting value being a velocity of Mach 1.0 at exhaust temperature for convergent nozzle. At this condition, the nozzle is said to be choked. Another factor that may limit the kinetic energy available at the nozzle is the maximum operating temperature of the nozzle.

- **Fuel flow rate.** Fuel flow determines the amount of energy transferred to the flow the more fuel is burned in the gas turbine combustion chamber the more heat energy will be available to the flow to be transformed into kinetic energy.

- **Bleed air and power extraction:** Most gas turbine equipped aircraft require pressurization, heating, and ventilation flow. This airflow is bled off from a given section of the compressor, which provides adequate pressure for the environmental control system. This results in a lower pressure in the burner and a lower mass flow at the nozzle, and, thus, a reduction in thrust for a given fuel flow. Other systems, such as the hydraulic system and the electrical system, require a power source. This power is usually extracted from the engine by using a shaft connected to the turbine. This results in less energy in the exhaust flow that can be used for thrust, or for a given thrust, a higher fuel consumption.

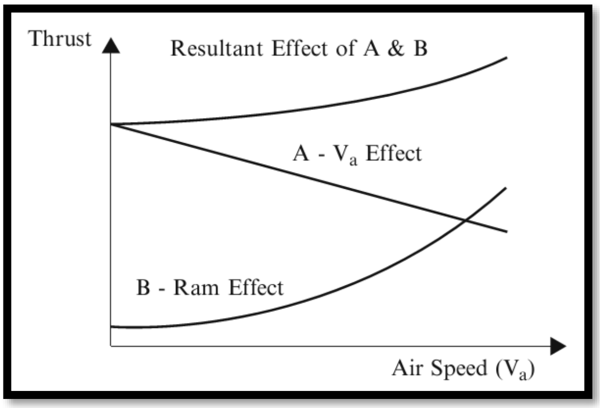
- **Turbine Inlet Temperature:** The total temperature at the turbine inlet (To,) must be limited by the thermal/mechanical limits of the turbine blades this results in less energy in the exhaust flow that can be used for thrust, To maintain an acceptable nozzle temperature under the various combinations of aircraft speed and altitude, the fuel flow may have to be reduced, thus reducing the available thrust This is usually done automatically by the fuel control system

- **Water or Water/Alcohol Injection:** Fluid injection systems were introduced primarily to cool the boosted air to the engine, thereby reducing detonation, also termed pre-ignition, knocking, Pre-ignition can cause severe damage to the engine such as splitting and holing pistons. Adding fluid, either at the inlet to the intercooler, beginning of the inlet manifold or just before the inlet valve, reduces the heat of the inlet air to the engine. This protects against detonation and allows more fuel to be added. The extra fuel can be either the alcohol additive or an increase in the normal fuel. Fluid injection systems have disadvantages which outweigh the advantages, except for specific applications, such as military aircraft or racing aircraft.

Based on operating medium

- Speed of the air
- Temperature of the air
- Pressure of the air
- Humidity of the air

- Speed of the Air: The air speed, or the flight speed has a direct effect on the net thrust. If the exhaust gas velocity is constant and the air velocity is increased, then the difference between both velocities leading to a decrease also in the net thrust. If the air mass flow and the fuel to-air ratio are assumed constants, then a linear decrease in the net thrust is enhanced RAM EFFECT: The movement of the aircraft relative to the outside air causes air to be rammed into the engine inlet duct. Ram effect increases the airflow to the engine, which in turn, means more gross thrust. However, it is not so easy: ram effects combine two factors, namely. The air speed increase and, at the same time, increase in the pressure of the air and the airflow into the engine. As described earlier, the increase of air speed reduces the thrust, which is sketched in Figure as the Curve "A". Moreover, the increase of the airflow will increase the thrust. Which is sketched by the curve "B" in the same. The curve "C" is the result of combining curves "A" and the increase of thrust due to ram becomes significant as the air speed increases, Which Will compensate for the loss in thrust due to the reduced pressure at high altitude. Ram effect is thus important in high-speed fighter aircraft. Also

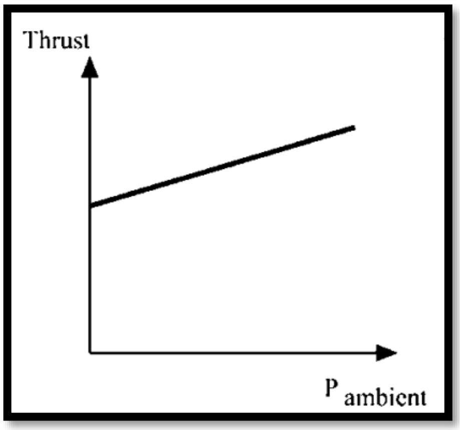


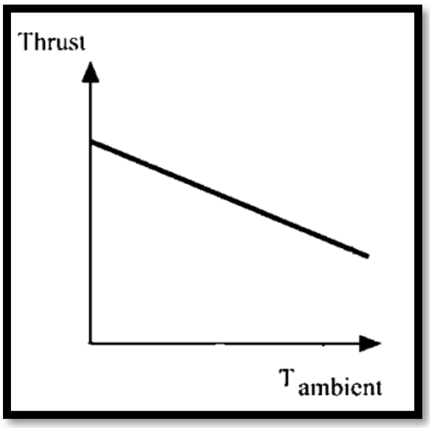
modern subsonic jet-powered aircraft fly at high subsonic speeds and higher altitudes to make use of the ram effect.

- **Air Density & the effect of Temperature & Pressure**

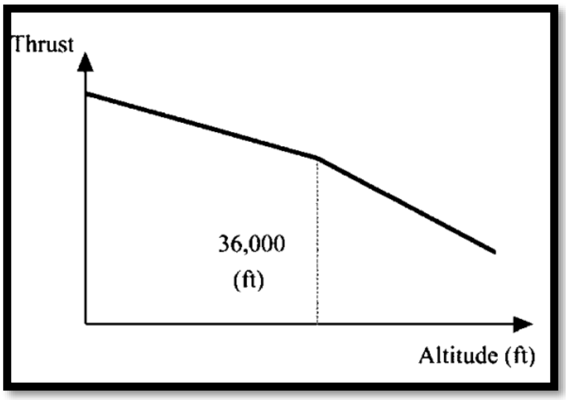
Air density has a profound effect on the thrust produced. The volume of the air flowing through the engine is relatively fixed for any particular rpm by the size and geometry of the inlet duct system. But since the thrust is determined by mass, not the volume of air, any increases in its density increases the mass and thus the thrust.

As the temperature of the air increases its density decreases. Therefore the thrust produced by the engine decreases. As the air pressure increases, its density increases, causing thrust produced by the engine to increase.





- **Altitude:** Altitude has a double effect on thrust. As the altitude increases, the air becomes colder and denser, up to the beginning of stratosphere. This causes the thrust to increase. But at the same time, the increase in altitude causes decrease in pressure, thus a decrease in density and corresponding decrease in thrust. Since the loss of thrust caused by decreasing pressure is greater than the increase caused by decreasing temperature. Thus the thrust decreases as the aircraft ascends. At the beginning of stratosphere at approximately 36,000 feet temperature stabilizes at -56.5^{0}C and remains at this temperature up to around 85000 feet. The pressure continues to fall above the 36,000 feet and the thrust therefore drops off at a faster rate than it does at the lower altitudes. This increased drop off in thrust makes 36,000 feet a chosen altitude for a long range cruise in jet powered aircraft.

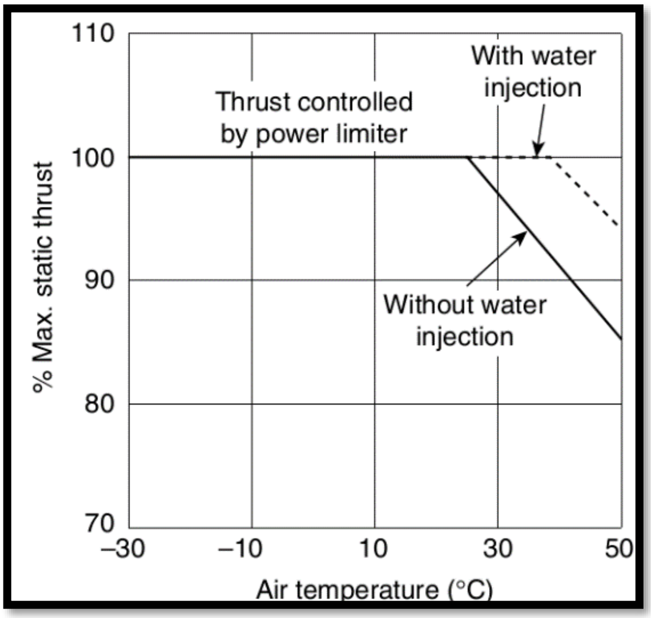


Thrust augmentation:

To achieve better take-off performance, higher rates of climb and increased performance at altitude during combat manoeuvres, there has been a demand for increasing the thrust output of aircraft for short intervals of time. The method of thrust increases the jet velocity is called "**Thrust Augmentation.** One of the following methods may be employed:

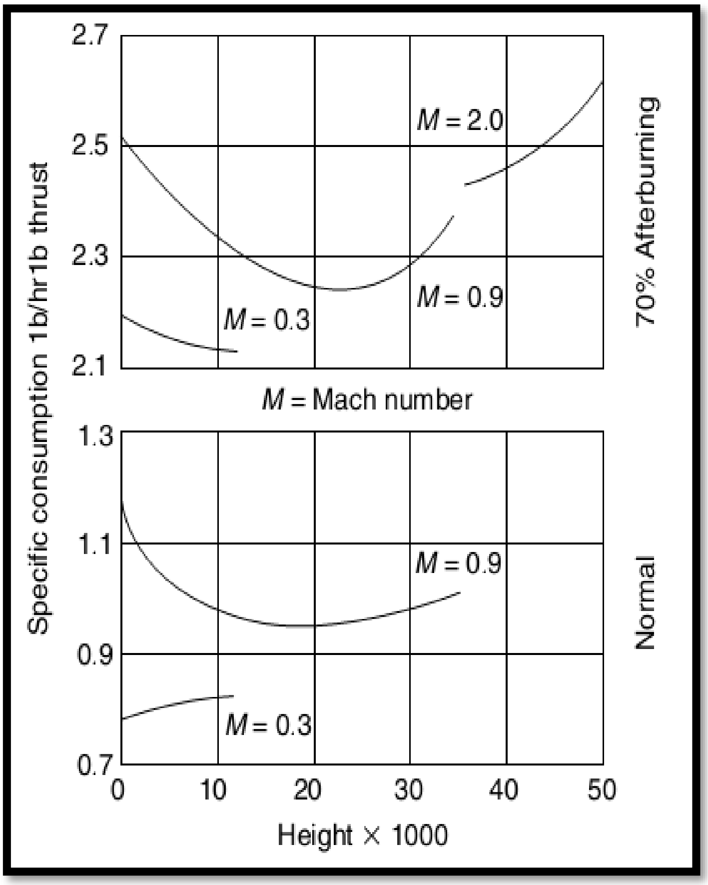
1. Increase of turbine-inlet temperature, which will increase the specific thrust and hence the thrust for a given engine size.
2. Increase of the mass flow rate through the engine without altering the cycle parameter

➢ Water injection
➢ Reheat or afterburning



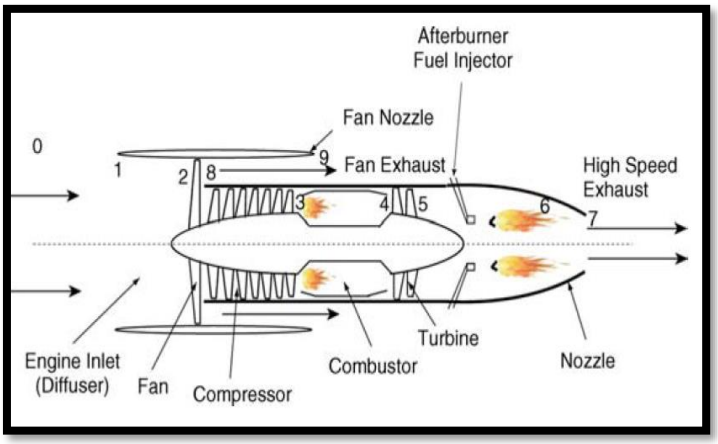
WATERINJECTION: The thrust may be restored or even boosted as much as 10%–30% for takeoff by cooling the airflow with water or water–methanol (methyl alcohol) injection. The technique is to inject a finely atomized spray of water or W/M (a mixture of water and methanol) into either the inlet of compressor or into the combustion chamber inlet. The compressor inlet water injection system is either all water or a mixture of water and methanol, but the combustion chamber water injection is always water and methanol. When methanol is added to water it gives anti freezing properties and also provides an additional source of fuel. The maximum thrust of an engine using water or W/M injection is called the "wet rating." Water has to be very pure since any impurities may cause rapid build up of hard deposits on vanes and rotor blades. It vaporizes rapidly causing intense cooling Turbojet engines usually employ water injection into the combustion chamber inlet, while in turboprop engines water or W/M is injected into the compressor inlet. Once injected into the compressor inlet, the water increases the air density and cools the turbine gas temperature, enabling extra fuel to be burned, which further adds to the power. Injection into the combustion chamber gives better distribution and makes higher water flow rates possible. It increases the mass flow through the turbine, relative to that through the compressor. The pressure and temperature drop across the turbine is reduced as depicted from the following compressor—turbine energy balance.

➢ Reheat or afterburning:



Afterburning (or reheat) is a method of augmenting the basic thrust of an engine to improve the take-off, climb, and combat performance (for military aircraft). Afterburning provides the best method for thrust augmentation for short periods. Afterburning is another combustion chamber located between the LPT and the jet pipe propelling nozzle. Fuel is burnt in this second combustion chamber utilizing the unburned oxygen in the exhaust gas. this will increase the temperature and velocity of the exhaust gases leaving the propelling nozzle and therefore increases the engine thrust. The burners of the afterburner are arranged so that the flame is concentrated around the axis of the jet pipe. Thus, a portion of the discharge gases flow along the walls of the jet pipe and protect these walls from the afterburner flame, the temperature of which is in excess of 1700.The afterburning jet is larger than that of a normal jet pipe would be for the same engine, to reduce the velocity of the gas stream. The afterburning jet pipe is fitted with either a two-position or a variable-area propelling nozzle.

