SAEX 1018	PROPULSION II	L	T	P	Credits	Total Marks
		3	0	0	3	100

UNIT I TURBINES **10 hrs.**

Introduction to turbine analysis, mean radius stage calculations, stage parameters, stage loading and flow coefficients degree of reaction, stage temperature ratio and pressure ratio, blade spacing, radial variation, velocity ratio. Axial flow turbine, stage flow path, Dimensional stage analysis.; Steps of turbine design: single stage and two stages. Turbine performance. Blade cooling

UNIT II COMPRESSOR **10 hrs.**

Principal of operation of centrifugal compressor - Work done and pressure rise Euler's Turbo machinery equations. Axial flow compressor analysis, cascade action, flow field. Euler's equation, velocity diagrams, flow annulus area stage parameters. Degree of reaction, cascade airfoil nomenclature and loss coefficient, diffusion factor, stage loading and flow coefficient, stage pressure ratio, Blade Mach No., repeating stage, repeating row

UNIT III RAMJET PROPULSION **10 hrs.**

Operating principle – Subcritical, critical and supercritical operation - Combustion in ramjet engine – Ramjet performance - Sample ramjet design calculations - Introduction to scramjet - Preliminary concepts in supersonic combustion - Integral ram rocket - Numerical problems.

UNIT IV FUNDAMENTALS OF ROCKET PROPULSION **10 hrs.**

Operating principle - Specific impulse of a rocket - Internal ballistics - Rocket nozzle classifications – Rocket performance considerations - Numerical problems. Electric, ion, Nuclear rocket propulsion -Principles

UNIT V CHEMICAL ROCKETS **10 hrs.**

Solid propellant rockets - Selection criteria of solid propellants - Important hardware components of solid rockets - Propellant grain design considerations - Liquid propellant rockets - Selection of liquid propellants Thrust control in liquid rockets - Cooling in liquid rockets - Limitations of hybrid rockets - Relative advantages of liquid rockets over solid rockets –Numerical Problems

(Computational problems must be given as assignments for each unit)

UNIT I - TURBINES

TURBINES

Introduction

1. The high temperature, high pressure gases leaving the combustion system contain a large amount of energy, most of which needs to be extracted as efficiently as possible to drive the compressor and engine driven accessories. The remainder is available for output, either by driving a power turbine or by forming a propelling jet.

2. The extraction of this energy is done by the turbine which, like the axial flow compressor, consists of stages of fixed blades, known as turbine stator blades or more usually nozzle guide vanes (NGV), and rotor blades. Each turbine 'stage' contains one set of NGVs followed by one set of rotor blades.

3. The turbine differs from the compressor, however, in that by expanding the gas flow it is moving it in the direction of decreasing pressure, i.e. the gas's direction of natural flow, so the tendency to incur losses is much reduced. This fundamental difference between compressor and turbine is useful to the engine designer, as it allows the use of a turbine with few stages to drive a multi-stage compressor spool.

Energy Transfer from Gas Flow to Turbine

4. **Reaction and Impulse Turbines.** Turbine stages may be designed as predominantly impulse or predominantly reaction with a considerable change in contour from blade root to tip.

(a) **Impulse.** In a turbine stage where the blading is of the impulse type, a pressure drop occurs only in the convergent NGV passages, together with a corresponding velocity

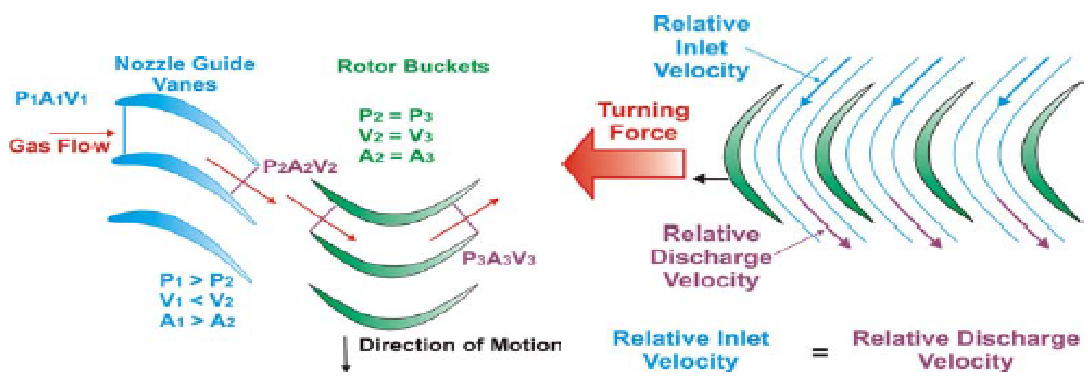


Fig 5-1: Impulse Blading

increase. The resultant stream of high velocity gas is directed at the rotor blades (Fig 5-1) where the passages are constant in area and therefore there is no further pressure drop. However, although its scalar element remains constant, the direction of the air velocity is changed, producing an impulse on the turbine which causes it to rotate. This is the oldest system and can be likened to the water wheel. It is used for starter turbines and APUs, but not in its purest form, in gas turbine engines.

(b) **Reaction.** Exactly the opposite series of events takes place in a reaction turbine. The entire pressure drop takes place between the rotor blades which have convergent passages in the direction of flow - the NGVs do no more than alter or guide the flow for the rotors (Fig 5-2). The turbine is driven round by the reaction force resulting from the acceleration of the gas through the converging blade passage where both the direction and the magnitude of the gas velocity are changed. Again, in practice, pure reaction blading is not used in gas turbine engines as it is inefficient.

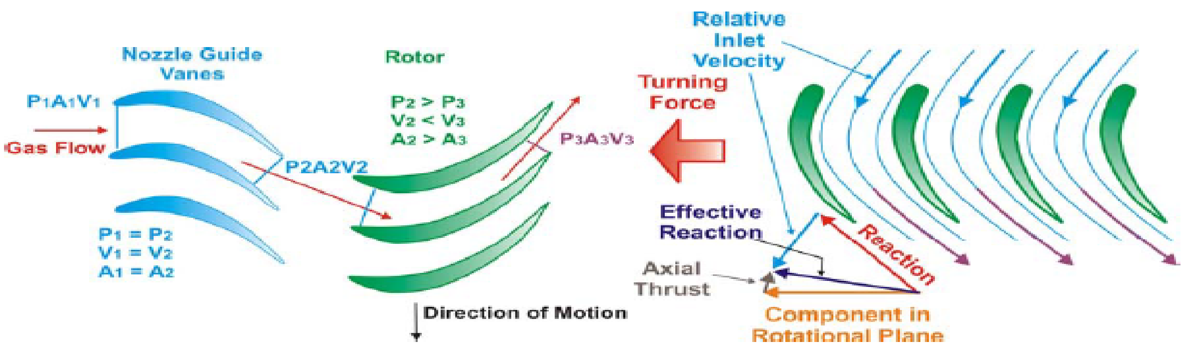


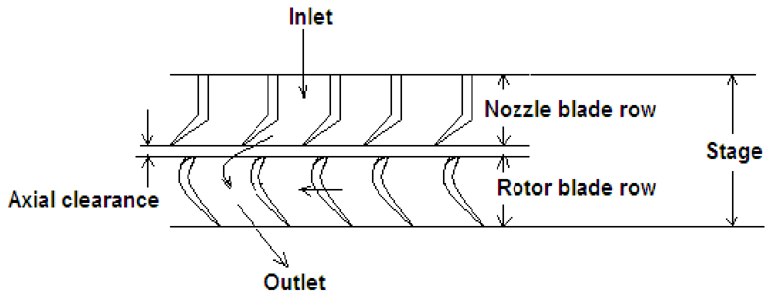
Fig 5-2: Reaction Blading

(c) **Practical Aircraft Engine Blading.** In a turbojet aircraft, a compromise between impulse and reaction blading is used, where the blade root is largely impulse blading and the blade tip mainly reaction blading, with a smooth transition along the length of the blade.

Axial and radial flow turbines

As with the compressor, there are two basic types of turbine—radial flow and axial flow. The vast majority of gas turbines employ the axial flow turbine.The radial turbine can handle low mass flows more efficiently than the axial flow machine and has been widely used in the cryogenic industry as a turbo-expander, and in turbochargers for reciprocating engines. Although for all but the lowest powers the axial flow turbine is normally the more efficient, when mounted back-to-back with a centrifugal compressor the radial turbine offers the benefit of a very short and rigid rotor. This configuration is eminently suitable for gas turbines where compactness is more important than low fuel consumption. Auxiliary power units for aircraft (APUs), generating sets of up to 3 MW, and mobile power plants are typical applications.

Impulse and Reaction Turbine



A turbine stage

Work can be extracted from a gas at a higher inlet pressure to the lower back pressure by allowing it to flow through a turbine. In a turbine as the gas passes through, it expands. The work done by the gas is equivalent to the change of its enthalpy. It is a well known fact that the turbines operate on the momentum principle. Part of the energy of the gas during expansion is converted into kinetic energy in the flow nozzles. The gas leaves these stationary nozzles at a relatively higher velocity. Then it is made to impinge on the blades over the turbine rotor or wheel. Momentum imparted to the blades turns the wheel. Thus, the two primary parts of the turbine are,

(i) The stator nozzles, and

(ii) the turbine rotor blades.

Normally a turbine stage is classified as

(i) an impulsion stage, and

(ii) a reaction stage

An impulse stage is characterized by the expansion of the gas which occurs only in the stator nozzles. The rotor blades act as directional vanes to deflect the direction of the flow. Further, they convert the kinetic energy of the gas into work by changing the momentum of the gas more or less at constant pressure. A reaction stage is one in which expansion of the gas takes place both in the stator and in the rotor.

The function of the stator is the same as that in the impulse stage, but the function in the rotor is two fold. (i) the rotor converts the kinetic energy of the gas into work, and

(ii) contributes a reaction force on the rotor blades.

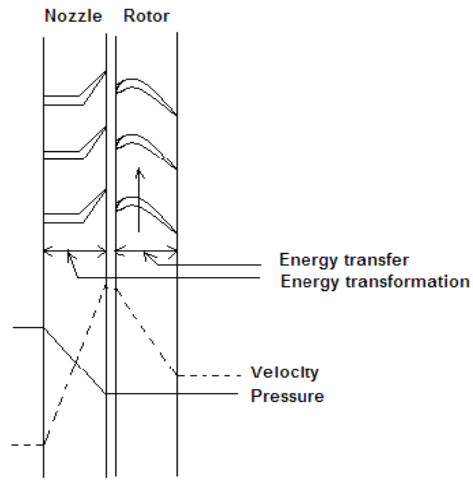
The reaction force is due to the increase in the velocity of the gas relative to the blades. This results from the expansion of the gas during its passage through the rotor.

A Single Impulse Stage

Impulse machines are those in which there is no change of static or pressure head of the fluid in the rotor. The rotor blades cause only energy transfer and there is no energy transformation. The energy transformation from pressure or static head to kinetic energy or vice versa takes place in fixed blades only. As can be seen from the below figure that in the rotor blade passage of an impulse turbine there is no acceleration of the fluid, i.e., there is no energy transformation. Hence, the chances are greater for separation due to boundary layer growth on the blade surface.

Due to this, the rotor blade passages of the impulse machine suffer greater losses giving lower stage efficiencies.

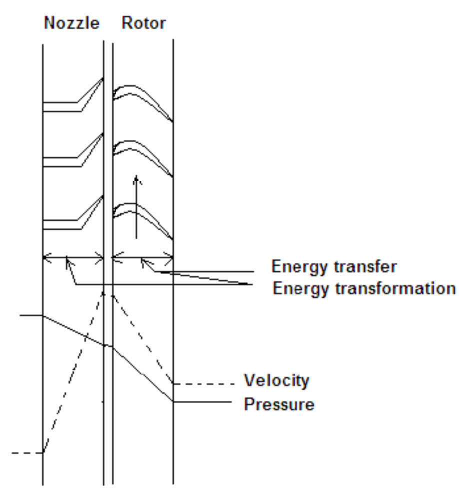
The paddle wheel, Pelton wheel and Curtis stem turbine are some examples of impulse machines.



An Impulse turbine stage

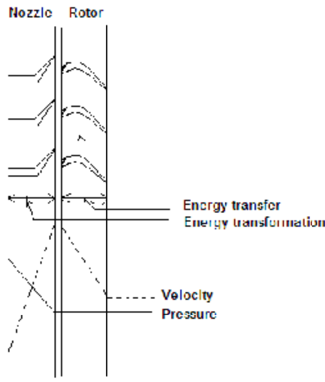
A Single Reaction Stage

The reaction stages are those, in which, changes in static or pressure head occur both in the rotor and stator blade passages. Here, the energy transformation occurs both in fixed as well as moving blades. The rotor experiences both energy transfer as well as energy transformation. Therefore, reaction turbines are considered to be more efficient. This is mainly due to continuous acceleration of flow with lower losses.

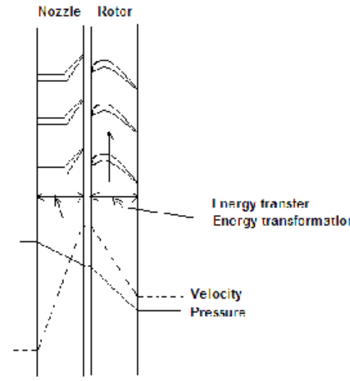


A reaction turbine stage

The degree of reaction of a turbomachine stage may be defined as the ratio of the static or pressure head change occurring in the rotor to the total change across the stage.



An Impulse turbine stage



A reaction turbine stage

<u>Velocity Triangles of a Single Stage Machine</u>

The flow geometry at the entry and exit of a turbomachine stage is described by the velocity triangles at these stations. The velocity triangles for a turbomachine contain the following three components.

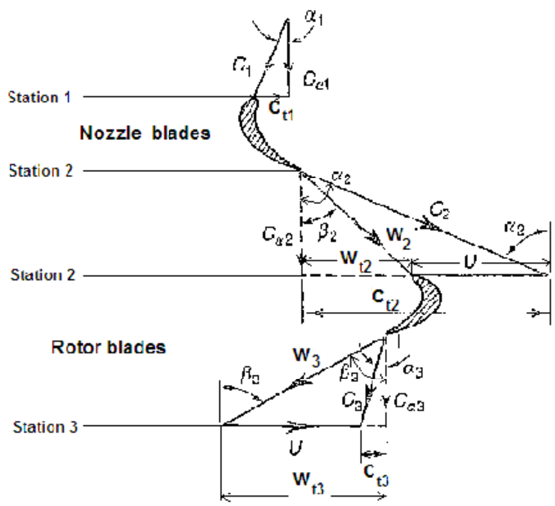
1. The peripheral / whirl / tangential velocity (u) of a rotor blades
2. The absolute velocity (c) of the fluid and
3. The relative velocity (w or v) of the fluid

These velocities are related by the following well-known vector equation

$$\vec{c} = \vec{u} + \vec{w}$$

This simple relation is frequently used and is very useful in drawing the velocity triangles for turbomachines.

The notation used here to draw velocity triangles correspond to the x-y coordinates; the suffix (a or α) identifies components in the axial direction and suffix (t) refers to tangential direction. Air angles in the absolute system are denoted by alpha (α), where as those in the relative system are represented by beta (β).



Velocity triangle for a turbine stage

Since the stage is axial, the change in the mean diameter between its entry and exit can be neglected. Therefore, the peripheral or tangential velocity (u) remains constant in the velocity triangles.

It can be proved from the geometry that

$$c_{t2} + c_{t3} = w_{t2} + w_{t3}$$

It is often assumed that the axial velocity component remains constant through the stage. For such condition,

$$c_a = c_{a1} = c_{a2} = c_{a3}$$

For constant axial velocity yields a useful relation, $\tan\alpha_2 + \tan\alpha_3 = \tan\beta_2 + \tan\beta_3$

Expression for Work Output

Though force and torque are exerted on both stationary and moving blades alike, work can only be done on the moving rotor blades. Thus the rotor blades transfer energy from the fluid to the shaft.

The stage work in an axial turbine ($u_3 = u_2 = u$) can be written as, $W = u_2 c_{t2} - u_3 c_{t3} = u\{c_{t2} - (-c_{t3})\} = u(c_{t2} - c_{t3})$

This equation can also be expressed in another form,

$$W = u^2\left(\frac{c_{t2}}{u} + \frac{c_{t3}}{u}\right)$$

The first term $\left(\frac{c_{t2}}{u}\right)$ in the bracket depends on the nozzle or fixed angle (α_2) and the ratio $\sigma = \frac{u}{c_2}$. The contribution of the second term $\left(\frac{c_{t3}}{u}\right)$ to the work is generally small. It is also observed that the kinetic energy of the fluid leaving the stage is greater for larger values of c_{t3}. The leaving loss from the stage is minimum when $c_{t3} = 0$, i.e., when the discharge from the stage is axial ($c_3 = c_{a3}$). However, this condition gives lesser stage work as can be seen from the above two equations.

Blade loading and Flow coefficients

Performance of turbomachines are characterized by various dimensionless parameters. For example, loading coefficient (ψ) and the flow coefficient (Φ) have been defined as,

$$\Psi = \frac{W}{u^2}$$

$$\phi = \frac{c_a}{u}$$

Since the work, W in the above equation is frequently referred to as the blade or stage work, the coefficient, ψ would also be known as the blade or stage loading coefficient.

For constant axial velocity (c_a), it can be shown that

$$\Psi = \Phi(\tan\alpha_2 + \tan\alpha_3) = \Phi(\tan\beta_2 + \tan\beta_3)$$

The $\Phi - \psi$ plots are useful in comparing the performances of various stages of different sizes and geometries.

Blade and Stage efficiencies

Even though the blade and stage work (outputs) are the same, the blade and the stage efficiencies need not be equal. This is because the energy inputs to the rotor blades and the stage (fixed blade ring plus the rotor) are different. The blade efficiency is also known as the utilization factor (ε) which is an index of the energy utilizing capability of the rotor blades. Thus,

$\varepsilon = \eta_b$

$= \text{Rotor blade work / Energy supplied to the rotor blades}$

$= W / E_{rb}$

$W = u_2 c_{t2} + u_3 c_{t3}$

$$= \frac{1}{2}\left(c_2^2 - c_3^2\right) + \frac{1}{2}\left(u_2^2 - u_3^2\right) + \frac{1}{2}\left(w_3^3 - w_2^2\right)$$

The energy supplied to the rotor blades is the absolute kinetic energy in the jet at the entry plus the kinetic energy change within the rotor blades.

$$E_{rb} = \frac{1}{2}c_2^2 + \frac{1}{2}\left(w_3^2 - w_2^2\right) + \frac{1}{2}\left(u_2^3 - u_3^2\right)$$

For axial machines, $u = u_2 = u_3$

$$\varepsilon = \eta_b = \frac{\left(c_2^2 - c_3^2\right) + \left(w_3^2 - w_2^2\right)}{c_2^2 + \left(w_3^2 - w_2^2\right)}$$

Maximum utilization factor for a single impulse stage.

$$\varepsilon = \frac{u\left(c_{t2} + c_{t3}\right)}{\frac{1}{2}c_2^2}$$

After rearranging the terms, we have

$$\eta_b = \varepsilon = 4\left(\sigma \sin\alpha_2 - \sigma^2\right)$$

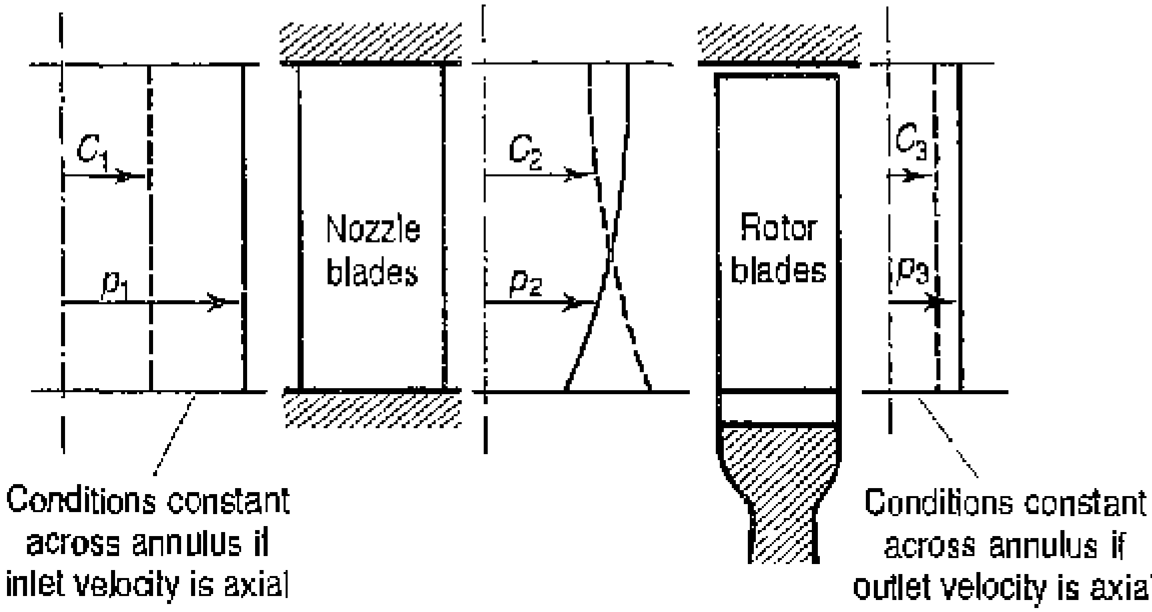
This shows that the utilization factor is a function of the blade-to-gas speed ratio and the nozzle angle.

Vortex theory

It was pointed out earlier that the shape of the velocity triangles must vary from root to tip of the blade because the blade speed U increases with radius. Another reason is that the whirl component in the flow at outlet from the nozzles causes the static pressure and temperature to vary across the annulus. With a uniform pressure at inlet, or at least with a much smaller variation because the whirl component is smaller, it is clear that the pressure drop across the nozzle will vary giving rise to a corresponding variation in efflux velocity C_2. Twisted blading designed to take account of the changing gas angles is called vortex blading.

