Module 2

PROPELLER THRORIES & JET PROPULSION

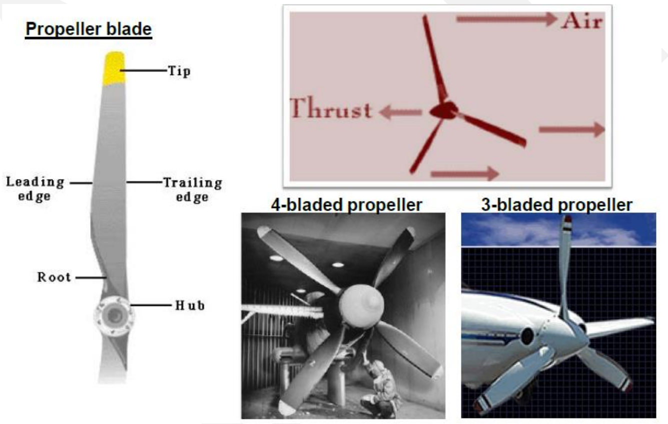
Syllabus:

Propeller Theories: Types of propeller, Propeller thrust: momentum theory, Blade element theories, propeller blade design, Propeller selection.

Jet Propulsion: Illustration of working of gas turbine engine – The thrust equation – Factors affecting thrust – Effect of pressure, velocity and temperature changes of air entering compressor – Methods of thrust augmentation – Characteristics of turboprop, turbofan and turbojet – Performance characteristics.

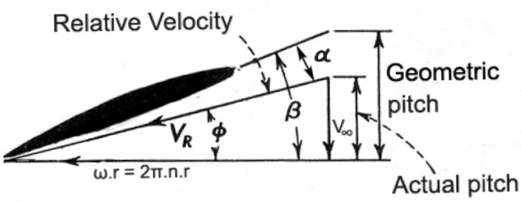
2.1 Propeller Fundamentals

A propeller is an interface between an engine and an aircraft. It creates thrust for flying an aircraft.



The performance of a propeller is dependent on the local aerodynamics on the blade elements, integrated over the blade length. For efficient operation each blade element should be at an Angle of Attack, α, optimized to a value near the maximum elemental lift to drag (L/D) ratio.

Propeller blade --- sectional geometry and local flow details



$$\textbf{Geometric Pitch},\quad p = 2\pi r.\tan\beta$$

The AoA($\boldsymbol{\alpha}$) is a function of the blade element geometric pitch (blade setting) angle, β and effective pitch angle (flow angle) φ.

The rotational speed, ($\mathbf{U = \omega.r}$) of each blade element is different, but as the forward speed, **Va**, is same, the pitch setting needs to be varied from hub to tip so as to maintain the best AoA for each blade element.

The blade section in fig shows that the section makes an angle $\boldsymbol{\beta}$ with the rotational direction, is known as pitch angle (blade setting angle), defined with respect to either (i) zero lift line, or (ii) chord line, or (ii) the flat undersurface of the blade section.

The "pitch" refers to the forward movement of the propeller for one revolution of the blade (section).

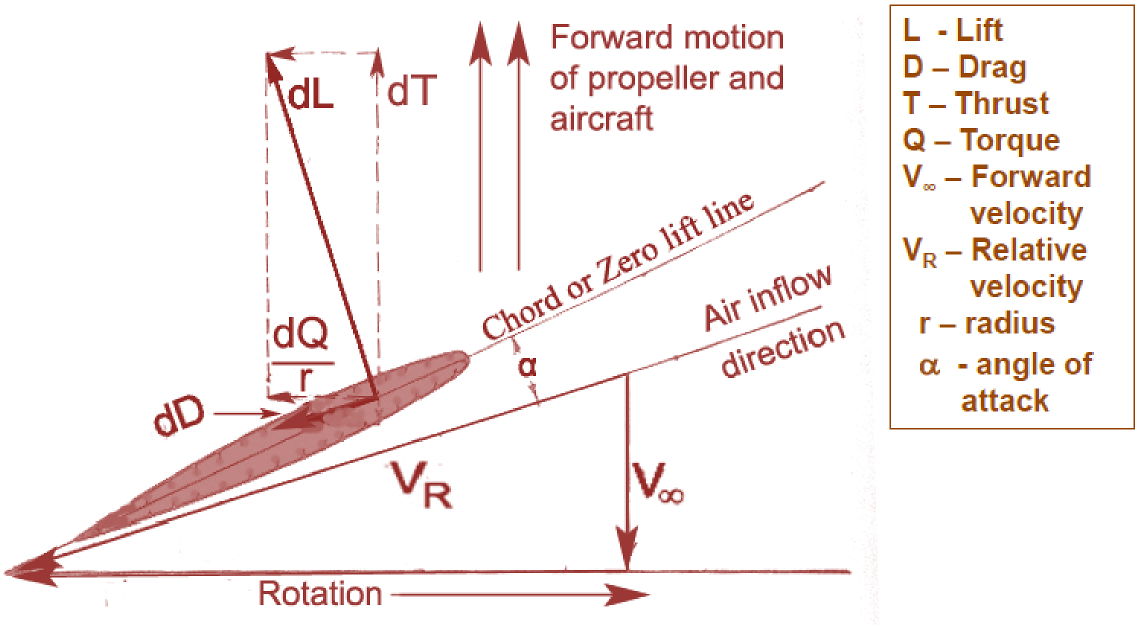
Theoretically, each section of the propeller may have its own pitch value. However, since all the blades sections of each blade of a propeller are assembled into one solid body, all the sections must move forward by the same amount per revolution of the propeller.

Thus, a difference between the geometric pitch, **p**, for a blade section, and the actual pitch, for the same section (when the body of the propeller moves forward) arises.

The lift and the drag of a blade element are perpendicular and parallel respectively to the relative wind direction coming on the blade element.

These may be projected to the forces: Tangential force (for **Torque**) and axial force (**Thrust**), in the planes normal and parallel respectively to the axis of rotation of the propeller.

Blade section geometry , local flow details and aerodynamic forces created



2.2 Propeller Thrust

All propulsors moving in air produce propulsive force, called thrust, by effecting a net change in momentum to a propulsive fluid in the direction of motion.

Propellers create thrust by introducing a small change in velocity to a relatively large mass of air, compared to those of various jet propulsion devices.

$$T_{net} = \dot{m}.(V_e - V_a)$$

2.3 Types of Propellers

There are three Pitch setting arrangements:

A fixed pitch propeller, in which the geometric pitch cannot be varied, must be matched to the various operating conditions of the engine and of the aircraft.

A variable pitch propeller, either variable manually, or through hydro-mechanical control system, usually offer at least two or more blade settings, one fine and the other coarse, to maximize the propeller efficiency, during take-off and during cruise respectively.

A constant speed propeller, automatically changes propeller pitch according to a built in control law (floating pitch) so as to maintain proper torque such that the speed of the propeller shaft is maintained constant with the help of a governor and a electro-hydro-mechanical control system. Most modern propellers are constant speed propellers.

2.4 Propeller performance parameters

Advance ratio: J the *advance ratio* is the ratio of the free stream fluid speed to the propeller cyclo-rotor tip speed. When a propeller-driven vehicle is moving at high speed relative to the fluid, or the propeller is rotating slowly, the advance ratio of its propeller(s) is a high number; and when it is moving at low speed, or the propeller is rotating at high speed, the advance ratio.The advance ratio J is a non-dimensional term given by

$$J = V_\infty / (n.D)$$

V∞ is the free stream fluid velocity, typically the true airspeed of the aircraft or the water speed of the vessel, **n** is the rotor rotational speed, **r** is the rotor radius.