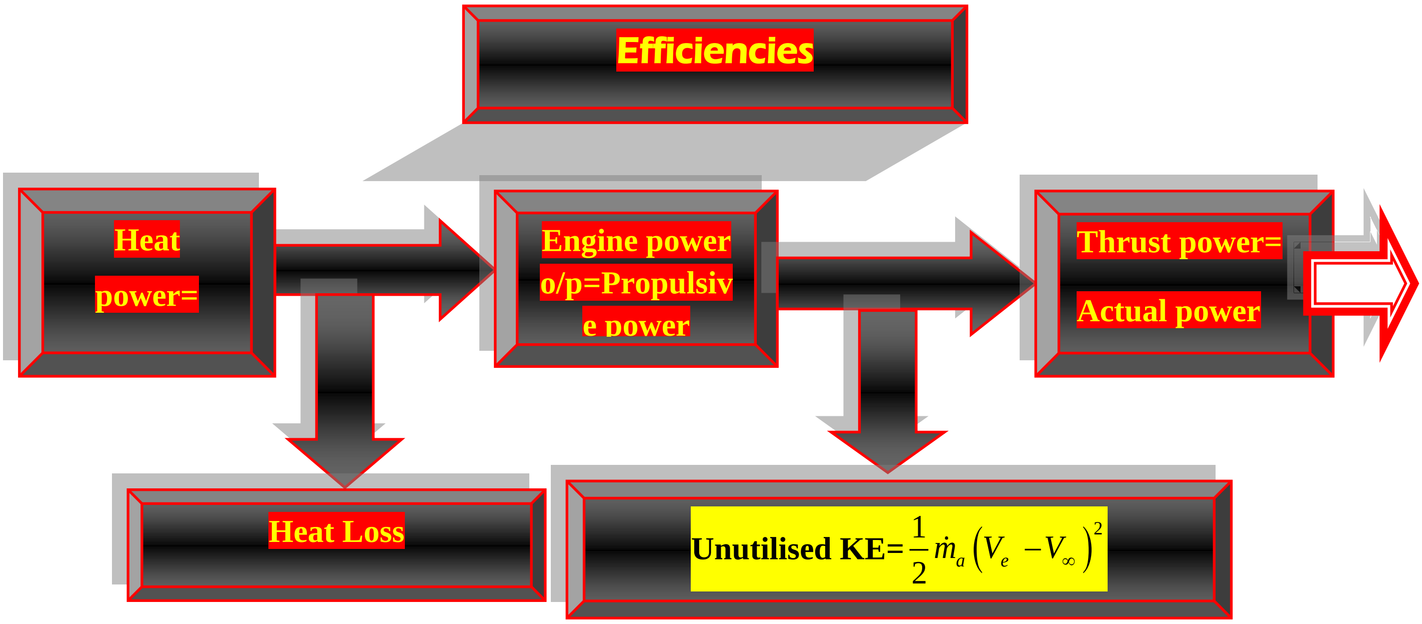
The increase of thrust in afterburning engine is also accompanied by an increase in specific fuel consumption. Therefore, afterburning is limited to periods of short duration. Fuel is not burnt in the afterburner as efficiently as in the engine combustion chamber, since the pressure in the afterburner is not the peak pressure in the cycle.



Engine Performance Parameters

The engine performance parameters are identified as

1. Propulsive efficiency
2. Thermal efficiency
3. Propeller efficiency
4. Overall efficiency
5. Take-off thrust
6. Specific fuel consumption
7. Specific Thrust
8. Non dimensional thrust



Propulsive efficiency

- It can be defined as the ratio of the useful propulsive energy or thrust power (FV_∞)to the sum of that energy and the unused kinetic energy of the jet. η_p is often called the Froude efficiency.
- The propulsion efficiency is a measure of the effectiveness with which the propulsive duct is being used for propelling the aircraft.

- Propulsive efficiency is the conversion of the kinetic energy of air when it passes through

  the engine into a propulsive power.
- It is influenced by the amount of the energy wasted in the propelling nozzle(s) and denoted by η_p.

$$\eta_p = \frac{Thrust\ power}{Power\ imparted\ to\ engine\ airflow}$$

$$\eta_p = \frac{Thrustpower}{Thrust\ power + Power\ wasted\ \text{in}\ \ \text{the}\ exhaust}$$

$$\eta_p = \frac{FV_\infty}{FV_\infty + \frac{1}{2}\dot{m}_a\left(V_e - V_\infty\right)^2}$$

After solving the above equation we will have following equation.

$$\eta_p = \frac{2\,{}^{V_\infty}\!/\!{}_{V_e}}{1 + {}^{V_\infty}\!/\!{}_{V_e}}$$

Say, If

$$\alpha = speed\ \ ratio = \frac{V_\infty}{V_e} = \frac{C_a}{C_j}$$

$$\boxed{\eta_p = \frac{2\alpha}{1+\alpha}}$$

Explanation:

$$\eta_p = \frac{2\alpha}{1+\alpha}$$

$$\eta_p = \frac{2V_\infty}{V_e + V_\infty} = \frac{2}{1+\frac{V_e}{V_\infty}}$$

Case-1

If the exhaust speed is much greater than the air (flight) speed, $V_e >>> V_\infty$, then the
thrust force $T \to$ maximum, $\eta_p \to 0$.
This case represents the take-off condition where $V_\infty = 0$.

Case-2

If the exhaust speed is nearly equal to the flight speed, $\frac{V_e}{V_\infty} \approx 1$, then the thrust force $T \to 0$, $\eta_p \to$ maximum (100%).

For this reason, turboprop engines have higher propulsive efficiency compared to turbojet engines, as in the former the exhaust speed is close to the flight

speed, while for turbojet engines, the exhaust speed is much higher than the flight speed.

Bypass engines (turbofan and propfan

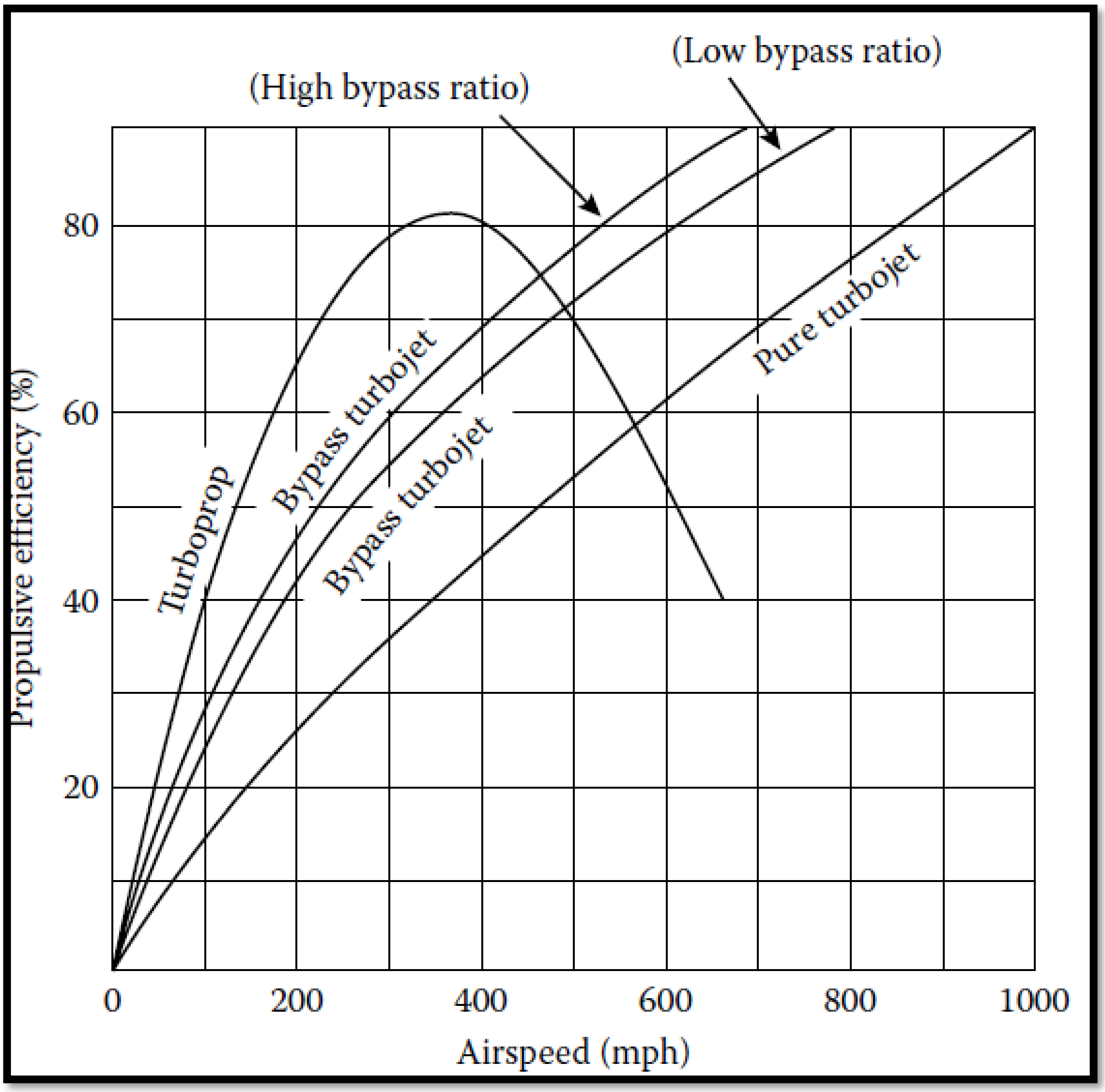
The air coming into the engine is split into two streams: the first passes through the fan/propfan, and is known as the cold stream V_{eCOLD}, while the other passes through the engine core—compressor, combustion chamber, and subsequent modules—and is known as the hot stream V_{eHOT}

$$\eta_p = \frac{2V_\infty\left[V_{eHOT} + \beta V_{eCOLD} - (1+\beta)V_\infty\right]}{V_{eHOT}^2 + \beta V_{eCOLD}^2 - (1+\beta)V_\infty^2}$$

Where (β) is the bypass ratio (BPR), which is the ratio between the mass flow rates of the
cold air and hot air,

or

$$\beta = \frac{\dot{m}_C}{\dot{m}_H}$$


(High bypass ratio)
(Low bypass ratio)
Turboprop
Bypass turbojet
Bypass turbojet
Pure turbojet
Propulsive efficiency (%)
80
60
40
20
0
200
400
600
800
1000
Airspeed (mph)

Thermal Efficiency

Thermal efficiency is the efficiency of energy conversion within the power plant itself. So, it
is considered as an internal efficiency while the propulsive efficiency resembles an external
efficiency. The thermal efficiency refers to the effectiveness of the engine in developing the excess kinetic energy in the exhaust stream.

$$\eta_{th} = \frac{\text{Power imparted to engine airflow}}{\text{Rate of energy supplied in the fuel}}$$

$$\eta_{th} = \frac{\frac{1}{2}\left[\left(\dot{m}_a + \dot{m}_f\right)V_e^2 - \dot{m}_a V_\infty^2\right]}{\dot{m}_f \times HV}$$

Or

$$\eta_{th} = \frac{\dot{m}_a\left[\left(1+f\right)V_e^2 - V_\infty^2\right]}{2\times \dot{m}_f \times HV}$$

Or

$$\dot{m}_a >>>> \dot{m}_f$$

fuel air ratio is neglected

$$f = \frac{\dot{m}_f}{\dot{m}_a} \cong 0$$

$$\boxed{\eta_{th} = \frac{\left[V_e^2 - V_\infty^2\right]}{2 \times f \times HV}}$$

Thermal efficiency for a two-stream engine (i.e. turbofan and propfan),

$$\boxed{\eta_{th} = \frac{V_{eHOT}^2 + \beta V_{eCOLD}^2 - (1+\beta)V_\infty^2}{2f \times HV}}$$

Thermal efficiency: Turboprop and Turboshaft Engines

The output of a turboprop or turboshaft engine is largely a shaft power. In this case, thermal efficiency is defined as

$$\boxed{\eta_{th} = \frac{SP}{\dot{m}_f \times HV}}$$

Where SP is devoted to shaft power

Propeller efficiency

Propellers are used in two types of aeroengines: piston and turboprop. In both cases, shaft power is converted to thrust power. Propeller efficiency (η_{pr}) is defined as the ratio of the thrust power generated by the propeller ($P_T = T_{pr} \times V_\infty$) to the shaft power (*SP*):

$$\eta_{pr} = \frac{Thrust\ power}{Shaft\ power} = \frac{TP}{SP} = \frac{P_T}{SP} = \frac{T_{pr} \times V_\infty}{SP}$$

Overall Efficiency:

The product of the propulsive and thermal efficiencies i.e. ($\eta_p \times \eta_{th}$) is called the overall efficiency.

$$\eta_o = \eta_p \times \eta_{th}$$

Or

$$\eta_o = \frac{TV_\infty}{\dot{m}_f \times HV}$$

Or

$$\eta_o = \left(\frac{2V_\infty}{V_\infty + V_e}\right)\eta_{th}$$

Takeoff Thrust

Take-off thrust is an important parameter that defines the ability of an aeroengine to provide a static and low-speed thrust that enables the aircraft to take off under its own power. Both ramjet and scramjet engines are not self-accelerating propulsion systems from static conditions: they require acceleration to an appreciable velocity by a boost system before they are capable of providing net positive thrust. Thus, they are excluded from this discussion.

The static thrust of a turbojet engine with an unchoked nozzle is expressed by the relation

$$T_{takeoff} = \dot{m}_a \left(1 + f\right) V_e \quad \text{..........(1)}$$

Where for static thrust

$$V_\infty = V_{aircraft} = 0$$

$$\eta_{th} = \frac{\frac{1}{2}\left[\left(\dot{m}_a + \dot{m}_f\right) V_e^2\right]}{\dot{m}_f \times HV} = \frac{\dot{m}_a \left(1 + f\right) V_e^2}{2 \times \dot{m}_f \times HV} \quad \text{..........(2)}$$

By solving equation (1) and (2)

$$T_{takeoff} = \frac{2 \times \eta_{th} \times \dot{m}_f \times HV}{V_e} \quad \text{..........(3)}$$

For a given mass flow rate of fuel and thermal efficiency, from equation (3) we have concluded that:

$$T_{takeoff} \ \alpha \ \frac{1}{V_e}$$

This relation outlines one of the advantages of turboprop engines over turbojet and turbofan engines. Turboprop engines accelerate a large mass flow rate of air to a small exhaust velocity, which in turn increases the takeoff thrust. Consequently, aircraft powered by turboprop engines can take off from a very short runway.

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).

AEROSPACE ADDA
SKY IS THE LIMIT

Thrust specific fuel consumption:

$$\boxed{TSFC = SFC = \frac{\dot{m}_f}{T}}$$

.