It has been common steam turbine practice, except in low-pressure blading where the blades are very long, to design on conditions at the mean diameter, keep the blade angles constant from root to tip, and assume that no additional loss is incurred by the variation in incidence along the blade caused by the changing gas angles. Comparative tests have been conducted by the earlier researchers on a single-stage gas turbine of radius ratio 1-37, using in turn blades of constant angle and vortex blading. The results showed that any improvement in efficiency obtained with vortex blading was within the margin of experimental error. This contrasts with similar tests on a 6-stage axial compressor, by another researcher, which showed a distinct improvement from the use of vortex blading. This was, however, not so much an improvement in efficiency (of about 1-5 per cent) as in the delay of the onset of surging which of course does not arise in accelerating flow. It appears, therefore, that steam turbine designers have been correct in not applying vortex theory except when absolutely necessary at the LP end. They have to consider the additional cost of twisted blades for the very large number of rows of blading required, and they know that the Rankine cycle is relatively insensitive to component losses. Conversely, it is not surprising that **the gas turbine designer, struggling to achieve the highest possible component efficiency, has consistently used some form of vortex blading which it is felt intuitively must give a better performance however small.**

Vortex theory has been outlined earlier by Cohen and others where it was shown that if the elements of fluid are to be in radial equilibrium, an increase in static pressure from root to tip is necessary whenever there is a whirl component of velocity. Figure 7.8 shows (see below) why the gas turbine designer cannot talk of impulse or 50 per cent reaction stages. The proportion of the stage pressure or temperature drop which occurs in the rotor must increase from root to tip. Although Fig. 7.8 refers to a single-stage turbine with axial inlet velocity and no swirl at outlet, the whirl component at inlet and outlet of a repeating stage will be small compared with CW2- the reaction will therefore still increase from root to tip, if somewhat less markedly.



Changes in pressure and velocity across the annulus

Choice of blade profile, pitch and chord

The next step is to choose stator and rotor blade shapes which will accept the gas incident upon the leading edge, and deflect the gas through the required angle with the minimum loss. An overall blade loss coefficient Y (or A) must account for the following sources of friction loss.

(a) Profile loss—associated with boundary layer growth over the blade profile (including separation loss under adverse conditions of extreme angles of incidence or high inlet Mach number).

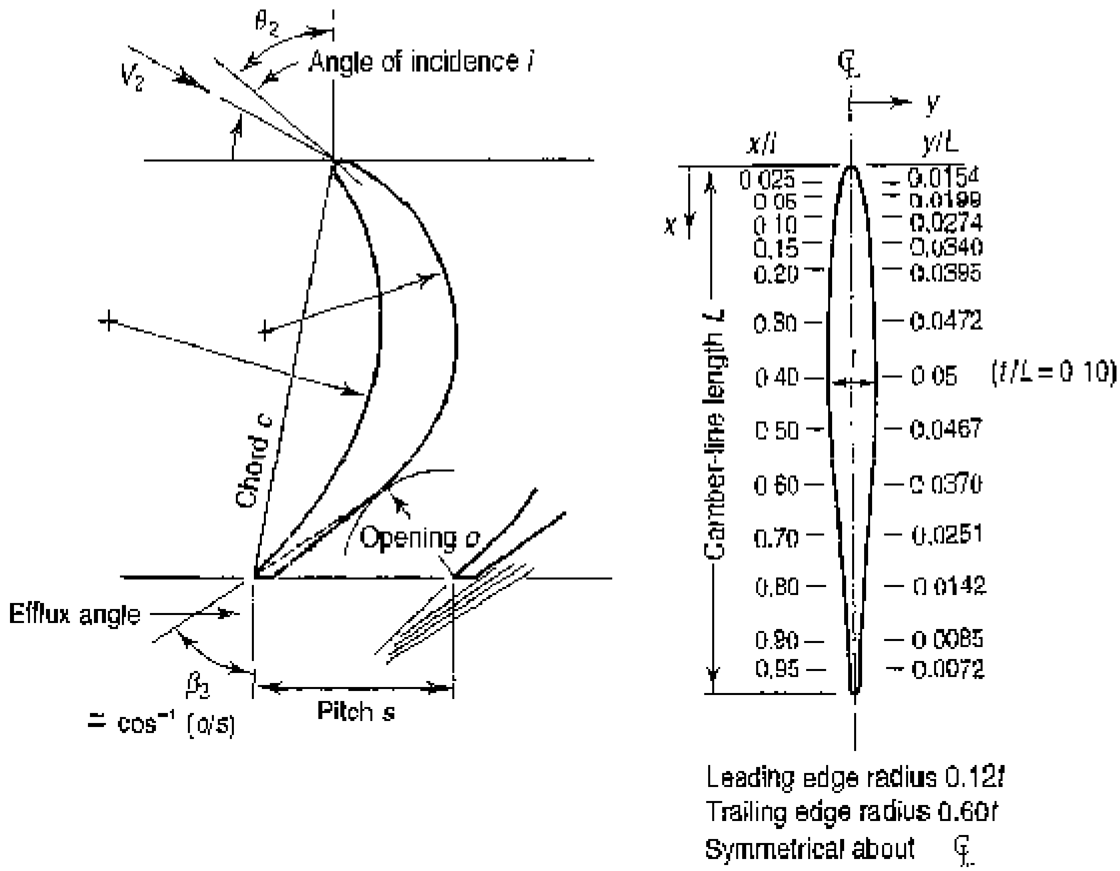
(b) Annulus loss—associated with boundary layer growth on the inner and outer walls of the annulus.

(c) Secondary flow loss—arising from secondary flows which are always present when a wall boundary layer is turned through an angle by an adjacent curved surface.

(d) Tip clearance loss—near the rotor blade tip the gas does not follow the intended path, fails to contribute its quota of work output, and interacts with the outer wall boundary layer.

The profile loss coefficient Yp is measured directly in cascade tests similar to those described for compressor blading. Losses (b) and (c) cannot easily be separated, and they are accounted for by a secondary loss coefficient Ys.

The tip clearance loss coefficient, which normally arises only for rotor blades, will be denoted by Y_k. Thus the total loss coefficient Y comprises the accurately measured two-dimensional loss Y_p, plus the three-dimensional loss (Y_s+Y_k) which must be deduced from turbine stage test results. All that is necessary for our present purpose for finding the choice of blade profile is limited to the knowledge of the sources of loss.



'Conventional' blade profiles: steam turbine section and T6 aerofoil section

Figure 7.11 shows a conventional steam turbine blade profile constructed from circular arcs and straight lines. Gas turbines have until recently used profiles closely resembling this, although specified by aerofoil terminology.

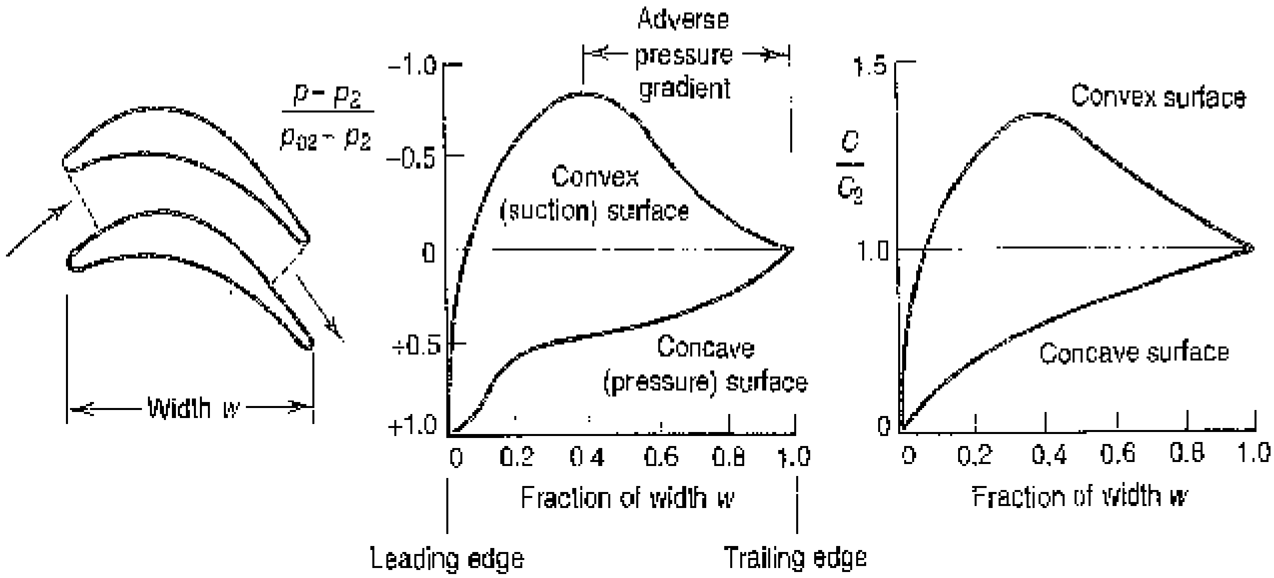


FIG. 7.20 Pressure and velocity distributions on a conventional turbine blade

Note that the blade profile will be completely determined when (a) the pitch/width ratio (s/w) is established, and (b) both the camber line angle α' and blade thickness/pitch ratio have been calculated for various values of x between 0 and 1.

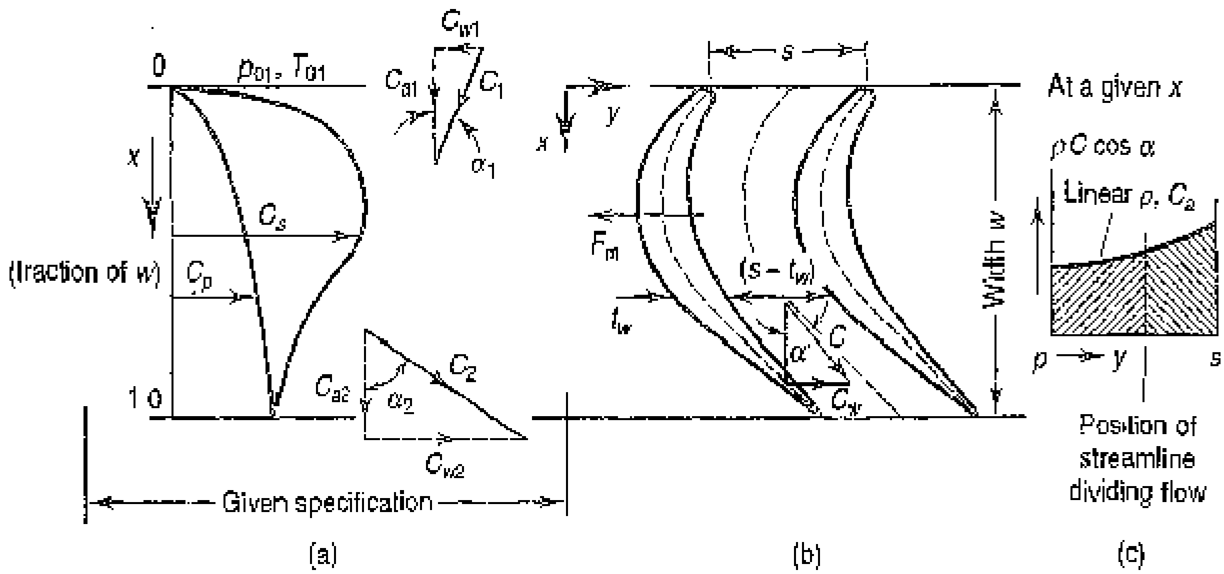


FIG. 7.22 The 'indirect' problem for two-dimensional compressible flow in a turbine cascade

Turbine Performance

The performance of turbine is limited principally by two factors: compressibility and stress. Compressibility limits the mass flow that can pass through a given turbine and, stress limits the wheel speed U. The work per stage depends on the square of the wheel speed. However, the performance of the engine depends very strongly on the maximum temperature. Of course, as the maximum temperature increases, the allowable stress level diminishes; hence in the design of the engine there must be a compromise between maximum temperature and maximum rotor tip speed U.

For given pressure ratio and adiabatic efficiency, the turbine work per unit mass is proportional to the inlet stagnation temperature. Since, in addition, the turbine work in a jet or turboshaft engine is commonly two or three times the useful energy output of the engine, a 1% increase in turbine inlet temperature can produce a 2% or 3% increase in engine output. This considerable advantage has supplied the incentive for the adoption of fairly elaborate methods for cooling the turbine nozzle and rotor blades.

Estimation of stage performance

The last step in the process of arriving at the preliminary design of a turbine stage is to check that the design is likely to result in values of nozzle loss coefficient and stage efficiency which were assumed at the outset. If not, the design calculations may be repeated with more probable values of loss coefficient and efficiency. When satisfactory agreement has been reached, the final design may be laid out on the drawing board and accurate stressing calculations can be performed. Before proceeding to describe a method of estimating the design point performance of a stage, however, the main factors limiting the choice of design, which we have noted during the course of the worked example, will be summarized. The reason we considered a turbine for a turbojet engine was simply that we would thereby be working near those limits to keep size and weight to a minimum. The designer of an industrial gas turbine has a somewhat easier task: he will be using lower temperatures and stresses to obtain a longer working life, and this means lower mean blade speeds, more stages, and much less stringentaerodynamic limitations. A power turbine, not mechanically coupled to the gas generator, is another case where much less difficulty will be encountered in arriving at a satisfactory solution. The choice of gear ratio between the power turbine and driven component is normally at the disposal of the turbine designer, and thus the rotational speed can be varied to suit the turbine, instead of the compressor as we have assumed here.

Limiting factors in turbine design

(*a*) *Centrifugal stresses* in the blades are proportional to the square of the rotational speed N and the annulus area: when N is fixed they place an upper limit on the *annulus area*.

(*b*) *Gas bending stresses* are (1) inversely proportional to the number of blades and blade section moduli, while being (2) directly proportional to the blade height and specific work output.

(1) The *number of blades* cannot be increased beyond a point set by blade fixing considerations, but the *section moduli* are roughly proportional to the cube of the *blade chord* which might be increased to reduce σ_{gb}. There is an aerodynamic limit on the *pitch/chord ratio*, however, which if too small will incur a high loss coefficient (friction losses increase because a reduction in s/c increases the blade surface area swept by the gas).

(2) There remains the *blade height*: but reducing this while maintaining the same annulus area (and therefore the same axial velocity for the given mass flow), implies an increase in the *mean diameter of the annulus*. For a fixed N, the mean diameter cannot be increased without increasing the *centrifugal disc stresses*. There will also be an aerodynamic limit set by the need to keep the *blade aspect ratio* (h/c) and *annulus radius ratio* (r_t/r_r) at values which do not imply disproportionate losses due to *secondary flows*, *tip clearance* and *friction on the annulus walls* (say not less than 2 and 1·2 respectively). The blade height might be reduced by reducing the annulus area (with the added benefit of reducing the centrifugal blade stresses) but, for a given mass flow, only by increasing the *axial velocity*. An aerodynamic limit on C_a will be set by the need to keep the maximum *relative Mach number at the blade inlet* (namely at the root radius), and the *Mach number at outlet from the stage*, below the levels which mean high friction losses in the blading and jet pipe respectively.

(*c*) *Optimizing the design*, so that it just falls within the limits set by all these conflicting mechanical and aerodynamic requirements, will lead to an efficient turbine of minimum weight. If it proves to be impossible to meet one or more of the limiting conditions, the required work output must be split between *two stages*. The second design attempt would be commenced on the assumption that the efficiency is likely to be a maximum when the work, and hence the temperature drop, is divided equally between the stages.

(*d*) The velocity triangles, upon which the rotor blade section depends, are partially determined by the desire to work with an average *degree of reaction of 50 per cent* to obtain low blade loss coefficients and *zero swirl* for minimum loss in the jet pipe. To avoid the need for two stages in a marginal case, particularly if it means adding a bearing on the downstream side, it would certainly be preferable to design with a lower degree of reaction and some swirl. An aerodynamic limit on the minimum value of the reaction at mean diameter is set by the need to ensure some *positive reaction at the blade root radius.*

The cooled turbine

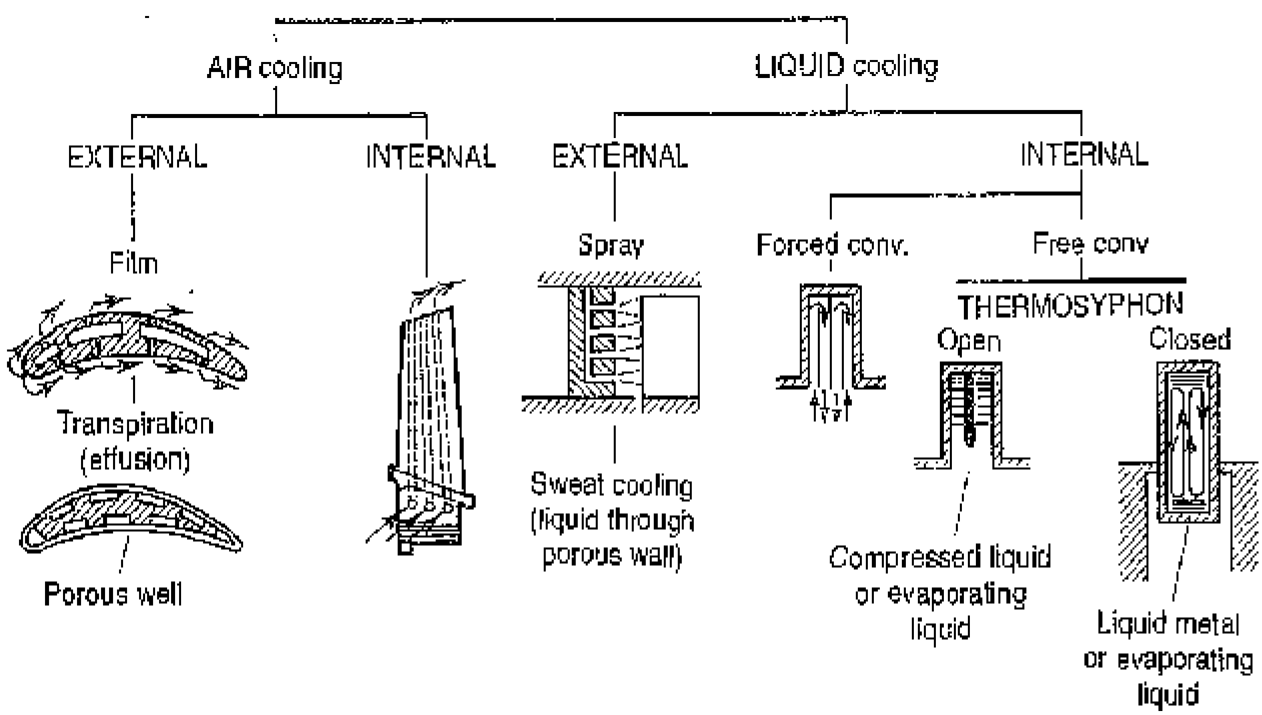


FIG. 7.29 Methods of blade cooling

Figure 7.29 illustrates the methods of blade cooling that have received serious attention and research effort. Apart from the use of spray cooling for thrust boosting in turbojet engines, the liquid systems have not proved to be practicable. There are difficulties associated with channelling the liquid to and from the blades—whether as primary coolant for forced convection or free convection open thermosyphon systems, or as secondary coolant for closed thermosyphon systems. It is impossible to eliminate corrosion or the formation of deposits in open systems, and very difficult to provide adequate secondary surface cooling area at the base of the blades for closed systems. The only method used successfully in production engines has been internal, forced convection, air cooling. With 1-5-2 per cent of the air mass flow used for cooling per blade row, the blade temperature can be reduced by between 200 and 300 °C. Using current alloys, this permits turbine inlet temperatures of more than 1650 K to be used. The blades are either cast, using cores to form the cooling passages, or forged with holes of any desired shape produced by electrochemical or laser drilling.

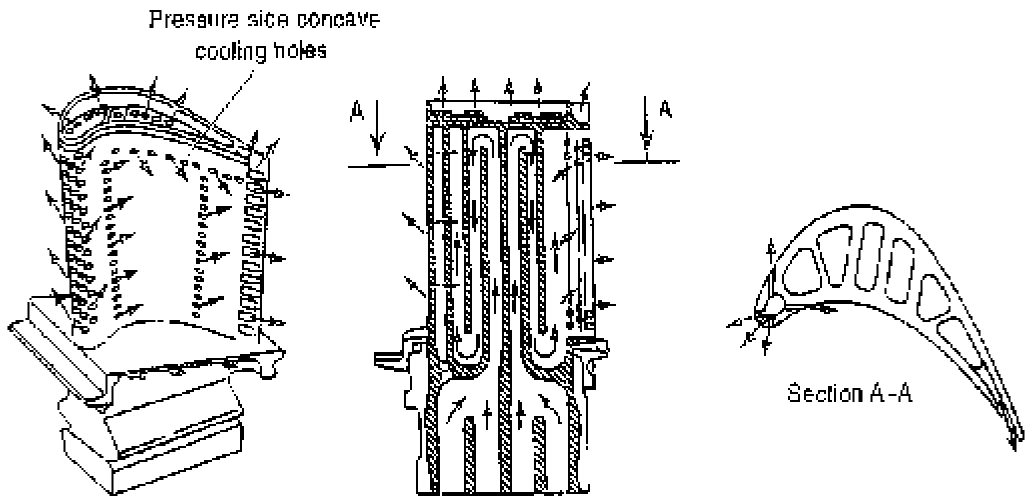
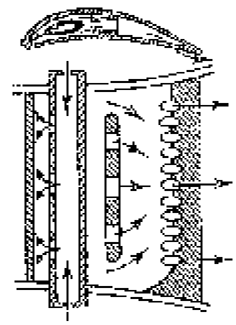


FIG. 7.30 Cooled turbine rotor blade [courtesy General Electric]

Figure 7.30 shows the type of turbine rotor blade introduced in the 1980s. The next step forward is likely to be achieved by transpiration cooling, where the cooling air is forced through a porous blade wall. This method is by far the most economical in cooling air, because not only does it remove heat from the wall more uniformly, but the effusing layer of air insulates the outer surface from the hot gas stream and so reduces the rate of heat transfer to the blade. Successful application awaits further development of suitable porous materials and techniques of blade manufacture. We are here speaking mainly of rotor blade cooling because this presents the most difficult problem. Nevertheless it should not be forgotten that, with high gas temperatures, oxidation becomes as significant a limiting factor as creep, and it is therefore equally important to cool even relatively unstressed components such as nozzle blades and annulus walls.

Figure 7.31 (a) illustrates the principal features of nozzle blade cooling. The air is introduced in such a way as to provide jet impingement cooling of the inside surface of the very hot leading edge. The spent air leaves through slots or holes in the blade surface (to provide some film cooling) or in the trailing edge.