The propeller efficiency: The propeller efficiency is given by the usual output power to input power ratio,

$$\eta_P = (T.V_\infty)/P = (T.V_\infty)/(2.\pi.n.Q)$$

Propeller tip speed: The propeller tip speed is given by

$$V_{tip,\ helical} = \sqrt{(\{\pi\ nD\}^2 + V_\infty{}^2)}$$

When propeller starts rotating at high speed. The propellers would experience tip speeds, which are supersonic and then you would have shocks. Those shocks would then reduce the efficiency of the propellers, the propellers could become of lower efficiency

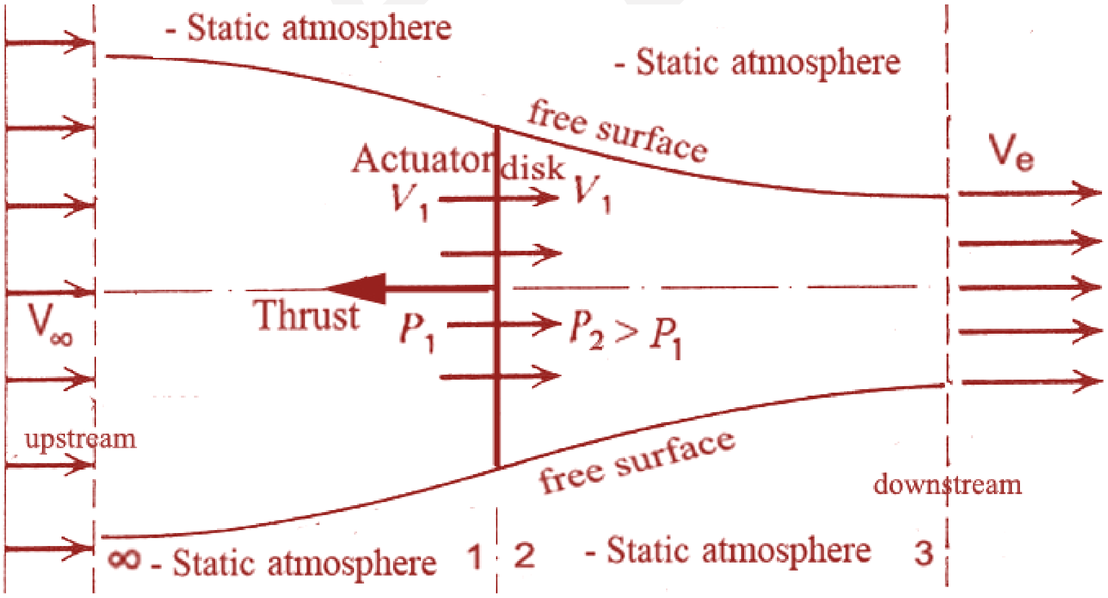
- Metal propellers are limited to M_{tip} of 0.85,
- Wooden ones are limited to M_{tip} of 0.75

2.5 Propeller Theories

There are two classical approaches to propeller theory

1) Momentum Theory
2) Blade Element Theory

1. Momentum Theory for Propeller (or Actuator Disk Theory)



Assumptions for *conceptual modeling* of a propeller

1) The propeller is assumed to be replaced by an '*actuator disk*', a flow energizer.

2) The 'disk' is assumed to be of very small thickness and is a continuous and 100% porous body of no mass, with a projected frontal area 'A' (*swept area*) equal to the annulus of the rotating propeller blades.

3) There is no 'resistance' (i.e. drag) of the air passing through the 'actuator disk', (since there are no propeller blades).

4) The axial velocity, V1through the 'disk' is uniform over the 'actuation' area and is considered to be smooth across the disk i.e. no abrupt changes are 'experienced'.

5) The received energy manifests itself in the working medium (i.e. air) finally in the form of differential pressure (**p2–p1**), a jump change across the actuator disk, uniformly distributed across the disk surface.

6) The fluid medium, air, is a assumed to be a perfect incompressible fluid. Flow is assumed 'irrotational' in front of and behind the disk, but not through it. *and*

7) The static pressures far from the disk, i.e. far upstream and far downstream, are both assumed equal to the atmospheric pressure. The corresponding velocities are independent values, to be determined separately.

The mass flow through the disk from continuity, is

$$\acute{m} = \rho.A.V$$

The thrust produced by the disk from Newton's II and III laws resulting in reaction force, thrust.

$$T = \acute{m}.\partial V = \rho.A.V.(V_e - V_\infty)$$

From simple fluid statics, thrust is produced by the differential static pressure on either side of the disk, multiplied by its surface area (swept area)

$$T = A\ (P_2 - P_1)$$

Applying Bernoulli's equation on either side of the dis

$$P_\infty + \tfrac{1}{2}\rho\ V_\infty^{\ 2} = P_1 + \tfrac{1}{2}\rho\ V_1^2 \text{ –upstream}$$

$$P_2 + \tfrac{1}{2}\rho\ V_2^{\ 2} = P_\infty + \tfrac{1}{2}\rho\ V_e^2 \text{ –downstream}$$

Using, $V_1 = V_2$ = constant through the disk,

$$P_2 - P_1 = \tfrac{1}{2}.\rho.\ (V_e^2 - V_\infty^2)$$

From above equations,

$$V_1 = \tfrac{1}{2}.(V_e + V_\infty)$$

This simple analysis shows that the air flow velocity through the actuator disk is the mean of the velocities far upstream and far downstream of the propeller.

Thus thrust

$$T = \tfrac{1}{2}.\rho.(V_e^2 - V_\infty^2).A$$

The velocity at the disk comes out to be the free stream axial velocity,

$$\mathbf{V_1 = V_\infty + v \; ; \quad and\ V_e = V_\infty + 2.v}$$

Therefore, $$\mathbf{T = \rho\ A\ (V_\infty + v)\ 2.v}$$

From the equation the *induced velocity*, *v*, can be found as,

$$v = \frac{[-V_\infty + \sqrt{\{V_\infty^2 - (2T/\rho.A)\}}]}{2}$$

For a static thrust, where the propeller is not in forward motion (at take off), V_∞ = zero,

$$v = \sqrt{\frac{T}{2\ \rho.A}}$$

The ideal efficiency can be calculated by using classical definition of eficiency ,

$$\boldsymbol{\eta_p = P_{out} / P_{in}}$$

$$\boldsymbol{P_{out} = T\ V_\infty \textbf{ and } \quad P_{in} = T\ V_1}$$

Therefore, $$\boldsymbol{\eta_i = Pout / P_{in}}$$

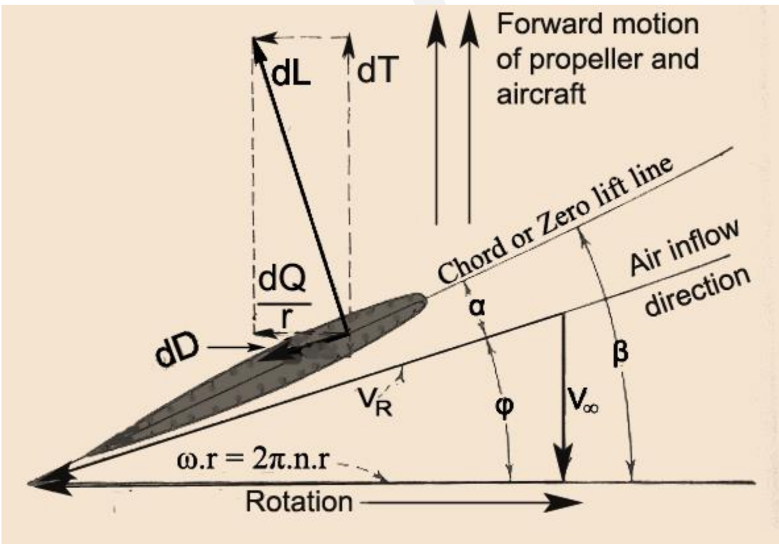
$$\boldsymbol{= T.V_\infty / T.V_1}$$

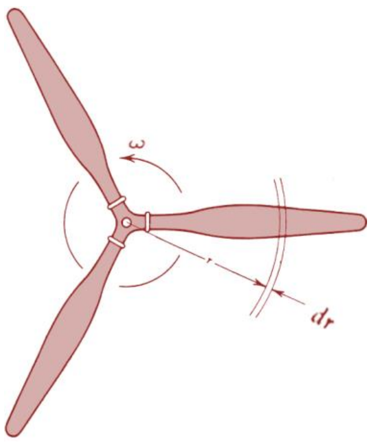
$$\boldsymbol{= V_\infty / [½(V_\infty + V_e)]}$$

Therefore, $$\boldsymbol{\eta_i = 1 / [1 + (v / V_\infty)]}$$

2. Blade element theory

The blade elements are assumed to be made up of airfoil shapes of known lift, Cl and drag, Cd characteristics. In practice a large number of different airfoils are used to make up one propeller blade. Each of these elements shall have its own lift, Cl and drag, Cd coefficient characteristics.





