AEROSPACE PROPULSION

M. Tech I semester (IARE R-18)

BY

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CO’s	Course Outcomes
CO1	Describe the various types, basic function, and performance analysis of air-breathing engine.
CO2	Understand the various inlets and combustion chamber performance parameters affecting it.
CO3	Describe principle operations of compressors, with work done and pressure rise explaining the design and performance parameters of turbine, understand configuration associated
CO4	Discuss the working principle of solid and liquid propellant rockets.
CO5	Demonstrate the working principle of liquid propellant rockets and gain basic knowledge of rocket propulsion and its feed systems.

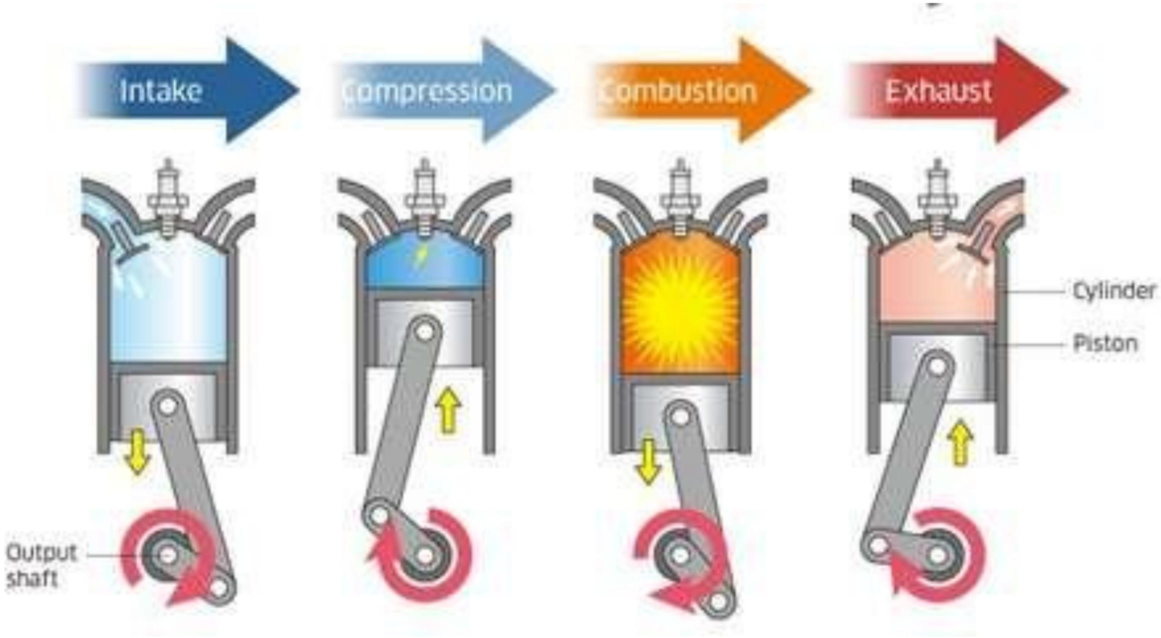
UNIT - I

AIR-BREATHING ENGINES

UNIT – I

CLO’s	Course Learning Outcomes
CLO1	Demonstrate different type’s aircraft engine operating principle.
CLO2	Understand steps involved in performance analysis of all aircraft engine.
CLO3	Analyze the engine performance parameters and parameters influencing them.

IC Engine



Various stages of an IC Engine

Propulsion

- Propulsion means to push forward or drive an object forward. The term is derived from two Latin words: **pro**, meaning before or forward; and **pellere**, meaning to drive.
- An aircraft propulsion system generally consists of an aircraft engine and some means to generate thrust, such as a **propeller** or a **propulsive nozzle.**

An aircraft propulsion system must achieve two things.