For ramjet, turbojet, and turbofan engines

Brake-specific fuel consumption:

$$\boxed{BSFC = SFC = \frac{\dot{m}_f}{SP}}$$

for turboprop and piston-propeller engines

Specific thrust:

The thrust developed per unit mass flow rate of air is known as specific thrust. This provides an indication of the relative size of engines producing the same thrust because the dimensions of the engine are primarily determined by the airflow.

$$F_s = \frac{Thrust}{mass\ flow\ rate\ of\ air} = \frac{T}{\dot{m}_a} = \frac{\dot{m}_a\left(V_e - V_\infty\right)}{\dot{m}_a} = V_e - V_\infty$$

Non-Dimensional thrust

It is defined as the ratio of Thrust to speed of sound is called non dimensional thrust

$$Non\ dimensional\ thrust = \frac{Thrust}{Speed\ of\ sound} = \frac{T}{a}$$

AEROSPACE ADDA SKY IS THE LIMIT

Condition For maximum thrust pow- (Turbojet engine)

We know that

$$Thrust\ power\left(P_T\right)=TV_\infty$$

$$P_T=\dot{m}_a\left(V_e-V_\infty\right)\times V_\infty$$

$$P_T=\dot{m}_a\left(V_e-V_\infty\right)\times V_\infty$$

$$P_T=\dot{m}_a\left(1-\frac{V_\infty}{V_e}\right)\frac{V_\infty}{V_e}V_e^2$$

Or

$$P_T=\dot{m}_a V_e^2\alpha\left(1-\alpha\right)$$

To get maximum thrust power condition for given mass flow rate of air and jet velocity. Differentiate thrust power with respect to α and put equal to zero

$$\frac{dP_T}{d\alpha}=0$$

After solving the above equation, we will have the value of:

$$\alpha = \frac{1}{2}$$

$$\frac{V_\infty}{V_e} = \frac{1}{2}$$

$$\boxed{V_e = 2V_\infty}$$

➢ Now Propulsive efficiency at maximum thrust power condition
➢ By putting the value of α we will get the value as follows

$$\boxed{\eta_p = \frac{2\alpha}{1+\alpha}}$$

Problem: 1

An aircraft fly at 960kmph. One of its turbojet engines takes in 40kg/s of air and expands the gases to the ambient pressure. The air-fuel ratio is 50 and the lower calorific value of the fuel is 43MJ/kg. For maximum thrust power determine

1. Jet velocity
2. Thrust
3. Specific thrust
4. Thrust power
5. Propulsive, thermal and overall efficiency
6. TSFC

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ANSWER

1. 533.4 m/s
2. 11.094 kN
3. 277.35 N/kg/s
4. 2958.77 Kw
5. 66.6%,12.65%,8.42%
6. 0.2596 kg/kN.

Problem: 2

A turbojet engine is powering a fighter airplane. Its cruise altitude and Mach number are 10 km and 0.8, respectively. The exhaust gases leave the nozzle at a speed of 570 m/s and a pressure of 0.67 bar. The exhaust nozzle is characterized by the ratio $\frac{A_e}{\dot{m}_a}$=0.006 $m^2 s/kg$.The fuel-to-air ratio is 0.02. It is required to calculate the specific thrust $\frac{T}{\dot{m}_a}$

Solution:

At altitude 10 km, the ambient temperature and pressure are

$$T_a = 223.3 \text{ K} \quad \text{and} \quad P_a = 0.265 \quad \text{bar}$$

The flight speed $u = M\sqrt{\gamma R T_a} = 239.6$ m/s.

$$\frac{T}{\dot{m}_a} = \left[(1+f)V_e - V_\infty\right] + \frac{A_e}{\dot{m}_a}(P_e - P_\infty)$$

$$\frac{T}{\dot{m}_a} = 584.77 N.s/kg$$ - **Ans**

Problem :3

A turbofan engine is powering an aircraft flying at Mach number 0.9, at an altitude of 33,000 ft, where the ambient temperature and pressure are −50.4°C and 26.2 kPa. The engine BPR is 3, and the hot airflow passing through the engine core is 22.7 kg/s. Preliminary analysis provided the following results:

$$f = 0.015$$

$$P_{eCOLD} = 55.26 kPa \qquad P_{eHOT} = 32.56 kPa$$

$$V_{eCOLD} = 339.7 m/s \qquad V_{eHOT} = 452 m/s$$

$$A_{eCOLD} = 0.299 m^2 \qquad A_{eHOT} = 0.229 m^2$$

Calculate the thrust force

The flight speed is

$$u = M\sqrt{\gamma RT_a} = 269.2 \text{ m/s}$$

The cold airflow $\dot{m}_c = 3 \times 22.7 = 68.1$ m/s.

Fuel flow rate $\dot{m}_f = f \times \dot{m}_h = 0.015 \times 22.7 = 0.3405$ kg/s.

The hot exhaust flow rate $= \dot{m}_h + \dot{m}_f = 23.04$ kg/s.

The thrust force is calculated from the relation

$$\begin{aligned}
T &= \dot{m}_h\left[(1+f)u_{eh} - u\right] + \dot{m}_c\left(u_{ec} - u\right) + A_{eh}\left(P_{eh} - P_a\right) + A_{ec}\left(P_{ec} - P_a\right) \\
&= (22.7)\left[(1.015)(452.1) - 269.2\right] + (68.1)[339.72 - 269.2] \\
&\quad + (0.229)[32.56 - 26.2] \times 10^3 + (0.299)[55.267 - 26.2] \times 10^3 \\
T &= 19.257 \text{ kN}
\end{aligned}$$

$\boxed{T = 19.257 kN \text{ -Ans}}$

Problem :4

The Boeing 747 aircraft is powered by four CF-6 turbofan engines manufactured by General Electric Company. Each engine has the following data:

Thrust force	24.0 kN
Air mass flow rate	125 kg/s
BPR	5.0
Fuel mass flow rate	0.75 kg/s
Operating Mach number	0.8
Altitude	10 km
Ambient temperature	223.2 k
Ambient pressure	26.4 kPa
Fuel heating value	42,800 kJ/kg

If the thrust generated from the fan is 75% of the total thrust, determine

1. The jet velocities of the cold air and hot gases
2. The specific thrust
3. The TSFC
4. The propulsive efficiency
5. The thermal efficiency
6. The overall efficiency

(Assume that the exit pressures of the cold and hot streams are equal to the ambient pressure.)

Solution:

Since the total air mass flow rate through the engine is $\dot{m}_a = 125$ kg/s, then the flow rates are calculated from the BPR as follows:

$$\dot{m}_c \equiv \dot{m}_{Fan} = \frac{5}{6} \times 125 = 104.2 \text{ kg/s}$$
$$\dot{m}_h \equiv \dot{m}_{core} = (1/6) \times 125 = 20.8 \text{ kg/s}$$

$$\text{Fuel-to-air ratio } f = \dot{m}_{fuel} / \dot{m}_h = \frac{0.75}{20.8} = 0.036$$

$$T_{fan} = 0.75T = 18 \text{ kN}$$
$$T_{core} = 0.25T = 6 \text{ kN}$$

$$\text{Flight speed } U = M\sqrt{\gamma R T_a} = 0.8\sqrt{1.4 \times 287 \times 223.2}$$

$$U = 240 \text{ m/s}$$

Exit velocity from fan (cold air)

$$\text{Since } T_{Fan} = \dot{m}_{Fan}\left(U_{e_{Fan}} - U_{Flight}\right)$$

$$\therefore U_{e_{Fan}} = \frac{T_{Fan}}{\dot{m}_{Fan}} + U_{Flight} = \frac{18,000}{104.2} + 240 = 173 + 240$$
$$U_{e_{Fan}} = 413 \text{ m/s}$$

Exhaust velocity from engine core (hot gases)

$$\text{since } T_{Core} = \dot{m}_{Core}\left[(1+f)U_{eC} - U\right]$$

$$\therefore U_{eC} = \frac{\left[T_{Core}/\dot{m}_{Core} + U\right]}{1+f} = \frac{1}{1.036}\left[6000/20.8 + 240\right]$$

$$U_{eCore} = 510 \text{ m/s}$$

2. Specific thrust $= \dfrac{T}{\dot{m}_a} = \dfrac{24}{125} = 0.192 \text{ kN}\cdot\text{s/kg} = 192 \text{ m/s}.$

3. $TSFC = \dfrac{\dot{m}_{Fuel}}{T} = \dfrac{0.75}{24} = 0.03125 \text{ kg}\cdot\text{Fuel/kN/s}.$

4. The propulsive efficiency
Since both cold (fan) and hot (core) nozzles are unchoked, then the propulsive efficiency can be expressed as

$$\eta_P = \frac{T \times U}{\frac{1}{2}\dot{m}_h\left[(1+f)U_{eh}^2 - U^2\right] + \frac{1}{2}\dot{m}_c\left[U_c^2 - U^2\right]}$$

$$\eta_P = \frac{2 \times 24{,}000 \times 240}{20.8\left[1.036 \times (510)^2 - (240)^2\right] + (104.2)\left[(413)^2 - (240)^2\right]}$$

$$\eta_P = \frac{11.52 \times 10^6}{4.409 \times 10^6 + 11.77 \times 10^6} = 0.712 = 71.2\%$$

5. The thermal efficiency

$$\eta_{th} = \frac{\dot{m}_{Core}\left[(1+f)U_{eCore}^2 - U^2\right] + \dot{m}_{Fan}\left(U_{eFan}^2 - U^2\right)}{2\dot{m}_{fuel}Q_{HV}}$$

$$\eta_{th} = \frac{16.179\times10^6}{2\times0.75\times42.8\times10^6} = 0.2546$$

$$\eta_{th} = 25.46\%$$

6. The overall efficiency

$$\eta_o = \eta_P\eta_{th} = 17.93\%$$

Problem:5

An aircraft flies at a speed of 520 kmph at an altitude of 8000 m. The diameter of the
propeller of an aircraft is 2.4 m and flight to jet speed ratio is 0.74. Find the following:
(i) The rate of air flow through the propeller
(ii) Thrust produced
(iii) Specific thrust
(iv) Specific impulse
(v) Thrust power

Solution:

$$\text{Area of the propeller disc A} = \frac{\pi}{4} d^2$$

$$= \frac{\pi}{4} (2.4)^2$$

$$\boxed{A = 4.52 \text{ m}^2}$$

Aircraft speed or flights peed

$$V_\infty = 520 kmph = 144.44 m/s$$

$$Velocity\ ratio\ \alpha = \frac{V_\infty}{V_e} = \frac{u}{u_e}$$

=0.74

Velocity of jet= V_e **=195.19m/s**

Velocity of airflow at the propeller= $c = \frac{1}{2}(V_\infty + V_e)$ **=169.81 m/s**

Mass flow rate of air-fuel mixture

$$\dot{m} = \rho A c$$

$$= 0.525 \times 4.52 \times 169.81$$

$$\boxed{\dot{m} = 402.96 \ \text{kg/s}}$$

We know that $\dot{m} = \dot{m}_a + \dot{m}_f$

Since mass flow rate of fuel ($\dot{m}_f$) is not given, let us take

$$\dot{m} = \dot{m}_a$$

$$\boxed{\text{Mass flow rate of air } \dot{m}_a = 402.96 \ \text{kg/s}}$$

Thrust produced

Ans: $T = 20.45 \times 10^3 N$

Specific thrust:

Ans: $F_s = \frac{T}{\dot{m}_a} = 50.75 Ns/kg$

Specific impulse

$$\text{Specific Impulse } (I_{sp}) = \frac{F}{W}$$

$$= \frac{F}{\dot{m}\,g}$$

$$= \frac{F}{\dot{m}_a \times g}$$

$$= \frac{20.45 \times 10^3}{402.96 \times 9.81}$$

$$\boxed{I_{sp} = 5.17 \text{ s}}$$

Thrust power

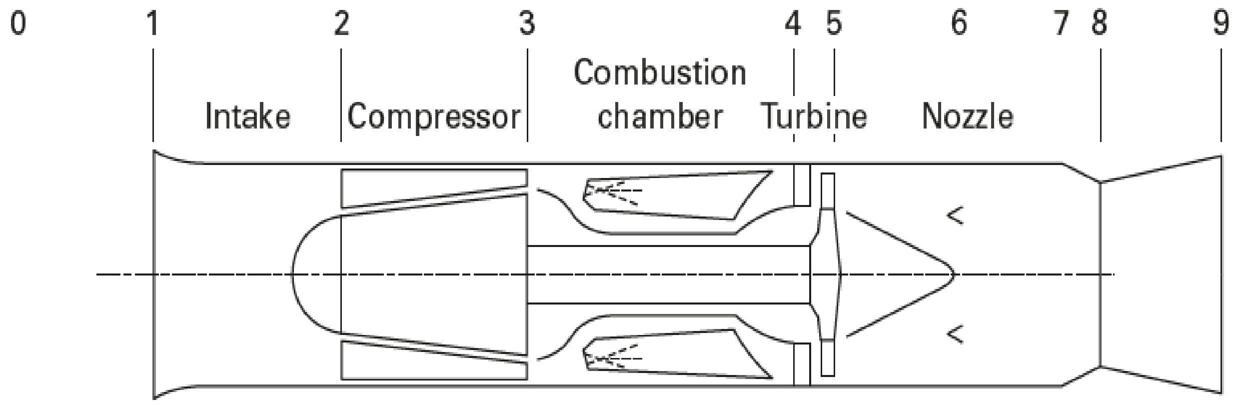
$$P_T = T \times V_\infty$$

Ans:

$$P_T = 2.95MW$$

Standard Station

Manufacturers use a standardized system of numbering based on the Society of Automotive Engineers (SAE) Aerospace Recommended Practice (ARP) 755. This system allows for standardized numbering for specific locations within the engine, irrespective of its configuration. The basis of this system is as follows



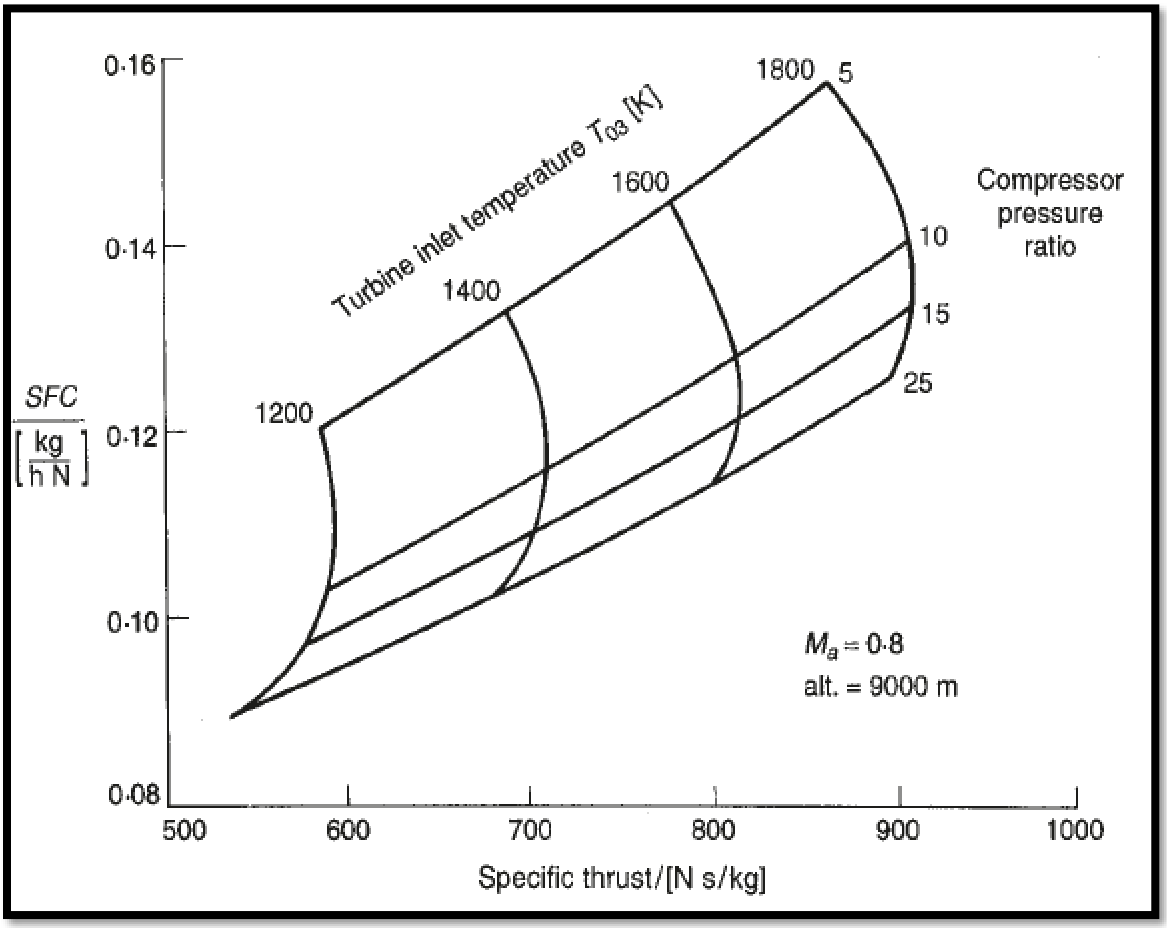
Station	Location
0	Free stream
1	Engine intake
2	Compressor inlet
3	Compressor delivery
4	Turbine inlet
5	Turbine exit
6	Front face of mixer or afterburner
7	Nozzle inlet
8	Nozzle throat
9	Nozzle exit (divergent only)