Using the blade elemental lift and drag characteristics the working capacity of the blade element may be found as:

Thrust produced,

$$dT = dL.cos\varphi - dD.sin\ \varphi$$
$$= \tfrac{1}{2}.\rho\ .V_R^2\ .c.dr.\ (Cl\ cos\varphi - Cd\ sin\ \varphi)$$

Torque to be supplied ,

$$dQ = (dL.sin\ \varphi + dD.cos\varphi).\ r$$
$$= \tfrac{1}{2}.\rho.V_R^2\ .c.dr.(Cl.sin\ \varphi + Cd.cos\varphi)$$

Substituting for Resultant inflow velocity Incident and aligned to the blade element,

$$\boldsymbol{V_R = V_\infty / Sin\ \varphi}$$

and for Incoming flow Dynamic head based on forward velocity of the element

$$\boldsymbol{q = \tfrac{1}{2}\ \rho\ V_\infty^2}$$

The elemental thrust is:

$$dT = \frac{q.c.dr}{\sin^2\phi}(C_l\ \cos\phi - C_d\ \sin\phi)$$

The elemental torque is:

$$dQ = \frac{q.c.r.dr}{\sin^2\phi}(C_l\ \sin\phi + C_d\ \cos\phi)$$

Propeller thrust and torque are now computed by integrating from the root to the tip of the blade and for number of blades, B

$$T = q.B.\int_0^R \frac{c.dr}{\sin^2\phi}(C_l\ \cos\phi - C_d\ \sin\phi)$$

$$Q = q.B\int_0^R \frac{c.r.dr}{\sin^2\phi}(C_l\ \sin\phi + C_d\ \cos\phi)$$

Thus, the net thrust and the torque are seen to be directly proportional to the number of blades, B and the chord, c.

This is not quite true in practice, as more is the number of blades and wider the blade chord -it shall result in more surface area, more flow blockage and higher consequent aerodynamic losses.

The optimum number of blades need to be found separately and not from the blade element theory.

The blade element efficiency,

$$\eta_{el} = \frac{\text{Thrust power produced}}{\text{Torque power supplied}}$$

In terms of elemental airfoil characteristics C_l and C_d , blade efficiency is :

$$\eta_{el} = \frac{v.dT}{2\pi n.dQ} = \frac{V}{2\pi nr} \cdot \frac{C_l \cos\phi - C_d \sin\phi}{C_l \sin\phi + C_d \cos\phi} = \frac{C_l \cos\phi - C_d \sin\phi}{C_l \sin\phi + C_d \cos\phi} . \tan\phi$$

2.6 Propeller Selection

Propeller manufacturers offer propellers covering a range of diameters Pitch Values, and solidities. The choice of these parameters can depend on considerations other than the aerodynamics. Aerodynamically propeller should have a higher efficiency and sufficient thrust for cruise and a high static thrust &r take-off

These two requirements are easier to satisfy with an automatically variable pitch (constant speed) propeller. A fixed pitch propeller is usually a compromise between these two operating regimes. Given the results of a series of propeller tests, one can utilize these data to select the best propeller diameter and blade angle combination. One approach that is sometimes used is based on a coefficient

C_s, the speed power coefficient, defined by,

$$Cs = (\rho.V^5/P.n^2)^{1/5}$$

Is often used for design / selection of propeller

If coeff of power, Cp as a function of J, is known, Cs can be obtained from

$$Cs = J/Cp^{1/5}$$

The *usefulness of C_s is in the process of defining it -- diameter was eliminated.* Thus the propeller design or selection related flow parameters may be estimated even before the propeller size is fixed.

Recent propeller designs are using highly swept blades, which are reminiscent of swept wings. But in propellers the use of sweep is for different purpose — it is more to control the loss of energy in secondary flow (radial and other non-axial flows). These propellers have started using transonic blade elements to allow the tips to go supersonic and thus permit use of designs for high subsonic aircraft propulsion. Both the aerodynamic design and the mechanical design of these Propellers are posing real challenges to the designers. These propellers are directly coup1ed to the low pressure turbines, with a possibility of direct drive. Additional problems that come in the way are the noise problems

2.7 Illustration of working of gas turbine engine (Turbojet)

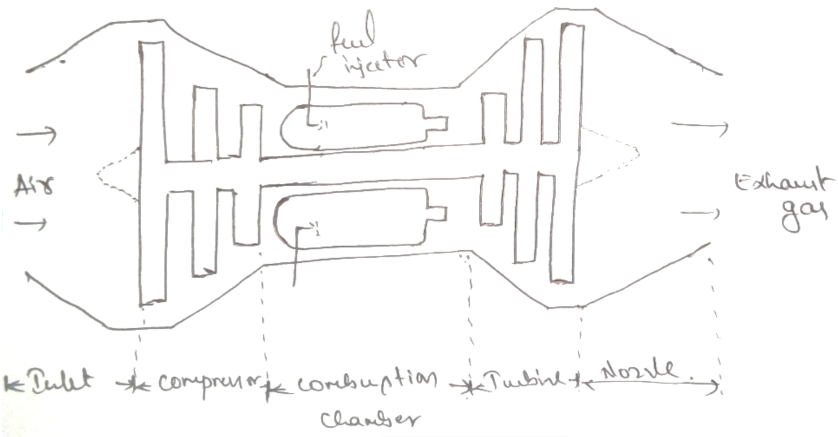


Fig: The turbojet engine

Working principle:

- The turbojet engine is a reaction engine. In a reaction engine, expanding gases push hard against the front of the engine.
- Turbojet engine derives its thrust by accelerating a mass of air through the core engine.
- The air taken in from an opening in the front of the engine is compressed to about 3-12 times its original pressure in a centrifugal or axial compressor.
- Fuel is added to the air and burned in a combustion chamber to raise the temperature of the mixer to about 1100^{0}C. The resulting hot air is passed through a turbine, which drives the compressor.
- If the turbine and compressor are efficient, the pressure at the turbine discharge will be nearly twice the atmospheric pressure.
- This excess pressure is sent to the nozzle to produce a high velocity stream of gas which produces the thrust. Thus all the propulsive force produced by a jet engine is derived from exhaust gases.
- An afterburner (or a reheat) is an additional component added to some jet engines. Primarily those on military supersonic aircrafts.
- Its purpose is to provide a temporary increase in thrust at the time of supersonic flight as well as takeoff.
- On military aircraft, the extra thrust is also useful for combat situations. This is achieved by injecting additional fuel into the jet pipe downstream of (after) the turbine.

Thermodynamic cycle analysis of Turbojet:

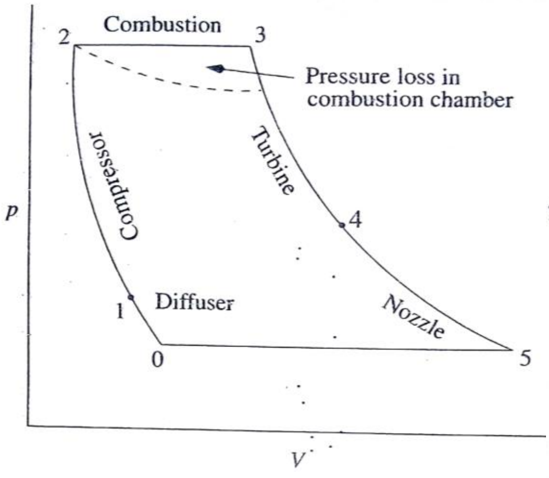


Fig. 7.10 p-V diagram of a turbojet engine

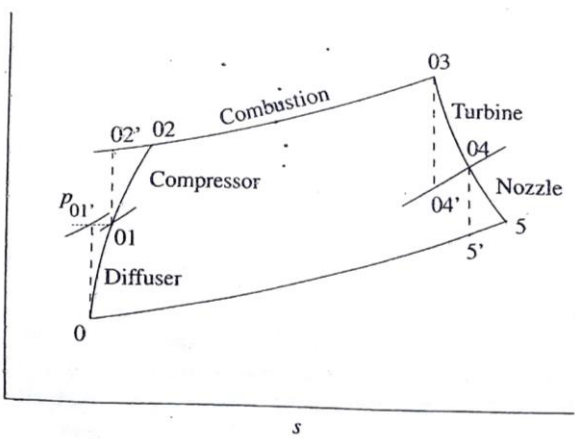


Fig. 7.11 T-s diagram of a turbojet engine

Figures 7.10 and 7.11 show the basic thermodynamic cycle of a turbojet engine of p-V and T-s diagrams. This is Joule or Brayton cycle. In the analysis of turbojet cycle following assumptions are made:

(i) There is no loss of pressure in the combustion chamber.