1. First, the thrust from the propulsion system must balance the drag of the airplane when the airplane is cruising.
2. The thrust from the propulsion system must exceed the drag of the airplane for the airplane to accelerate. The greater the difference between the thrust and the drag, called the excess thrust, the faster the airplane will accelerate.

What is thrust

Thrust can be defined as the reaction force produced by the accelerated mass ejected from the propeller or nozzle .

Basic principle

Newton's third law

How ?

High thrust can be produced by accelerating a large mass of gas by a small amount (propeller), or by accelerating a small mass of gas by a large amount (propulsive nozzle).

What is meant by air breathing engine?

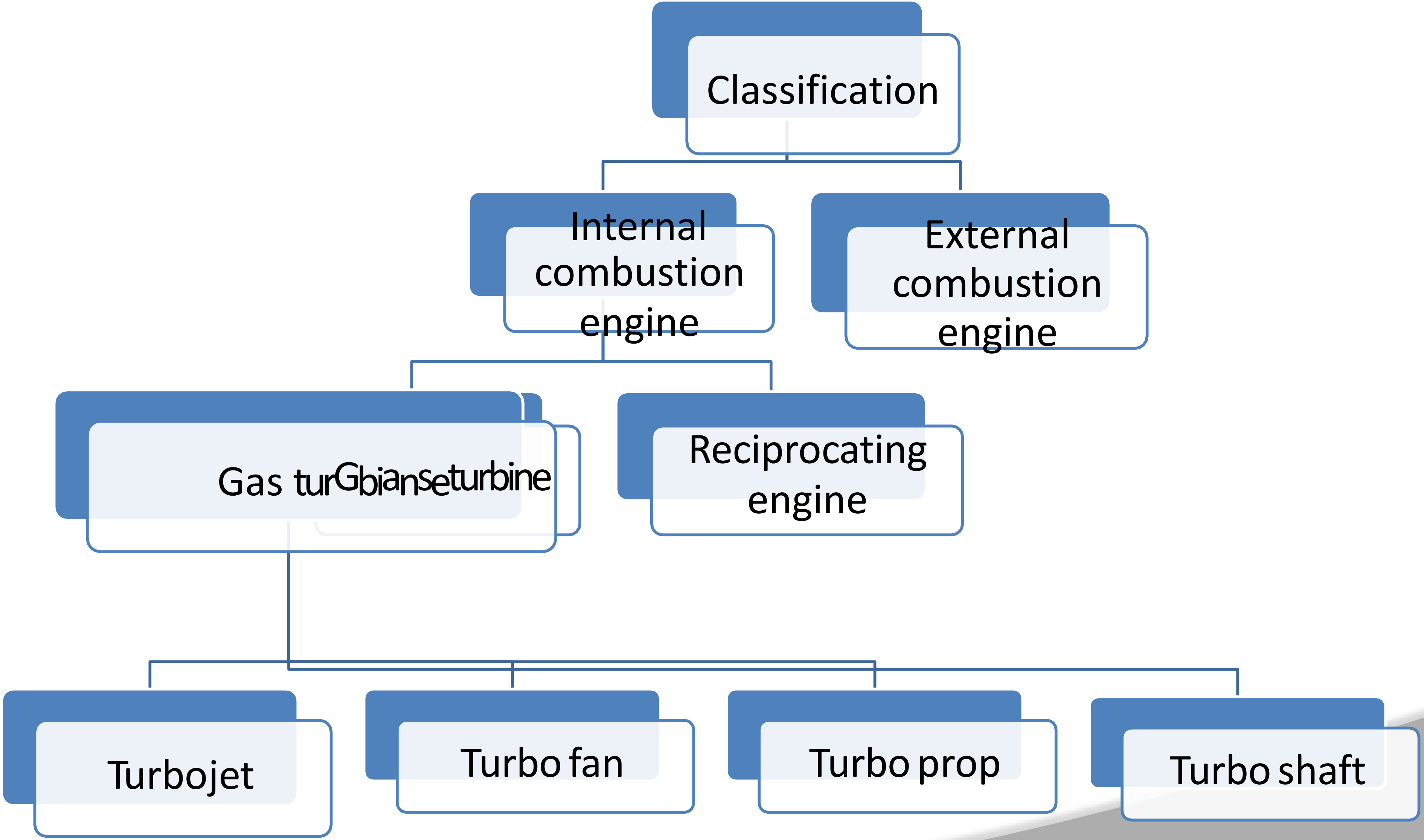
A propulsive unit or engine uses atmospheric air as oxidizer for the combustion process are called air breathing engines.