Optimization of the turbojet Engine/Cycle

Subsonic cruise condition:

- **When considering the design of a turbojet the basic thermodynamic parameters at the disposal of the designer are the turbine inlet temperature and the compressor pressure ratio means basic thermodynamic parameters for the design of turbojet engine are turbine inlet temperature and compressor pressure ratio.**
- **Thrust is strongly dependent on the value of TIT.**
- **At constant pressure ratio an increase in TIT will cause some increase in SFC.**

- The gain in F_s with increasing temperature is invariably more important than the penalty in increased SFC particularly at high flight speeds where small engine size is essential to reduce both weight and drag.
- The effect of increase in pressure ratio is to reduce SFC.
- At fixed value of T_{03}, increasing the pressure ratio initially results in an increase in Fs but eventually leads to decrease.
- The optimum pressure ratio for maximum specific thrust increases as value of TIT increases



Higher cruising speed at the same altitude

- It is found that in general, for any given values of pressure ratio (r_c) and T_{03} , *SFC* is increased and the specific thrust is reduced.
- These effects are due to the combination of increase in inlet momentum drag and an increase in compressor work due to rise in compressor inlet temperature.
- Compressor curves for different altitudes show an increase in F_s and on decrease in SFC with increasing altitude due to fall in temperature and the resultant reduction in compressor work.
- Notable effect of increase in design cruise speed is that the optimum compressor pressure ratio for maximum specific thrust is reduced .This is because of the larger ram compression in the intake
- The higher temperature at the compressor inlet and the need for a higher jet velocity make the use of a high turbine inlet temperature desirable and indeed essential for economic operation of supersonic aircraft.

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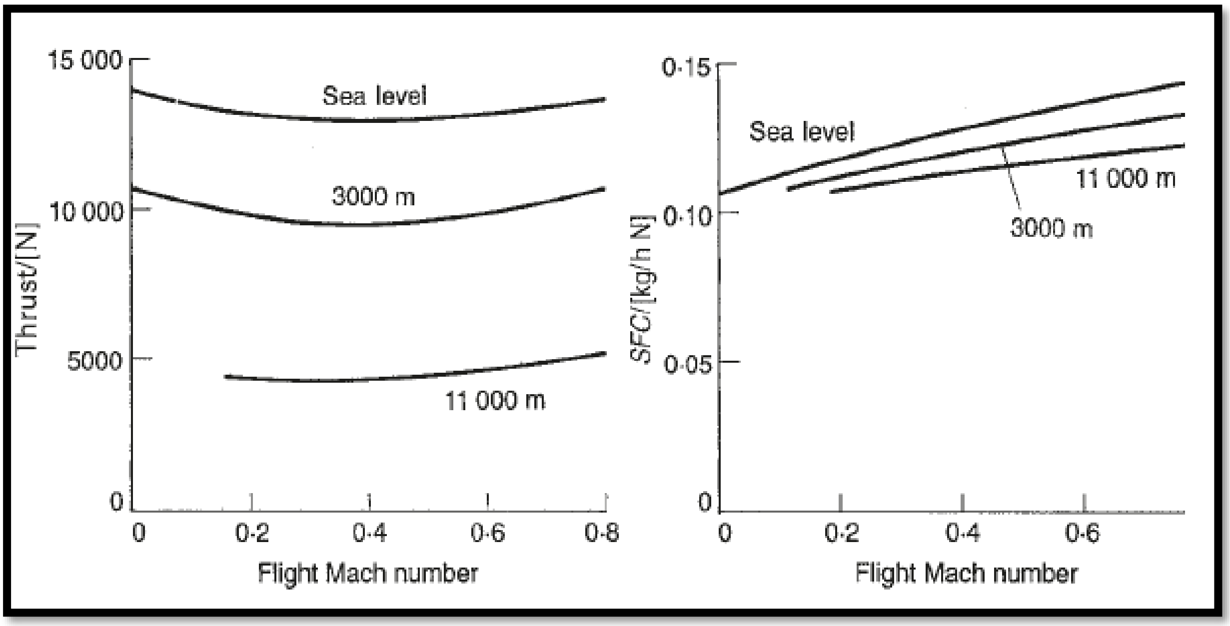
Variation of thrust and *SFC* with Mach number and altitude

- At different flight conditions both the thrust and *SFC* will vary, owing to the change in air mass flow with density and the variation of momentum drag with forward speed.
- If the engine were run at a fixed rotational speed, the compressor pressure ratio and turbine inlet temperature would change with intake conditions.
- Typical variations of thrust and *SFC* with change in altitude and Mach number, for a simple turbojet operating at its maximum rotational speed, are shown in following figure



- It can be seen that thrust decreases significantly with increasing altitude, owing to the decrease in ambient pressure and density, even though the specific thrust increases with altitude owing to the favourable effect of the lower intake temperature.

- SFC, however, shows some improvement with increasing altitude.
- SFC is dependent upon ambient temperature, but not pressure, and hence its change with altitude is not so marked as that of thrust. It is obvious from the variation in thrust and SFC that the fuel consumption will be greatly reduced at high altitudes.
- With increase of Mach number at a fixed altitude the thrust initially decreases, owing to increasing momentum drag, and then starts to increase owing to the beneficial effects of the ram pressure ratio; at supersonic Mach numbers this increase in thrust is substantial.

Optimization of the turbofan

The turbofan engine was originally conceived as a method of improving the propulsion efficiency of the jet engine by reducing the mean jet velocity, particularly for operation at high subsonic speeds.

Turbofan engines are usually described in terms of *bypass ratio,* defined as the ratio of the flow through the bypass duct (cold stream) to the flow at entry to the HP compressor (hot stream). bypass ratio β is given by:

$$\beta = \frac{\dot{m}_{COLD}}{\dot{m}_{HOT}}$$

- Total mass flow rate to the engine is:

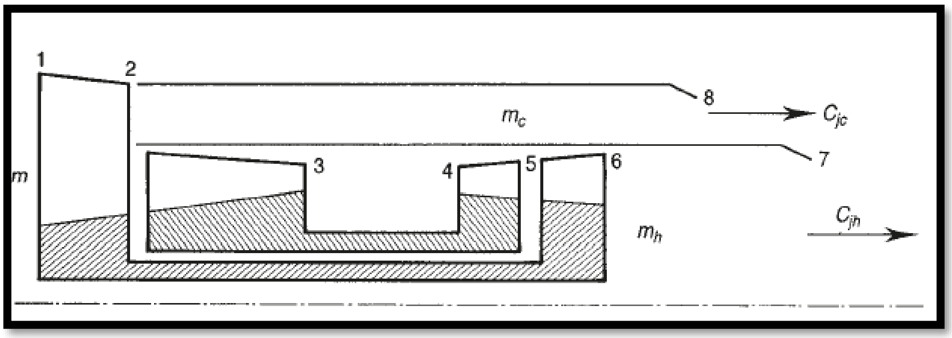
$$\dot{m}_{total} = \dot{m}_{COLD} + \dot{m}_{HOT}$$

➢ Hence primary mass flow rate through core or hot section

$$\dot{m}_{HOT} = \frac{\dot{m}_{total}}{1+\beta}$$

➢ Secondary mass flow rate through fan or cold section

$$\dot{m}_{COLD} = \left(\frac{\beta}{1+\beta}\right)\dot{m}_{total}$$



For the particular case where both streams are expanded to atmospheric pressure in the propelling nozzles, the net thrust is given by

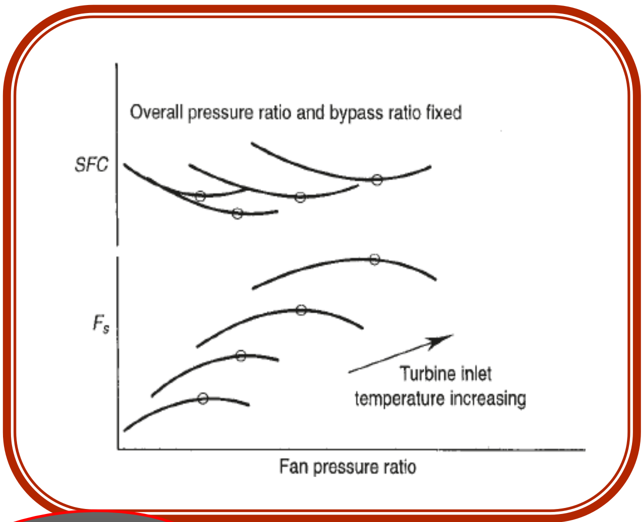
$$T = \dot{m}_H\left[(1+f)V_{eHOT} - V_\infty\right] + \dot{m}_C\left(V_{eCOLD} - V_\infty\right)$$

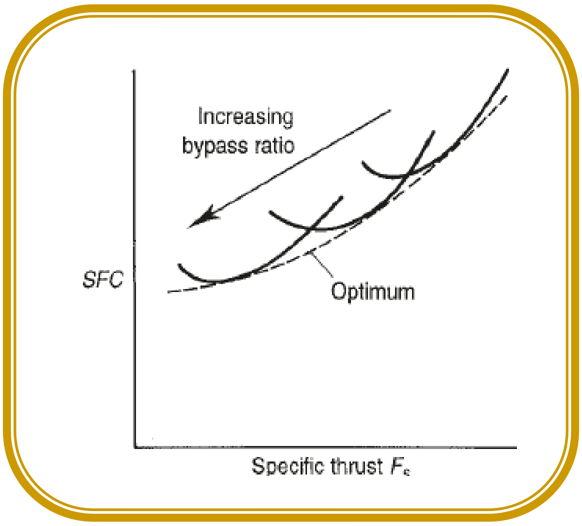
Designers of turbofans have four thermodynamic parameters:

➢ Overall pressure ratio
➢ Turbine inlet temperature
➢ Bypass ratio

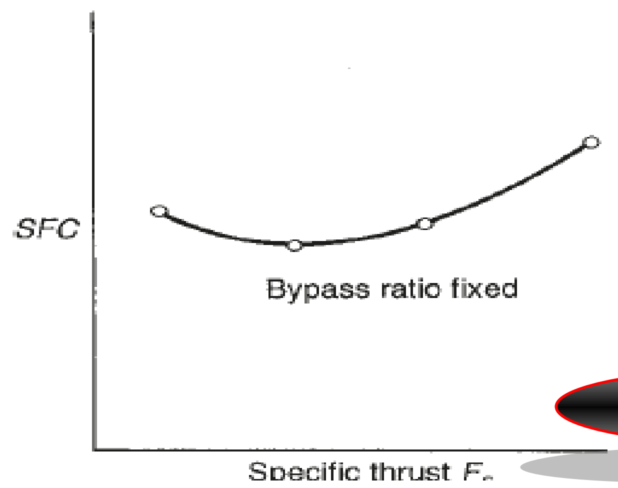
- Fan pressure ratio.

- ✓ The overall pressure ratio and turbine inlet temperatures can be thought of as determining the quality of the engine cycle.
- ✓ While the bypass ratio and fan pressure ratio characterize the effectiveness with which the available energy is converted to thrust





Optimization of fan pressure ratio



For fixed overall pressure ratio and bypass ratio

- ✓ Consider turbofan engine with OPR & BPR fixed.
- ✓ For a particular value of TIT the energy input is fixed.
- ✓ For low value of FPR the fan thrust will be small and the work extracted.

- ➢ From low pressure turbine will also be small. Thus, it energy will be extracted from hot stream and large values of SFC as KE loss in jet is high. As, FPR increases fan thrust increases and hot thrust decreases.

- For a given value of TIT, there will be optimum value of FPR for minimum SFC and maximum specific thrust coincides with minimum SFC.

Variation of SFC with specific Thrust:

- Increasing BPR improves SFC at the expense of a significant reduction in specific thrust.
- The optimum fan pressure increases with turbine inlet temperature.
- The optimum fan pressure decreases with the increase of BPR.
- The optimum SFC requires low specific thrust, and this is particularly important for high bypass turbofans.

Performance characteristics

Turbojet (subsonic)

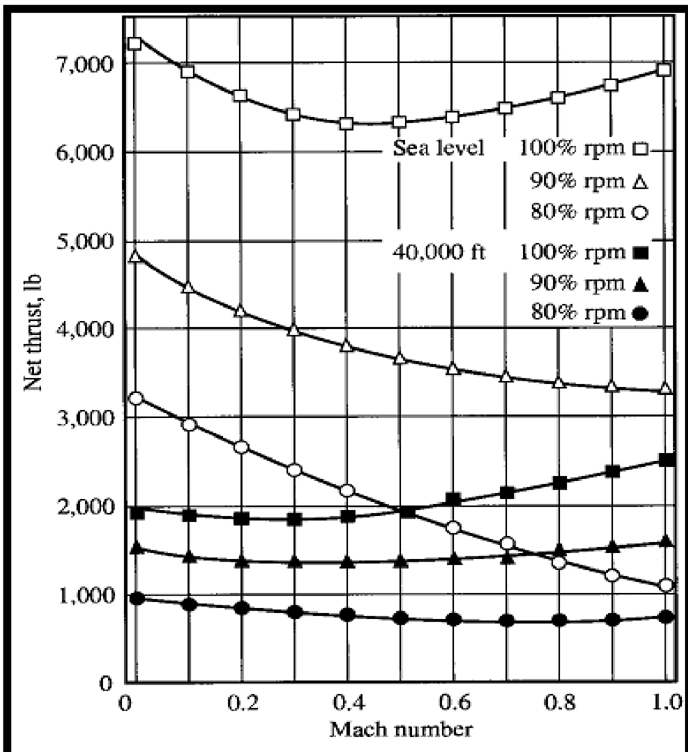


Figure 3.13 Typical results for the variation of thrust with subsonic Mach number for a turbojet.