(ii) The specific heat is constant.

(iii) Power developed by the turbine is just sufficient to drive the compressor.

Diffuser Efficiency: It's defined as the ratio of an enthalpy change that occurs between the entrance to exit stagnation pressure to the kinetic energy.

At the inlet to the diffuser air enters with a velocity equal to the forward speed of the aircraft. In diffuser air velocity is decreased and pressure is increased. In the ideal case the pressure will rise such that the velocity at the exit of the diffuser is zero. However, in actual practice the air will have a velocity of about 60–90 m/s at diffuser exit. If η_d is the efficiency of the diffuser then the total pressure at the end of diffusion process is given by

$$\frac{p_{01}}{p_0} = \left(1 + \eta_d \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}}$$

Compressor Efficiency: It is defined as the ratio of ideal work of compression for given pressure ratio to the actual work of compression for given pressure ratio.

From the diffuser air goes into a compressor. If η_C is the compressor efficiency and $\frac{p_{02}}{p_{01}}$ the pressure ratio, we get

$$\eta_C = \frac{h_{02'} - h_{01}}{h_{02} - h_{01}}$$

or

$$h_{02} - h_{01} = \frac{1}{\eta_C}(h_{02'} - h_{01}) = \frac{C_p}{\eta_C}(T_{02'} - T_{01})$$

$$= \frac{C_p T_{01}}{\eta_C}\left(\frac{T_{02'}}{T_{01}} - 1\right) = \frac{C_p T_{01}}{\eta_C}\left[\left(\frac{p_{02}}{p_{01}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]$$

Burner or combustion efficiency: It is a measure of how effectively the heat content of a fuel is transferred into useable heat.

$$\eta_b = \frac{(\dot m + \dot m_f)\, C_{pg}\, T_{03} - \dot m\, C_{pa}\, T_{04}}{\dot m_f\, Q_f}$$

Turbine Efficiency: It is defined as the ratio of actual work of expansion for given pressure ratio to the ideal work of expansion for given pressure ratio.

We assume that there is no pressure loss in the combustion chamber so that the pressure remains constant and full pressure is available for expansion in the turbine and nozzle. Since we have assumed that turbine and compressor work are same.

$$h_{02} - h_{01} = h_{03} - h_{04}$$

If η_T is the turbine efficiency and $\frac{p_{04}}{p_{03}}$ is the turbine pressure ratio,

$$h_{03} - h_{04} = \eta_T C_p T_{03}\left[1 - \left(\frac{p_{04}}{p_{03}}\right)^{\frac{\gamma-1}{\gamma}}\right]$$

By equating the turbine and compressor work, we get

$$\frac{C_p T_{01}}{\eta_C}\left[\left(\frac{p_{02}}{p_{01}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] = \eta_T C_p T_{03}\left[1 - \left(\frac{p_{04}}{p_{03}}\right)^{\frac{\gamma-1}{\gamma}}\right] \qquad (7.3)$$

Nozzle Efficiency: it is defined as the ratio of the actual kinetic energy at exit to the kinetic energy at the exit when the process is isentropic for the same inlet and exit pressure.

From Eq. 7.3, knowing the ram compression and the compressor pressure ratio, the required turbine pressure ratio to produce a power equal to that absorbed by the compressor can be obtained and from the turbine pressure ratio the nozzle pressure can be obtained.

$$\eta_2 = \frac{\text{Actual enthalpy drop}}{\text{ideal enthalpy drop}}$$

If η_{noz} is the nozzle efficiency then,

$$\eta_{noz} = \frac{h_{04} - h_5}{h_{04} - h_{5'}} \qquad (7.4)$$

It is assumed that there is no loss in passing the gas from turbine exhaust to the nozzle. It should be noted that in Eq. 7.4, h_5 is used instead of total enthalpy h_{05} because the exhaust nozzle efficiency is an indication of the percentage of total energy converted into velocity energy. Therefore,

$$h_{04} - h_5 = \eta_{noz} C_p T_{04}\left[1 - \left(\frac{p_5}{p_{04}}\right)^{\frac{\gamma-1}{\gamma}}\right]$$

Performance of a Turbojet Engine:

With the help of the above analysis it is possible to estimate the performance of a turbojet engine taking into account the component efficiencies and other parameters. Figure 7.12 shows the thrust specific fuel consumption for various compressor pressure ratios at two different Mach numbers. It is evident that for a given Mach number there is only one compressor pressure ratio which gives best fuel economy for given values of component efficiencies and maximum allowable temperature. As the pressure ratio increases with a given maximum temperature fuel consumption decreases to a minimum. After that further increase in pressure ratio will not improve the fuel economy until maximum temperature is not raised. For a given pressure ratio higher maximum temperature will result in more thrust. Maximum thrust per unit mass of fuel is achieved at a lower pressure ratio than the one which produces minimum specific fuel consumption at the same turbine inlet temperature. In addition to the above parameters, three more variables – flight speed, altitude (inlet temperature and pressure), and fuel flow rate – greatly affect the performance of a jet engine.

The turbojet is almost a constant thrust engine. The specific fuel consumption based on thrust power reduces because with almost constant thrust, the thrust power increases as shown in Fig. 7.13. Therefore, the maximum speed is the most efficient operational point for the turbojet.

The static thrust of such engines is very low as compared to propeller engine aircraft for which cruise thrust is about 60 per cent of the takeoff thrust. The thrust at first decreases with increase in speed because the velocity c_j in the thrust equation increases. After a minimum value, the thrust starts increasing (see Figs. 7.14 and 7.15) due to increased ram compression at higher speeds.

As the altitude increases, the thrust decreases due to decrease in density, pressure and temperature of the air (see Fig. 7.15). However, the rate of

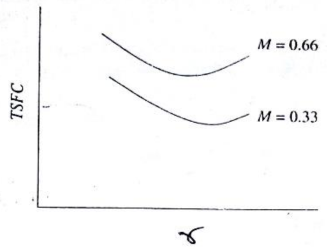


Fig. 7.12 Thrust specific fuel consumption vs compressor pressure ratio for a turbojet engine

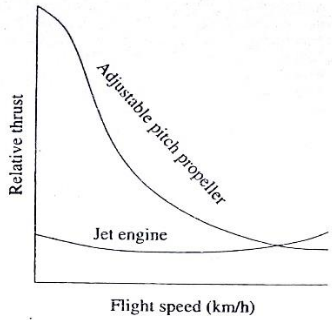


Fig. 7.14 Thrust vs speed for propeller driven and turbojet engine

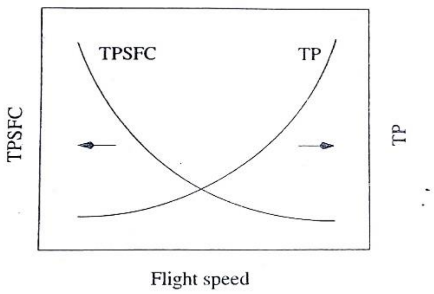


Fig. 7.13 Thrust specific fuel consumption and thrust power vs flight speed

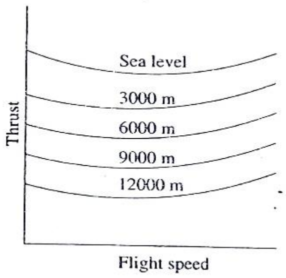


Fig. 7.15 Effect of altitude on thrust at maximum rpm

decrease of thrust is less than the rate of decrease of density with altitude because some loss due to reduced density is compensated by lesser drag. The thrust is maximum at sea level. Due to considerable reduction in drag (at an altitude of 8000 m the drag is reduced to less than 25 per cent of sea level drag), the turbojet is most efficient when flown at high altitudes and at relatively high speeds. The fuel consumption on fuel mass per km of travel increases with speed as power output also increases with speed.

The operational range of turbojet engine is about 800 to 1100 km/h and the specific fuel consumption is about 1.0 to 1.5 kg/thrust h at cruising speeds and are still greater at lower speeds. The altitude limit is about 10000 m.

2.8 The Thrust Equation

Let us consider the control volume of a schematic propulsive device shown in Fig. 7.16. A mass $\dot{m}_i$ of air enters the control volume with a velocity c_i and pressure p_i and the products of combustion of mass $\dot{m}_j$ leaves the control volume with a velocity c_j and pressure p_j. The flow is assumed to be steady and reversible outside the control volume, the pressure and velocity being constant over the entire control volume except that at the exhaust area A_j. Force F is the force necessary to balance the thrust produced due to change in momentum of the fluid as it passes through the control volume.