e.g. Turbojet engine

Non air breathing engine

A propulsive unit or engine has its own oxidizer stored in it is called non-air breathing engines.

e.g. Rocket engine



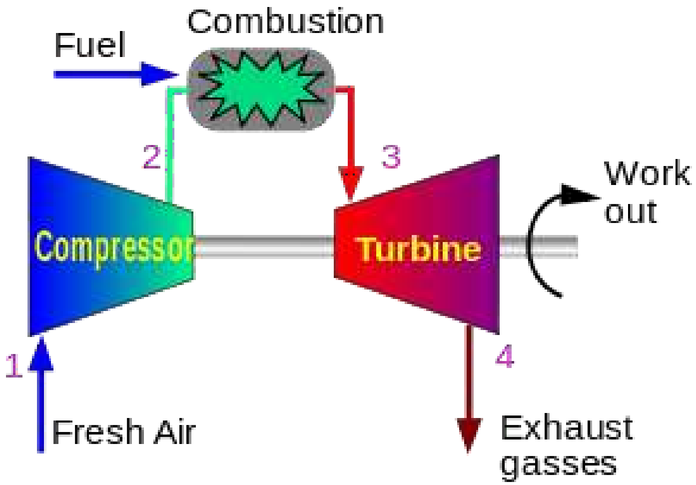
- Classification
  - Internal combustion engine
    - Gas turbine
      - Turbojet
      - Turbo fan
      - Turbo prop
      - Turbo shaft
    - Reciprocating engine
  - External combustion engine

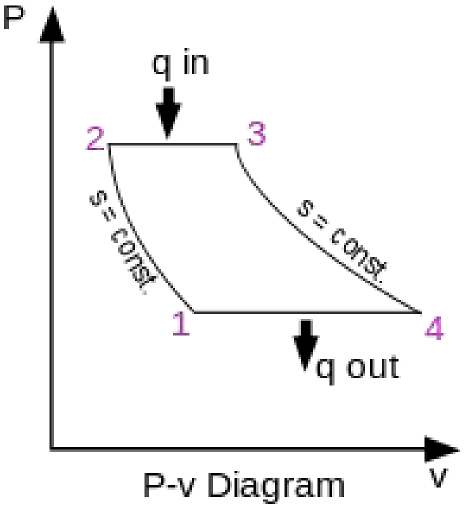
Gas turbine

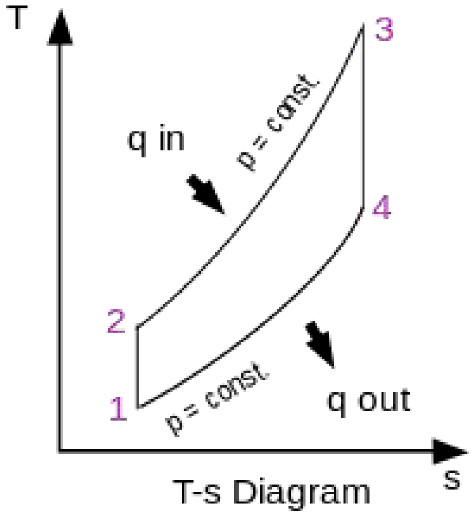
A gas turbine, also called a combustion turbine, is a type of continuous combustion, internal combustion engine. There are three main components:

1. An upstream rotating gas compressor;
2. A downstream turbine on the same shaft;
3. A combustion chamber or area, called a combustor, in between 1. and 2. above.