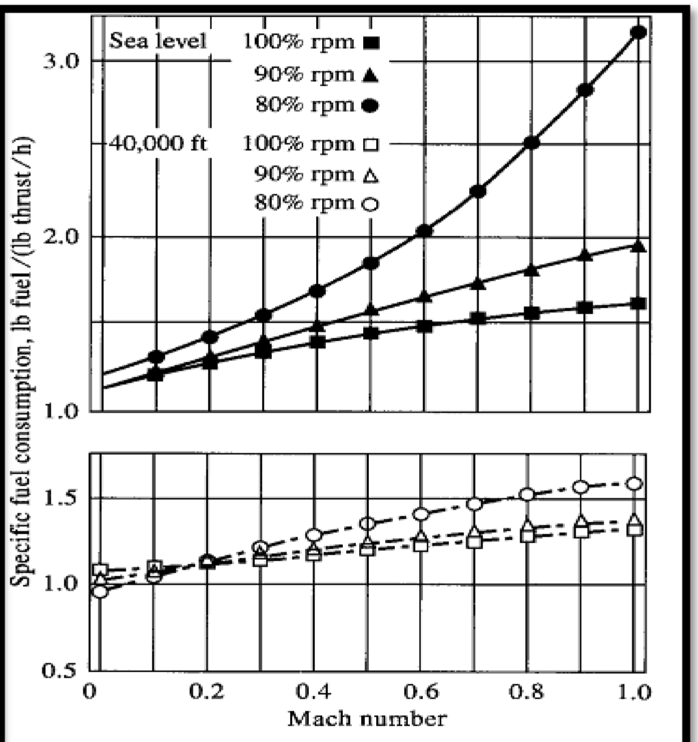


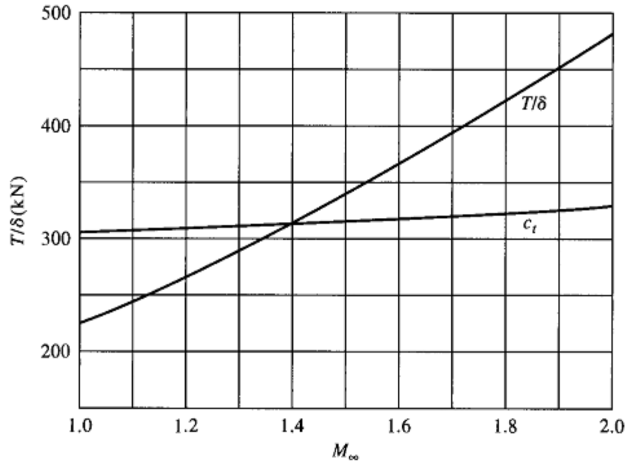
Figure 3.14 Typical results for the variation of thrust specific fuel consumption with subsonic Mach number for a turbojet.

Conclusion:

- Thrust is reasonably constant with increase in Forward velocity
- Thrust decreases with increase in altitude
- TSFC is reasonably constant with increase in forward velocity
- TSFC reasonably constant with increase in altitude

Turbojet (supersonic)

- TSFC is reasonably constant with forward velocity
- The total pressure recovered at the exit of the diffuser is large as M increases, particularly for supersonic values, Plot becomes quite large. This is essentially the pressure of the flow as it enters the compressor, through which the pressure is further increased considerably. The net effect of these higher-pressure levels inside the engine for supersonic flight is that Thrust is greatly increased;



Typical results for the variation of thrust and thrust specific fuel consumption with supersonic Mach number for a turbojet.

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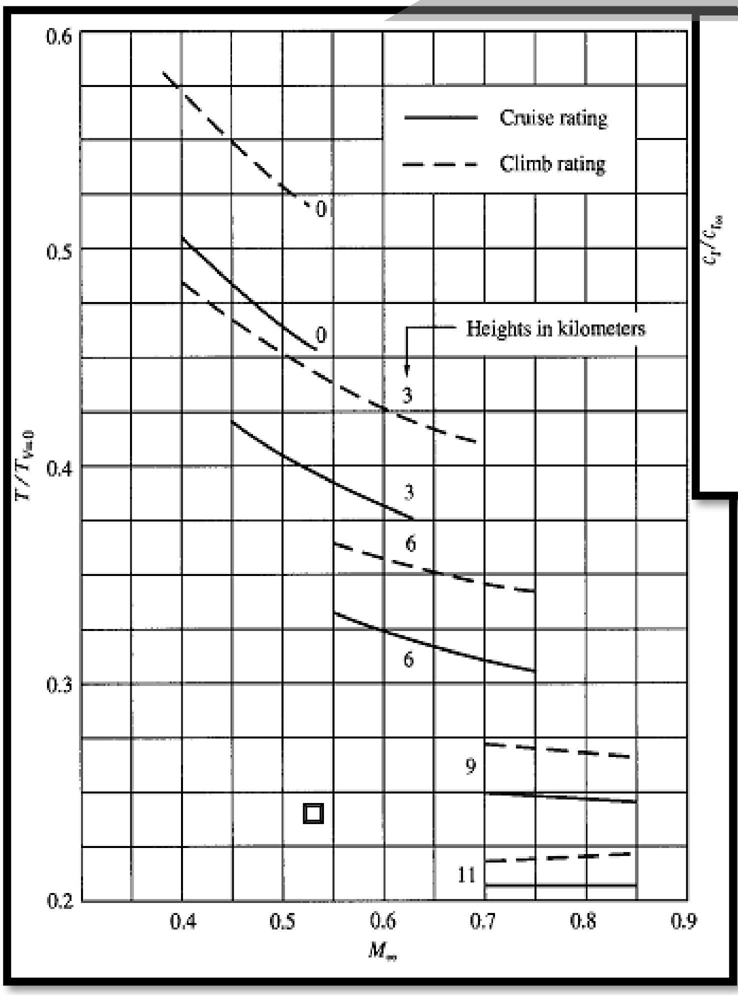
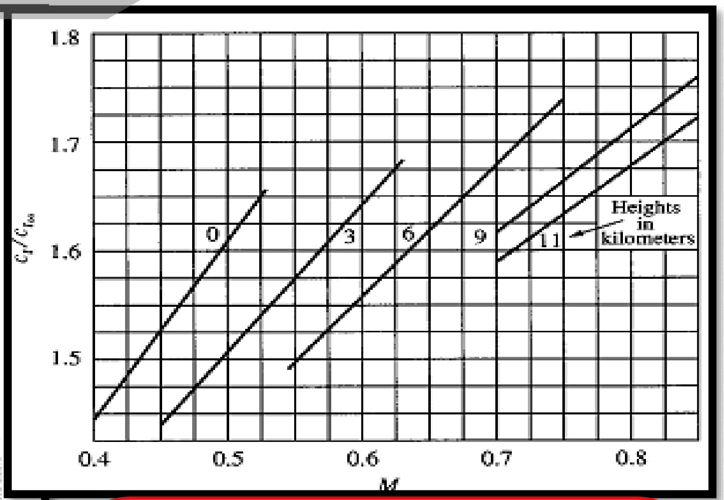
Turbofan (High Bypass)





Conclusions:

- Thrust decreases with increase in Forward velocity, but remains reasonably constant at 0.7<M<0.85
- Thrust decreases with increase in altitude
- TSFC increases increase in forward velocity

Turbofan (Low Bypass)

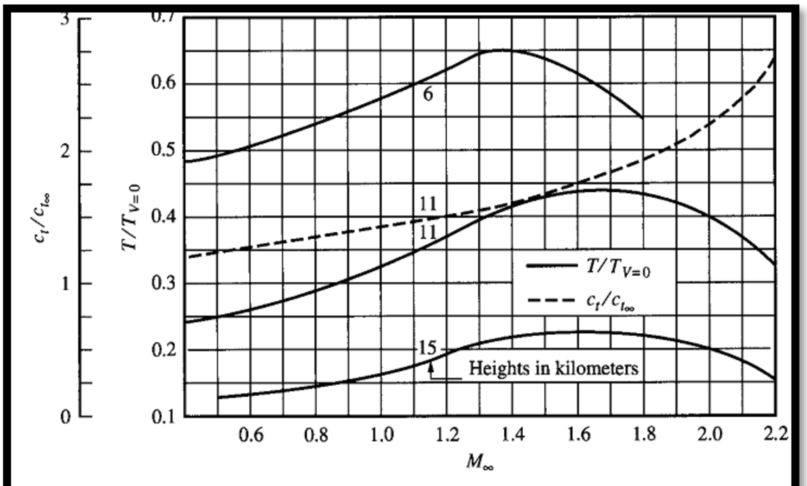


Figure 3.22 Variation of thrust and thrust specific fuel consumption with subsonic and supersonic Mach number and altitude for a generic military turbofan.

Conclusions:

- Thrust decreases with increase in altitude
- Thrust increases with increase in forward velocity(M>1)
- TSFC increases with increase in forward velocity

GATE-AE

Turboprop Engine

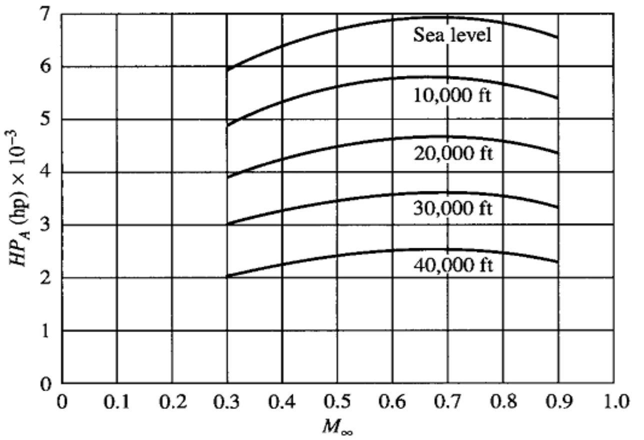
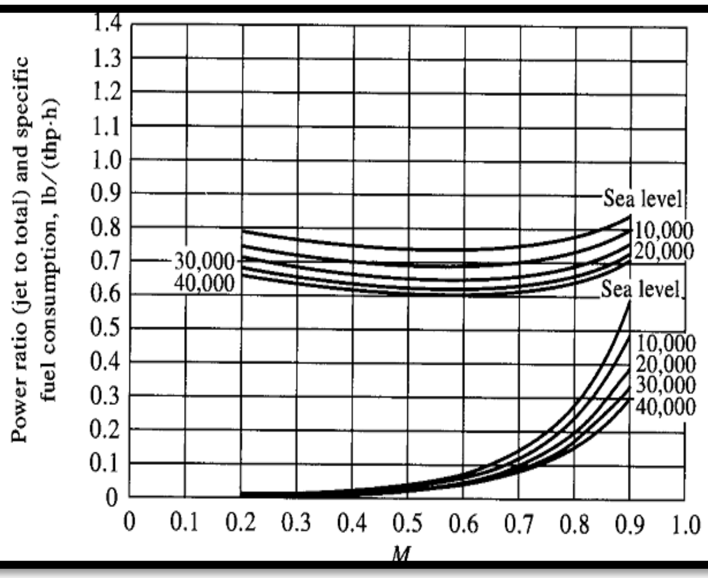


Figure 3.25 Variation of maximum horsepower available HP_A as a function of Mach number and altitude for a typical turboprop engine.



Conclusions:

- Thrust is reasonably constant with increase in Forward velocity
- Thrust decreases with increase in altitude
- TSFC is reasonably constant with increase in forward velocity(M<0.85)
- TSFC reasonably constant with increase in altitude

Inlet/Intake

- The primary purpose of the inlet is to bring the air required by the engine from free stream conditions to the conditions required at the entrance of the fan or compressor with minimum total pressure loss.
- The inlet interchanges the organized kinetic and random thermal energies of the gas in an essentially adiabatic process. The perfect (no-loss) inlet would thus correspond to an isentropic process
- The prime requirement is to minimize the pressure loss up to the compressor face while ensuring that the flow enters the compressor with a uniform pressure and velocity, at all flight conditions. Non-uniform, or distorted, flow may cause compressor surge which can result in either engine flame-out or severe mechanical damage due to blade vibration induced by unsteady aerodynamic effects. Even with a well-designed intake, it is difficult to avoid some flow distortion during rapid maneuvering.
- The intake is a critical part of an aircraft engine installation, having a significant effect on both engine efficiency and aircraft safety.

The Requirements of the inlets:

- High total pressure ratio π_d,
- Controllable flow matching of requirements,
- Good uniformity of flow,
- Low installation drags,
- Good starting and stability,
- Low signatures (acoustic, radar, etc.),
- Minimum weight and cost while meeting life and reliability goals

Note: The fan or compressor works best with a uniform flow of air at a Mach number of about 0.5

A list of the major design variables for the inlet and nacelle includes the following:

- Inlet total pressure ratio and drag at cruise
- Engine location on wing or fuselage (avoidance of foreign-object damage, Inlet flow up wash and downwash, exhaust gas re ingestion, ground clearance)
- Aircraft attitude envelope (angle of attack, yaw angle, cross-wind takeoff)
- Inlet total pressure ratio and distortion levels required for engine operation
- Engine-out wind milling airflow and drag (nacelle and engine)
- Integration of diffuser and fan flow path contour
- Integration of external nacelle contour with thrust reverser and accessories
- Flow field interaction between nacelle, pylon, and wing
- Noise suppression requirements.

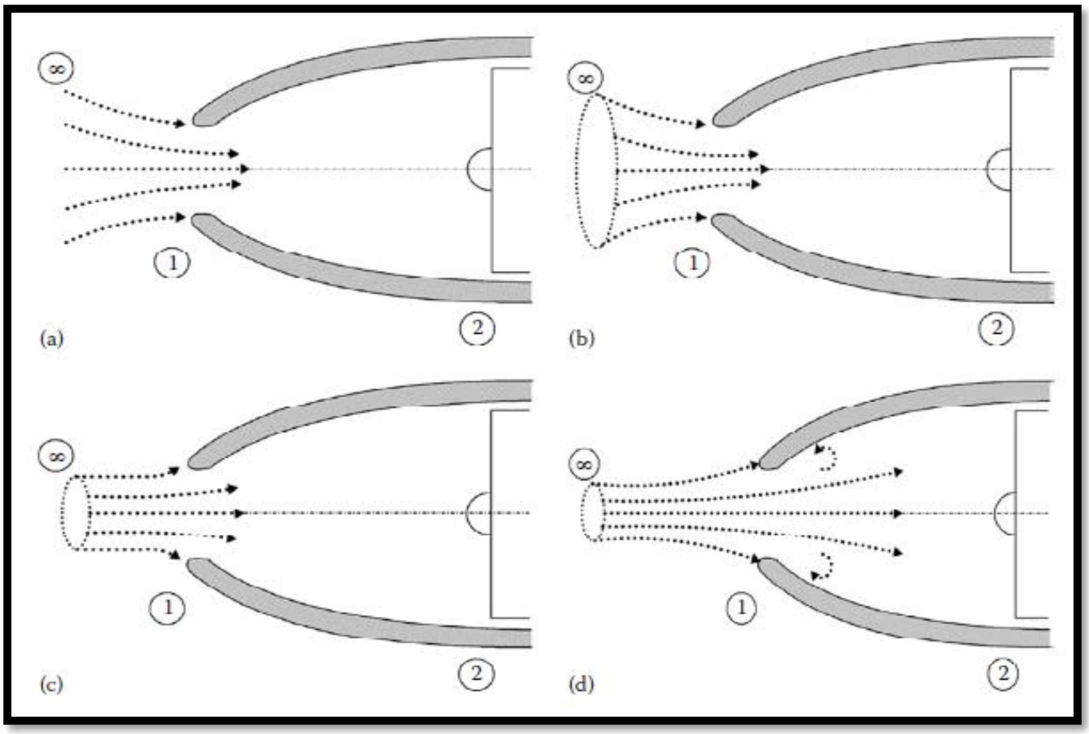
Design considerations:

- The airflow entering the compressor or fan must have low Mach number, in the range 0.4 to 0.7, Part of this deceleration occurs upstream of the inlet entrance plane.
- The inlet must be designed to prevent boundary layer separation, even when the axis of the intake is not perfectly aligned with the streamline direction far upstream of the inlet.
- It is important that the stagnation pressure loss in the inlet be small.
- It is even more important that the flow velocity and direction leaving the inlet be uniform, since distortions in the velocity profile at the compressor inlet can severely upset the compressor aerodynamics and may lead to failure of the blades due to vibrations.