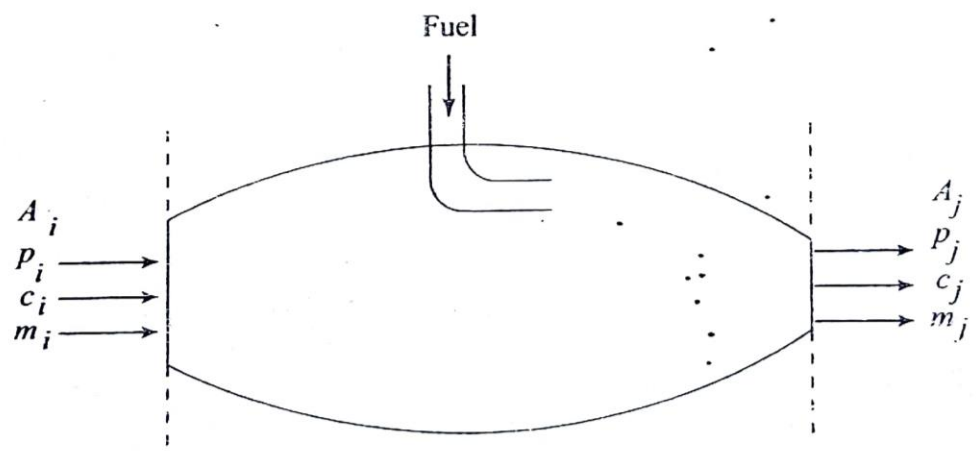


Fig. 7.16 General schematic diagram of a propulsive device

If p_a is the atmospheric pressure, then writing the momentum equation, we get

$$\dot{m}_j c_j - \dot{m}_i c_i = F + (p_i - p_a)A_i - (p_j - p_a)A_j \tag{7.6}$$

or thrust,

$$F = (\dot{m}_j c_j - \dot{m}_i c_i) + (p_j - p_a)A_j - (p_i - p_a)A_i$$

We have by mass balance,

$$\dot{m}_j = \dot{m}_i + \dot{m}_f \tag{7.7}$$

where $\dot{m}_j, \dot{m}_i$ and $\dot{m}_f$ are the mass flow rates of exhaust gases, air, and fuel respectively. If

$$f = \text{Fuel air ratio} = \frac{\dot{m}_f}{\dot{m}_i}$$

$$\dot{m}_j \doteq \dot{m}_i(1+f)$$

$$\text{Thrust} = \underbrace{\dot{m}_i[(1+f)c_j - c_i]}_{\text{Momentum thrust}} + \underbrace{(p_j - p_a)A_j - (p_i - p_a)A_i}_{\text{Pressure thrust}} \tag{7.8}$$

From Eq. 7.8 it is clear that the net thrust produced is made up two parts, viz., momentum thrust and the pressure thrust . If the exhaust velocity c_j from the control volume is subsonic, then $p_j \approx p_a$ and also $p_i \approx p_a$ so that the pressure thrust is quite small. Similar is the case for propeller engines. For supersonic exhaust velocity the pressure p_j may differ from p_a. However, the pressure thrust developed is so small as compared to the momentum thrust that it can safely be neglected for simple calculations and the net thrust is given by

$$\text{Thrust} = \dot{m}_i[(1+f)c_j - c_i] \tag{7.9}$$

The thrust, given by Eq. 7.9 can be increased by increasing the mass flow or increasing the velocity of the exhaust jet for a given c_i. Equation 7.9 has been derived for a simple control volume shown in Fig. 7.16, but is equally applicable to an aircraft flying at a forward speed c_i. In the latter case the velocities are considered relative to the aircraft. The air has a velocity c_i equal to aircraft forward speed, relative to the engine. Thus, we see that a large amount of thrust can be obtained either by propelling a large mass of air and increasing its velocity by a small amount, or by increasing the velocity of a small mass of air to a high value. In the case of aircraft power plants the fuel-air ratio f is very small (about 0.01 to 0.02) and hence the mass of fuel can be neglected safely without causing much error in the performance calculations.

The thrust is, then, given by

$$F = \dot{m}_i(c_j - c_i) \tag{7.10}$$

the effective speed ratio, α, is given by

$$\alpha \equiv \frac{c_i}{c_j} \tag{7.11}$$

and

$$F = \dot{m}_i c_i\left(\frac{1}{\alpha} - 1\right) \tag{7.12}$$

The product $\dot{m}_i c_j$ is called the gross thrust and $\dot{m}_i c_i$ is called the inlet drag or inlet momentum. Equation 7.12 is also applicable to turboprop engines.

The thrust developed by the engine overcomes the drag on the aircraft and in doing so it develops power, called the thrust power which is given by

$$P_T = Fc_i = \dot{m}_i(c_j - c_i)c_i \tag{7.13}$$

Note that turboprop engine is rated in kilowatts while turbojet and ramjet are rated on the basis of thrust developed.

2.9 Factors affecting thrust

As has been already mentioned that two parameters which affect the performance of the jet propulsion cycle are -- the forward speed of the aircraft and the altitude at which the aircraft flies. These two do not enter into the analysis of shaft power cycle. The effect of the two parameters are discussed in the following sections.

7.10.1 Effect of Forward Speed

The forward speed affects the inlet pressure and temperature of the compressor. The inlet duct to the compressor acts as a diffuser. The air which enters the diffuser at flight speed is slowed down to the speed acceptable to compressor, and at the same time raising its pressure and temperature. This increase of pressure and temperature due to aircraft speed is known as *ram effect*, or simply ram. It becomes more and more prominent as the flight speed increases. For a given aircraft speed and ram efficiency, the ram pressure ratio increases as the ambient temperature decreases at high altitudes.

specific impulse

$F/\dot{m}_a$

I_{sp}

The second effect of aircraft forward speed is in relation to propulsive efficiency. As flight velocity increases, the inlet drag also increases. If there were no ram effect, the net specific thrust, I_{sp}, would decrease. This is because the jet velocity remains the same. With ram, the increase of inlet temperature reduces the gross-thrust to some extent. However, the increase of inlet pressure more than compensates for this. Because of this, the cycle pressure ratio increases without shaft work being necessary. The overall effect of forward speed on inlet drag and ram is to reduce somewhat the net specific thrust.

7.10.2 Effect of Altitude

The effect of altitude on a turbojet is by virtue of reduction of ambient pressure and temperature. The temperature of atmosphere varies considerably and continuously with location and time, so that the standard atmosphere is used for calculating the performance at various altitudes. The most general data used are those for International Standard Atmosphere or ICON atmosphere (International Commission On Navigation). It corresponds approximately to average values found in the middle latitudes. It is normalized by using a linear decrease of temperature or lapse rate of 1.98°C per 300 m of altitude, starting with a ground level temperature of 15°C. With the temperature fixed, the pressure can be calculated according to the principles of hydrostatics using 1.03 bar as the ground level pressure. Table 7.1 provides values of International Standard Atmosphere.

2.10 Thrust augmentation

The poor take-off characteristic of the turbojet engine can be improved by augmenting the thrust. The thrust from a turbojet is given by

$$F = \dot{m}_a[(1+f)c_j - c_i]$$

in which the exhaust c_j, is the function of the maximum temperature in the cycle. Higher the maximum temperature higher is the value of c_j. Another method of increasing thrust is to increase the mass flow rate. Improved thrust results in shorter take-off distances, high climb rate and good manoeuvrability at high altitudes. The thrust augmentation can be effected by the following methods:

(i) Burning of additional fuel in the tail pipe between the turbine exhaust section and entrance section of the exhaust nozzle. This method of thrust augmentation increases the jet velocity and is known as *afterburning*. The device used is called the afterburner.

(ii) Injecting refrigerants, water or water-alcohol mixture at some point between inlet and exit sections of the air compressor. This method of thrust augmentation increases the mass flow rate and decreases the work of compression.

(iii) Bleeding off air in excess of that required for stoichiometric combustion in the main combustion chamber – at the entrance section of the combustion chamber, and burning it with the stoichiometric fuel-air ratio in a separate one. The combustion products from the latter combustor are expanded in a separate auxiliary exhaust nozzle. The bled air is replaced by water which is injected in the main combustors. This method of thrust augmentation is known as the bleed burn cycle.