(a)

FIG. 7.31 (a) Turbine nozzle cooling [(b) courtesy Rolls-Royce]

Blade Cooling

Blade cooling is the most effective way of maintaining high operating temperatures making use of the available material. Blade cooling may be classified based on the cooling site as external cooling and internal cooling. Another classification based on the cooling medium is liquid cooling and air cooling.

External Cooling

The external surface of the gas turbine blade is cooled by making use of compressed air from the compressor.

Other methods of external cooling are film cooling and transpiration cooling.

Internal Cooling

Internal cooling of blades is achieved by passing air or liquid through internal cooling passages from hub towards the blade tip. The cooling of the blades is achieved by conduction and convection.

Hollow blades can also be manufactured with a core and internal cooling passage. Based on the cooling medium employed, blade cooling may be classified into liquid cooling and air cooling.

Requirements for Efficient Blade Cooling

In a conventional cooled blade, cooling is obtained due to convection by passing cooling air through internal passages within the blade. The success in obtaining the large reduction in metal temperature at the expense of a small quantity of cooling flow is governed by the skill in devising and machining the cooling passages. Because the internal cooling relies on the cooling air scrubbing against the cooling surface, the internal surface area must be large and the velocity of the cooling air must be high. This implies that the cross-sectional flow area of the passage must be small.

The design of the blade internal geometry for cooling is more complex because of the various aerodynamic, heat transfer, stress and mechanical design criteria that must be satisfied. The most successful designs have incorporated radial passages through which cooling air passes, escaping at the tip.

The radial flow turbine

In a radial flow turbine, gas flow with a high tangential velocity is directed inwards and leaves the rotor with as small a whirl velocity as practicable near the axis of rotation. The result is that the turbine looks very similar to the centrifugal compressor, but with a ring of nozzle vanes replacing the diffuser vanes as in Fig. 7.37. Also, as shown there would normally be a diffuser at the outlet to reduce the exhaust velocity to a negligible value.

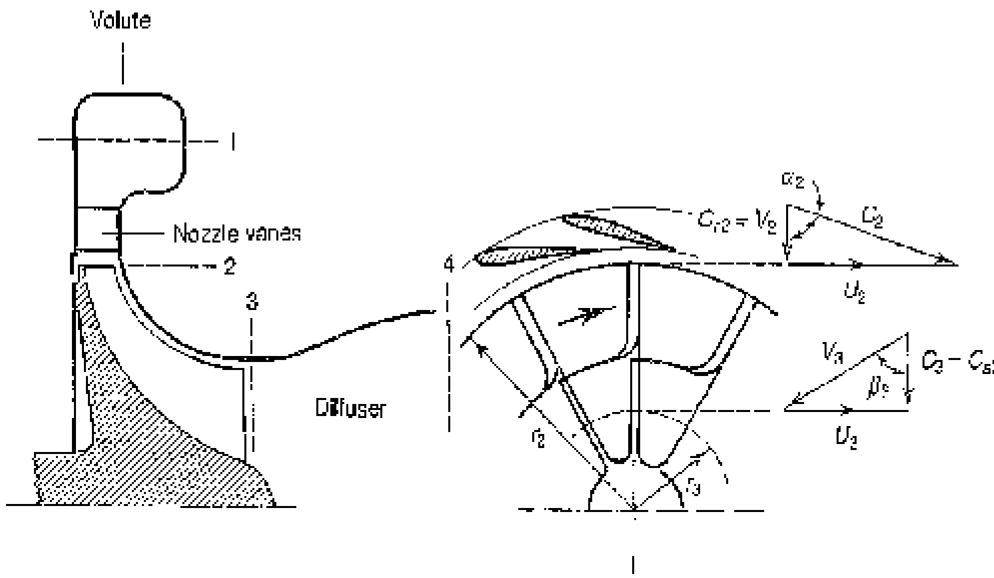


FIG. 7.37 Radial inflow turbine

The velocity triangles are drawn for the normal design condition in which the relative velocity at the rotor tip is radial (i.e. the incidence is zero) and the absolute velocity at exit is axial. Because C_{w3} is zero, the specific work output W becomes simply

$$W = c_p(T_{01} - T_{03}) = C_{w2}U_2 = U_2^2 \tag{7.46}$$

In the ideal isentropic turbine with perfect diffuser the specific work output would be

$$W' = c_p(T_{01} - T_4') = C_0^2/2$$

Turbine and Compressor Matching

The problem of matching turbine and compressor performance has great importance for jet engines, which must operate under conditions involving large variations in thrust, inlet pressure, and temperature, and flight Mach number. Matching the components of turbofan and turboprop engines involves similar considerations and procedures.

Essentially the matching problem is simple, though the computation can be length. The steady-state engine performance at each speed is determined by two conditions: continuity of flow and a power balance. The turbine mass flow must be the sum of the compressor mass flow and the fuel flow, minus compressor bleed flow. Also the power output of the turbine must be equal to that demanded by the compressor.

In principle, the matching computations could proceed as follows:

1. Select operating speed
2. Assume turbine inlet temperature
3. Assume compressor pressure ratio
4. Calculate compressor work per unit mass
5. Calculate turbine pressure ratio required to produce this work
6. Check to see if compressor mass flow plus fuel flow equals turbine mass flow; if not, assume a new value of compressor pressure ratio and repeat steps 4, 5, and 6 until continuity is satisfied.
7. Now calculate the pressure ratio across the jet nozzle from the pressure ratios across the diffuser, compressor, combustor, and turbine.
8. Calculate the area of jet nozzle outlet necessary to pass the turbine mass flow calculated in step 6 with pressure ratio calculated in step 7 and the stagnation temperature calculated. If the calculated area does not equal the actual exit area, assume a new value of turbine inlet temperature (step-2) and repeat the entire procedure.

The designer will try to match turbine and compressor so that the compressor is operating near its peak efficiency through the entire range of operation, as shown in the below figure, where the operating line (i.e., the locus of stead-state matching condition) runs through the centers of the islands defined by the constant-efficiency lines.

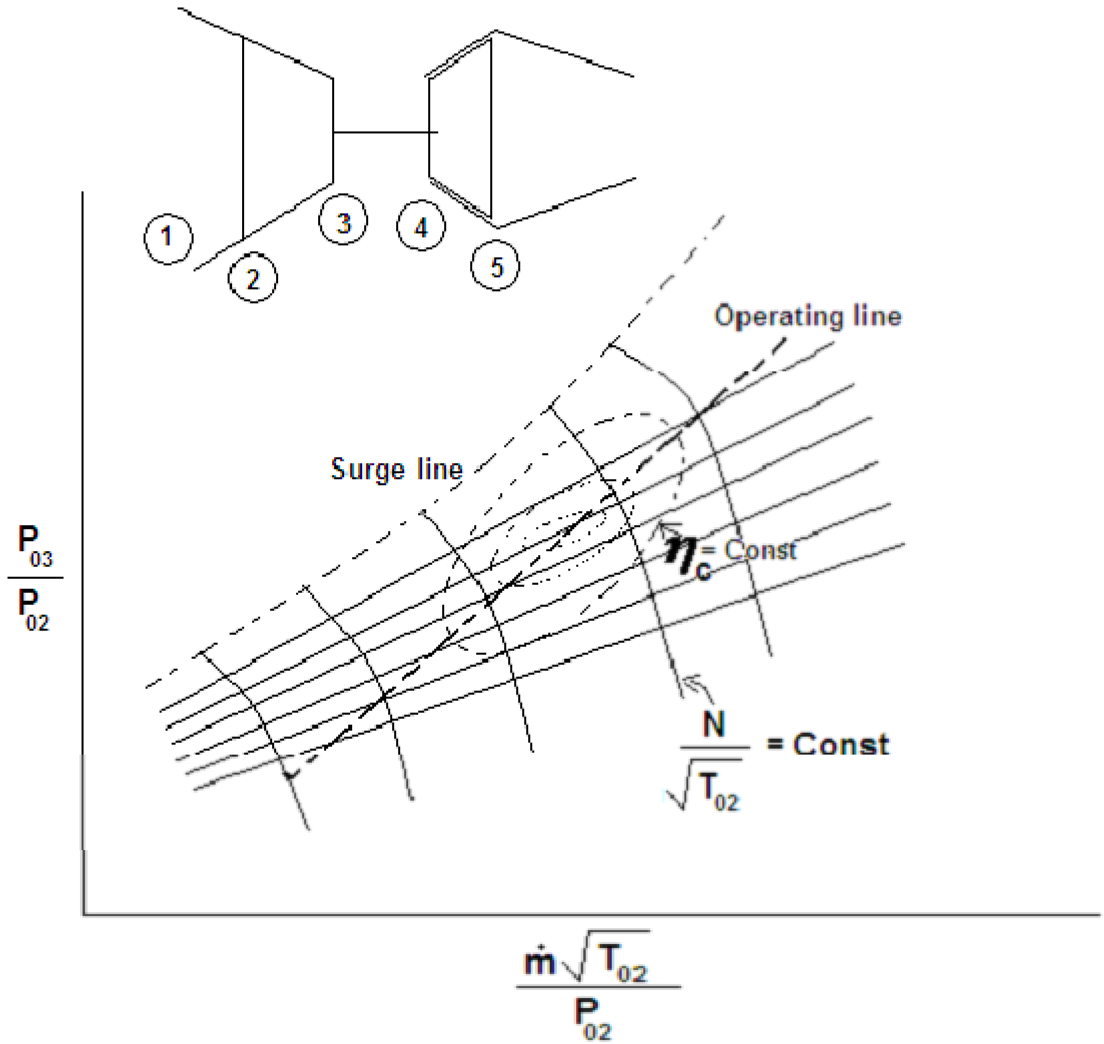


Fig. Operating line on a compressor map

UNIT – II

COMPRESSORS

Introduction

1. Two types of compressor, the centrifugal flow and the axial flow compressors are used in gas turbine engines to compress the ingested air prior to it being fed into the combustion system. Both centrifugal and axial compressors are continuous flow machines which function by imparting kinetic energy to the air by means of a rotor, subsequently diffusing the velocity into static pressure rise. In the centrifugal compressor the airflow is radial, with the flow of air from the centre of the compressor outwards. This type of compressor was used extensively in the early days of gas turbines, the technology being based upon piston engine superchargers. In the axial compressor the flow of air is maintained parallel to the compressor shaft. Either type, or a combination of both, may be used in gas turbines and each has its advantages and disadvantages. Axial / centrifugal compressor combinations are used extensively in turboshaft and turboprop engines, while axial flow compressors are used in turbofan and turbojet engines. Centrifugal compressors are limited to the small gas turbine 'gas generators' for engine air starters and missile engines. Centrifugal compressors generally need to operate at much higher rpm than axial compressors.

2. Compressor design is a balance of the aerodynamic considerations. Some of the principle factors affecting the performance being the aerofoil sections, pitch angles and the length / chord ratios of the blades. Another important detail is the clearance between the blade tips and the compressor annulus.

Requirements of a Compressor

3. The efficiency of a compressor is one of the factors directly influencing the specific fuel consumption (SFC) achieved by the engine. For maximum efficiency to be realized a compressor must satisfy a number of requirements. These are:

(a) High Mass Flow. Jet engine air mass flows are becoming much larger and, apart from any ram-compression contributed by the intake, these must be matched by the swallowing capacity of the compressor. Thus, for a large subsonic transport type the required air mass flow at altitude requires the use of high by-pass turbofan engines.

(b) High Pressure Ratio. The thermal efficiency and the work output of the constant pressure cycle are both proportional to the compressor pressure ratio. In this respect the centrifugal compressor has a maximum pressure ratio of about 4.5:1 for a single stage. This pressure ratio can be raised to approximately 6:1 by using a two stage, single-entry centrifugal compressor. The upper limit for axial compressors is more a matter of stability and complexity, with current values of approximately 10:1 for single-spool compressors, and in excess of 25:1 for multi-spool compressors. Although higher pressure ratios give higher engine efficiency due to an improved SFC, as shown in Fig 3-1, a balance must be struck between efficiency and the power needed from the turbine to drive the compressor. Sufficient power must remain to propel the aircraft, and the turbine has a finite limit to the power which it can generate.

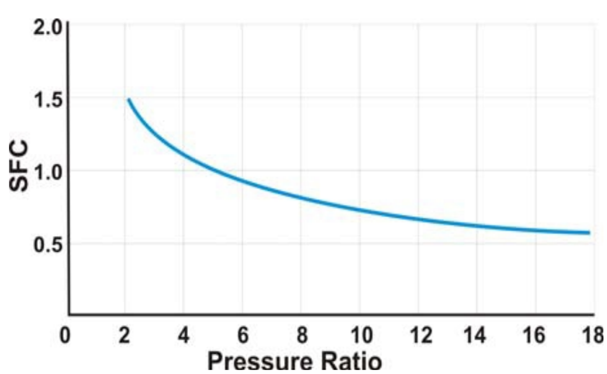


Fig 3-1: SFC and Pressure Ratio

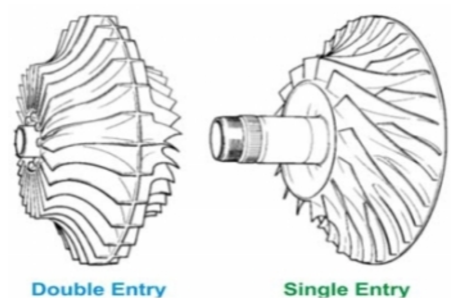
(c) **Stable Operation Under All Conditions.** Both centrifugal and axial compressors are liable to exhibit unstable characteristics under certain operating conditions. The centrifugal type is less likely to stall and surge than the axial but it is not capable of the high pressure ratios now required. In high pressure ratio axial compressors anti-stall/surge devices are often a design requirement to guard against unstable conditions. These devices are discussed more fully in para 23 et seq.

4. Compressor design in most engines is a compromise between high performance over a narrow band of rpm, and moderate performance over a wide band of rpm. Consequently, although it is possible for the compressor to be designed so that very high efficiency is obtained at the highest power, any deviation from the design conditions may cause significant changes in the aerodynamic flow leading to a loss of efficiency and unstable conditions within the engine. Because flow varies with operating conditions, it is usual to compromise by designing for a lower efficiency giving greater flexibility, thus optimising performance over a wider range of rpm.

CENTRIFUGAL COMPRESSORS

Introduction

5. The rotating part of a centrifugal compressor, known as an impeller, can be either single or double-sided (Fig 3-2). Although normally used singly to give a single compression stage, two impellers can be linked together in a two stage, single-sided arrangement. The single stage compressor unit consists of three main components: the compressor casing, which



embodies the fixed air inlet guide vanes and outlet ports, the impeller and the diffuser (Fig 3-3). The main features of the single stage centrifugal compressor are:

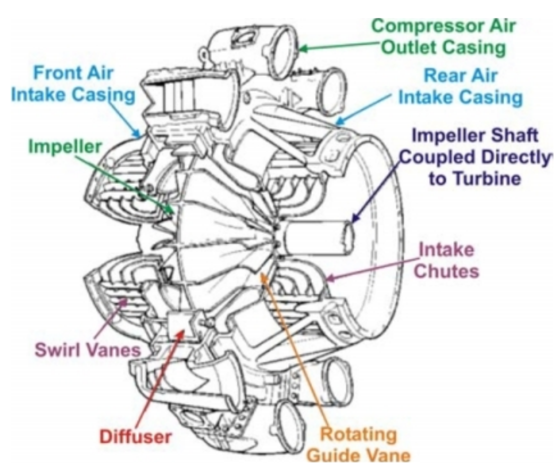
(a) For a given useful capacity and pressure ratio it can be made comparatively small in size and weight.

(b) Reasonable efficiency can be maintained over a substantial range of operating conditions.

(c) Very robust.

(d) Simple and cheap to manufacture.

(e) Tolerant to foreign object damage (FOD).



Principles of Operation

6. The impeller is rotated at high speed by the turbine, and air entering the intake at atmospheric temperature and pressure passes through the fixed intake guide vanes, which direct the air smoothly into the centre of the impeller. The impeller is designed to admit the air without excessive velocity. The air is then picked up by the rotating guide vanes of the impeller, and centrifugal force causes the air to flow outwards along the vanes to the impeller tip. The air leaves the impeller tip approximately at right angles to its intake direction, and at an increased velocity. On leaving the impeller vane passages, the air acquires a tangential velocity which represents about half the total energy acquired during its passage through the impeller. The air then passes through the diffuser where the passages form divergent nozzles which convert most of the velocity energy into pressure energy. Work is done by the compressor in compressing the air and since the process involves adiabatic heating (no heat transfer), a rise in air temperature results.

7. It can be seen that the air mass flow and the pressure rise through the compressor depend on the rotational speed of the impeller and its diameter. Impellers operate at tip speeds of up to 500 m/s to give high tangential air velocity at the impeller tip for

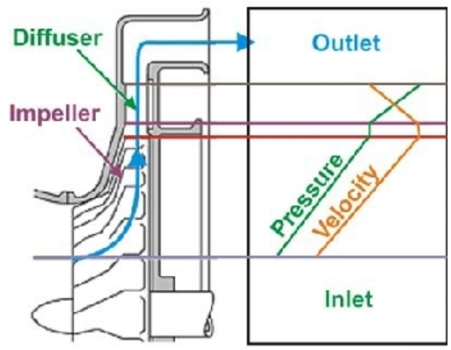


Fig 3-4: Pressure and Velocity Changes through a Centrifugal Compressor

conversion to pressure. Intake air temperature also influences the pressure rise obtained in the compressor. For a given amount of work done by the impeller, a greater pressure rise is obtained from cold than from warm air. Fig 3-4 shows the changes in velocity and pressure through a centrifugal compressor.

8. Efficiency losses in the compressor are caused by friction, turbulence and shock, and these are proportional to the rate of airflow through the system. Consequently the actual pressure rise is lower than the ideal value of 4.5:1 and a constant pressure ratio for a given rpm, with varying inlet flow conditions, is not obtained. Therefore :

(a) The pressure obtained for a given impeller design is less than the ideal value and is dependent on the impeller rpm and variations of the mass airflow.

(b) The temperature rise depends mainly on the actual work capacity of the impeller and on frictional losses.

Another source of loss is caused by leakage of air between the impeller and its casing. This is minimized by keeping their clearances as small as possible during manufacture.

Impellers

9. Airflows through the two main types of impeller for centrifugal compressors, the single-entry and the double-entry, are shown in Fig 3-5a and b respectively. If a double-entry impeller is used, the airflow to the rear side is reversed in direction and a plenum chamber is required, which encircles the rear inlet region with an opening directed towards the oncoming airflow (Fig 3-5b). The impeller consists of a forged disc with radially disposed vanes on one or both sides forming divergent passages. At high tip speeds the velocity of the air relative to the vane at entry approaches the speed of sound, and it is essential for maximum efficiency that there is the minimum shock wave formation at entry. Therefore on most compressors the pick-up portions of the blades are curved and blended into the radial portions at the tip. There are consequently no secondary bending stresses in the vanes from the effects of rotation alone and the loads that arise from imparting angular motion to the air are negligible. The vanes may be swept back to increase the pressure rise slightly, but radial vanes are more commonly employed because they are more easily manufactured, and are stable in their action.

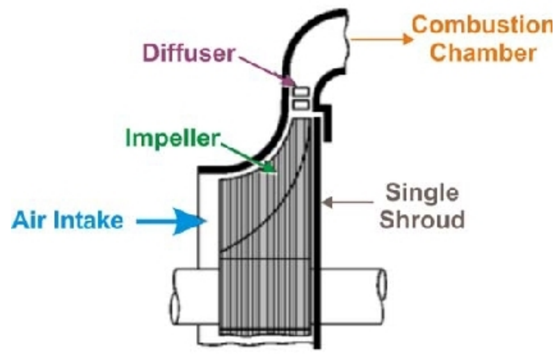


Fig 3-5a: Airflow through Single Entry Impeller

AXIAL FLOW COMPRESSORS

Introduction

13. The axial flow compressor converts kinetic energy into static pressure energy through the medium of rows of rotating blades (rotors) which impart kinetic energy to the air and alternate rows of stationary diffusing vanes (stators) which convert the kinetic energy to pressure energy.

Construction

14. The axial flow compressor consists of an annular passage through which the air passes, and across which are arranged a series of small blades of aerofoil section, alternately rotating on a central shaft assembly or fixed to the outer case. Each pair of rotor and stator rings is termed a stage, and a typical gas turbine engine may have between 10 and 15 stages on a single spool or divided between multiple spools. Each rotating ring is mounted on either a separate disc, or on an axial drum attached to the turbine drive shaft. Some of the rotating stages may be manufactured with integral blades and discs (BLISKS). BLISKS are used in turboshaft and turbofan engines. An additional row of stator blades may be fitted to single spool-engines to direct the incoming air onto the first row of rotor blades at the optimum angle. These are the inlet guide vanes (IGVs), which may be at a fixed pitch but are more usually automatically adjusted to suit prevailing intake conditions. The final set of stator blades situated in front of the combustion chamber are called the outlet guide vanes (OGVs), and these straighten the airflow into the combustion stage.

15. The cross-sectional area of the annulus is progressively reduced from the front to the rear of the compressor in order to maintain an almost constant axial velocity with increasing density. Consequently the rotors and stators vary in length according to the pressure stage, becoming progressively smaller towards the rear of the compressor. 16. As the pressure increases throughout the length of the compressor unit, each stage is working against an increasingly adverse pressure gradient. Under such conditions, it becomes more difficult to ensure that each consecutive stage operates efficiently, and this limits the flexibility of the single-spool engine (Fig 3-7). A more flexible system is achieved by dividing the compressor into separate pressure sections operating independently and driven on coaxial shafts by separate turbines. Such arrangements are termed multi-spool compressors and the construction and layout are shown in Figs 3-8 and 3-9.

17. Multi-spool compressors may be used in both turbojets and turbofan engines. In the turbofan engine the multi-spool layout enables the low pressure compressor or fan to handle a large mass flow, a proportion of which is fed into the subsequent compressors, while the remainder is ducted to the rear of the engine. The ratio of bypass to core-flow air may vary to suit the changing conditions of the engine. Turbofan engines exhibit an improved SFC over normal turbojets.