- **Design of inlets that must operate efficiently in both supersonic and subsonic flight poses special problems.**

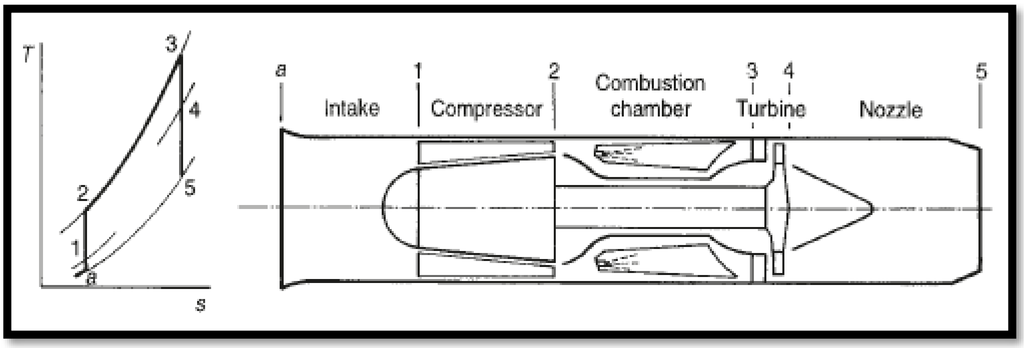
Subsonic Intakes

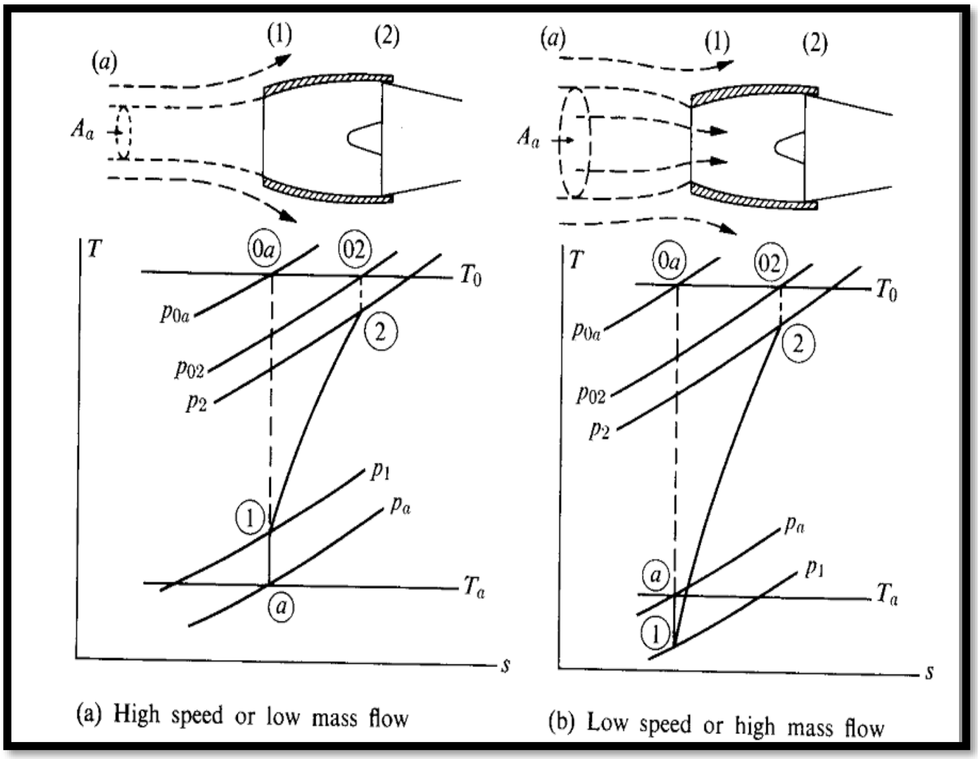
Subsonic intakes are found in the turbojet or turbofan engines powering most of the current civil transports (commercial and cargo aircraft).
A subsonic aircraft has an intake with a relatively thick lip. Concerning turboprop engines, the intakes are much complicated by the propeller and gear box at the inlet to the engine.
Subsonic inlets have fixed geometry, although inlets for some high-BPR turbofan engines
are designed with blow-in-doors. These doors are spring-loaded parts installed in the
perimeter of the inlet duct designed to deliver additional air to the aero engine during
takeoff and climb conditions, as this is when the highest thrust is needed and the aircraft speed is low



Flow characteristics of podded intakes: (a) ground run, (b) climb, (c) high-speed cruise, and (d) top speed

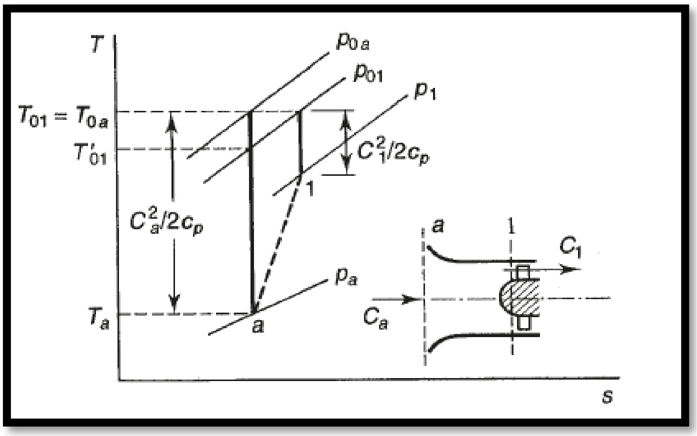
Performance Parameters:





Performance Parameters

The intake efficiency can be expressed in a variety of ways, but the two most commonly used are the isentropic efficiency η_i (defined in terms of temperature rises) and the ram efficiency η_r (defined in terms of pressure rises).



The isentropic efficiency of diffuser is given by as follows:

$$\eta_i = \frac{T'_{01} - T_a}{T_{01} - T_a}$$

The ram efficiency of diffuser is given by as follows:

$$\eta_r = \frac{p_{01} - p_a}{p_{0a} - p_a}$$

Key point of subsonic diffuser

- **Total temperature in diffuser is constant**

$$T_{01} = T_{0a} = T_{\infty} + \frac{V_{\infty}^2}{2c_p}$$

- *or*

$$T_{01} = T_{0a} = T_0 + \frac{C_a^2}{2c_p}$$

- **For isentropic flow**
- $$\frac{p_{01}}{p_a} = \left(\frac{T'_{01}}{T_a}\right)^{\frac{\gamma}{\gamma-1}}$$

$$\eta_i = \frac{T'_{01} - T_a}{T_{01} - T_a}$$

It follows that

$$T'_{01} - T_a = \eta_i \frac{C_a^2}{2c_p}$$

$$\frac{p_{01}}{p_a} = \left[1 + \frac{T'_{01} - T_a}{T_a}\right]^{\gamma/(\gamma-1)}$$

$$= \left[1 + \eta_i \frac{C_a^2}{2c_p T_a}\right]^{\gamma/(\gamma-1)}$$

$$\frac{p_{01}}{p_a} = \left[1 + \eta_i \frac{\gamma - 1}{2} M_a^2\right]^{\gamma/(\gamma-1)}$$

- **Pressure rise in diffuser (Pressure ratio)**
- $$\frac{p_{01}}{p_a} = \left(1 + \eta_i \frac{\gamma - 1}{2} M_a^2\right)^{\frac{\gamma}{\gamma-1}}$$
- **Temperature ratio at inlet of diffuser(Stagnation temperature ratio)**
- $$\frac{T_{01}}{T_a} = \left(1 + \frac{\gamma - 1}{2} M_a^2\right)$$
- **For subsonic :take diffuser isentropic efficiency as 0.93**

Diffuser ram efficiency

The ram efficiency η_r hr is defined by the ratio of the ram pressure rise to the inlet dynamic head, namely

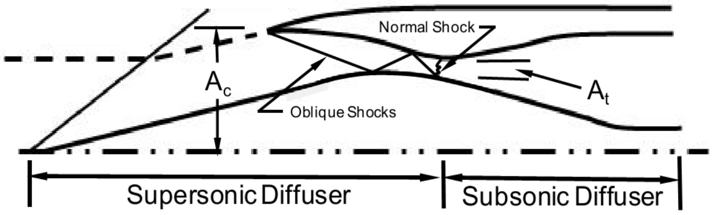
$$\eta_r = \frac{p_{01} - p_a}{p_{0a} - p_a}$$

- $p_{01} - p_a$ is referred as ram pressure rise

- For subsonic intakes, both η_i and η_r are found to be independent of inlet Mach number up to a value of about 0.8 and thence their suitability for cycle calculations.

Supersonic Inlets

- The supersonic inlet is required to provide the proper quantity and uniformity of air to the engine over a wider range of flight conditions than the subsonic inlet is.
- In addition, the nature of supersonic flow makes this inlet more difficult to design and integrate into the airframe.
- In supersonic flight, the flow is decelerated by shock waves that can produce a total pressure loss much greater than, and in addition to, the boundary-layer losses.



- A supersonic inlet is made up of two distinct parts. First the flow is compressed supersonically from the velocity of the flight vehicle or, in other words, the free stream Mach number.
- This is done by reducing the flow area as the flow proceeds downstream. In this region the flow velocity is reduced through a series of compression waves and/or oblique shocks.

- Flow velocity is reduced to a minimum speed at the duct minimum area, called the throat of the inlet, where the flow approaches sonic velocity or a Mach number of one. At this point the flow Mach number will be reduced from supersonic, above one, to subsonic, below one, through a normal shock. This begins the second part of the inlet, the subsonic diffuser.
- In this region the velocity is reduced as the flow area is increased. The result of this process is conditioned air, smooth, subsonic air at high pressure, which is then delivered to the engine.

Key points of supersonic diffuser

- η_i is less than 0.93 in case of supersonic diffuser
- The existence of shock waves, which lead to large decrease in stagnation pressure even in the absence of viscous effects.
- It must operate efficiently both during the subsonic flight phases (takeoff, climb and subsonic cruise) and at supersonic design speed.
- The intake efficiency will depend upon the location of the engine in the aircraft (in wing, pod or fuselage), the value of η_i decreasing with increase in inlet Mach number.
- In practice, neither η_i nor η_r is used for supersonic intakes and it is more usual to quote values of stagnation pressure ratio p_{01}/p_{0a} as a function of Mach number.
- p_{01}/p_{0a} is called the **pressure recovery factor of the intake (PRF).**
- Pressure recovery factor should be as high as possible
- Knowing the pressure recovery factor, the pressure ratio p_{01}/p_a can be found from

where p_{0a}/p_a is given in terms of

$$\frac{p_{0a}}{p_a} = \left[1 + \frac{\gamma - 1}{2} M_a^2\right]^{\gamma/(\gamma-1)}$$

$\frac{p_{01}}{p_{0a}}$ = Pressure recovery factor (PRF)

If PRF is not given then we can take

$$\left(\frac{p_{01}}{p_{0a}}\right)_{\text{shock}} = 1.0 - 0.075(M_a - 1)^{1}$$

➤ $\frac{p_{01}}{p_a} = \frac{p_{01}}{p_{0a}} \times \frac{p_{0a}}{p_a}$

Nozzles

The task of the exhaust nozzle is to convert gas potential energy into kinetic energy (i.e. gas velocity) necessary for the generation of thrust. Propelling nozzle' to refer to the component in which the working fluid is expanded to give a high-velocity jet. This is accomplished solely by the geometrical shape of the nozzle, which is basically a tube of varying cross-section. Not every nozzle type performs in the same manner. Depending on the type of aircraft, and design flight speed, different types of nozzles are employed.

Functions of the Nozzles:

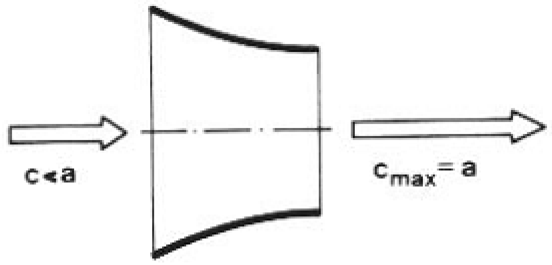
- Accelerate the flow to a high velocity with minimum total pressure loss
- Match exit and atmospheric pressure as closely as desired
- Permit afterburner operation without affecting main engine operation
- Allow for cooling of blades
- Mix core and bypass streams of turbofan if necessary
- Allow for thrust vectoring
- Suppress jet noise and infrared radiation
- Thrust vector control

The exhaust nozzles may be classified as

1. Convergent nozzle

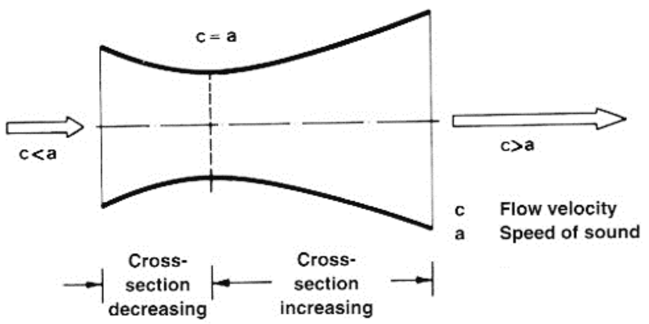
2. C-D nozzle

Convergent nozzle



In a convergent nozzle that the cross-section of a duct decreases in the stream wise direction if a subsonic fluid flow is to be accelerated. A convergent nozzle is fitted to all airliners which fly at subsonic or transonic speeds. Thus it is either of the axisymmetric or annular geometry. All subsonic/ transonic turbojets and turboprop engines have one axisymmetric convergent nozzle

Convergent-Divergent Nozzles



For higher exhaust velocities above Mach 1.5. Convergent-Divergent nozzle shape is required. The geometric characteristic of this nozzle is a decreasing cross- sectional area in its forward part (much like a convergent nozzle), followed by a cross-sectional increase in its rearward portion (the divergent section).

In this nozzle, the subsonic flow is accelerated in the converging section up to the minimum area or throat. It reaches a sonic speed exactly at the throat in the divergent section, pressure is allowed to decrease below its critical value, with fluid velocity continuing to accelerate to supersonic values.