7.11.1 The Afterburner

This method of thrust augmentation is being widely applied for obtaining high thrust for short duration. It is known that turbine blade material considerations limit the combustion chamber temperature rise. This in turn, limits the basic engine fuel- air ratio to values of about 0.017. As a result, the products of combustion leaving the turbine contain enough unutilized oxygen to support further combustion. Thus if a suitable burner is installed between the turbine and exhaust nozzle, a considerable amount of fuel can be burned in this section to produce temperatures entering the nozzle as high as 2000°C. This increases the gas velocity, and hence provides a thrust increase. A boost of about 30 per cent can be obtained in this manner. However, the fuel consumption increases rapidly. For about 20 per cent thrust increase by use of reheat the overall fuel consumption may be increased by more than 100 per cent and this additional mass of fuel has to be carried by the turbojet. Therefore, it is used only for take-off or for high climbing rates and for a very short duration. Because of the temperature rise in the afterburner, there is a large increase in specific volume of gases, and to keep the pressure drop as small as possible, the tail pipe of the afterburning area is more than that of the normal engine. Furthermore, the afterburning engine must be equipped with a variable exit area exhaust nozzle so that by varying its area with the afterburner operating, the normal conditions at inlet to the afterburner will be unaffected.

Another way of seeing the need for an increase of nozzle exit area is to examine the compressor performance chart. Figure 7.21 shows the sketch of the typical thrust variation with speed for an afterburner engine.

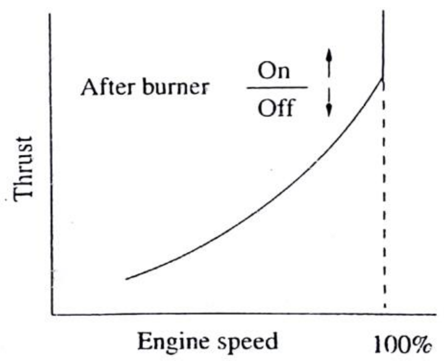


Fig. 7.21 Variation of thrust with speed

7.11.2 Injection of Water-Alcohol Mixture

This method of thrust augmentation is probably the simplest one to achieve. Mixture of water and alcohol or just water is injected at the combustion chamber or compressor inlet section through a series of suitable spray nozzles to produce an increase in thrust.

The effect which produces the greater gain in thrust is the cooling effect within the compressor through water evaporation, which brings about rise in compressor discharge pressure. The rise in pressure can be seen from the

The factors which contribute to thrust augmentation by water-alcohol mixture injection can be summarized such as:

(i) evaporative cooling which produces higher pressure and higher mass flow;

(ii) additional mass of injected fluid; and

(iii) possibility of burning of alcohol.

The first factor determines whether a water injection system is practical on a given engine installation. The first factor also provides the key to determine where water injection produces the higher augmentation thrust ratios. Cooling will be accomplished by the injected fluid until the air at compressor discharge is saturated. Thus to get this saturation point, the amount of cooling increases as the thrust ratio is greater on a hot day than on a cold day and the augmented thrust ratio decreases with altitude and increases the flight Mach number.

The water injected into an axial flow compressor tends to be centrifugally separated from the air. To eliminate this problem, water injection into the combustion chamber has been developed. The principle of opera-

at the turbine and nozzle, these components will handle more mass, the additional mass being the water injected into the burner. Effectively water injection into the combustion chamber produces a thrust increase by

(i) increasing the compressor pressure ratio due to reduced compressor air flow, and

(ii) increasing the total mass flow through turbine and exhaust nozzle. The magnitude of thrust increase is entirely dependent upon compressor operating characteristics.

7.11.3 Bleed Burn Cycle

Since in a turbine excess air is also present, a small percentage of high-pressure air from the compressor is bled to an auxiliary combustion chamber by by-passing the turbine. In auxiliary combustion chamber the bled air is heated by an additional fuel supply to a higher temperature than would be permissible in the main engine on account of the limiting temperature at the turbine blades. The hot gases are then discharged forming an additional jet. A shut off valve is used to bring the engine to normal position. Water is injected into main combustion chamber to replace the mass of the extracted air, thus maintaining the discharge of main jet at the same level. This method is usually used for take-offs only due to high rate of liquid consumption which cannot be carried with engine during its flight. The augmented thrust ratio is highest for this method among the three methods.

Of the three methods discussed, afterburning seems to be the only practical method for thrust augmentation during flight. The air bleed-off system gives maximum thrust augmentation but at the expense of large fuel consumption. This is used only when a large take-off thrust is needed. For smaller thrust augmentation ratio water injection is used because of simplicity and light weight. Afterburner is used for medium thrust augmentation ratio. Afterburner combined with water injection can also be used.

2.11 Effect of pressure, velocity and temperature changes in Compressor

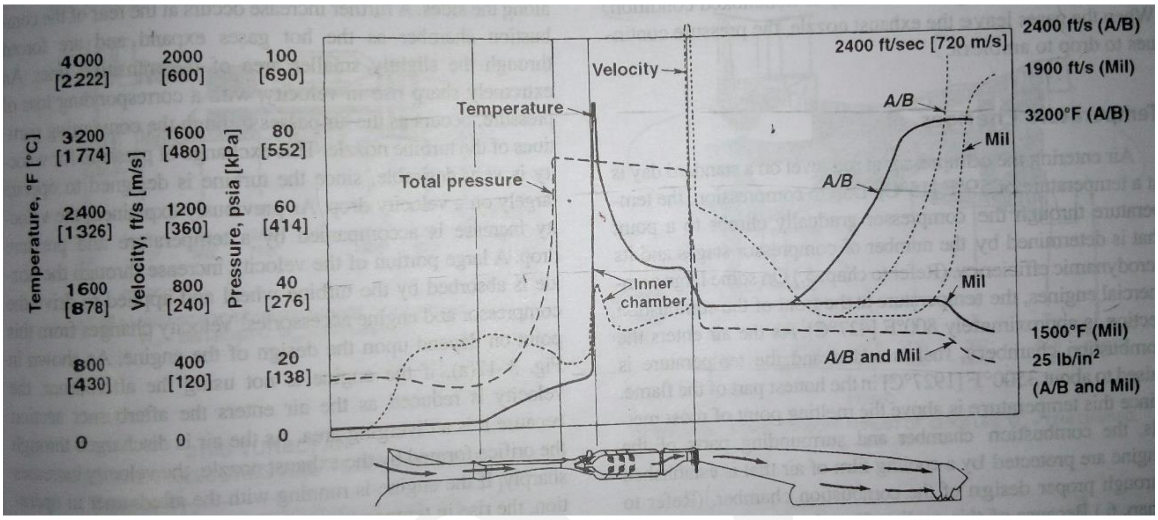


Fig: Temperature, pressure and velocity diagram for a typical turbojet engine with and without afterburner operation

Pressure changes:

- Air usually enters the front of the compressor at a pressure that is less than ambient, indicating that there is considerable suction at the inlet to the engine. This somewhat negative pressure at the engine inlet may be partly or completely overcome by ram pressure as the airplane speed increases.
- From this point on, there is a considerable increasing in the later stages of compression. A final static pressure rise is accomplished in the divergent section of the diffuser.

Temperature changes:

- Air entering the compressor at sea level on a standard day is at a temperature of 15^0C.
- Due to compressor gradually climbs to a point that is determined by the number of compressor stages and its aerodynamic efficiency. On some large commercial engines, the temperature at the front of the combustion section is approximately 427^0C.

Velocity changes:

- The velocity of the air at the front of the compressor must be less than sonic for most present-day compressors.

- If the ambient air velocity is zero (aircraft stationary), the air velocity in front of the duct increases as it is drawn into the compressor. Because the incoming air at zero aircraft forward velocity has no kinetic energy relative to the engine intake before entering, it does not contribute to the total compression ratio. This situation changes as the ram recovery point of the inlet is reached. From this point on, the relative kinetic energy does contribute to the total pressure ratio in the form of ram compression. In a good inlet duct, this compression will occur early and efficiently, with a minimum temperature rise.
- On the other hand, if the airplane speed is high subsonic or supersonic, the air's velocity is slowed in the duct. Airflow velocity through the majority of compressors is almost constant, and in most compressors may decrease slightly. A fairly large drop in airspeed occurs in the enlarging diffuser passage.