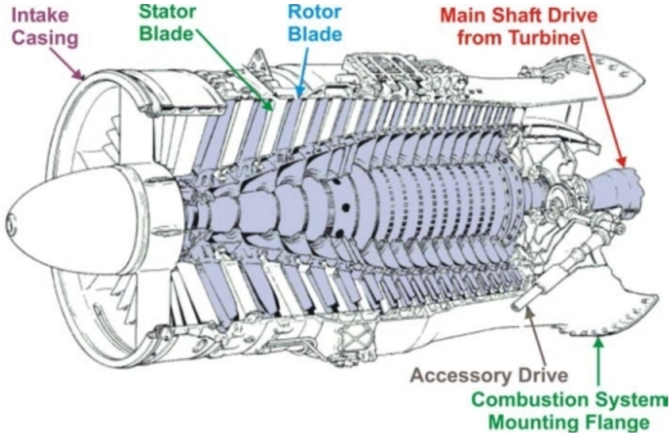


Fig 3-7: Single-Spool Compressor

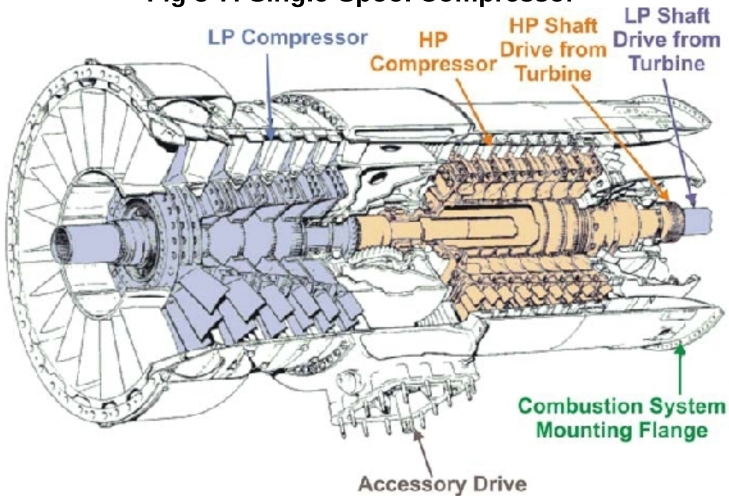


Fig 3-8: Twin-Spool Compressor

18. In the quest for improved efficiency, engines with by-pass ratios greater than existing turbofans have been designed and are currently being developed. These engines are termed prop-fans or ultra high by-pass (UHB) ratio engines.

19. The axial compressor provides a convenient supply of air at various pressures and temperatures which can be tapped off at the appropriate stages and used to provide engine intake and IGV anti-icing, cooling of high temperature components (Chapter on cooling and lubrication) and, combined with the cooling, provide a system of pressure balancing to reduce the end-loads throughout the engine. End-loads are caused by the rotor stages, consisting of numerous areofoil sections, creating a forward thrust of several kilo-newtons on the front end bearings. Similarly, the gas stream impinging on the turbine assembly imposes a rearwards load. Although the loads can be reduced considerably by careful design of the turbine arrangement, this is only effective at a given power setting. Departure from design power requires the addition of compressor bleed air to achieve adequate pressure balance.

Principles of Operation

20. Air is continuously induced into the engine intake, and is encountered by the first stage rotor or LP fan. If fitted, the IGVs direct the flow onto the first row of rotor blades. The rotor and fan blades are rotated at high speed by the turbine, and impart kinetic energy to the airflow. At the same time, the divergent passage between consecutive rotor blades diffuses the flow to give a pressure rise. The airflow is then swept rearwards through a ring of stator blades, which converts the kinetic energy of

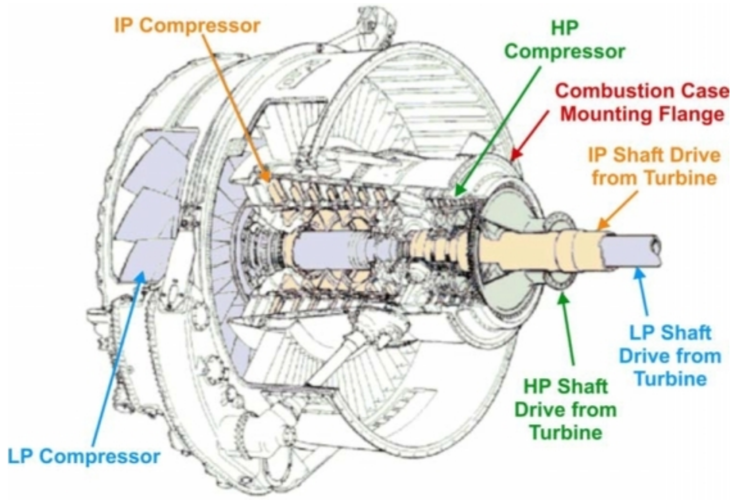


Fig 3-9: Three Spool Compressor

the stream to pressure energy by diffusing arrangements of the blades. The stator blades also direct the airflow at the correct angle onto the next stage rotor blades, where the sequence is repeated. Thus, at each compressor stage, the airflow velocity is increased by the rotor, and then converted to a pressure increase through the diffusing action of both rotor and stator. The net effect is an approximately constant mean axial velocity with a small, but smooth, pressure increase at each stage (Fig 3-10). As mentioned previously, the cross-sectional area of the compressor annulus is progressively reduced from front to rear of the compressor to maintain constant axial velocity with increasing pressure in accordance with the Equation of Continuity:

$M = AV\rho = \text{Constant}$

Where M = Air mass flow

A = Annulus (decreasing)

V = Axial velocity (constant)

ρ = Airflow density (increasing)

21. A vector analysis will help to show airflow, pressure, and velocity changes through a typical axial-flow compressor. Starting with inlet air (Fig 3-11), notice that the length of arrows (vectors) A and B are the same. This indicates that no change in velocity occurred at this point. The inlet guide vanes only deflect the air to a predetermined angle toward the direction of rotation of the rotor. At points C and D the vectors are of different lengths showing that work is being done upon the air in the form of a velocity loss and a pressure gain. The stator entrance (vector E) and the stator discharge (vector F) show another velocity loss and pressure gain exactly like that occurring through the rotor. The discharge air (D) seems to be at an incorrect angle to enter the first-stage stator, but due to the presence of rotary air motion caused by the turning compressor, the resultants E and G are produced which show the true airflow through the compressor. Notice that these vectors are exactly in line for entrance into the next stage of the compressor. One final aerodynamic point to note is that the stator entrance (vector E) is longer than stator discharge (vector F) because of the addition of energy to the air by the rotor rotation X. Thus, as each set of blades, rotors, and stators, causes a pressure rise to

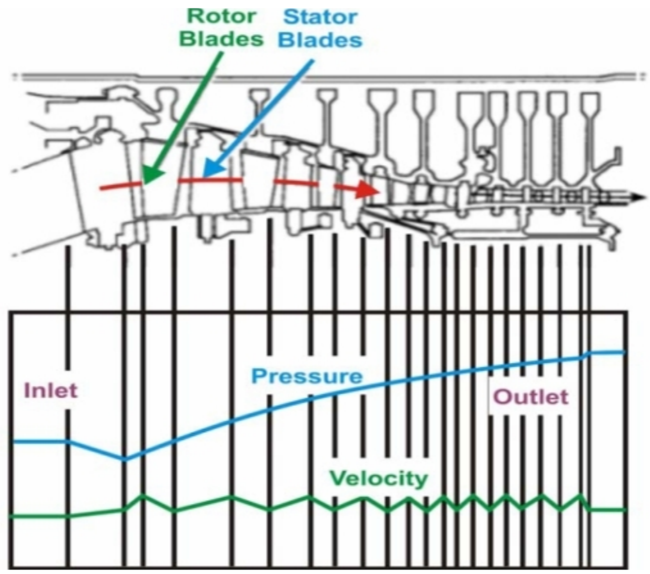


Fig 3-10: Flow through an Axial Compressor

occur at the expense of its discharge velocity the air's rotary motion restores the velocity energy at each blade's entrance for it in turn to convert to pressure energy. The pressure ratio across each stage of the compressor is in order of 1:1.1 or 1.2. This small pressure rise at each stage assists in reducing the possibility of blade stalling by reducing the rate of diffusion and blade deflection angles.

22. You will notice that both the rotor blades and stator blades are diffusing the airflow. It is much more difficult to obtain an efficient deceleration (diffusion or pressure increase) of airflow than it is to get efficient acceleration, because there is a natural tendency in a diffusion process for the air to break away from the walls of the diverging passage, reverse its direction and flow back in the direction of the pressure gradient lower pressure. It has been determined that a pressure ratio of approximately 1:1.2 is all that can be handled by a single compressor stage since higher rates of diffusion and excessive turning angles on the blades result in excessive air instability, hence low efficiency. A desired overall compression ratio of the engine is achieved by simply adding more stages onto the compressor. The amount of pressure rise or compression ratio depends on the mass of air discharged by the compressor, the restrictions to flow imposed by the parts of the engine through which the air must pass, and the operating conditions (pressures) inside the engine compared with the ambient air pressure at the compressor intake. The final pressure is the result of multiplying the pressure rise in each stage.

COMPRESSOR STALL AND SURGE

23. A compressor is designed to operate between certain critical limits of rpm, pressure ratio and mass flow, and, if operation is attempted outside these limits, the flow around the compressor blades breaks down to give violent turbulent flow. When this occurs, the compressor will stall or surge. The greater the number of stages in the compressor, the more complex the problem becomes because of the variety of interactions that are possible between stages. The phenomenon of compressor stall and surge is complex one and the following paragraphs are only intended as a brief introduction to the causes of stall and surge, and to explain how the onset of such is alleviated by careful compressor design and the incorporation of anti-stall/surge devices within the engine.

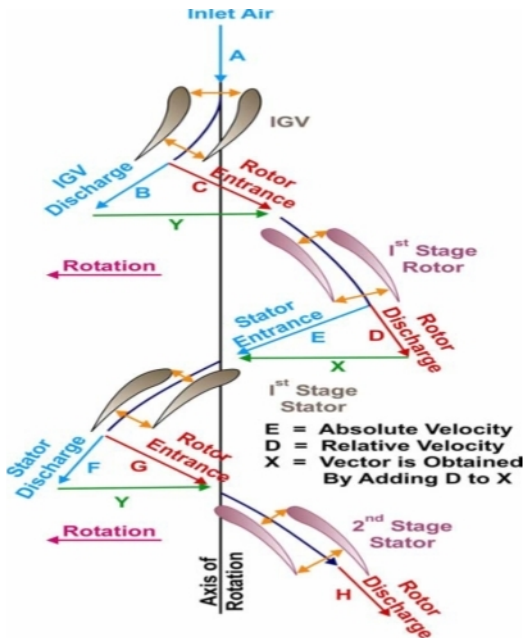


Fig 3-11: Vector Analysis of Airflow through an Axial Flow Compressor

Compressor Performance

24. The wide ranges of rpm, altitude and flight speeds over which gas turbines engine must function satisfactorily, produce problems which affect the design and limit the operation of the compressor. Fig 3-12 shows a schematic static test rig which may be used to determine the compressor characteristics.

25. In the test rig shown in Fig 3-12, the compressor is driven by a variable speed motor providing independent rpm control. The valve system on the compressor delivery allows the back pressure to be varied, thus simulating the engine demands. Fig 3-13 shows a plot of the pressure ratios (P_{t2}/P_{t1}) and the air mass flow produced at a fixed compressor rpm.

26. Closing the valve progressively from the fully open position (A) will increase the backpressure and hence reduce the mass flow and increase the pressure ratio. Operation on the negative slope of the curve is stable. Point B represents the maximum pressure ratio attainable at the given rpm. Further closing of the valve will reduce the mass flow and the pressure ratio, and the compressor enters a region of unstable operation since the valve setting is demanding a pressure ratio which cannot be met. Operation to the left at B is possible, but with continued closing of the valve the falling flow and pressure ratio become mutually conducive. Flow will momentarily cease, reverse, and, if the valve restriction is not removed, oscillate between point C and some point E. Loud hammering sounds and violent vibration identify this situation and, if the valve is not released, mechanical failure will occur. This is compressor surge. Fig 3-14 shows a set of constant rpm (N) lines with the surge points joined to give the compressor surge line.

27. Operation to the left of this line is to be avoided. Lines of constant compressor efficiency, ηc have also been plotted on Fig 3-14, and it can be seen that there is a point D where the efficiency is a

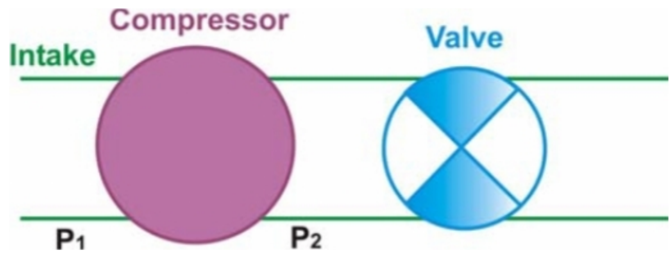


Fig 3-12: Static Test Rig for Compressors

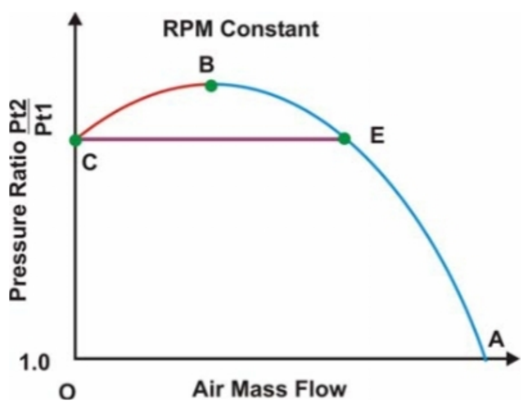


Fig 3-13: Compressor Performance at Fixed RPM

maximum. This operating point is called the Design Point and the corresponding values of rpm, air mass flow and pressure ratios are the design values. Compressor operation should be as near to this point as possible since ηc is reduced at higher or lower rpm. For off-design conditions the operating point will be determined by the engine demands and the operating line shown on the figure represents stable compressor operation for a given engine geometry. The engine demands, in terms of flow resistance, depend on the combustion chamber, turbine and nozzle conditions. If there is any change in the geometry or operating characteristics of these components a new operating line will result. In particular, if there is any increase in flow restriction, the operating line will be moved towards the surge line.

Compressor Stall and Surge

28. Compressor rotor blades, which are small aerofoils, stall in the same way as an aircraft wing by an increase in the angle of attack to the point where flow breakaway occurs on the upper surface. Since the pitch of the blades is fixed, this condition is brought about by a change in direction of the relative airflow (V1). This is shown in Fig 3-15.

29. A reduction in the axial velocity V_a to a value V_a' (Fig 3-15), while the rpm and hence blade speed U remains constant, increases the angle of attack. If the fall in V_a is sufficient the blade stalling angle will be reached. The fall in V_a at constant rpm is associated with a reduction in mass flow from the stable value on the operating line and is due to a variety of causes which will be discussed shortly. Generally, when the critical condition is approached, due to velocity gradients and local effects, a group of blades will be affected first rather than the complete blade row. Flow breakaway on the upper surfaces will reduce the available air space between the blades. Air will be deflected to adjacent blades causing an increase in angle of attack for those on one side and a decrease on the other. Thus the stall "cell" moves around the blade row, the movement being about half rotor speed.

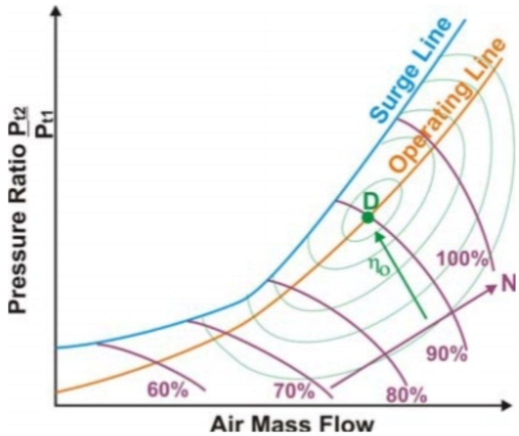


Fig 3-14: Compressor Performance Graph

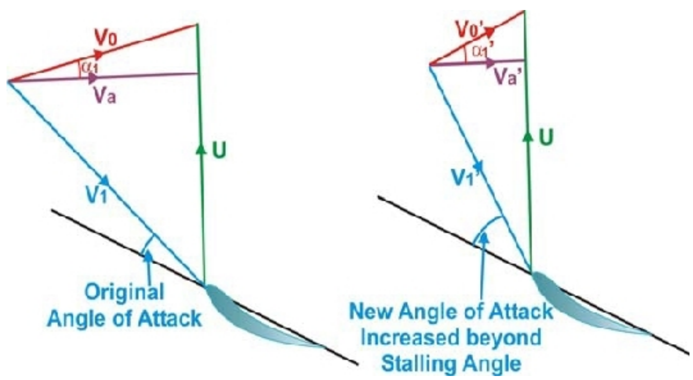


Fig 3-15: Relative Airflow on Compressor Blades

There may be several “cells” which can coalesce and eventually stall the whole row, or they may die out. The “rotating” stall is only one example of the development of the unstable operation which can result from numerous situations.

Causes of Compressor Stall and Surge

30. As already indicated, there is a departure from the stable operating line relationship between air mass flow, pressure ratio and rpm. Many conditions can cause this, but all can be discussed on the basis of Fig 3-16. The graph is plotted in terms of pressure ratio, and corrected rpm (N) and mass flow (m).

31. For high efficiency, the operating line will be close to the surge line and the distance between the two is a measure of the stall / surge margin. Reduction in air mass flow at constant N will cause the operating point to move towards the surge line. The fall in air mass flow can be caused by distorted intake conditions due to aircraft manoeuvres, ingestion of gun or rocket gases, flying through the jet stream of another aircraft etc. Increase in flow resistance due to choking of the turbine or nozzle will have a similar result and can occur on start-up due to the sudden ignition of fuel producing, momentarily, an excessive gas volume. Throttle slamming can produce the same conditions if the Acceleration Control Unit (ACU) does not adequately restrict fuel flow, as will any malfunctioning of a variable nozzle mechanism. A stall produced by down-stream flow restriction causes a drop in rpm which the ACU will sense and attempt to correct by increasing the fuel flow, thus making matters worse and probably producing engine surge. One common problem, particularly on high pressure ratio engines, is to achieve stall-free acceleration. In Fig 3-16 the bend or "knee" in the surge line is typical of high pressure ratio engines and reduces the stall margin in a region through which the operating point must pass during acceleration. Every point on the operating line represents a stable condition where the compressor supply just balances engine demand. Thus when the throttle is opened to accelerate the engine from A to the design point D, the compressor accelerates along a line ABD above the operating line. The vertical separation of the two lines is proportional to the rate of acceleration and this is

very limited at the "knee". If an acceptable rate of stall-free acceleration is not possible then special measures will be necessary. These are discussed later.

32. High altitude operation produces an increase in N for a given value of rpm thus moving the operating point upwards and reducing the stall margin. The reduction in Reynold's Number at high altitude increases compressor blade drag reducing ηc, and the thickened boundary layer reduces the effective air space, tending to increase the angle of attack. Flight at high Mach numbers produces an

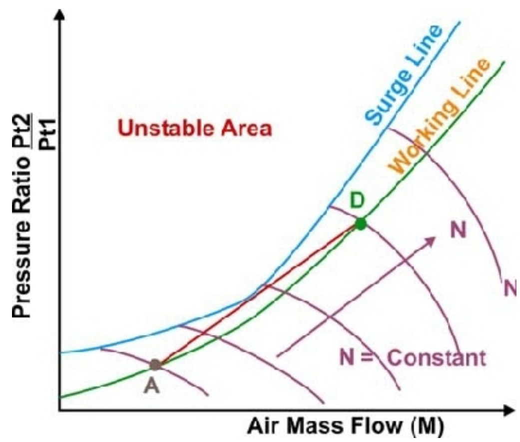


Fig 3-16: Relationship between Pressure Ratio and Air Mass Flow *FIS*

increase in compressor inlet temperature and a fall in N moving the operating point downwards towards the "knee" where the stall margin is small.

33. **<u>Compressor Stall at Low Rpm.</u>** With a reduction in mass airflow at low rpm, the angle of attack of the first low pressure stages is greater than that of the high pressure stages, so that the low pressure stages are the first to stall, the succeeding stages not necessarily being affected. Stalling of the initial compressor stages may be indicated by an audible rumbling noise and a rise in turbine gas temperature (TGT). The stall of the first stage may affect the whole compressor, or confine itself to the one stage. In the latter case, a further reduction in mass flow would cause a successive breakdown of the remaining stages.

34. **<u>Compressor Acceleration Stall.</u>** On starting, or during rapid acceleration from low rpm, the sudden increase in combustion pressure caused by additional fuel can cause a momentary back pressure which affects the compressor by reducing the mass airflow thus causing the same conditions as described above.

35. **<u>Compressor Stall at High Rpm.</u>** At high compressor rpm the angle of attack of all the stages is about the same, so that a sudden reduction in mass flow causes a simultaneous breakdown of flow through all the stages. This type of stall is usually initiated by airflow interference at the intake during certain manoeuvres or gun firing. The compressor may be unstalled by throttling fully back, but in some cases it may be necessary to stop the engine.

<u>Compressor Surge</u>

36. In an axial compressor, surging indicates a complete instability of flow through the compressor. Surging is a motion of airflow forwards and backwards through the compressor, which is accompanied by audible indications ranging from muffled rumblings to an abrupt explosion and vibration, depending on the degree of severity. A rapid rise in TGT and fluctuation or fall of rpm are the instrument indications of this condition. Compressor surge causes very severe vibrations and excessive temperatures in the engine, and should therefore be avoided or minimized.

<u>Surge Point</u>

37. The combinations of airflow and pressure ratio at which surge occurs is called the surge point and such a point can be derived for each combination of mass airflow and pressure for given value of rpm. If these points are then plotted on a graph of pressure ratio against airflow, the line joining them is known as the surge line which defines the minimum value of stable airflow that can be obtained at various rpm (Fig 3-16). The safety margin shown is designed into the engine. In a good axial

compressor the operating line is as near to the surge line as possible to maximize the efficiency for each value of rpm, whilst being far enough away to give a reasonable safety margin for control of the air mass flow.