At operating pressure ratios less than the design value, a convergent–divergent nozzle of fixed proportions would certainly be *less* efficient because of the loss incurred by the formation of a shock wave in the divergent portion. For these reasons aircraft gas turbines normally employ a convergent propelling nozzle. A secondary advantage of this type is the relative ease with which the following desirable features can be incorporated:

(a)Variable area: which is essential when an afterburner is incorporated. Earlier engines sometimes used variable area to improve starting performance, but this is not necessary today.
(b) Thrust reverser: to reduce the length of runway required for landing, used almost universally in civil transport aircraft.
(c) Noise suppression: Most of the jet noise is due to the mixing of the high velocity hot stream with the cold atmosphere, and the intensity decreases as the jet velocity is reduced. For this reason the jet noise of the turbofan is less than that of the simple turbojet. In any given case, the noise level can be reduced by accelerating the mixing process and this is normally achieved by increasing the surface area of the jet.

The main limitations on the design are:

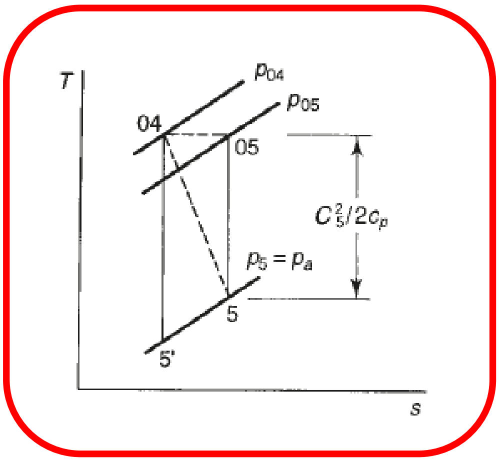
- The exit diameter must be within the overall diameter of the engine otherwise the additional thrust is offset by the increased external drag.

- In spite of the weight penalty, the included angle of divergence must be kept below about 30° because the loss in thrust associated with divergence (nonaxiality) of the jet increases sharply at greater angles.
- The design of the convergent–divergent nozzle was especially critical for supersonic transport aircraft, such as the Concorde, which spent extended periods at high supersonic speeds.

Losses in a Nozzle

- Thrust loss due to exhaust velocity vector angularity.
- Thrust loss due to the reduction in velocity magnitude caused by friction in the boundary layers
- Thrust loss due to loss of mass flow between nozzle entry and exit from leakage through the nozzle walls
- Thrust loss due to flow no uniformities

Isentropic efficiency of nozzle:



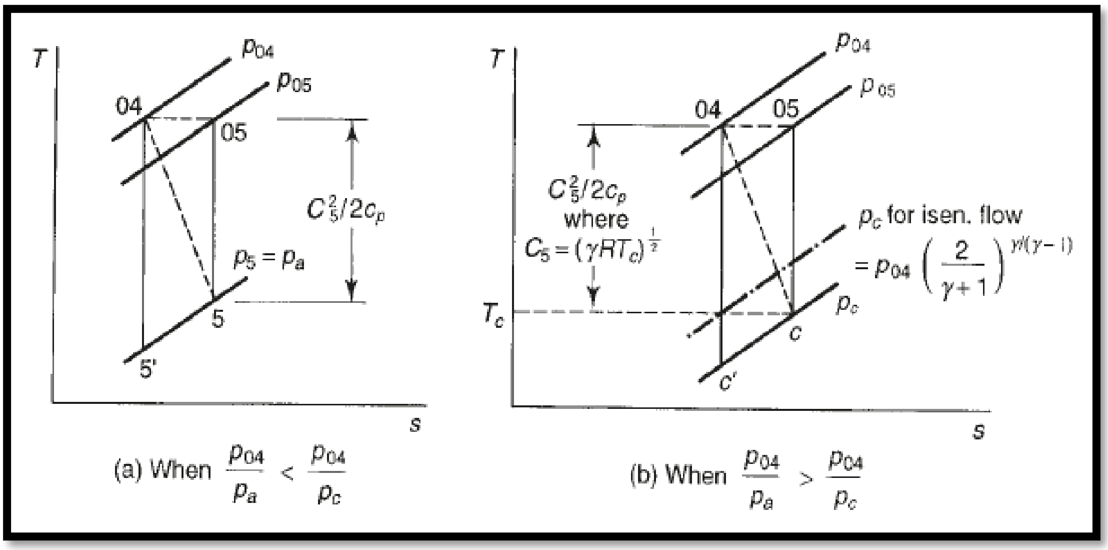
- $\eta_{iN} = \dfrac{Actual\ enthalpy\ drop}{Isentropic\ enthalpy\ drop} = \dfrac{T_{04} - T_5}{T_{04} - T_5'}$

- The value of η_{iN} is obviously dependent on a wide variety of factors, such as the length of the jet pipe, and whether the various auxiliary features mentioned earlier are incorporated because they inevitably introduce additional frictional losses.

- Another factor is the amount of swirl in the gases leaving the turbine, which should be as low as possible
- We shall assume a value of 0.95 for η_{iN} hence $\eta_{iN} = 0.95$

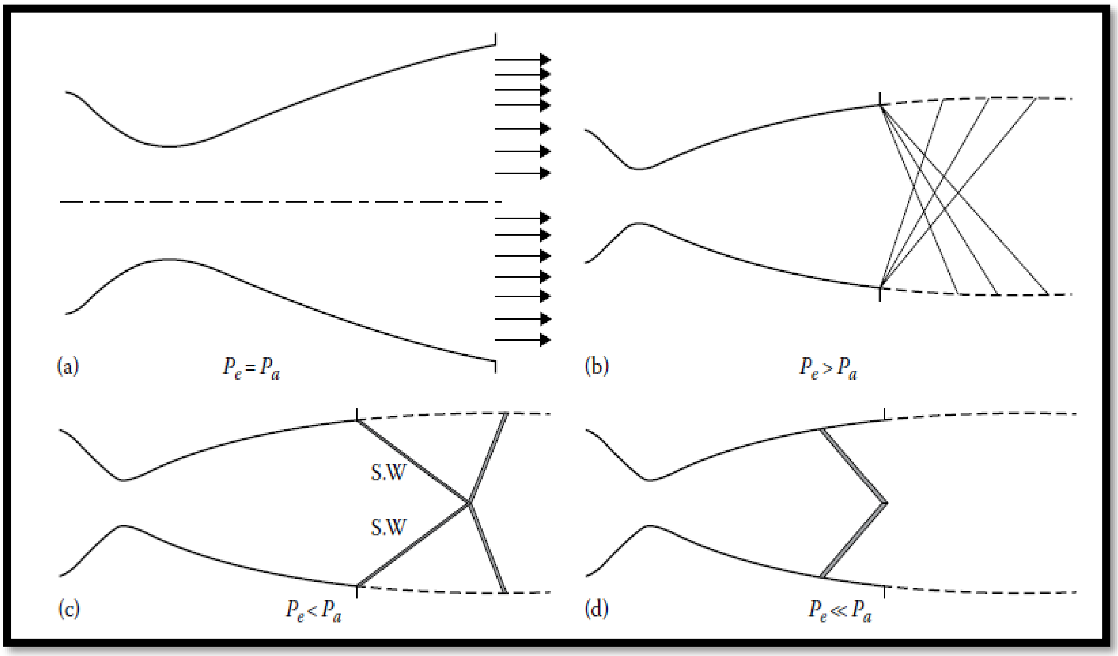
Note: Convergent nozzles are used in almost all the present subsonic transports. Moreover, in most cases these convergent nozzles are also choked, and incomplete expansion of the flowing gases to the ambient pressure is encountered. At nozzle outlet, the gases exit at sonic speed while the pressure is greater than the ambient pressure. Thus a pressure-thrust force is developed. On the contrary, a convergent-divergent nozzle (C-D) satisfies a full expansion to the ambient pressure.

Nozzle loss for unchoked and choked flows



- If $\dfrac{p_{04}}{p_c} > \dfrac{p_{04}}{p_5}$ Then nozzle is unchoked
- If $\dfrac{p_{04}}{p_c} < \dfrac{p_{04}}{p_5}$ Then nozzle is choked

Where p_c is critical pressure



Behaviour of convergent-divergent nozzle. (a) Design condition. (b) Exit pressure exceeds ambient pressure. (c) Ambient pressure exceeds exit pressure. (d) Ambient pressure greatly exceeds exit pressure

$$\eta_{iN} = \frac{T_{04} - T_5}{T_{04} - T_5'} \quad \text{.......... (1)}$$

From equation (1) we can find pressure ratio

$$\eta_{iN} = \frac{T_{04} - T_5}{T_{04} - T_5'} = \frac{T_{04} - T_5}{T_{04}\left(1 - \dfrac{T_5'}{T_{04}}\right)}$$

$$T_{04} - T_5 = \eta_{iN} T_{04}\left(1 - \frac{T_5'}{T_{04}}\right)$$

$$T_{04} - T_5 = \eta_{iN} T_{04}\left(1 - \frac{1}{\dfrac{T_{04}}{T_5'}}\right)$$

$$T_{04} - T_5 = \eta_{iN} T_{04} \left[1 - \left(\frac{1}{\frac{p_{04}}{p_5}} \right)^{\frac{\gamma-1}{\gamma}} \right]$$

$$1 - \left(\frac{1}{\frac{p_{04}}{p_5}} \right)^{\frac{\gamma-1}{\gamma}} = \frac{T_{04} - T_5}{\eta_{iN} T_{04}}$$

$$\left(\frac{1}{\frac{p_{04}}{p_5}} \right)^{\frac{\gamma-1}{\gamma}} = 1 - \frac{T_{04} - T_5}{\eta_{iN} T_{04}}$$

$$\frac{p_5}{p_{04}} = \left(1 - \frac{T_{04} - T_5}{\eta_{iN} T_{04}} \right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{p_{04}}{p_5} = \left[\frac{1}{\left(1 - \frac{T_{04} - T_5}{\eta_{iN} T_{04}} \right)} \right]^{\frac{\gamma}{\gamma-1}}$$

where $\frac{p_{04}}{p_5}$ is nozzle pressure ratio and $p_5 = p_a$

$$\frac{p_{04}}{p_5} = \left[\frac{1}{1-\frac{1}{\eta_{iN}}\left(1-\frac{T_5}{T_{04}}\right)}\right]^{\frac{\gamma}{\gamma-1}} \quad \text{....................(1)}$$

Now calculation of stagnation pressure to static pressure in terms of mach number

$$\frac{T_{04}}{T_5} = \frac{T_{05}}{T_5} = \left(1+\frac{\gamma-1}{2}M_5^2\right)$$

$$\frac{T_5}{T_{04}} = \frac{1}{\left(1+\frac{\gamma-1}{2}M_5^2\right)}$$

$$\frac{p_{04}}{p_5} = \left[\frac{1}{1-\frac{1}{\eta_{iN}}\left(1-\frac{1}{\left(1+\frac{\gamma-1}{2}M_5^2\right)}\right)}\right]^{\frac{\gamma}{\gamma-1}}$$

For If the nozzle is choked, then the exit Mach number is unity (M_5 =1) and all conditions are denoted by subscript (*c*). The temperature ratio is then $T_5 = T_c$

$$\frac{T_c}{T_{04}} = \frac{1}{\left(1+\frac{\gamma-1}{2}\right)} = \frac{2}{\gamma+1}$$

Or

$$\frac{T_c}{T_{04}} = \frac{2}{\gamma+1}$$ putting this value in above equation (1)

$$\frac{p_{04}}{p_c} = \left[\frac{1}{1-\frac{1}{\eta_{iN}}\left(1-\frac{2}{\gamma+1}\right)}\right]^{\frac{\gamma}{\gamma-1}}$$

$$\frac{p_{04}}{p_c} = \left[\frac{1}{1-\frac{1}{\eta_{iN}}\left(\frac{\gamma-1}{\gamma+1}\right)}\right]^{\frac{\gamma}{\gamma-1}} \quad \text{..........(2)}$$

Case 1: If nozzle is unchoked take $\frac{p_{04}}{p_5}$ from equation (1) for further calculation and we have optimum expansion, for unchoked nozzle the exit/jet velocity can be calculated by:

$$V_e = V_j = C_j = \sqrt{2c_p\left(T_{04} - T_5\right)}$$

Case 2 If the nozzle is choked $\frac{p_{04}}{p_c}$ from equation (2) for further calculation, for choked nozzle the exit/jet velocity can be calculated by:

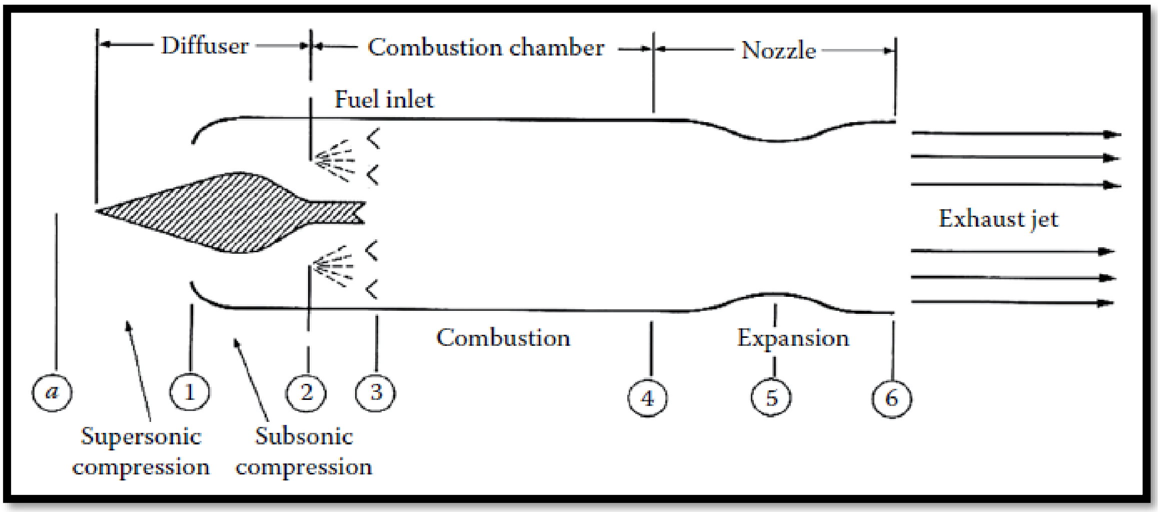
$$V_e = V_j = C_j = \sqrt{2c_p\left(T_{04} - T_c\right)} = \sqrt{\gamma R T_c}$$