The basic operation of the gas turbine is a Brayton cycle with air as the working fluid







Ideal Brayton Cycle:

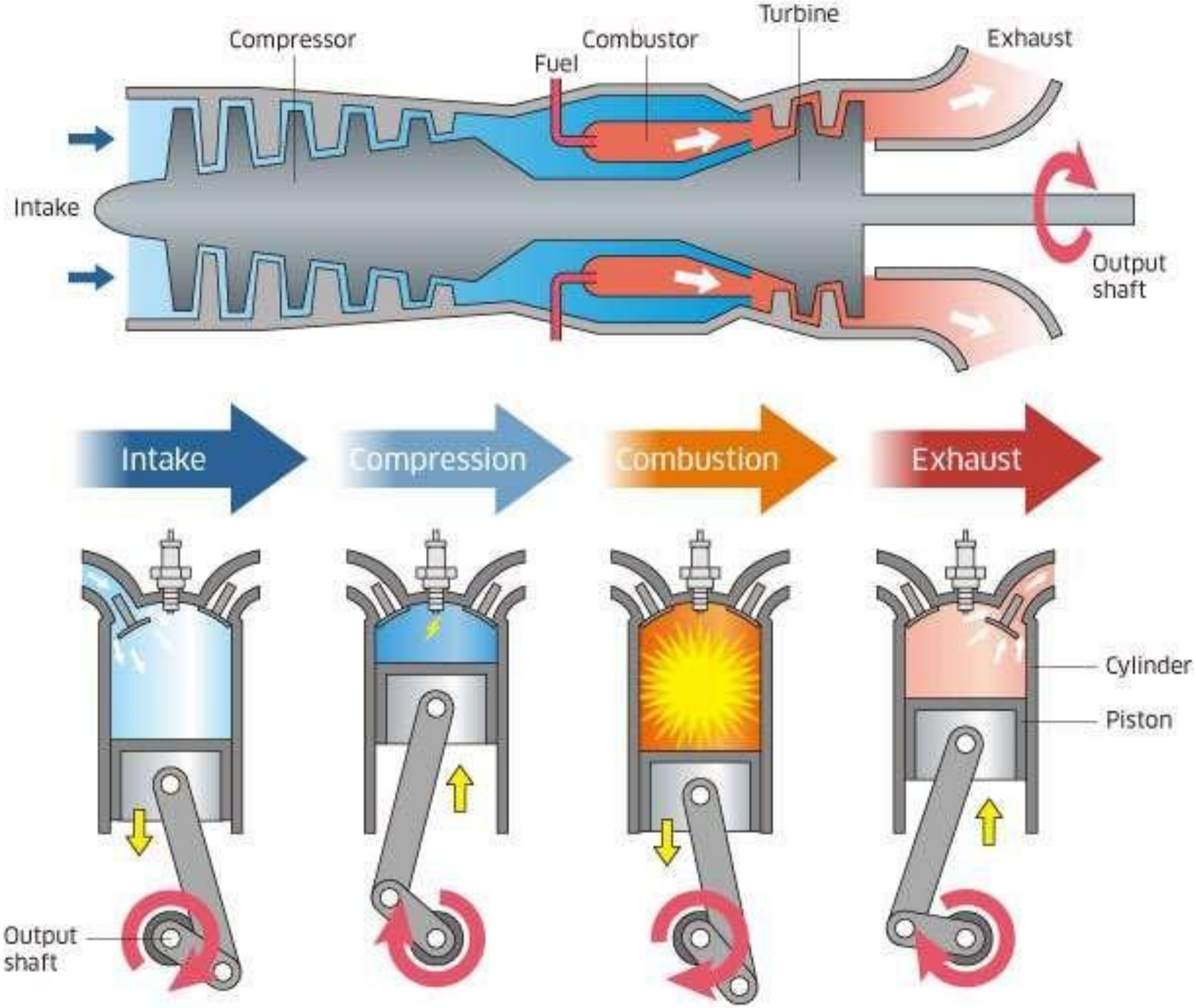
1-2 isentropic process – ambient air is drawn into the compressor, where it is pressurized.