Avoidance of Compressor Stall and Surge

38. In the case of a high pressure ratio engine with an inadequate stable acceleration capability there is a need for stall/surge protection. Anti-stall/surge devices can be added retrospectively, but are normally incorporated in the original design.

39 **Variable Inlet Guide Vanes and/or Stators.** To suit off-design operation such as start-up and acceleration from idling, variable angle guide vanes and sometimes variable stator blades are fitted. The function of these is to match the air angles to the rotor speed and avoid the stalling condition. The blade mechanism is actuated by rpm and outside air temperature (OAT) signals. The effect is to move the operating line further from the surge line (Fig 3-16), thus increasing the stall margin and acceleration capability. The control system is set to move the blades in response to engine speed to avoid low rpm and acceleration difficulties. From earlier comments on the need to operate near the surge line for high efficiency and pressure ratio, it will be evident that there is a loss of blade efficiency when the angles are not the design values.

40. **Blow-off or Bleed Valves.** Since the air mass flow, and hence the axial velocity, at the front of the compressor depends on the flow resistance, relief of the resistance will prevent high angles of attack during off-design operation. Blow-off or bleed valves at an intermediate stage are activated by an rpm or OAT signal to relieve the back-pressure. The effect will be to reduce the angle of attack at the front and relieve the choking tendency at the back. Again, the effect is to move the operating line away from the surge line (Fig 3-16). When the valves are in operation there is not only a fall in compressor efficiency but also a spill of airflow, which means an increase in SFC.

41. **Multi-Spool Engines.** The difficulty of matching the compressor rpm to the off-design flow conditions in high pressure ratio engines is relieved by rotating the front low pressure, intermediate, and high pressure sections at different speeds. Multi-spool design enables the front stages to run at a lower rpm more suited to the low pressure air angles, and the higher pressure sections to run at higher rpms to avoid choking.

42. **Variable Area Nozzle.** Engines having an afterburner, are fitted with variable area nozzles. Whilst afterburning is in operation, the nozzle control system varies the nozzle area to maintain a constant pressure drop across the turbines for a given engine rpm. This avoids undue backpressure being felt by the compressor section and subsequent surge occurring. On some engines, a limited nozzle variation is allowed in the non-afterburning range to increase dry nozzle area for taxiing and reduced nozzle area for emergency operation.

COMPARISON OF AXIAL FLOW AND CENTRIFUGAL FLOW ENGINES

Factors

43. **Power.** For a given temperature of air entering the turbine, power output is a function of the quantity of air handled. The axial flow engine can handle a greater mass of air per unit frontal area than the centrifugal type.

44. **Weight.** Most axial flow engines have a better power/weight ratio, achieving a given thrust for a slightly lower structural weight.

45. **Efficiency.** The centrifugal compressor may reach an efficiency of 75 to 90% up to pressure ratios of 4.5:1. Above this ratio efficiency falls rapidly. The axial flow compressor may have an efficiency of 80 to 90% over a wide range of compression ratios and is more economical in terms of fuel used per kN of thrust per hour.

46. **Design.** The power of the centrifugal compressor engine can be increased by enlarging the diameter of the impeller, thus increasing the rotor stresses for a given rpm, and increasing the rotational speed of the rotor up to a maximum of 500 m/s. This increases the diameter and frontal area of the fuselage or nacelle. The power of the axial flow engine on the other hand can be increased by using more stages in the compressor without a marked increase in diameter. The smaller frontal area of the axial flow engine leads to low drag which is an important fact in engines designed for high speed aircraft.

47. **Construction and Durability.** The centrifugal compressor is easier and cheaper to manufacture, and has better FOD resistance than the axial compressor.

Materials

48. Compressors rotate at high rpm, and the materials chosen must be capable of withstanding considerable stresses, both centrifugal and aerodynamic. The aerodynamic stresses arise mainly from the buffeting imparted by the pulsating pressure concentration between the impeller tip and the leading edge of the diffuser.

49. The centrifugal impeller is cast and drop stamped in aluminium alloy, which is then milled to the required shape, heat treated, and polished to resist cracking and aid crack detection. Production methods using powder metallurgy techniques and ceramic based materials are also available. The rotating intake guide vanes at the eye of the impeller are sometimes

edged with steel to resist against erosion and FOD. The diffuser for centrifugal compressors is usually cast in aluminium or magnesium alloy. Because of the limited pressure ratio of the centrifugal compressor, the temperature rise across the impeller and through the diffuser is within the 200°C limit for aluminium alloy.

50. The stator and rotor blades of the axial compressor are made from variety of materials depending on the pressure, temperature and centrifugal force encountered at the various stages. Aluminium alloy can be used for the low pressure stages, although titanium is often used for the first stage of the LP compressor, because of its superior strength and FOD resistance. Steel, titanium, carbon fibre composites and advanced ceramics may be used on the higher pressure stages where the temperature due to compression exceeds 200°C. Indeed, ceramic blades have been tested successfully to 1300°C. Modern blades are usually manufactured hollow with, or without, a honeycomb core (Fig 3-17). One method of manufacture uses rolled titanium side panels assembled in dies, hot twisted in a furnace and hot pressure formed to achieve the precise required configuration. The centre is milled to accommodate the honeycomb. Both panels and the honeycomb are finally joined using automated furnaces for diffusion bonding. In another method, the two machined and contoured halves of the blades are diffusion bonded under high heat and pressure. The resulting homogeneous piece of defect-free material is then given its aerodynamic shape by superplastic forming. In a vacuum furnace, the flat, bonded piece is placed over a special fixture which is shaped like the finished blade. The blade is heated to a superplastic state and then, by the force of gravity, settles on the curved fixture. This process gives the blade about 90% of its twist. The final shape is created in a heat die where argon gas pressure is applied to the blade. The drums or discs which support the rotor blades are often made from steel forgoings. However, powder metallurgy is sometimes used. As with the centrifugal compressor, the compressor casing for axial compressors is usually manufactured from aluminium or magnesium alloys.

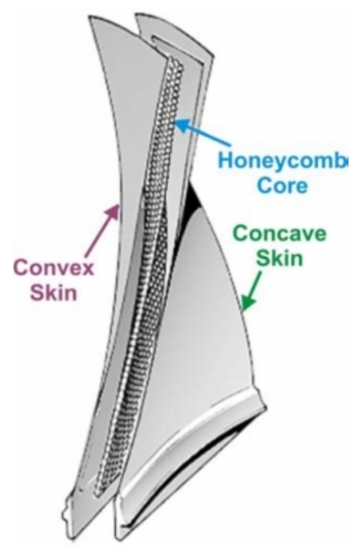


Fig 3-17: Axial Compressor Blade Construction

UNIT III

RAMJET

Basic Principle

Ramjet is simply a duct of a special shape, which faces the airflow caused by the forward motion and relies on the ram effect to collect the air, add heat to it by combusting suitable fuel and then exit through a nozzle at higher velocity and mass to create ever increasing thrust. There are no

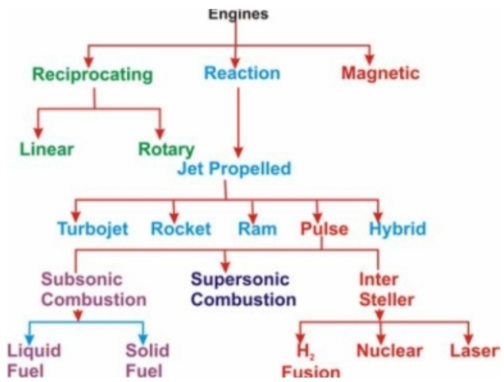


Fig 13-5: Engines *Rockets,*

moving parts, no need for lubrication, and no energy losses in trying to run something. To put it simply, it is an 'Aerodynamic engine' or to make it sound more complicated we can call it Athodyd short for 'Aero- thermo-dynamic duct'. In Fig 13-6 a type of ramjet engine is shown in which the injectors spray the mist of fuel into the ram compressed air stream and a spark ignites the mixture. The grill-type flame holder provides a type of barrier to the burning mixture while allowing, expanding hot gases to escape through the exhaust nozzle. The high-pressure air coming into the combustion chamber keeps the burning mixture from effectively reacting toward the intake end of the engine. It is important to note that ramjets will not function until enough air is coming through the intake to create a high-pressure flow. Otherwise, the expanding gases of the burning fuel-air mixture would be expelled from both ends of the engine.

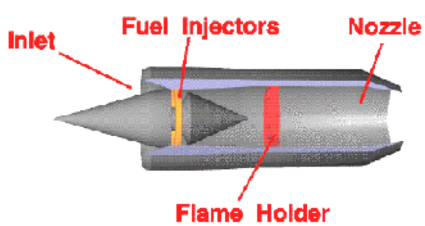


Fig 13-6: Ramjet Engine

RAM EFFECT

Ram Effect. By a suitable design of intake the additional compression and therefore pressure rise can be achieved at the air intake which is called as ram effect. The ram effect increases with the increase in forward speed. At 1.0M the external compression caused by the ram effect in the engine intake is approximately equal to that of the engine. At higher Mach No the contribution of ram effect increases markedly as compared to the turbojet engine. Thus now we can dispense off the compressor and achieve the necessary compression by pure ram effect. The above three components viz. Ram effect, compressing the air, Engine, converting the chemical energy into heat and pressure energy and finally the jet converting heat and pressure energy into forward push make what is called as ram jets.

Increase in the Compression

Fig 13-7 shows the compression ratio due to ram pressure at various Mach numbers. From around Mach 1 there is sufficient compression to operate a ram jet, but until Mach 2 is reached the operation is not really efficient. From the graph it can be seen that, at Mach 2, the compression ratio due to ram effect is about 7:1 increasing at Mach 3 to 28:1 and

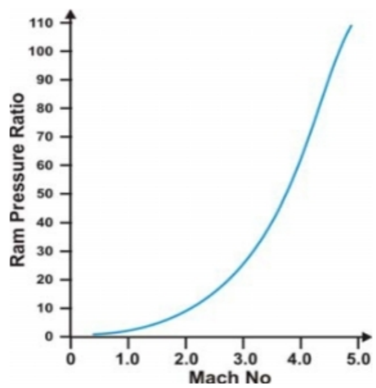


Fig 13-7: Compression Ratio Due to Ram Pressure

at Mach 4 to 70:1. (The values quoted are those attainable in practice.)

16. Because of the absence of severe centrifugal stresses in the ram jet, the ram temperature rise does not begin to become a limiting factor until about Mach 5. At speeds beyond Mach 5 however, techniques and materials are severely tested.

Increase in the Thrust

17. If we now compare the thrust obtained by the reaction engines, against the speed of the aircraft (TAS), as shown in Fig 13-8, it can be clearly seen that the trust without intake ram effect would be a straight line and will show a steady drop as the TAS increases and tends to equal the exhaust jet velocity. The ram effect however starts to increase as speed goes past 300 kts (500 kmph) and continues to increase the thrust till about 3.0M for the Turbojets. The subsequent drop in Turbojet thrust in this graph is again due to TAS approaching jet exhaust velocities (V_e) and the difference $V_e - V_0$ reduces drastically (V_0 being the TAS). This can be increased slightly by using Reheat augmentation engines, however the turbojets have to now make way for ramjets to take over from here onwards.

Thrust of a Ramjet

18. The thrust equation for a ramjet contains three terms: gross thrust, ram drag, and a pressure correction. If the free stream conditions are denoted by a "o" subscript and the exit conditions by an "e" subscript, the thrust (F) is equal to the mass flow rate (Me) times the velocity at the exit (Ve) minus the free stream mass flow rate (Mo) times the velocity (Vo) plus the pressure difference (Pe - Po) times the nozzle exit area (Ae)

F = Me Ve – Mo Vo + (Pe – Po) Ae (13.1)

The term Me Ve (exit mass flow rate times exit velocity) is often referred to as the gross thrust, since this term is largely associated with conditions in the nozzle.

The second term Mo Vo (free stream mass flow rate times free stream velocity) is called the ram drag. This is the resistance felt to the exhaust as it leaves the nozzle.

19. In the ramjet, the exit velocity is supersonic, and the exit pressure depends on the area ratio between the throat of the nozzle (minimum area) and the exit of the nozzle. Only for a unique design

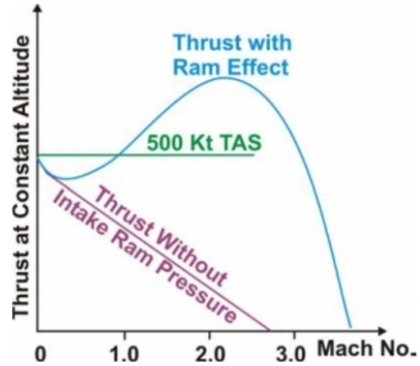


Fig 13-8: Ram Effect on Thrust of Jet Engine *Rockets*

condition will give the exit pressure equal the free stream static pressure. For all other conditions, we must include the third term of the thrust equation (Pe – Po) Ae (exit pressure minus free stream pressure times the exit area). This pressure correction is usually small compared to the first term of the thrust equation. But for completeness, this term is usually included in the gross thrust. The flow characteristics through a ram jet are as follows:

(a) **Static Pressure.** The dynamic pressure of the free stream is progressively converted to static pressure through a set of Oblique Shock Wave (critical operation) at the intake to a value close to1.0 M at the intake lip. At the intake lip the flow becomes subsonic through a weak NSW. The divergent nature of the intake further decelerates the subsonic flow and increases the static pressure. The static pressure at the end of the intake is at its maximum value and the flow now enters the combustion chamber. This process of compressing the airflow without any mechanical means is called Ram compression. During the ignition and combustion process the flow is accelerated again and the pressure drops further. At the beginning of con-divergent exhaust the pressure stabilizes in the later half of the jet pipe again and drops further at the exhaust owing to the higher velocity of the flow as compared to intake.

(b) **Velocity.** The velocity is dropped to 1.0 M through the critical operation of the intake by placing series of oblique shock waves and normal shock waves at the intake lip. The flow decelerates further through the intake to subsonic speeds and the velocity is minimum at the end of intake section. The combustion imparts high velocity to the flow again and through a con-di duct the gases leave the duct at much higher velocity than the intake, thereby producing all the necessary thrust to propel the aircraft forward.

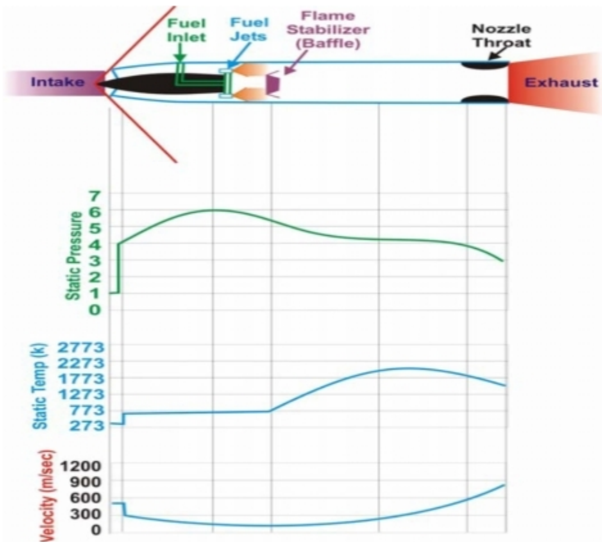


Fig 13-9: Ramjet Lay Out and Operation

(c) **Temperature.** As discussed earlier the temperature at the intake (T_1) is a function of ambient temperature (T), square of mach no. (M) and a ratio of specific heat (σ) of the medium at constant pressure and volume. σ is 1.4 for air

$$T_1 = T\ 1 + \frac{\sigma - 1}{2} M^2$$

$$\text{Therefore, } T_{air} = T\,(1 + 0.2\,M^2)$$

This is a stagnation temperature at the intake lip if the flow is suddenly stopped from high Mach number to subsonic speed and is called Ram temperature rise. However, through a system of oblique shock wave when a flow is decelerated the temperature rise becomes the function of mach no. of the flow just entering the engine.

The temperature thus suddenly rises at the lip due to presence of normal shock wave and ensuing deceleration. The subsequent rise in temperature through intake and ignition up to the pilot zone is gradual owing to the steady drop in the velocity. After combustion begins the temperature rapidly rises and achieves its maximum value towards the end of the jet pipe. Subsequent fall of the temperature is due to acceleration of the flow through the con-di exhaust nozzle. The temperature thereafter drops due to dissipation to ambient air. The ramjets, because of the ram temperature rise, are limited to a maximum speed of about 4.5M. Beyond that they now have to make way for the Scramjets or supersonic ramjets. Thus temperature rise like turbine limits on turbojets has forced us to change the engine again. We will study about it later when we look into Scramjets.

PERFORMANCE OF RAMJET

Compression Ratio. Compression ratio in the ramjets is the ratio of volume of air received at the intake to the volume of air after the intake. Diesel engines have a compression ratio of about 20:1, most car engines operate at compression ratios as high as 11:1, a pulse jet runs at a compression ratio of less than 2:1 and ramjets are about the same at their low speeds regime. Very high compression ratio is therefore required to produce thrust for better fuel efficiently. As the compression is totally dependent on ram effect the efficiency is high only at high speeds. Even though sufficient compression is available at 1.0M, till about 2.0M the required ram effect is not obtained for satisfactory operation (Compression ratio between 5 to 7). But beyond 4.5M the rise in temperature due to ram compression causes disassociation in the molecules and combustion process suffers.

Thrust vs. Altitude. The sea level thrust varies from 40kN for 2.0M to 200kN for 4.5M ramjet engine. But 4.5M ram jet thrust drops to 17kN at 60,000ft. The maximum altitude is governed by the amount of minimum pressure at which combustion can be sustained. This altitude increases with increase in mach no (for e.g., for 2M, ramjet maximum altitude is 285,000ft.)

Specific Fuel Consumption (SFC). The graph shows max thrust and min self-sustaining obtained at various M no. and corresponding SFC. Following deductions can be arrived from the graph:

(a) At max thrust condition the SFC steadily drops along with increase in mach no. till 4M and thereafter flattens out to increase at 4.5M. This is the limiting speed for ramjet.

(b) At min self-sustaining thrust the SFC progressively increases from 2.5M to 4.5M and attains the max value of SFC equal to that of max thrust condition.

(c) Thus at 4.5M the ramjet has to fly at max thrust condition to self-sustain the operation. This happens due to high temperature rise associated with increase in mach no. The hypersonic speed can be negotiated therefore by supersonic ramjets.

Ram Jet Applications

24. The ram jet has potential applications for both missiles and aircraft operating in the Earth's atmosphere. The pure ram jet is used for missile application, using a booster system, such as a solid rocket, to accelerate the ram jet to self sustaining speed. For aircraft applications the turbo-ram jet can be employed. The turbo-ram jet engine combines the turbo-jet for low speed flight, whilst the ram jet takes over for flight conditions up to about Mach 3.

25. The choice of a ram jet power unit over other available power units for certain applications may be influenced by the following considerations:

(a) A superior thrust/weight ratio and specific fuel consumption at speeds in excess of Mach 2.5, within the atmosphere.

(b) Simple and relatively cheap construction.

(c) Light weight.

(d) Simplicity of fuel systems giving a greater measure of reliability.

(e) Its thrust is controllable over a wide range.

(f) The fuel used (Kerosene) is non-corrosive and readily available.

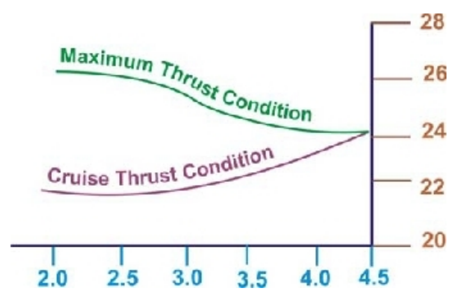


Fig 13-10: Ramjet SFC

(g) It is simple to service and maintain.

SUPERSONIC RAMJETS (SCRAMJETS)

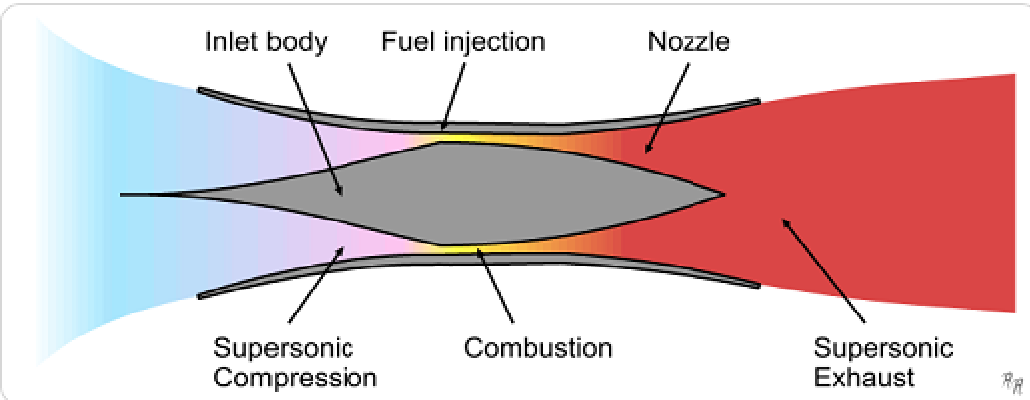
Many of the low-orbit reusable space vehicles are now considering the use of scramjets for their power plant while still in the earth's atmosphere but even scramjets remains largely the domain of the drawing board.

Scramjet is an acronym for Supersonic Combustion Ramjet. The scramjet differs from the ramjet in that combustion takes place at supersonic air velocities through the engine. It is mechanically simple (Fig 13-11), but vastly more complex aerodynamically than a jet engine. Hydrogen is normally the fuel used.

In the hypersonic regime (> 5 M) the ram temperature rise is sufficient to cause breakdown of O_2, N_2 and CO_2 molecules into individual atoms, the phenomenon called as dissociation. This leads to the decrease in engine efficiency.

29. To keep the temperature of Ram Air below dissociation values at hypersonic speeds the flow at the intake is not decelerated to subsonic speeds. Thus keeping the flow supersonic through the intake compromises a maximum pressure recovery. The combustion is achieved by burning the fuel at supersonic speeds The scram jet remains a viable power unit till the escape velocity is of 24M.