2.12 Turboprop engine

Working principle:

- A turboprop engine is a jet engine attached to a propeller. The turbine at the back is turned by the hot gases and this turns a shaft that drives the propeller.
- Like the turbojet engine, the turboprop engine consists of a compressor, combustion chamber and turbine, which than creates the power to drive the compressor.
- Compared to a turbojet engine, the turboprop engine has better propulsion efficiency. Modern turboprop engines are equipped with propellers that have a smaller diameter but a larger number of blades for efficient operation at much higher flight speeds.
- Turboprop engine drives its propulsion by conversion of gas stream energy into mechanical power to drive the compressor, accessories, etc.
- A free turbine is incorporated in the turboprop engine. The shaft in which the free turbine is mounted drives the propeller through the propeller reduction gear system.
- Approximately 90% of thrust comes from propeller and about only 10% comes from the exhaust gases.

Characteristics:

- High propulsive efficiency at low airspeeds, which results in shorter takeoff rolls but fall rapidly as airspeed increases.
- More complicated design and heavier weight than a turbojet.
- Lowest TSFC.

- Large frontal area of propeller and engine combination that necessitates longer landing gears for low wing air planes but does not necessarily increase parasitic drag .
- Possibility of efficient reverse thrust.

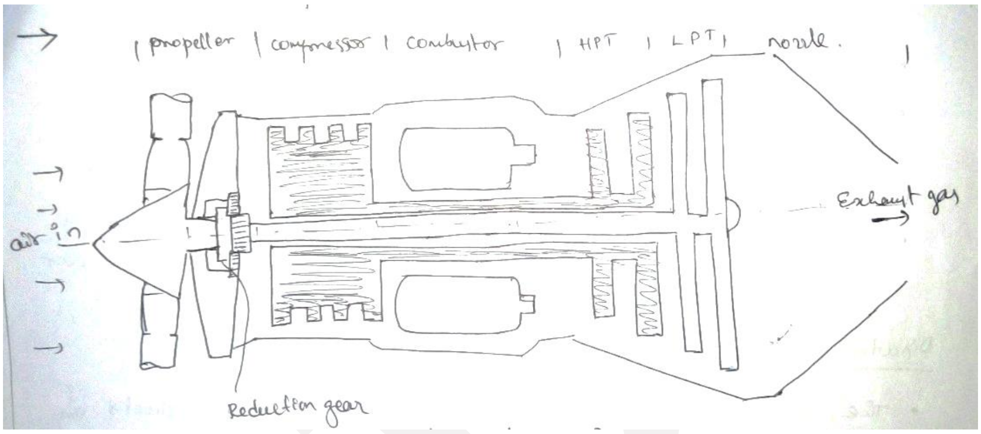


Fig: Turboprop engine

Performance:

Turboprop engines combine in them the high take-off thrust and good propeller efficiency of the propeller engines at speeds lower than 800 km/h and the small weight, lower frontal area, and reduced vibration and noise of the pure jet engine. Its operational range is between that of propeller engines and turbojets though it can operate in any speed upto 800 km/h.

The power developed by the turboprop remains almost same at high altitudes and high speeds as that under sea-level and take-off conditions because as speed increases ram effect also increases. The specific fuel consumption increases with increase in speed and altitude. The thrust developed is high at take-off and reduces at increased speed.

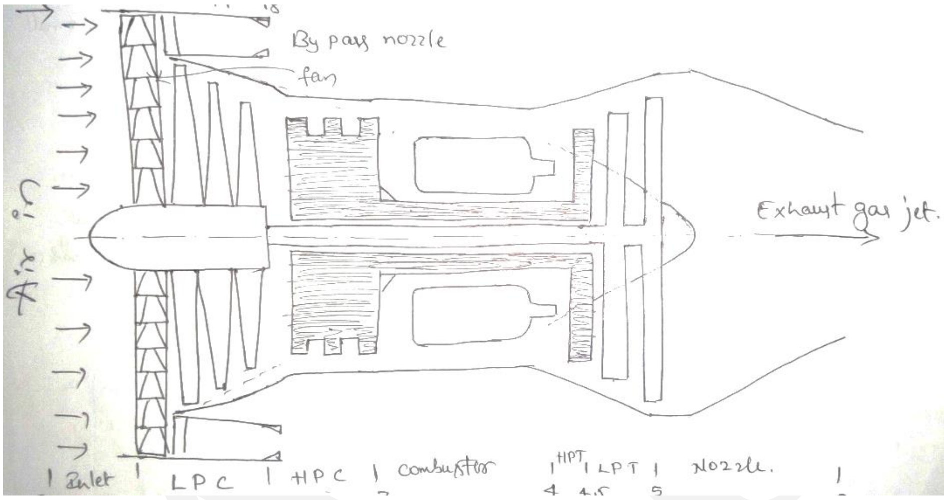
Advantages:

- Turboprop engines have a higher thrust at takeoff and better fuel economy.
- The frontal area is less than propeller engines so that drag is reduced.
- The turboprop can operate economically over a wide range of speeds ranging from low speeds where pure jet engine is uneconomical to high speeds of about 800 km/h where the propeller engine efficiency is low.
- It is easy to maintain and has lower vibrations and noise.
- The power output is not limited as in the case of propeller engines.
- The multishaft arrangement allows a great flexibility of operation over a wide range of speeds.

Disadvantages:

- The main disadvantage is that at high speeds, due to shocks and flow separation. The propeller efficiency decreases rapidly, thereby, putting up a maximum speed limit on the engine.
- It requires a reduction gear which increases the cost and also consumes certain amount of energy developed by the turbine in addition to requiring more space.

2.13 Turbofan engine



Working principle:

- A turbofan engine has a large fan at the front, which sucks in air. Most of the air flows around outside of core engine, making it quieter and giving more thrust at low speeds.
- In a turbojet engine, all the air entering the intake passes through the gas generator, which is composed of the compressor, the combustion chamber and the turbine. However, in a turbofan engine only a portion of the incoming air goes into the combustion chamber.
- The remaining air or fan air (or secondary air) either leaves separately from the primary engine air, or ducted back to mix with the primary air through the engine core at the rear.
- The objective of bypass system is to increase thrust without increasing fuel consumption. This is achieved by increasing the total air mass flow and reducing the velocity within the same total energy supply.
- The increased efficiency of a turbofan engine is combined with a substantial noise reduction, typically 10-20%, which is a very important consideration.

- Turbofan engines are generally classified based on the bypass ratio i.e, low bypass (1:1), medium bypass (2-3:1) and high bypass (4:1 or greater).
- In a low bypass engine, the fan and compressor sections handle approximately the same mass of air flow.
- A medium bypass engine produces thrust ratio which is approximately the same as its bypass ratio. The fan of medium bypass ratio engine has a larger diameter compared to that on a low bypass engine of comparable power.
- A high bypass turbofan engine utilizes even wider diameter fan in order to push more air. In this type of engine about 80% of the thrust is provided by the fan and remaining only 20% by the core engine.

Characteristics:

- Increased thrust at forward speeds similar to turboprop results in a relatively short takeoff. However, unlike the turboprop, the turbofan thrust is not penalized with increasing airspeed, up to approximately Mach 1 with current fan designs.
- Weight falls between turbojet and turboprop.
- Ground clearances are less than turboprop but not as good as turbojet.
- TSFC and specific weight falls between turbojet and turboprop, resulting in increased operating economy and aircraft range over the turbojet.
- Considerable noise level reduction of 10 to 20 percent over the turbojet reduces acoustic fatigue in surrounding aircraft parts and is less objectionable to the people on the ground.

Advantages:

- Higher thrust at lower airspeeds.
- Lower TSFC.
- Shorter takeoff distance.
- Considerable noise reduction.

Disadvantages:

- Higher specific weight.
- Larger frontal area.
- Inefficient at high altitudes.

2.14 Turbojet engine

Working principle:

- The turbojet engine is a reaction engine. In a reaction engine, expanding gases push hard against the front of the engine.
- Turbojet engine derives its thrust by accelerating a mass of air through the core engine.
- The air taken in from an opening in the front of the engine is compressed to about 3-12 times its original pressure in a centrifugal or axial compressor.
- Fuel is added to the air and burned in a combustion chamber to raise the temperature of the mixer to about 1100^0C. The resulting hot air is passed through a turbine, which drives the compressor.