2-3 isobaric process – the compressed air then runs through a combustion chamber, where fuel is burned, heating that air—a constant-pressure process, since the chamber is open to flow in and out.

3-4 isentropic process – the heated, pressurized air then gives up its energy, expanding through a turbine (or series of turbines). Some of the work extracted by the turbine is used to drive the compressor.

4-1 isobaric process – heat rejection (in the atmosphere).

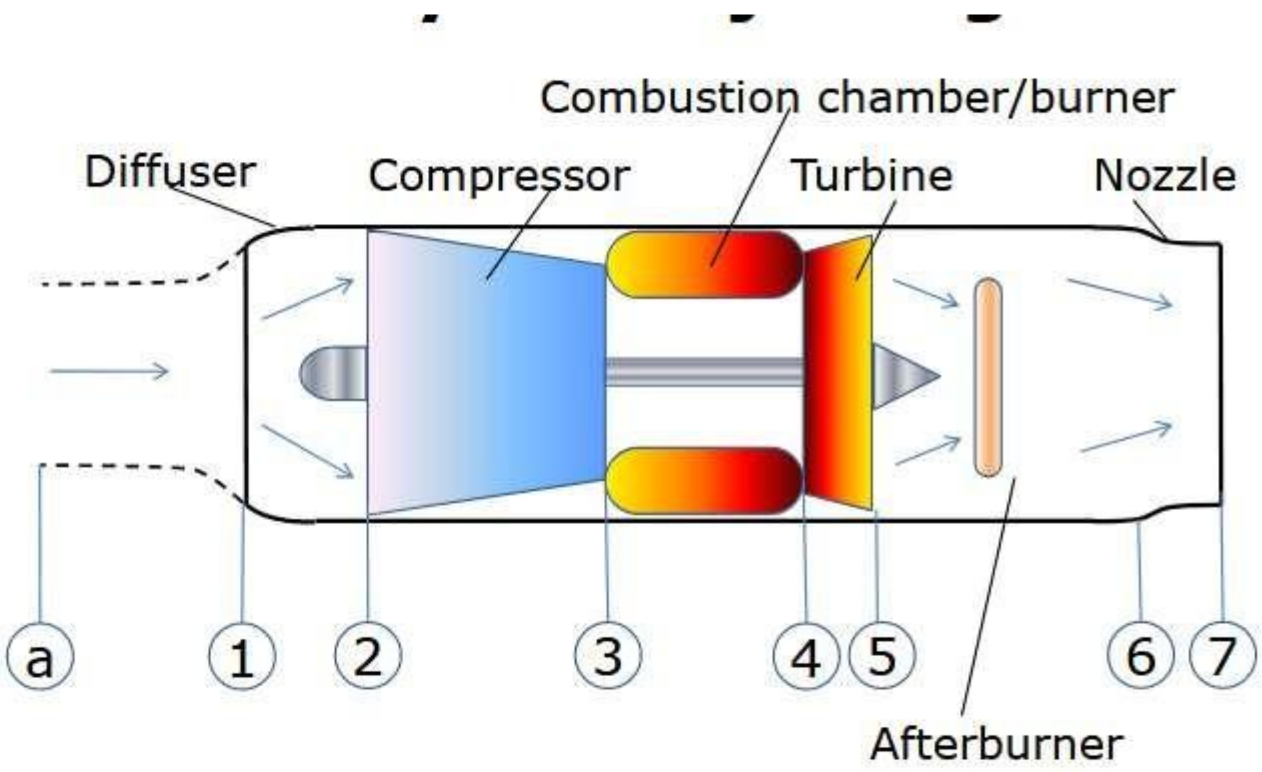
Comparison between Jet Engine & IC Engine