SCRAM JET ENGINES



- A scram jet engine is an engine that is much lighter than a conventional jet engine, can propel an object at speeds of over 5000 miles per hour and has no moving parts.
- If you could get it to work, the trip from London to Sydney would only take two hours!
- This technology would also be very useful to launch small satellites.
- The engine runs on oxygen, which it gets from the atmosphere, and a small amount of hydrogen.
- The engine would save a fantastic amount on the cost of fuel.
- This technology has been around since the 1950s but the problem is the motor will only become efficient at five times the speed of sound or Mach 5.
- Because of this the plane would need two engines, an engine capable of getting it to Mach 5 and a Scram Jet.

- A ramjet has no moving parts and achieves compression of intake air by the forward speed of the air vehicle.
- Air entering the intake of a supersonic aircraft is slowed by aerodynamic diffusion created by the inlet and diffuser to velocities comparable to those in a turbojet augmenter.
- The expansion of hot gases after fuel injection and combustion accelerates the exhaust air to a velocity higher than that at the inlet and creates positive push.
- Scramjet is an acronym for Supersonic Combustion Ramjet.
- The scramjet differs from the ramjet is that combustion takes place at supersonic air velocities through the engine.
- It is mechanically simple, but vastly more complex aerodynamically than a jet engine.
- Hydrogen is normally the fuel used.

- It is mechanically simple, but vastly more complex aerodynamically than a jet engine.
- Hydrogen is normally the fuel used.
- A **scramjet** (**s**upersonic **c**ombustion **ramjet**) is a variation of a ramjet with the key difference being that the flow in the combustor is supersonic.
- At higher speeds it is necessary to combust supersonically to maximize the efficiency of the combustion process.
- Projections for the top speed of a scramjet engine (without additional oxidizer input) vary between Mach 12 and Mach 24 (orbital velocity), but the X-30 research gave Mach 17 due to combustion rate issues.
- By way of contrast, the fastest conventional air-breathing, manned vehicles, such as the U.S. Air Force SR-71, achieve slightly more than Mach 3.2 and rockets achieved Mach 30+ during Apollo.
- **Like a ramjet, a scramjet essentially consists of a constricted tube through which inlet air is compressed by the high speed of the vehicle, fuel is combusted, and then the exhaust jet leaves at higher speed than the inlet air.**
- Also like a ramjet, there are few or no moving parts. In particular there is no high speed turbine as in a turbofan or turbojet engine that can be a major point of failure.
- A scramjet requires supersonic airflow through the engine, thus, similar to a ramjet, scramjets have a minimum functional speed. This speed is uncertain due to the low number of working scramjets, relative youth of the field, and the largely classified nature of research using complete scramjet engines.
- However it is likely to be at least Mach 5 for a pure scramjet, with higher Mach numbers 7-9 more likely. Thus scramjets require acceleration to hypersonic speed via other means.
- A hybrid ramjet/scramjet would have a lower minimum functional Mach number, and some sources indicate the NASA X-43A research vehicle is a hybrid design.
- Recent tests of prototypes have used a booster rocket to obtain the necessary velocity.
- **Air breathing engines should have significantly better specific impulse while within the atmosphere than rocket engines.**
- However scramjets have weight and complexity issues that must be considered. While very short suborbital scramjets test flights have been successfully performed, perhaps significantly no flown scramjet has ever been successfully designed to survive a flight test.
- The viability of scramjet vehicles is hotly contested in aerospace and space vehicle circles, in part because many of the parameters which would eventually define the efficiency of such a vehicle remain uncertain.
- This has led to grandiose claims from both sides, which have been intensified by the large amount of funding involved in any hypersonic testing. Some notable aerospace gurus such as Henry Spencer and Jim Oberg have gone so far as calling orbital scramjets 'the hardest way to reach orbit', or even 'scramjets' due to the extreme technical challenges involved.

- Major, well funded projects, like the X-30 were cancelled before producing any working hardware.
- The scramjet is a proposed solution to both of these problems, by modifications of the ramjet design. The main change is that the blockage inside the engine is reduced, so that the air isn't slowed down as much. This means that the air is cooler, so that the fuel can burn properly. Unfortunately the higher speed of the air means that the fuel has to mix and burn in a very short time, which is difficult to achieve.
- To keep the combustion of the fuel going at the same rate, the pressure and temperature in the engine need to be kept constant. Unfortunately, the blockages which were removed from the ramjet were useful to control the air in the engine, and so the scramjet is forced to fly at a particular speed for each altitude. This is called a "constant dynamic pressure path" because the wind that the scramjet feels in its face is constant, making the scramjet fly faster at higher altitude and slower at lower altitude.
- The inside of a very simple scramjet would look like two kitchen funnels attached by their small ends. The first funnel is the intake, and the air is pushed through, becoming compressed and hot. In the small section, where the two funnels join, fuel is added, and the combustion makes the gas become even hotter and more compressed. Finally, the second funnel is a nozzle, like the nozzle of a rocket, and thrust is produced.
- Note that most artists' impressions of scramjet-powered vehicle designs depict waveriders where the underside of the vehicle forms the intake and nozzle of the engine. This means that the intake and nozzle of the engine are asymmetric and contribute directly to the lift of the aircraft. A waverider is the required form for a hypersonic lifting body
- A scramjet is a type of engine which is designed to operate at the high speeds normally associated with rocket propulsion.
- It differs from a classic rocket by using air collected from the atmosphere to burn its fuel, as opposed to an oxidizer carried with the vehicle.
- Normal jet engines and ramjet engines also use air collected from the atmosphere in this way.
- The problem is that collecting air from the atmosphere causes drag, which increases quickly as the speed increases.

Advantages and disadvantages of scramjets

Special cooling and materials

- Unlike a rocket that quickly passes mostly vertically through the atmosphere or a turbojet or ramjet that flies at much lower speeds, a hypersonic airbreathing vehicle optimally flies a "depressed trajectory", staying within the atmosphere at hypersonic speeds.
- Because scramjets have only mediocre thrust-to-weight ratios, acceleration would be limited. Therefore time in the atmosphere at hypersonic speed would be considerable, possibly 15-30 minutes.

- Similar to a reentering space vehicle, heat insulation from atmospheric friction would be a formidable task. The time in the atmosphere would be greater than that for a typical space capsule, but less than that of the space shuttle.
- Therefore studies often plan on "active cooling", where coolant circulating throughout the vehicle skin prevents it from disintegrating from the fiery atmospheric friction.
- Active cooling could require more weight and complexity. There is also safety concern since it's an active system.
- Often, however, the coolant is the fuel itself, much in the same way that modern rockets use their own fuel and oxidizer as coolant for their engines.
- Both scramjets and conventional rockets are at risk in the event of a cooling failure.

Half an engine

- The typical waverider scramjet concept involves, effectively, only half an engine. The shockwave of the vehicle itself compresses the inlet gasses, forming the first half of the engine. Likewise, only fuel (the light component) needs tankage, pumps, etc. This greatly reduces craft mass and construction effort, but the resultant engine is still very much heavier than an equivalent rocket or conventional turbojet engine of similar thrust.

Simplicity of design

- Scramjets have few to no moving parts. Most of their body consists of continuous surfaces. With simple fuel pumps, reduced total components, and the reentry system being the craft itself, scramjet development tends to be more of a materials and modelling problem than anything else.

Additional propulsion requirements

- A scramjet cannot produce efficient thrust unless boosted to high speed, around Mach 5, depending on design, although, as mentioned earlier, it could act as a ramjet at low speeds. A horizontal take-off aircraft would need conventional turbofan or rocket engines to take off, sufficiently large to move a heavy craft. Also needed would be fuel for those engines, plus all engine associated mounting structure and control systems. Turbofan engines are heavy and cannot easily exceed about Mach 2-3, so another propulsion method would be needed to reach scramjet operating speed. That could be ramjets or rockets. Those would also need their own separate fuel supply, structure, and systems. Many proposals instead call for a first stage of droppable solid rocket boosters, which greatly simplifies the design.

UNIT IV

ROCKET MOTORS

INTRODUCTION

1. The rocket is the oldest practical heat engine, dating back over seven centuries. It differs from all other types of engine in that it is entirely self-contained and can operate under water, in the atmosphere, or in space, because it does not rely on an outside source for oxygen.

2. As the weight of an oxidizer may be over six times the weight of fuel for combustion, it can be appreciated that the total weight of the propellant load is greatly in excess of that for other types of heat engine. In addition, the rocket consumes its propellant at a rate very much higher than the turbojet or ram jet. It is unlikely, therefore, that the rocket will ever take the place of the turbojet as a propulsive unit for "conventional" aircraft. It has its own range of applications - those requiring a high thrust/weight ratio for small frontal areas, flights at high supersonic speeds, and flights at extremely high altitudes or in space.

Solid Propellant Rocket Motors

3. **Motor Body.** The layout of a solid propellant rocket motor is shown in Fig 13-1. The propellant is housed in the combustion chamber which must be as light as possible but capable of withstanding very high temperatures and pressures. Thin walled tubes of steel or reinforced plastics,sometimes coated internally with a refractory material, are commonly used. The nozzle is attached directly to the combustion chamber and there are no pumps, valves, or other moving parts to complicate an essentially simple maintenance requirement.

4. **Propellant Charge.** Fuel and an oxidizer are combined to form the solid propellant, which is contained in the combustion chamber. It is physically and chemically stable at ambient temperatures, but burns smoothly when ignited, giving off hot gases continuously without depending on an atmosphere.

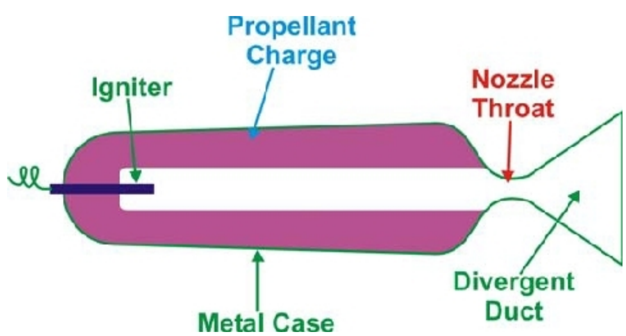


Fig 13-1: Typical Solid Propellant Rocket *Rockets,*

5. **Burning Rate and Grain Geometry.** Burning rate is the rate at which propellant is consumed, and is dependent on the chemistry of the propellant and the chamber pressure. The higher the chamber pressure, the faster the propellant burns. However, at any instant, rocket thrust is proportional to the burning surface area of the propellant. Thus the thrust can be pre-arranged to a large extent by a suitable choice of the charge shape. In most cases, the ideal is to obtain a constant thrust throughout the burning period, and when burning takes place over a constant area of charge, the configuration of the charge is said to be neutral. When an increase in burning area occurs, the configuration is termed progressive. Similarly, a decrease in burning area is termed regressive. Two basic configurations are end burning and radial burning:

(a) **End-burning or "Cigarette".** In the end-burning grain (Fig 13-2), burning is initiated at one end, and the surface area of burning remains constant until all the propellant is consumed. Thus a "cigarette" configuration is perfectly neutral. Because the burning area is small, thrust levels are low, although long burning times are possible. Since burning propellant is always in contact with the casing it can cause rapid over-heating, and the centre of gravity of the motor shifts during burning. Because of these problems, end burning grains are rarely used nowadays.

(b) **Radial or Internal Burning.** With a radial grain (Fig 13-3), burning is initiated along the whole surface of an internal hollow port. Because the burning area is large, high thrusts

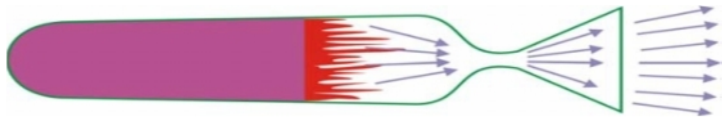


Fig 13-2: End-Burning or Cigarette

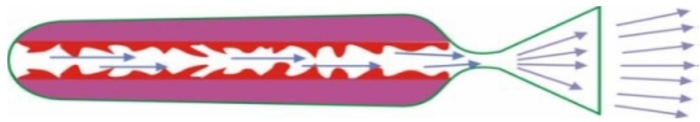


Fig 13-3: Radial or Internal Burning

can be obtained with short burning times. Case overheating does not occur because the case is insulated by the propellant itself, and the centre of gravity remains static throughout burning. A simple radial burning grain is strongly progressive, i.e. the burning area and therefore the thrust increases with time.

6. Propellant Characteristics. The desirable characteristics of a solid propellant are:

(a) High density (more propellant for a given volume).
(b) High release rate of chemical energy.
(c) Ease of handling.
(d) No deterioration in storage.
(e) High physical strength.
(f) Predictable burning rate.
(g) Safe - the propellant must be insensitive to impact and not subject to accidental ignition.
(h) Easily produced from cheap and available raw materials.
(j) Exhaust products should be smokeless, non-luminous, non-toxic, and undetectable.

Liquid Propellant Rocket Motors

7. Whilst the solid rocket is in general use for most kinds of tactical and strategic military missile, the liquid propellant rocket (Fig 13-4) has many advantages. The most important of these are the ability to stop, restart and throttle the rocket, higher energy levels, and the ability to cool the nozzle simply and easily. Consequently, the liquid propellant rocket is in general use as a space propulsion motor, and it is attractive for some military applications, providing that the storage and handling problems of the liquid propellants can be overcome.

8. Liquid Propellants. The constituents of liquid propellants are similar to solid propellants in that they contain a fuel and an oxidizer. In addition they may also contain a catalyst to aid chemical reaction, and an inert additive, such as water, for cooling purposes. They usually differ from solid propellants in that these constituents are commonly carried in separate tanks and are only brought together in the combustion chamber for the burning process. There are however, several types of propellant available:

(a) Monopropellant. A monopropellant is a propellant containing oxidant and fuel in a single substance. It may be a mixture of several compounds such as hydrogen peroxide and alcohol, or a homogeneous propellant such as nitromethane.

(b) **Bipropellant.** A bipropellant is a propellant in which an oxidant and fuel are carried separately from each other, and only mixed in the combustion chamber. Because more tanks are involved, this is a more complicated system than the monopropellant, but the majority of liquid propelled rockets use bipropellant systems because they are more energetic.

(c) **Cryogenic Propellant.** Liquid oxygen (at $–172^0$ C) and liquid hydrogen (at $–252^0$C) offer very high performance as rocket fuels, but storage is a problem.

(d) **Storable Propellants.** Storable propellants are in a liquid state at ambient temperature, and can be kept for long periods in sealed containers. Kerosene and nitric acid are examples.

9. **Combustion Chamber.** The combustion chamber and its associated convergent-divergent nozzle form the rocket motor itself, in which the propellants are burnt and exhausted to atmosphere. Usually the thrust chamber and nozzle are cooled by pumping the liquid propellants themselves through a jacket around the chamber and nozzle before they are burnt. Some small rockets may use ablative cooling, or may not be cooled at all.

10. **Nozzle Shape.** The simplest form of nozzle is of conical shape. This is still used on small nozzles, but there is a tendency for separation to occur with over or under expansion. With the bell nozzle (Fig 13-4) losses are kept to a minimum and an almost uniform velocity profile is obtained at the exit plane, but it is more expensive to manufacture.

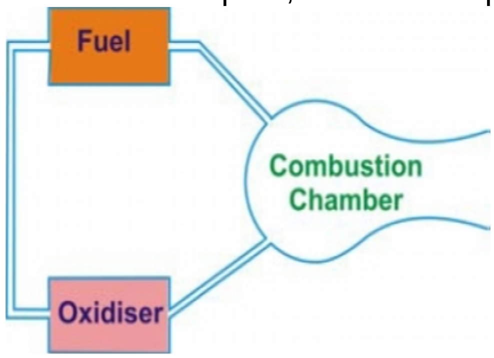


Fig 13-4: Typical Liquid Propellant Motor

Rockets – Special Features and Applications
Historical Reference

- The basic principles of all propulsive devices lie with the laws of motion due to Newton (17^{th} Century AD). These laws are phenomenological and therefore one can expect that even before Newton there may have existed many devices working on the principles of reaction.
- Rockets working directly on the principle of reaction are perhaps the simplest of the propulsive engines.
- The reciprocating engines and gas turbine engines are relatively more complex.
- The Chinese are credited with the invention of rockets probably in 12-14^{th} century AD.
- Indians used the rockets as effective weapons in late 18^{th} century against British and in 19^{th} century, the rockets became a part of the warfare in Europe. But it was only in the early part of the present century that man has recognized the full potential of rocket owing to the interests in space travel/satellite technology and like.
- Tsiolkovsky (USSR 1903) Goddard (USA, 1912) and Oberth (1921) are the pioneers of modern rocketry.
- The liquid propellant rocket owe their genesis to these people.
- The German V-2 rockets (25 tons, 65 sec, LOX-Alchol) and the post-second World war progress in rocketry are too familiar to all.

Some Special Features

The non-air breathing nature of rockets makes them very distinct among the propulsive devices.

(a) The reaction system does not depend on the surrounding atmosphere. There are no velocity limitations and altitude ceiling.

(b) Since it has to carry its own oxidizer required for combustion reaction, the specific propellant consumption is very high. Rockets consume approximately 15kg/kg-hr of propellant compared to about 1 kg/kg-hr of fuel by turbojet engine.

(c) High pressure operation is possible and hence the ratio of energy liberation per unit volume (and also unit weight of hardware) is very high.

(d) Main part of the rockets contains no moving element. Hence there is no constraint on internal aerodynamics and the reliability is high. This also implies quick response times, which makes them ideal control components.

With the above features, it is clear that the rockets are the most suitable power plants for

(i) High altitude and space applications where atmospheric oxygen is not available, eg. Launch vehicles and satellite control rockets.

(ii) All applications where high thrust are required for short duration: missiles, boosters, JATO etc.

Rockets in Space Applications

There are a variety of rockets when it comes to launching and satellite control. Many of these are non-chemical in nature but are restricted to extremely low thrust levels.

Sl No	Type	Order of Magnitude of Thrust (N)	F/W	Operational Time	I_{sp} (sec)	Applications
1	Solar sail (not a rocket in fact)	10^{-5}	10^{-4}	Years	∞	Satellite Altitude control
2	Electric Prop. (Electro thermal, Electro Static, Electro Magnetic)	10^{-6} - 10^{-2}	10^{-5}-10^{-3}	Years	150 - 6000	Satellite control, stabilization, orbit maneuver
3	Stored cold gas (N_2, NH_3 etc.)	10^{-2} - 10^{-1}	~ 10^{-3}	Years	50 - 100	-do-
4	Nuclear Rocket	upto 10^5	20-30	Minute to hours	800	Interplanetary and space travel
5	Chemical Rocket (Solid, Liquid and Hybrid)	upto 10^7	upto 80	Seconds to minutes@	150-450	Launch vehicles, Missiles, Control rockets, Sounding rockets, JATO etc

@ shuttle main engine operate for about 8 min at a time but over 7 hrs cumulatively.

Internal Ballistics

The parameters that govern the burning rate and mass discharge rate of rocket motors are called internal ballistic properties; they include

- r – propellant burning rate (velocity of consumption), m/sec or mm/sec or in/sec.
- K – ratio of burning surface to throat area, A_b/A_t
- σ_p – temperature sensitivity of burning rate, expressed as percent change of burning rate per degree change in propellant temperature at a particular value of chamber pressure.
- π_K - temperature sensitivity of pressure expressed as percent change of chamber pressure per degree change in propellant temperature at a particular value of K,

and the influences caused by pressure, propellant ingredients, gas velocity, or acceleration. The subsequent solid propellant rocket parameters are performance parameters; they include thrust, ideal exhaust velocity, specific impulse, propellant mass fraction, flame temperature, temperature limits and duration.

Propellant Burning Rate

The rocket motor's operation and design depend on the combustion characteristics of the propellant, its burning rate, burning surface, and grain geometry. The branch of applied science describing these is known as **internal ballistics.**

Solid propellant burns normal to its surface. The (average) burning rate, r, is defined as the regression of the burning surface per unit time. For a given propellant, the burning rate is mainly dependent on the pressure, p, and the initial temperature, T_i, of the propellant. Burning rate is also a function of propellant composition. For composite propellants it can be increased by changing the propellant characteristics:

1. Add a burning rate catalyst, often called burning rate modifier (0.1 to 3.0% of propellant) or increase percentage of existing catalyst.
2. Decrease the oxidizer particle size.
3. Increase oxidizer percentage
4. Increase the heat of combustion of the binder and/or the plasticizer
5. Imbed wires or metal staples in the propellant

The burning rate of propellant in a motor is a function of many parameters, and at any instant governs the mass flow rate $\dot{m}$ of hot gas generated and flowing from the motor (stable combustion);

$$\dot{m} = A_b \, r \, \rho_b$$

Here A_b is the burning area of the propellant grain, r the burning rate, and ρ_b the solid propellant density prior to motor start. The total mass m of effective propellant burned can be determined by integrating the above equation,

$$m = \int \dot{m} \, dt = \rho_b \int A_b \, r \, dt$$

Where A_b and r vary with time and pressure.

Burning Rate Relation with Pressure

Classical equations relating to burning rate are helpful in preliminary design, data extrapolation, and understanding the phenomena. Unless otherwise stated, burning rate is expressed for 70°F or 294 K propellant (prior to ignition) burning at a reference chamber pressure of 1000 psia or 6.895 MPa. For most production-type propellant the burning rate is approximated as a function of chamber pressure, at least for a limited range of chamber pressures, which is given as

$$r = a P^n$$

where r, the burn rate, is usually in centimeter per second and chamber pressure P is in MPa; a is an empirical constant influenced by ambient temperature. Also a is known as the temperature coefficient and it is NOT dimensionless. The burning rate exponent n, sometimes called the combustion index, is independent of the initial grain temperature and describes the influence of chamber pressure on the burning rate.

Burning Rate Relation with Temperature

Temperature affects chemical reaction rates and the initial ambient temperature of a propellant grain prior to combustion influences burning rate.

The sensitivity of burning rate to propellant temperature can be expressed in the form of temperature coefficient, the two most common being

The sensitivity of burning rate to propellant temperature can be expressed in the form of temperature coefficient, the two most common being

$$\sigma_p = \left(\frac{\partial \ln r}{\partial T}\right)_p = \frac{1}{r}\left(\frac{\partial r}{\partial T}\right)_p$$

$$\pi_K = \left(\frac{\partial \ln p}{\partial T}\right)_K = \frac{1}{P}\left(\frac{\partial P}{\partial T}\right)_K$$

with σ_p , temperature sensitivity of burning rate and π_K, temperature sensitivity of pressure.