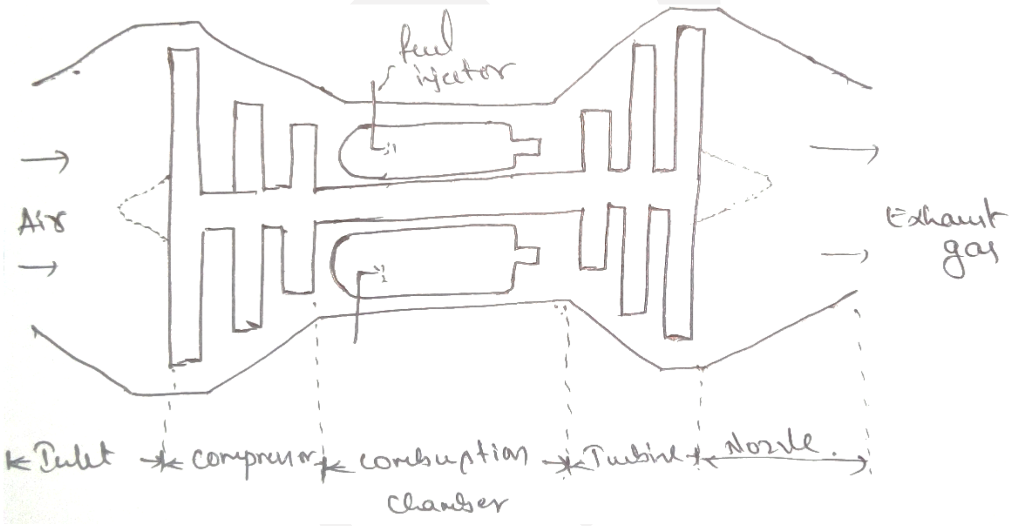


Fig: The turbojet engine

- If the turbine and compressor are efficient, the pressure at the turbine discharge will be nearly twice the atmospheric pressure.
- This excess pressure is sent to the nozzle to produce a high velocity stream of gas which produces the thrust. Thus all the propulsive force produced by a jet engine is derived from exhaust gases.
- An afterburner (or a reheat) is an additional component added to some jet engines. Primarily those on military supersonic aircrafts.
- Its purpose is to provide a temporary increase in thrust at the time of supersonic flight as well as takeoff.

- On military aircraft, the extra thrust is also useful for combat situations. This is achieved by injecting additional fuel into the jet pipe downstream of (after) the turbine.

Characteristics:

- Low thrust at low forward speed.
- Relatively high, thrust specific fuel consumption (TSFC) at low altitude and airspeeds, a disadvantage that decreases as altitude and airspeed increase.
- Long takeoff roll.
- Small frontal area, resulting in low drag and reduced ground clearance problems.
- Lightest specific weight.
- Ability to take advantage of high ram pressure ratios.

Advantages:

- The power to weight ratio of a turbojet engine is about 4 times that of a propeller system having reciprocating engines.
- It is simple, easy to maintain and requires lower lubricating oil consumption. Furthermore, complete absence of liquid cooling results in reduced frontal area.
- There is no limit to the power output which can obtained from a turbojet while the piston engines have reached almost their peak power and further increase will be at the cost of complexity and greater engine weight and frontal area of the aircraft.
- The speed of the turbojet engine is not limited by the propeller and it can attain higher flight speeds than engine propeller aircrafts.

Disadvantages:

- The fuel economy at low operational speeds is extremely poor.
- It has low takeoff thrust and hence poor starting characteristics.

2.15 Problems (Solve all the problems, whatever did in class)

7.1 A turbojet power plant uses aviation kerosene having a calorific value of 43 MJ/kg. The fuel consumption is 0.18 kg per hour per N of thrust, when the thrust is 9 kN. The aircraft velocity is 500 m/s the mass of air passing through the compressor is 27 kg/s. Calculate the air-fuel ratio and overall efficiency.

Solution

$$\dot{m}_f = \frac{0.18}{3600} \times 9000 = 0.45 \text{ kg/s}$$

$$\text{Air-fuel ratio} = \frac{27}{0.45} = \mathbf{60:1} \quad \stackrel{\text{Ans}}{\Longleftarrow}$$

$$\text{Thrust power, } P_T = F \times c_i$$

$$= 9 \times 500 = 4500 \text{ kW}$$

$$\text{Heat input, } Q = 0.45 \times 43000 = 19350 \text{ kW}$$

$= \dot{m}_f \times CV$

$$\eta = \frac{P_T}{Q}$$

$$= \frac{4500}{19350} \times 100$$

$$= \mathbf{23.26\%} \quad \stackrel{\text{Ans}}{\Longleftarrow}$$

7.4 The effective jet exit velocity from a jet engine is 2700 m/s. The forward flight velocity is 1350 m/s and the air flow rate is 78.6 kg/s. Calculate

(i) thrust,

(ii) thrust power, and

(iii) propulsive efficiency.

Solution

$$\alpha = \frac{c_i}{c_j} = \frac{1350}{2700} = 0.5$$

$$\text{Thrust, } F = \dot{m}_a(c_j - c_i)$$

$$= 78.6 \times (2700 - 1350)$$

$$= 106110 \text{ N}$$

$$\text{Thrust power} = F \times c_i$$

$$= 106110 \times 1350$$

$$= \mathbf{143.25 \times 10^6 \ W} \quad \stackrel{\text{Ans}}{\Longleftarrow}$$

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

$$= \frac{2 \times 0.5}{1.5} = 0.6667$$

$$= \mathbf{66.67\%} \quad \stackrel{\text{Ans}}{\Longleftarrow}$$

7.10 Air enters a turbojet engine at a rate of 12×10^4 kg/h at 15°C and 1.03 bar and is compressed adiabatically to 182°C and four times the pressure. Products of combustion enter the turbine at 815°C and leave it at 650°C to enter the nozzle. Calculate the isentropic efficiency of the compressor, the power required to drive the compressor, the exit speed of gases and thrust developed when flying at 800 km/h. Assume the isentropic efficiency of turbine is same as that of the compressor and the nozzle efficiency 90%.

Solution

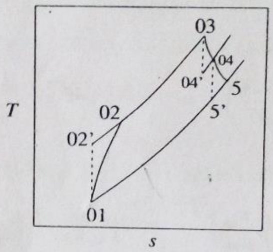


Fig. 7.32

$$T_{02} - T_{01} = \frac{T_{01}}{\eta_C}\left[r_c^{\left(\frac{\gamma-1}{\gamma}\right)} - 1\right]$$

$$455 - 288 = \frac{288}{\eta_C} \times (4^{0.286} - 1)$$

$$\eta_C = 84\% = \eta_T \quad \text{Ans}$$

Compressor work

$$\dot{m}_a = \frac{12 \times 10^4}{3600} = 33.33 \text{ kg/s}$$

$$W_C = \dot{m}_a C_p (T_{02} - T_{01}) = 33.33 \times 1.005 \times (455 - 288)$$

$$= \mathbf{5594\ kW} \quad \text{Ans}$$

$$T_{03} - T_{04} = \eta_T T_{03}\left(1 - \frac{1}{r_t^{0.248}}\right)$$

$$1088 - 923 = 1088 \times 0.84 \times \left(1 - \frac{1}{r_t^{0.248}}\right)$$

$$r_t = 2.23$$

$$p_{04} = \frac{p_{03}}{r_t} = \frac{4.12}{2.23} = 1.8475 \text{ bar}$$

$$\frac{p_{04}}{p_a} = \frac{1.8475}{1.03} = 1.79$$

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

$$= \frac{1}{\left[1 - \frac{1}{0.9} \times \left(\frac{0.33}{2.33}\right)\right]^{4.03}} = 1.99$$

Nozzle will not choke.

$$\frac{T_{5'}}{T_{04}} = \left(\frac{p_5}{p_{04}}\right)^{\frac{\gamma-1}{\gamma}}$$

$$T_{5'} = 923 \times \left(\frac{1.03}{1.8478}\right)^{\frac{0.33}{1.33}} = 798.46 \text{ K}$$

$$\eta_n = \frac{T_{04} - T_5}{T_{04} - T_{5'}}$$

$$T_5 = T_{04} - \eta_n (T_{04} - T_{5'})$$

$$= 923 - 0.9 \times (923 - 798.46) = 810.91 \text{ K}$$

$$c_j = \sqrt{2 \times 1147 \times (923 - 810.91)}$$

$$= \mathbf{507.1\ m/s} \quad \text{Ans}$$

$$F = \dot{m}(c_j - c_i) = 33.33 \times (507.1 - 222.2)$$

$$= \mathbf{9495.72\ N} \quad \text{Ans}$$