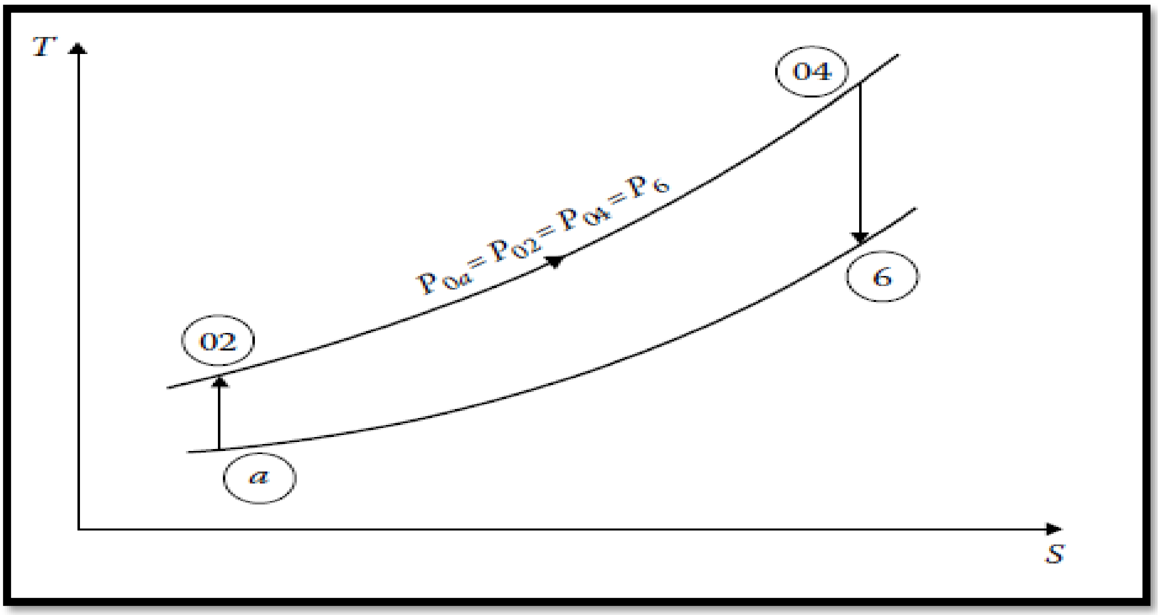
The engine performance is estimated by defining the adiabatic efficiencies as follows:

- Diffuser efficiency: $0.7 < \eta_d < 0.9$ (depending strongly on flight Mach number)
- Compressor efficiency: $0.85 < \eta_c < 0.90$
- Burner efficiency: $0.97 < \eta_b \leq 0.99$ (same value applies for the efficiency of afterburner)
- Turbine efficiency: $0.90 < \eta_t < 0.95$
- Nozzle efficiency: $0.95 < \eta_n < 0.98$

Cycle analysis

1. Ideal Ramjet





- For isentropic (reversible adiabatic) flow inside the engine, no pressure drop will been countered in the three modules of the engine; thus

$$p_{0a} = p_{02} = p_{04} = p_{06}$$

- Moreover, since neither work nor heat addition nor rejection takes place in the intake and nozzle, then from the first law of thermodynamics, equal total enthalpy (and thus, total temperature) is presumed. Thus

$$T_{0a} = T_{02}, \quad T_{04} = T_{06}$$

- Full expansion of the hot gases within the nozzle is assumed; thus

$$p_a = p_6 = p_e$$

$$\frac{T_{0a}}{T_a} = 1 + \frac{\gamma_a - 1}{2} M^2 = \frac{T_{02}}{T_a}$$

$$\frac{T_{0e}}{T_e} = \frac{T_{06}}{T_6} = 1 + \frac{\gamma_6 - 1}{2} M_e^2 = \frac{T_{04}}{T_e}$$

γ_a, γ_6 are the specific heat ratios for air and exhaust gases, respectively ? If we ignore the variations in fluid properties, (R, γ), then from

$$\frac{P_{0a}}{P_a} = \left(1 + \frac{\gamma_a - 1}{2} M^2\right)^{\gamma_a / \gamma_a - 1}$$

$$\frac{P_{06}}{P_e} = \left(1 + \frac{\gamma_6 - 1}{2} M_e^2\right)^{\gamma_6 / \gamma_6 - 1}$$

The relations between total and static conditions (temperature and pressure) at the inlet and outlet of the engine

$$\frac{P_{06}}{P_e} = \frac{P_{0a}}{P_a}$$

Or

$$M_a = M_6$$

$$V_e = \frac{a_e}{a} V_\infty = \sqrt{\frac{\gamma_6 R T_e}{\gamma_a R T_a}}\, V_\infty$$

The flight and exhaust Mach numbers are equal, but the flight and exhaust speeds are not. This difference generates the thrust force. Again, assuming constant (γ, R) within the engine,

$$V_e = \sqrt{\frac{T_e}{T_a}}\, V_\infty$$

$$V_e = V_\infty \sqrt{\frac{T_{04}}{T_{0a}}}$$

Fuel-to-air ratio (f): To get the fuel-to-air ratio, apply the energy equation to the combustion process

$$\dot{m}_a h_{02} + \dot{m}_f Q_R = \left(\dot{m}_a + \dot{m}_f\right) h_{04}$$

$$Q_R = \text{Heating value of the fuel}$$

$$h_{02} + fQ_R = \left(1 + f\right) h_{04}$$

$$C_{p_2} T_{02} + fQ_R = \left(1 + f\right) C_{p_4} T_{04}$$

Thrust force (T): Since full expansion in the nozzle is assumed, that is, *Pe* = *Pa*, then the thrust force is expressed as

$$T_{02} = T_{0a}$$

$$\therefore f = \frac{\left(C_{p_4} T_{04} / C_{p_2} T_{0a}\right) - 1}{\left(Q_R / C_{p_2} T_{0a}\right) - \left(C_{p_4} T_{04} / C_{p_2} T_{0a}\right)}$$

$$T = \dot{m}_a \left(1 + f\right) V_e - \dot{m}_a V_\infty$$

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Turbojet cycle

Inlet calculation

$$\eta_r = \frac{p_{01} - p_a}{p_{0a} - p_a}$$

$$\eta_i = \frac{T'_{01} - T_a}{T_{01} - T_a}$$

Compressor calculation

$$\eta_c = \frac{T'_{02} - T_{01}}{T_{02} - T_{01}}$$

Combustion chamber

$$f = \frac{c_p T_{03} - c_p T_{02}}{\eta_{cc} \times HV - c_p T_{03}}$$

$$P_{03} = P_{02} - \Delta P_{cc}$$

Turbine calculation

$$W_C = \eta_M W_T$$

$$\eta_T = \frac{T_{03} - T_{04}}{T_{03} - T'_{04}}$$

Afterburner operative/Inoperative

Nozzle calculation

$$\frac{p_{04}}{p_c} = \left[\frac{1}{1 - \frac{1}{\eta_{iN}}\left(\frac{\gamma - 1}{\gamma + 1}\right)}\right]^{\frac{\gamma}{\gamma - 1}}$$

Nozzle is choked/Unchoked

Choked:

$$V_e = V_j = C_j = \sqrt{2c_p\left(T_{04} - T_c\right)} = \sqrt{\gamma R T_c}$$

Unchoked:

$$V_e = V_j = C_j = \sqrt{2c_p\left(T_{04} - T_5\right)}$$

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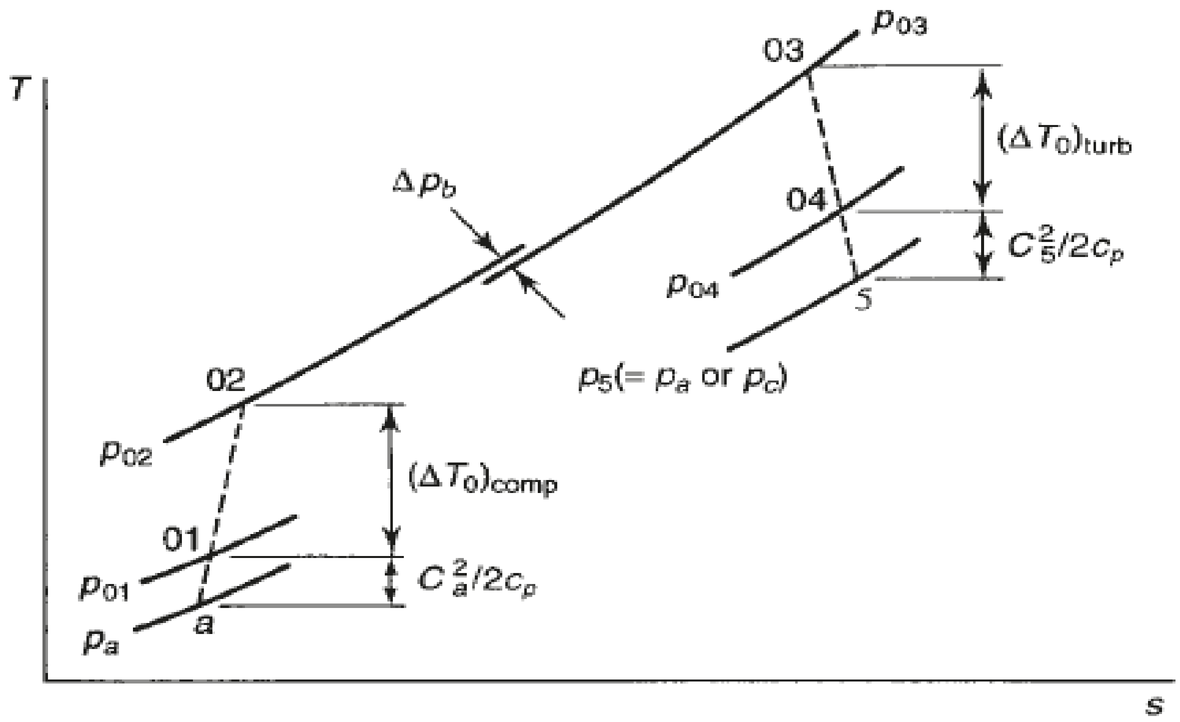
Problem: 1

PROBLEM:

Determination of the specific thrust and *SFC* for a simple turbojet engine, having the following component performance at the design point at which the cruise speed and altitude are *M* 0.8 and 10 000 m:

Compressor pressure ratio	8.0
Turbine inlet temperature	1200 K
Isentropic efficiency:	
of compressor, η_c	0.87
of turbine, η_t	0.90
of intake, η_i	0.93
of propelling nozzle, η_j	0.95
Mechanical transmission efficiency η_m	0.99
Combustion efficiency η_b	0.98
Combustion pressure loss Δp_b	4% comp. deliv. press.

Solution


T
s
p_{03}
03
$(\Delta T_0)_{turb}$
Δp_b
04
$C_5^2/2c_p$
p_{04}
5
$p_5(= p_a \text{ or } p_c)$
02
p_{02}
$(\Delta T_0)_{comp}$
01
p_{01}
$C_a^2/2c_p$
a
p_a

Inlet calculation

From the ISA table, at 10 000 m

$$p_a = 0.2650 \text{ bar}, \quad T_a = 223.3 \text{ K} \quad \text{and} \quad a = 299.5 \text{ m/s}$$

The stagnation conditions after the intake may be obtained as follows:

$$\frac{C_a^2}{2c_p} = \frac{(0.8 \times 299.5)^2}{2 \times 1.005 \times 1000} = 28.6 \text{ K}$$

$$T_{01} = T_a + \frac{C_a^2}{2c_p} = 223.3 + 28.6 = 251.9 \text{ K}$$

$$\frac{p_{01}}{p_a} = \left[1 + \eta_i \frac{C_a^2}{2c_p T_a}\right]^{\gamma/(\gamma-1)} = \left[1 + \frac{0.93 \times 28.6}{223.3}\right]^{3.5} = 1.482$$

$$p_{01} = 0.2650 \times 1.482 = 0.393 \; bar$$

Compressor calculation

At the outlet from the compressor,

$$p_{02} = \left(\frac{p_{02}}{p_{01}}\right) p_{01} = 8.0 \times 0.393 = 3.144 \text{ bar}$$

$$T_{02} - T_{01} = \frac{T_{01}}{\eta_c}\left[\left(\frac{p_{02}}{p_{01}}\right)^{(\gamma-1)/\gamma} - 1\right] = \frac{251.9}{0.87}\left[8.0^{1/3.5} - 1\right] = 234.9 \text{ K}$$

$$T_{02} = 251.9 + 234.9 = 486.8 \text{ K}$$

Combustion chamber calculation

$$p_{03} = p_{02}\left(1 - \frac{\Delta p_b}{p_{02}}\right) = 3.144(1 - 0.04) = 3.018 \text{ bar}$$

$$f = \frac{c_{p,gas}T_{03} - c_{p,air}T_{02}}{\eta_{cc} \times HV - c_{p,gas}T_{03}}$$

=0.0198

Turbine calculation

$W_t = W_c/\eta_m$ and hence

$$T_{03} - T_{04} = \frac{c_{pa}(T_{02} - T_{01})}{c_{pg}\eta_m} = \frac{1.005 \times 234.9}{1.148 \times 0.99} = 207.7 \text{ K}$$

$$T_{04} = 1200 - 207.7 = 992.3 \text{ K}$$

$$T'_{04} = T_{03} - \frac{1}{\eta_t}(T_{03} - T_{04}) = 1200 - \frac{207.7}{0.90} = 969.2 \text{ K}$$

$$p_{04} = p_{03}\left(\frac{T'_{04}}{T_{03}}\right)^{\gamma/(\gamma-1)} = 3.018\left(\frac{969.2}{1200}\right)^{4} = 1.284 \text{ bar}$$

Nozzle calculation

The nozzle pressure ratio is therefore

$$\frac{p_{04}}{p_a} = \frac{1.284}{0.265} = 4.845$$

The critical pressure ratio, from equation (3.14), is

$$\frac{p_{04}}{p_c} = \frac{1}{\left[1 - \frac{1}{\eta_j}\left(\frac{\gamma - 1}{\gamma + 1}\right)\right]^{\gamma/(\gamma-1)}} = \frac{1}{\left[1 - \frac{1}{0.95}\left(\frac{0.333}{2.333}\right)\right]^{4}} = 1.914$$

Since $p_{04}/p_a > p_{04}/p_c$ the nozzle is choking:†

If the nozzle is choked $\frac{p_{04}}{p_c}$ we will take for further calculation

$$T_5 = T_c = \left(\frac{2}{\gamma + 1}\right)T_{04} = \frac{2 \times 992.3}{2.333} = 850.7 \text{ K}$$

$$p_5 = p_c = p_{04}\left(\frac{1}{p_{04}/p_c}\right) = \frac{1.284}{1.914} = 0.671 \text{ bar}$$

$$\rho_5 = \frac{p_c}{RT_c} = \frac{100 \times 0.671}{0.287 \times 850.7} = 0.275 \text{ kg/m}^3$$

$$C_5 = (\gamma RT_c)^{\frac{1}{2}} = (1.333 \times 0.287 \times 850.7 \times 1000)^{\frac{1}{2}} = 570.5 \text{ m/s}$$

$$\frac{A_5}{m} = \frac{1}{\rho_5 C_5} = \frac{1}{0.275 \times 570.5} = 0.006\,374 \text{ m}^2 \text{ s/kg}$$

The specific thrust is

$$F_s = (C_5 - C_a) + \frac{A_5}{m}(p_c - p_a)$$

$$= (570.5 - 239.6) + 0.006\,374(0.671 - 0.265)10^5$$

$$= 330.9 + 258.8 = 589.7 \text{ N s/kg}$$

We find that the theoretical fuel/air ratio required is 0.0194. Thus the actual fuel/air ratio is

$$f = \frac{0.0194}{0.98} = 0.0198$$

The specific fuel consumption is therefore

$$SFC = \frac{f}{F_s} = \frac{0.0198 \times 3600}{589.7} = 0.121 \text{ kg/h N}$$

The End