The coefficient σ_p (typically 0.001 – 0.009 / K) for a new propellant is usually calculated from strand burner test data, and π_K (typically 0.067 – 0.278 % / °C) from small-scale or full-scale motors. Mathematically, these coefficients are the partial derivatives of the natural logarithm of the burning rate r or the chamber pressure p, respectively, with respect to propellant temperature T.

The values of π_K and σ_p depend primarily on the nature of the propellant burning rate, the composition, and the combustion mechanism of the propellant. It is possible to derive a relationship between the two temperature sensitivities, namely

$$\pi_K = \frac{1}{1-n}\sigma_p$$

This formula is usually valid when the three variables are constant over the chamber pressure and temperature range.

The temperature sensitivity σ_p can be also expressed as

$$\sigma_p = \left[\frac{\partial \ln (aP^n)}{\partial T}\right]_p = \frac{1}{a}\frac{da}{dT}$$

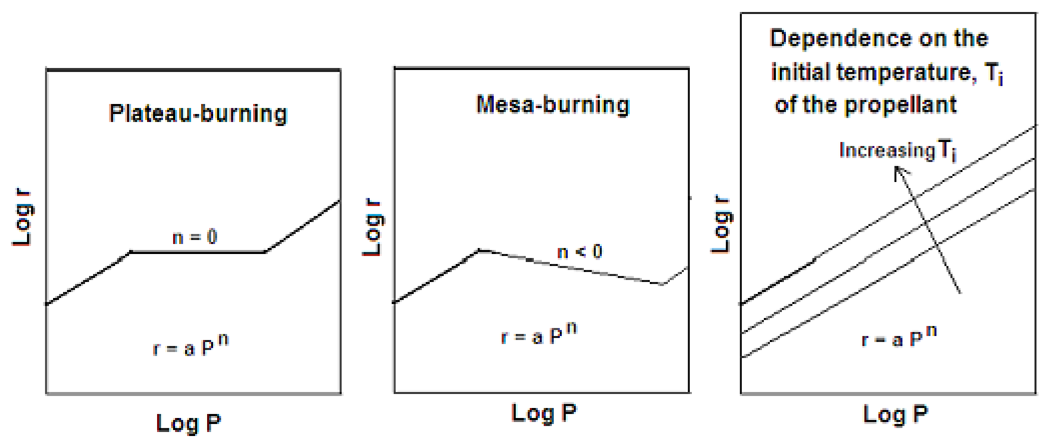


Fig. Burning rate versus pressure

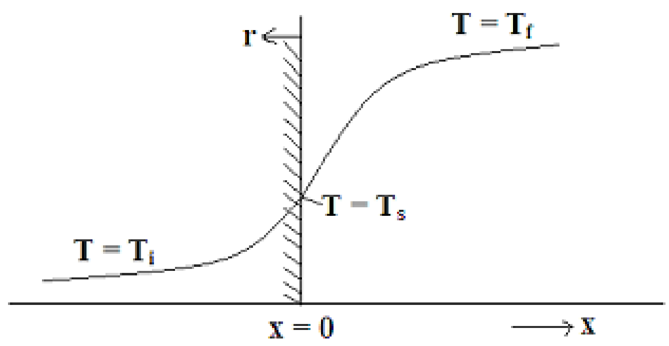


Fig. Coordinate system and temperature profile for a stationary burning solid propellant

Equilibrium chamber pressure

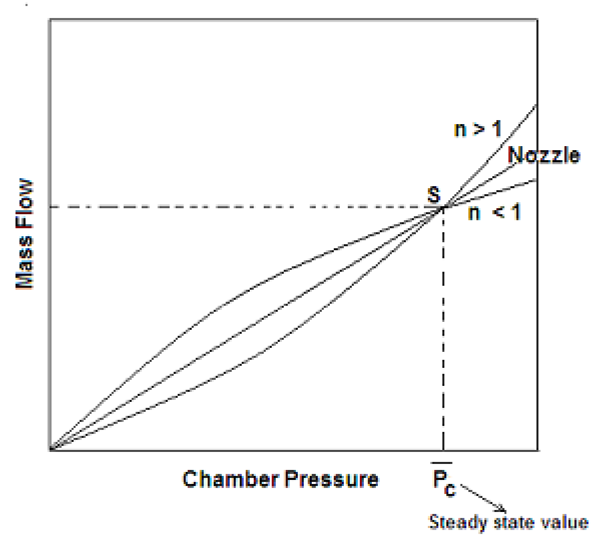


Fig. The balance of mass in a solid rocket motor with reference to the burning rate index, n

In the above figure the straight line through the origin and point 'S' depicts the mass flow through the nozzle as a function of P_c. At point S there is a balance between mass production and outflux of the mass. At higher pressures ($> \overline{P}_c$) the mass flow through the nozzle is larger than the production at the burning surface in case n < 1 and the reverse happens for n > 1. Thus if n < 1 the pressure will drop to its steady-state value $\overline{P}_c$. Note that when n < 1 even at higher chamber pressure rocket motor will back to its designed equilibrium chamber pressure and ensure a stable operation. On the other hand when n>1 these types of situations will possibly lead to over-pressurization and rupture of the rocket motor or depressurization and flame out.

Erosive Burning

Erosive burning refers to the increase in the propellant burning caused by the high-velocity flow of combustion gases over the burning propellant surface. It can seriously affect the performance of solid propellant rocket motors. It occurs primarily in the port passages or perforations of the grain as the combustion gases flow toward the nozzle; it is more likely to occur ehen the port passage cross-sectional area A is small relative to the throat area A_t with a port-to-throat area ratio of 4 or less. The high velocity near the burning surface and the turbulent mixing in the boundary layers increase the heat transfer to the solid propellant and thus increase the burning rate.

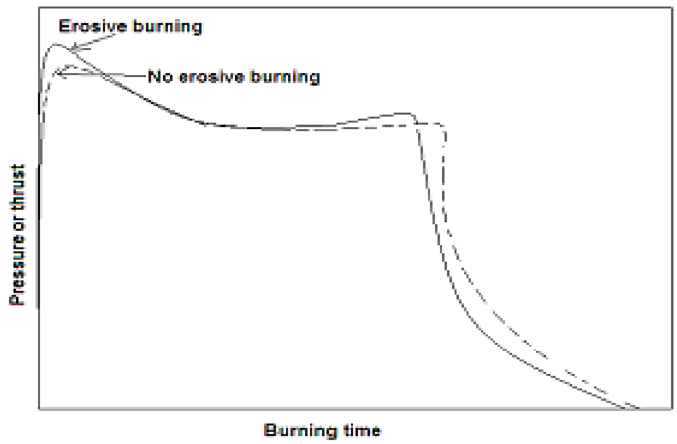


Fig. Typical pressure-time curve with and without erosive burning

Erosive burning increases the mass flow and thus also the chamber pressure and thrust during the early portion of the burning for a particular motor (see above Fig.). Erosive burning causes early burnout of the web, usually at the nozzle end, and exposes the insulation and aft closure to hot combustion gas for a longer period of time; this usually requires more insulation layer thickness (and more inert mass) to prevent local thermal failure. In designing motors, erosive burning is either avoided or controlled to be reproducible from one motor to the next.

Total burning rate = steady state burning rate (aP_c^n) + erosive burning

Basic Performance Relations

One basic performance relation derived from the principle of conservation of matter. The propellant mass burned per unit time has to equal the sum of the change in gas mass per unit time in the combustion chamber grain cavity and the mass flowing out through the exhaust nozzle per unit time.

$$A_b \, r \, \rho_b = \frac{d}{dt}(\rho_1 V_1) + A_t \, P_1 \sqrt{\frac{k}{RT_1}\left(\frac{2}{k+1}\right)^{(k+1)/(k-1)}}$$

The term on the left side of the equation gives the mass rate of gas generation. The first term on the right gives the change in propellant mass in the gas volume of the combustion chamber, and the last term gives the nozzle flow. The burning rate of propellant is r; A_b is the propellant burning area; ρ_b is the solid propellant density; ρ_1 is the combustion gas density; V_1 is the chamber gas cavity volume, which becomes larger as the propellant is expended; A_t is the throat area; P_1 is the chamber pressure; T_1 is the absolute chamber temperature, which is usually assumed to be constant; and k is the specific heat ratio of the combustion gases. During startup the changing mass of propellant in the grain cavity becomes important.

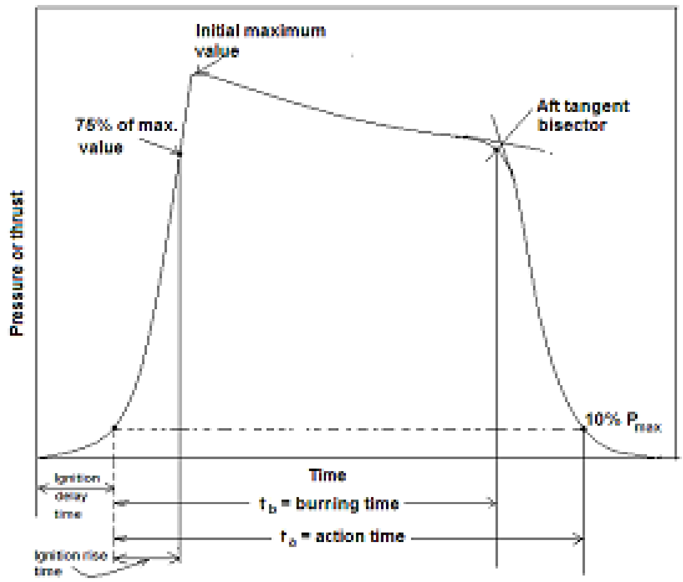


Fig. Definition of burning time and action time

Isentropic Flow through Nozzles

In a converging diverging nozzle a large fraction of the thermal energy of the gases in the chamber is converted into kinetic energy. As will be explained, the gas pressure and temperature drop dramatically and gas velocity can reach values in excess of around 3.2 km/sec. This is a reversible, essentially isentropic flow process.

If a nozzle inner wall has a flow obstruction or a wall protrusion (a piece of weld splatter or slag), then the kinetic gas energy is locally converted back into thermal energy essentially equal to the stagnation temperature and stagnation pressure in the chamber. Since this would lead quickly to a local overheating and failure of the wall, nozzle inner walls have to be smooth without any protrusion.

Nozzle exit velocity can be derived as,

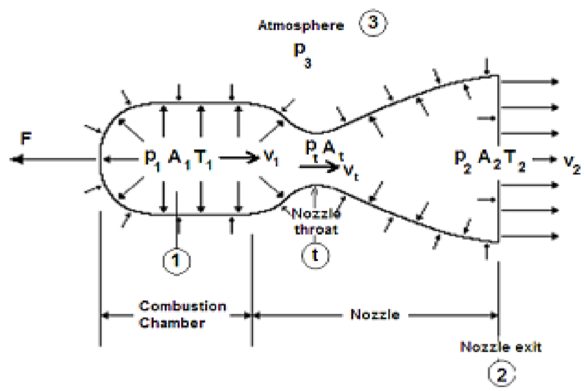
$$v_2 = \sqrt{\frac{2k}{k-1} RT_1 \left[1 - \left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right] + v_1^2}$$

This relation also holds for any two points within the nozzle. Note that when the chamber section is large compared to the nozzle throat section, the chamber velocity or nozzle approach velocity is comparatively small and the v_1^2 can be neglected. The chamber temperature T_1 is at the nozzle inlet and, under isentropic condition, differ little from the stagnation temperature or (for a chemical rocket) from combustion temperature. This leads to an important simplified expression of the exhaust velocity v_2, which is often used in the analysis.

$$v_2 = \sqrt{\frac{2k}{k-1} RT_1 \left[1 - \left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right]}$$

$$= \sqrt{\frac{2k}{k-1}\frac{R' T_o}{M}\left[1 - \left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right]}$$

Thrust and Thrust Coefficient



$$F = \dot{m} v_2 + (p_2 - p_3) A_2$$

$$F = C_F \, A_t \, P_1$$

Where C_F is the thrust coefficient, which can be derived as a function of gas property k, the nozzle area ratio (A_2/A_t), and the pressure ratio across the nozzle p_1/p_2 , but independent of chamber temperature. For any fixed pressure ratio (p_1/p_3) the thrust coefficient C_F and the thrust F have a peak when $p_2 = p_3$. This peak value is known as optimum thrust coefficient.

$$C_F = \sqrt{\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{(k+1)/(k-1)}\left[1 - \left(\frac{p_2}{p_1}\right)^{(k-1)/k}\right]} + \frac{p_2 - p_3}{p_1}\frac{A_2}{A_t}$$

Effective Exhaust Velocity

In a rocket nozzle the actual exhaust velocity is not uniform over the entire exit cross-section and does not represent the entire thrust magnitude. The velocity profile is difficult to measure accurately. For convenience a uniform axial velocity 'c' is assumed which allows a one-dimensional description of the problem. This *effective exhaust velocity* 'c' is the average equivalent velocity at which propellant is ejected from the vehicle. It is defined as

$$c = I_{sp}\, g_o = \frac{F}{\dot{m}}$$

It is usually given in meters per second.

The concept of weight relates to the gravitational attraction at or near sea level, but in space or outer satellite orbits, "weight" signifies the mass multiplied by an arbitrary constant, namely g_o. In system international (SI) or metric system of units I_{sp} can be expressed simply in "seconds", because of the use of the constant g_o.

Specific Propellant Consumption

Specific propellant consumption is the reciprocal of the specific impulse.

Mass Ratio

The mass ratio of a vehicle or a particular vehicle stage is defined to be the final mass m_f (after rocket operation has consumed all usable propellant) divided by initial mass m_o (before rocket operation).

$$\text{Mass Ratio, MR} = \frac{m_f}{m_o}$$

This applies to a single or multi-stage vehicle. The final mass m_f is the mass of the vehicle after the rocket has ceased to operate when all the useful propellant mass m_p has been consumed and ejected. The final vehicle mass m_f includes all those components that are not useful propellant and may include guidance devices, navigation gear, payload (e.g., scientific instruments or a military warhead), flight control systems, communication devices, power supplies, tank structure, residual or unusable propellant, and all the propulsion hardware. In some vehicles it can also include wings, fins, a crew, life support systems, reentry shields, landing gears etc. Typical value of Mass ratio can range from 60% for tactical missiles to less than 10 % for unmanned launch vehicle stages. This mass ratio is an important parameter in analyzing flight performance. Note that when mass-ratio is applied to a single stage of a multi-stage rocket, then its upper stages become the "payload"

Propellant Mass Fraction, ζ

Propellant mass fraction 'ζ' is defined as the ratio of propellant mass 'm_p' to the initial mass 'm_o'

$$\zeta = \frac{m_p}{m_o}$$

$$m_o = m_f + m_p$$

Characteristic Velocity

The characteristic velocity has been used frequently in the rocket propulsion literature. It is represented by a symbol C^*. It is defined as,

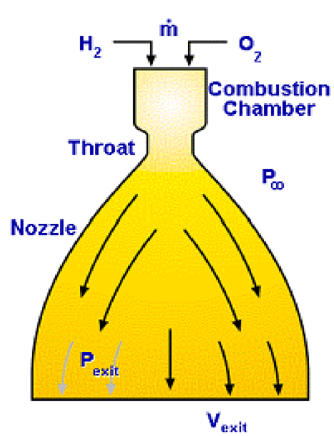
$$C^* = \frac{p_1 A_t}{\dot{m}}$$

The characteristic velocity is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants. It is basically a function of the propellant characteristics. It is easily determined from data of $\dot{m}$, p_1, and A_t. It relates to the efficiency of the combustion and is essentially independent of nozzle characteristics. However, the specific impulse and the effective exhaust velocities are functions of the nozzle geometry, such as the nozzle area ratio.

Rocket Nozzles

Purpose:

The nozzle is the component of a rocket or air-breathing engine that produces thrust. This is accomplished by converting the thermal energy of the hot chamber gases into kinetic energy and directing that energy along the nozzle's axis, as illustrated below.



Simple representation of a rocket nozzle

Although simplified, this figure illustrates how a rocket nozzle works. The propellant is composed of a fuel, typically liquid hydrogen (H_2), and an oxidizer, typically liquid oxygen (O_2). The propellant is pumped into a combustion chamber at some rate $\dot{m}$ (mdot) where the fuel and oxidizer are mixed and burned. The exhaust gases from this process are pushed into the throat region of the nozzle. Since the throat is of less cross-sectional area than the rest of the engine, the gases are compressed to a high pressure. The nozzle itself gradually increases in cross-sectional area allowing the gases to expand. As the gases do so, they push against the walls of the nozzle creating thrust.

Mathematically, the ultimate purpose of the nozzle is to expand the gases as efficiently as possible so as to maximize the exit velocity (v_{exit}). This process will maximize the thrust (F) produced by the system since the two are directly related by the equation

$$F = \dot{m}\, v_{exit} - (p_{exit} - p_{\infty})\, A_{exit}$$

where

F = thrust force

$\dot{m}$ = mass flow rate

v_{exit} = exhaust gas velocity at the nozzle exit

p_{exit} = pressure of the exhaust gases at the nozzle exit

p_{∞} = ambient pressure of the atmosphere

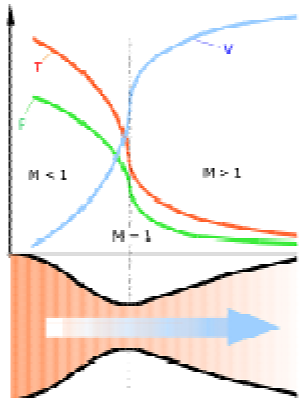
A_{exit} = cross-sectional area of the nozzle exit

Expansion Area Ratio:

In theory, the only important parameter in rocket nozzle design is the expansion area ratio (□), or the ratio of exit area (A_{exit}) to throat area (A_{throat}).

$$\varepsilon = \frac{A_{exit}}{A_{throat}}$$

Fixing all other variables (primarily the chamber pressure), there exists only one such ratio that optimizes overall system performance for a given altitude (or ambient pressure). However, a rocket typically does not travel at only one altitude. Thus, an engineer must be aware of the trajectory over which a rocket is to travel so that an expansion ratio that maximizes performance over a range of ambient pressures can be selected.



Typical temperatures (T) and pressures (p) and speeds (v) in a De Laval Nozzle

Maximum thrust for a rocket engine is achieved by maximizing the momentum contribution of the equation without incurring penalties from over expanding the exhaust. This occurs when $P_e = P_{amb}$. Since ambient pressure changes with altitude, most rocket engines spend very little time operating at peak efficiency.

Underexpanded nozzle

Ambient pressure nozzle

Overexpanded nozzle

Grossly overexpanded nozzle

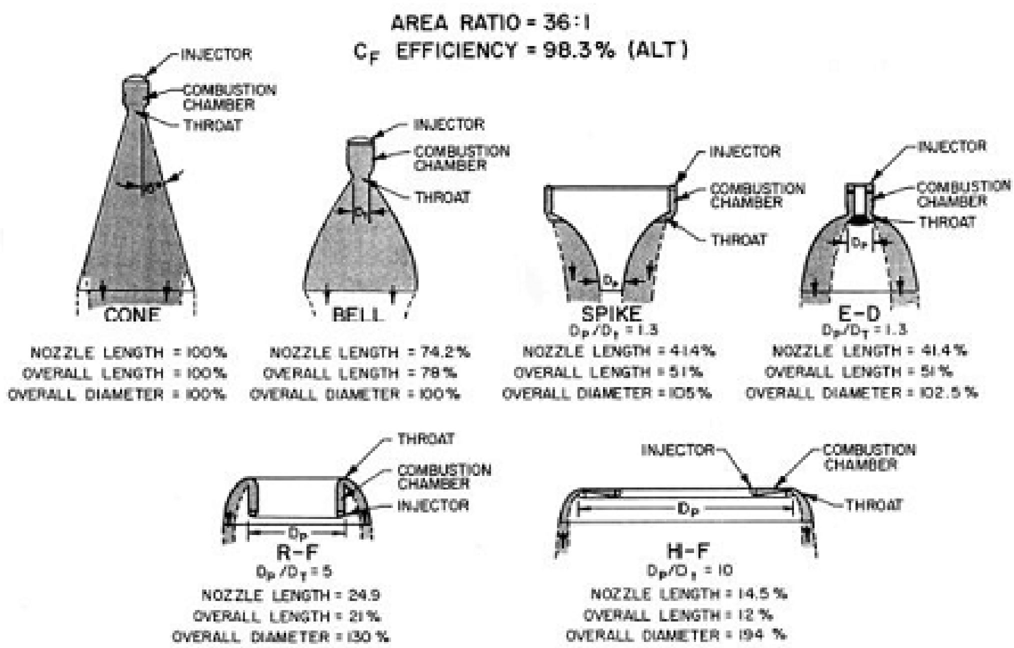
If the pressure of the exhaust jet varies from atmospheric pressure, nozzles can be said to be underexpanded, ambient or overexpanded. If under or overexpanded then loss of efficiency occurs, grossly overexpanded nozzles lose less efficiency, but the exhaust jet is usually unstable. Rockets become progressively more underexpanded as they gain altitude. Note that almost all rocket engines will be momentarily grossly overexpanded during startup in an atmosphere.

Rocket Nozzle Shapes

Not all rocket nozzles are alike, and the shape selected usually depends on the application. This section discusses the basic characteristics of the major classes of nozzles used today.

Nozzle Comparisons:

To date three major types of nozzles, the cone, the bell or contoured, and the annular or plug, have been employed. Each class satisfies the design criteria to varying degrees. Examples of these nozzle types can be seen below.



Size comparison of optimal cone, bell, and radial nozzles for a given set of conditions

Conical Nozzle:

The conical nozzle was used often in early rocket applications because of its simplicity and ease of construction. The cone gets its name from the fact that the walls diverge at a constant angle. A small angle produces greater thrust, because it maximizes the axial component of exit velocity and produces a high specific impulse (a measure of rocket efficiency). The penalty, however, is a longer and heavier nozzle that is more complex to build. At the other extreme, size and weight are minimized by a large nozzle wall angle. Unfortunately, large angles reduce performance at low altitude because the high ambient pressure causes overexpansion and flow separation.

Bell Nozzle:

The bell, the most commonly used nozzle shape, offers significant advantages over the conical nozzle, both in size and performance. Referring to the above figure, note that the bell consists of two sections. Near the throat, the nozzle diverges at a relatively large angle but the degree of diveregence tapers off further downstream. Near the nozzle exit, the diveregence angle is very small. In this way, the bell is a compromise between the two extremes of the conical nozzle since it minimizes weight while maximizing performance. The most important design issue is to contour the nozzle to avoid oblique shocks and maximize performance. However, we must remember that the final bell shape will only be the optimum at one particular altitude.

Annular Nozzles:

The annular nozzle, also sometimes known as the plug or "altitude-compensating" nozzle, is the least employed of those discussed due to its greater complexity. The term "annular" refers to the fact that combustion occurs along a ring, or annulus, around the base of the nozzle. "Plug" refers to the centerbody that blocks the flow from what would be the center portion of a traditional nozzle. "Altitude-compensating" is sometimes used to describe these nozzles since that is their primary advantage, a quality that will be further explored later.

Before describing the various forms of annular nozzles, it is useful to mention some key differences in design parameters from the conical or bell nozzles. The expansion area ratio for a traditional nozzle has already been discussed. When considering an annular nozzle, the area of the centerbody (A_{plug}) must also be taken into account.

$$\varepsilon = \frac{A_{exit} - A_{plug}}{A_{throat}}$$

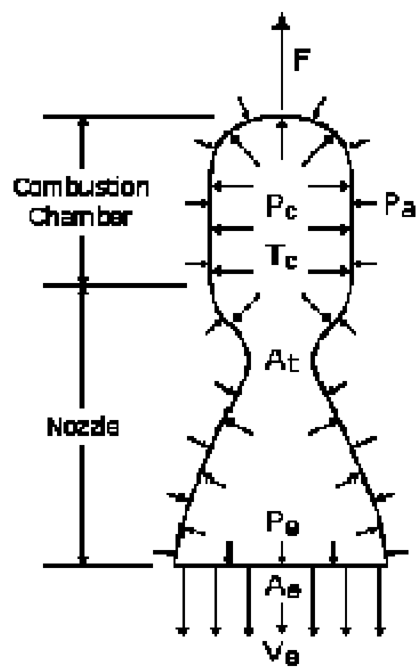
Another parameter particular to this type of nozzle is the annular diameter ratio, D_p / D_t, or the ratio of the centerbody diameter to that of the throat. The ratio is used as a measure of the nozzle geometry for comparison with other plug nozzle shapes. Typical values of this ratio appear in the above figure.

ROCKET PROPULSION

Isaac Newton stated in his third law of motion that "for every action there is an equal and opposite reaction." It is upon this principle that a rocket operates. Propellants are combined in a combustion chamber where they chemically react to form hot gases which are then accelerated and ejected at high velocity through a nozzle, thereby imparting momentum to the engine. The thrust force of a rocket motor is the reaction experienced by the motor structure due to ejection of the high velocity matter. This is the same phenomenon which pushes a garden hose backward as water flows from the nozzle, or makes a gun recoil when fired.

Thrust

Thrust is the force that propels a rocket or spacecraft and is measured in pounds, kilograms or Newtons. Physically speaking, it is the result of pressure which is exerted on the wall of the combustion chamber.

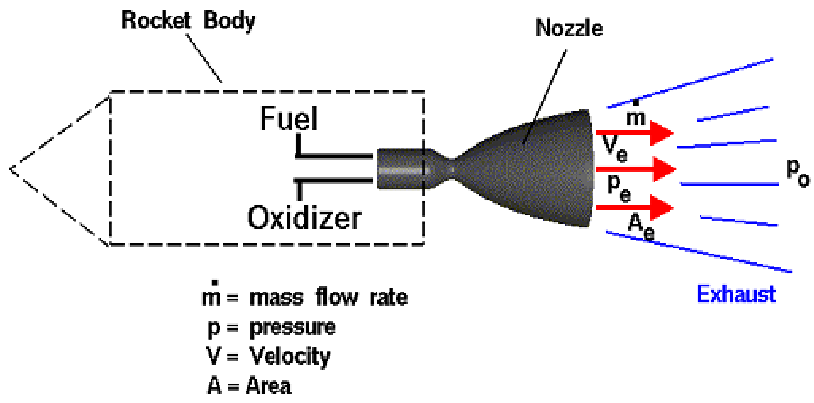


The figure to the right shows a combustion chamber with an opening, the nozzle, through which gas can escape. The pressure distribution within the chamber is asymmetric; that is, inside the chamber the pressure varies little, but near the nozzle it decreases somewhat. The force due to gas pressure on the bottom of the chamber is not compensated for from the outside. The resultant force *F* due to the internal and external pressure difference, the thrust, is opposite to the direction of the gas jet. It pushes the chamber upwards.

To create high speed exhaust gases, the necessary high temperatures and pressures of combustion are obtained by using a very energetic fuel and by having the molecular weight of the exhaust gases as low as possible. It is also necessary to reduce the pressure of the gas as much as possible inside the nozzle by creating a large section ratio. The section ratio, or expansion ratio, is defined as the area of the exit *Ae* divided by the area of the throat *At*.

Depending on the context, the chemical rockets are classified in many ways as follows:

(a) Type of propellant: Solid, Liquid (mono propellant / bipropellant and hybrid rockets)
(b) Application: Launch vehicle, ABM's, JATO's, ICBM, IRBM, SAM etc.
(c) Size of Unit (and thrust level sometimes): 10 ton, 100 kg etc.
(d) Type of subsystem: Turbopump fed, clustering, grain type etc.



$$\text{Thrust} = \quad F = \dot{m}\, V_e + (p_e - p_0)\, A_e$$

SOLID ROC ETS

Specific Impulse: 100-400 sec
Thrust: 10^3-10^7 N

- Solid rockets are the simplest and earliest types of rocket propulsion dating back to the first rockets used by the Chinese.
- Solid rockets are filled with a solid mixture of a propellant and an oxidizer. Little else is actually required for these rockets.
- The designs are very simple and therefore very reliable.
- The main drawback of solid rockets is that once ignited, they burn until all of the fuel is gone. Because of this, they aren't used often in space where propulsion systems are usually required to be turned on and off many times. However, they are good for getting things into space. In fact, the space shuttles use solid rocket boosters (SRBs) during takeoff.
- **Quick Fact** : The SRBs are the largest solid-propellant motors ever flown and the first designed for reuse. Each is 149.16 feet long and 12.17 feet in diameter.

MONOPROPELLANT ROCKETS

Specific Impulse: 100-300 sec
Thrust: 0.1-100 N

- Monopropellant rockets are simple propulsion systems that rely on special chemicals which, when energized, decompose. This decomposition creates both the fuel and an oxidizer (which allows the fuel to burn), which then react with each other. Because they only use a single propellant, monopropellant rockets are quite simple and reliable. Unfortunately, they are not very efficient. They are mainly used to make small adjustments such as attitude control. Main propulsion systems usually use some other technology.

BIPROPELLANT ROC ETS

Specific Impulse: 100-400 sec
Thrust: 0.1-10^7 N

- Bipropellant rockets separate the fuel and oxidizer and mix them in the chamber where they burn. Bipropellant rockets are widely used and more efficient than monopropellant rockets. The reaction given in the lesson on chemistry gives an example of a fuel(H_2)/oxidizer(O_2) combination. It's actually a very good combination in that it releases a large amount of energy. It's the combination used by the space shuttle's main engines. Unfortunately, large tanks kept at extremely low temperatures are required to

carry them. In fact, the main purpose of the giant red external tank attached to the space shuttle on take-off is to carry enough fuel to get the space shuttle into space.

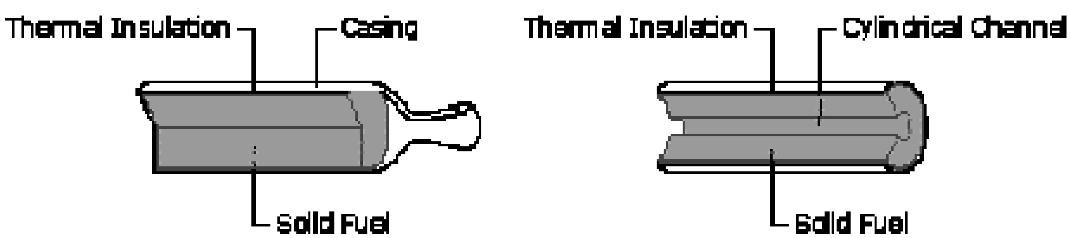
The main drawback of bipropellant rockets is that they are more complex than solid or monopropellant rockets. The fuel and oxidizer have to be stored separately and fed together in exactly the right ratios to achieve maximum efficiency. Despite the extra complexity, bipropellant rockets are still one of the preferred systems for primary propulsion

Solid Rocket Motors

Solid rockets motors store propellants in solid form. The fuel is typically powdered aluminum and the oxidizer is ammonium perchlorate. A synthetic rubber binder such as polybutadiene holds the fuel and oxidizer powders together. Though lower performing than liquid propellant rockets, the operational simplicity of a solid rocket motor often makes it the propulsion system of choice.

Solid Fuel Geometry

A solid fuel's geometry determines the area and contours of its exposed surfaces, and thus its burn pattern. There are two main types of solid fuel blocks used in the space industry. These are cylindrical blocks, with combustion at a front, or surface, and cylindrical blocks with internal combustion. In the first case, the front of the flame travels in layers from the nozzle end of the block towards the top of the casing. This so-called end burner produces constant thrust throughout the burn. In the second, more usual case, the combustion surface develops along the length of a central channel. Sometimes the channel has a star shaped, or other, geometry to moderate the growth of this surface.



The shape of the fuel block for a rocket is chosen for the particular type of mission it will perform. Since the combustion of the block progresses from its free surface, as this surface grows, geometrical considerations determine whether the thrust increases, decreases or stays constant.

Burn Rate

The burning surface of a rocket propellant grain recedes in a direction perpendicular to this burning surface. The rate of regression, typically measured in millimeters per second (or inches per second), is termed *burn rate*. This rate can differ significantly for different propellants, or for one particular propellant, depending on various operating conditions as well as formulation. Knowing quantitatively the burning rate of a propellant, and how it changes under various conditions, is of fundamental importance in the successful design of a solid rocket motor.

Propellant burning rate is influenced by certain factors, the most significant being: combustion chamber pressure, initial temperature of the propellant grain, velocity of the combustion gases flowing parallel to the burning surface, local static pressure, and motor acceleration and spin. These factors are discussed below.

- Burn rate is profoundly affected by chamber pressure. The usual representation of the pressure dependence on burn rate is the Saint-R

  a grain L/D ratio of 6. A greater A_{port}/A_t ratio should be used for grains with larger L/D ratios.

- In an operating rocket motor, there is a pressure drop along the axis of the combustion chamber, a drop that is physically necessary to accelerate the increasing mass flow of combustion products toward the nozzle. The static pressure is greatest where gas flow is zero, that is, at the front of the motor. Since burn rate is dependant upon the local pressure, the rate should be greatest at this location. However, this effect is relatively minor and is usually offset by the counter-effect of erosive burning.
- Burning rate is enhanced by acceleration of the motor. Whether the acceleration is a result of longitudinal force (e.g. thrust) or spin, burning surfaces that form an angle of about 60-90° with the acceleration vector are prone to increased burn rate.

It is sometimes desirable to modify the burning rate such that it is more suitable to a certain grain configuration. For example, if one wished to design an end burner grain, which has a relatively small burning area, it is necessary to have a fast burning propellant. In other circumstances, a reduced burning rate may be sought after. For example, a motor may have a large L/D ratio to generate sufficiently high thrust, or it may be necessary for a particular design to restrict the diameter of the motor. The web would be consequently thin, resulting in short burn duration. Reducing the burning rate would be beneficial.

There are a number of ways of modifying the burning rate: decrease the oxidizer particle size, increase or reduce the percentage of oxidizer, adding a burn rate catalyst or suppressant, and operate the motor at a lower or higher chamber pressure. These factors are discussed below.

- The effect of the oxidizer particle size on burn rate seems to be influenced by the type of oxidizer. Propellants that use ammonium perchlorate (AP) as the oxidizer have a burn rate that is significantly affected by AP particle size. This most likely results from the decomposition of AP being the rate-determining step in the combustion process.

- The burn rate of most propellants is strongly influenced by the oxidizer/fuel ratio. Unfortunately, modifying the burn rate by this means is quite restrictive, as the performance of the propellant, as well as mechanical properties, are also greatly affected by the O/F ratio.
- Certainly the best and most effective means of increasing the burn rate is the addition of a *catalyst* to the propellant mixture. A catalyst is a chemical compound that is added in small quantities for the sole purpose of tailoring the burning rate. A burn rate *suppressant* is an additive that has the opposite effect to that of a catalyst -- it is used to decrease the burn rate.
- For a propellant that follows the Saint-Robert's burn rate law, designing a rocket motor to operate at a lower chamber pressure will provide for a lower burning rate. Due to the nonlinearity of the pressure-burn rate relationship, it may be necessary to significantly reduce the operating pressure to get the desired burning rate. The obvious drawback is reduced motor performance, as specific impulse similarly decays with reducing chamber pressure.

Monopropellant Engines

By far the most widely used type of propulsion for spacecraft attitude and velocity control is monopropellant hydrazine. Its excellent handling characteristics, relative stability under normal storage conditions, and clean decomposition products have made it the standard. The general sequence of operations in a hydrazine thruster is:

- When the attitude control system signals for thruster operation, an electric solenoid valve opens allowing hydrazine to flow. The action may be pulsed (as short as 5 ms) or long duration (steady state).
- The pressure in the propellant tank forces liquid hydrazine into the injector. It enters as a spray into the thrust chamber and contacts the catalyst beds.
- The catalyst bed consists of alumina pellets impregnated with iridium. Incoming hydrazine heats to its vaporizing point by contact with the catalyst bed and with the hot gases leaving the catalyst particles. The temperature of the hydrazine rises to a point where the rate of its decomposition becomes so high that the chemical reactions are self-sustaining.
- By controlling the flow variables and the geometry of the catalyst chamber, a designer can tailor the proportion of chemical products, the exhaust temperature, the molecular weight, and thus the enthalpy for a given application. For a thruster application where specific impulse is paramount, the designer attempts to provide 30-40% ammonia dissociation, which is about the lowest percentage that can be maintained reliably. For gas-generator application, where lower temperature gases are usually desired, the designer provides for higher levels of ammonia dissociation.
- Finally, in a space thruster, the hydrazine decomposition products leave the catalyst bed and exit from the chamber through a high expansion ratio exhaust nozzle to produce thrust.

Monopropellant hydrazine thrusters typically produce a specific impulse of about 230 to 240 seconds.

Other suitable propellants for catalytic decomposition engines are hydrogen peroxide and nitrous oxide, however the performance is considerably lower than that obtained with hydrazine - specific impulse of about 150 s with H_2O_2 and about 170 s with N_2O.

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Monopropellant systems have successfully provided orbit maintenance and attitude control functions, but lack the performance to provide weight-efficient large ΔV maneuvers required for orbit insertion. Bipropellant systems are attractive because they can provide all three functions with one higher performance system, but they are more complex than the common solid rocket and monopropellant combined systems. A third alternative are *dual mode* systems. These systems are hybrid designs that use hydrazine both as a fuel for high performance bipropellant engines and as a monopropellant with conventional low-thrust catalytic thrusters. The hydrazine is fed to both the bipropellant engines and the monopropellant thrusters from a common fuel tank.

Cold gas propulsion is just a controlled, pressurized gas source and a nozzle. It represents the simplest form of rocket engine. Cold gas has many applications where simplicity and/or the need to avoid hot gases are more important than high performance. The Manned Maneuvering Unit used by astronauts is an example of such a system.

Staging

Multistage rockets allow improved payload capability for vehicles with a high ΔV requirement such as launch vehicles or interplanetary spacecraft. In a multistage rocket, propellant is stored in smaller, separate tanks rather than a larger single tank as in a single-stage rocket. Since each tank is discarded when empty, energy is not expended to accelerate the empty tanks, so a higher total ΔV is obtained. Alternatively, a larger payload mass can be accelerated to the same total ΔV. For convenience, the separate tanks are usually bundled with their own engines, with each discardable unit called a *stage*.

There are two families of solids propellants: **homogeneous and composite**. Both types are dense, stable at ordinary temperatures, and easily storable.

Homogeneous propellants are either simple base or double base. A simple base propellant consists of a single compound, usually **nitrocellulose**, which has both an oxidation capacity and a reduction capacity. Double base propellants usually consist of **nitrocellulose and nitroglycerine**, to which a plasticiser is added. Homogeneous propellants do not usually have specific impulses greater than about 210 seconds under normal conditions. Their main asset is that they do not produce traceable fumes and are, therefore, commonly used in tactical weapons. They are also often used to perform subsidiary functions such as jettisoning spent parts or separating one stage from another.

Modern composite propellants are heterogeneous powders (mixtures) which use a crystallized or finely ground mineral salt as an oxidizer, often ammonium perchlorate, which constitutes between 60% and 90% of the mass of the propellant. **The fuel itself is generally aluminum**. The propellant is held together by a polymeric binder, usually **polyurethane or polybutadienes**, which is also consumed as fuel. Additional compounds are sometimes included, such as a catalyst to help increase the burning rate, or other agents to make the powder easier to manufacture. The final product is rubberlike substance with the consistency of a hard rubber eraser.

Composite propellants are often identified by the type of polymeric binder used. The two most common binders are polybutadiene acrylic acid acrylonitrile (**PBAN**) and hydroxy-terminator polybutadiene (**HTPB**). PBAN formulations give a slightly higher specific impulse, density, and burn rate than equivalent formulations using HTPB. However, PBAN propellant is the more difficult to mix and process and requires an elevated curing temperature. HTPB binder is stronger and more flexible than PBAN binder. Both PBAN and HTPB formulations result in propellants that deliver excellent performance, have good mechanical properties, and offer potentially long burn times.

Hybrid Propellants

Hybrid propellant engines represent an intermediate group between solid and liquid propellant engines. One of the substances is solid, usually the fuel, while the other, usually the oxidizer, is liquid. The liquid is injected into the solid, whose fuel reservoir also serves as the combustion chamber. The main advantage of such engines is that they have high performance, similar to that of solid propellants, but the combustion can be moderated, stopped, or even restarted. It is difficult to make use of this concept for vary large thrusts, and thus, hybrid propellant engines are rarely VESFVE) WB05SD5)hWB05S5==E)tWBFS&EF=

Hybrid Rocket Engines are those which use liquid oxidizer and a solid fuel. Below figure shows typical elements of an Hybrid Rocket Engine.

The liquid oxidizer is atomized and sprayed over the fuel block. In hypergolic systems, only gas phase reactions occur. The oxidizer content of the hot product gases decreases along the port and the length of the grain.

- Two of the issues in this combustion process are
- (i) mixing of the oxidizer rich and fuel rich gases across the diffusion flame occurs much later than the length of the fuel grain and
- (ii) fuel regression rate is small.

The first issue is resolved by adding mixing devices and second issue is solved by adding a certain amount of oxidizer into the fuel.

- Hybrid rocket engines retain the advantage of controllability like liquid rockets. The added safety is an attraction for use of hybrid rockets in situations calling for safety similar to civil aircraft operations. There may be possibilities for their use in single stage-to-orbit vehicles providing low cost access to space.

Solid Propellant Rockets

Advantages	Disadvantages
Simple Design (few or no moving parts)	Explosion and fire potential is larger; failure can be catastrophic
Easy to operate (little preflight checkout)	Many require environmental permit and safety features for transport on public conveyances
Ready to operate quickly	Under certain conditions some solid propellants can detonate
Will not leak, spill, or slosh	Cumulative grain damage occurs through temperature cycling or rough handling; this limits useful life
Can be stored for 5 to 25 years	If designed for reuse, it requires extensive factory rework and new propellants
Usually, higher overall density; this allows a more compact package, a small vehicle (less drag)	Requires an ignition system
Can provide TVC, but at increased complexity	Once ignited, cannot change predetermined thrust or duration
Some propellants have nontoxic, clean exhaust gases, but at a performance penalty	Integrity of grain (cracks, unbounded areas) is difficult to determine in the field
Some grain and case design can be used with several nozzles	Large boosters take a few seconds to start
Thrust termination devices permit control over total impulse	Cannot be tested prior to use
Can be designed for recovery and reuse	Thermal insulation is required in almost all rocket motors
Some tactical missile motors have been produced in large quantities	Needs a safety provision to prevent inadvertent ignition, which would lead to an unplanned motor firing. Can cause a disaster.

Liquid Propellant Rockets

Advantages	Disadvantages
High specific impulse than solid propellant rockets	Relatively complex design, more parts or components, more things to go wrong!
Can be randomly throttled and randomly stopped and restarted.	Cryogenic propellants cannot be stored for long periods except when tanks are well insulated and escaping vapours are recondensed. Propellant loading occurs at the launch stand and requires cryogenic propellant storage facilities
Thrust-time profile can be randomly controlled; this allows a reproducible flight trajectory	Spills or leaks of several propellants can be hazardous, corrosive, toxic, and cause fires, but this can be minimized with gelled propellants
Cutoff impulse can be controllable with thrust termination device (better control of vehicle terminal velocity)	More overall weight for most short-duration, low-total-impulse applications.
Can be tested at full thrust on ground or launch pad prior to flight	Non-hypergolic propellants require an ignition system
Can be designed for reuse after field services and checkout	Tanks need to be pressurized by a separate pressurization system. This can require high pressure inert gas storage for long periods of time.
Thrust chamber (or some part of the vehicle) can be cooled and made lightweight	Bullet impact will cause leaks, sometimes a fire, but usually no detonations; gelled propellants can minimize or eliminate these hazards.
Storable liquid propellants have been kept in vehicle for more than 20 years and engine can be ready to operate quickly.	Usually requires more volume due to lower average propellant density and relatively inefficient packaging of engine components
Most propellants have nontoxic exhaust, which is environmentally acceptable	Sloshing in tank can cause a flight stability problem, but can be minimized with baffles.
Can modify operating conditions during firing to prevent some failures that would otherwise result in the loss of the mission or vehicle	Smoky exhaust (soot) plume can occur with some hydrocarbon fuels
Can provide component redundancy (e.g., dual check valves or extra thrust chamber) to enhance reliability	Needs special design provisions for start in zero gravity
Plume radiation and smoke are usually low	High-thrust unit requires several seconds to start