Gas Turbine Cycles

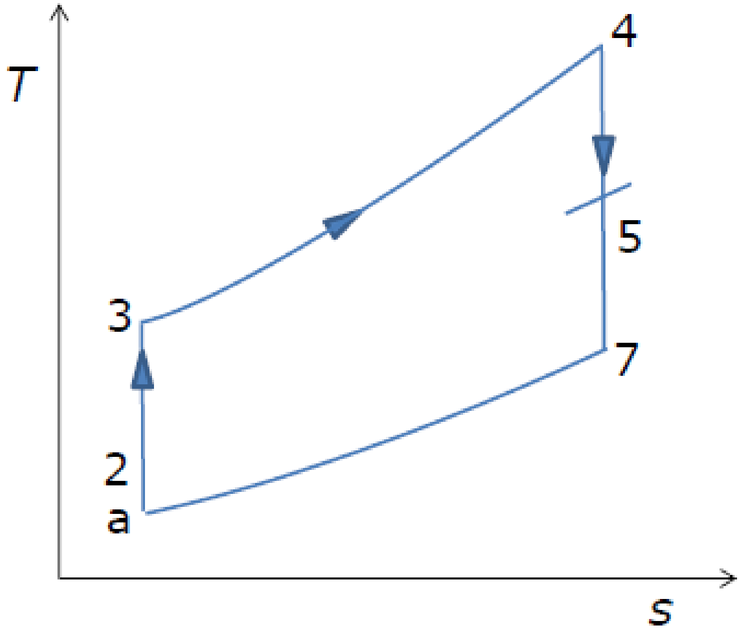
- Gas turbine engines operate on Brayton cycles.
- Ideal Brayton cycle is a closed cycle, whereas gas turbines operate in the open cycle mode.
- Ideal cycle assumes that there are no irreversibility's in the processes, air behaves like an ideal gas with constant specific heats, and that there are no frictional losses.
- All air-breathing jet engines operate on the Brayton cycle (open cycle mode).
- The most basic form of a jet engine is a turbojet engine.

Ideal cycle for turbo jet engines

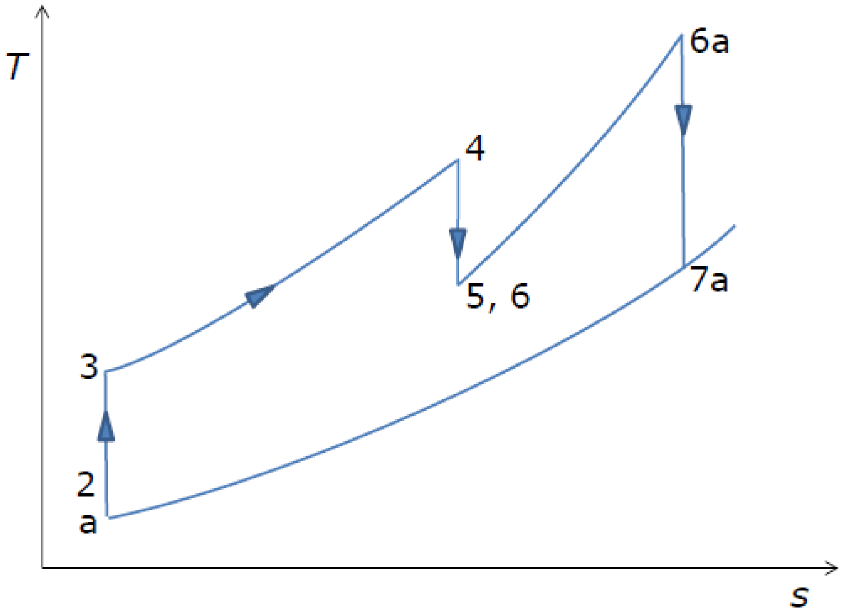


The different processes in a turbojet cycle are the following:

- a-1:Air from far upstream is brought to the air intake (diffuser) with some acceleration/deceleration
- 1-2:Air is decelerated as is passes through the diffuser
- 2-3:Air is compressed in a compressor (axial or centrifugal)
- 3-4:The air is heated using a combustion chamber/burner
- 4-5:The air is expanded in a turbine to obtain power to drive the compressor
- 5-6:The air may or may not be further heated in an afterburner by adding further fuel
- 6-7:The air is accelerated and exhausted through the nozzle to produce thrust.



Ideal turbojet cycle (without afterburning)



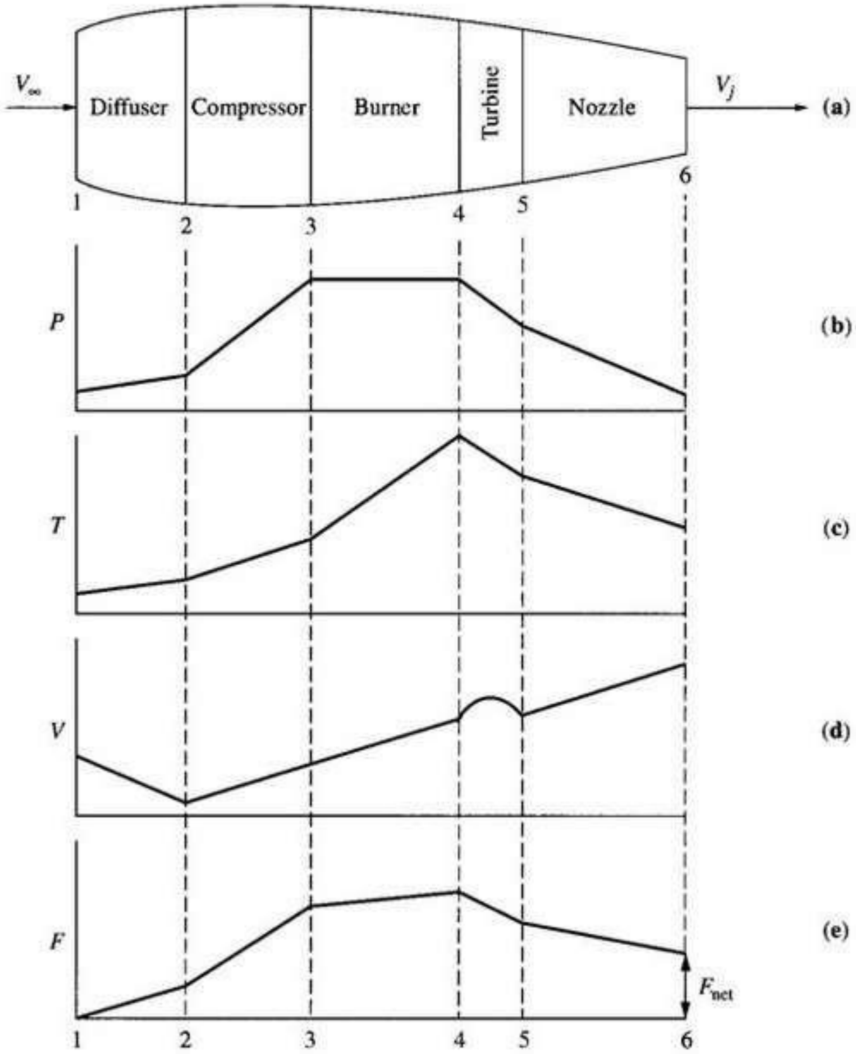
Ideal turbojet cycle (with afterburning)

- Afterburning: used when the aircraft needs a substantial increment in thrust. For eg. to accelerate to and cruise at supersonic speeds.

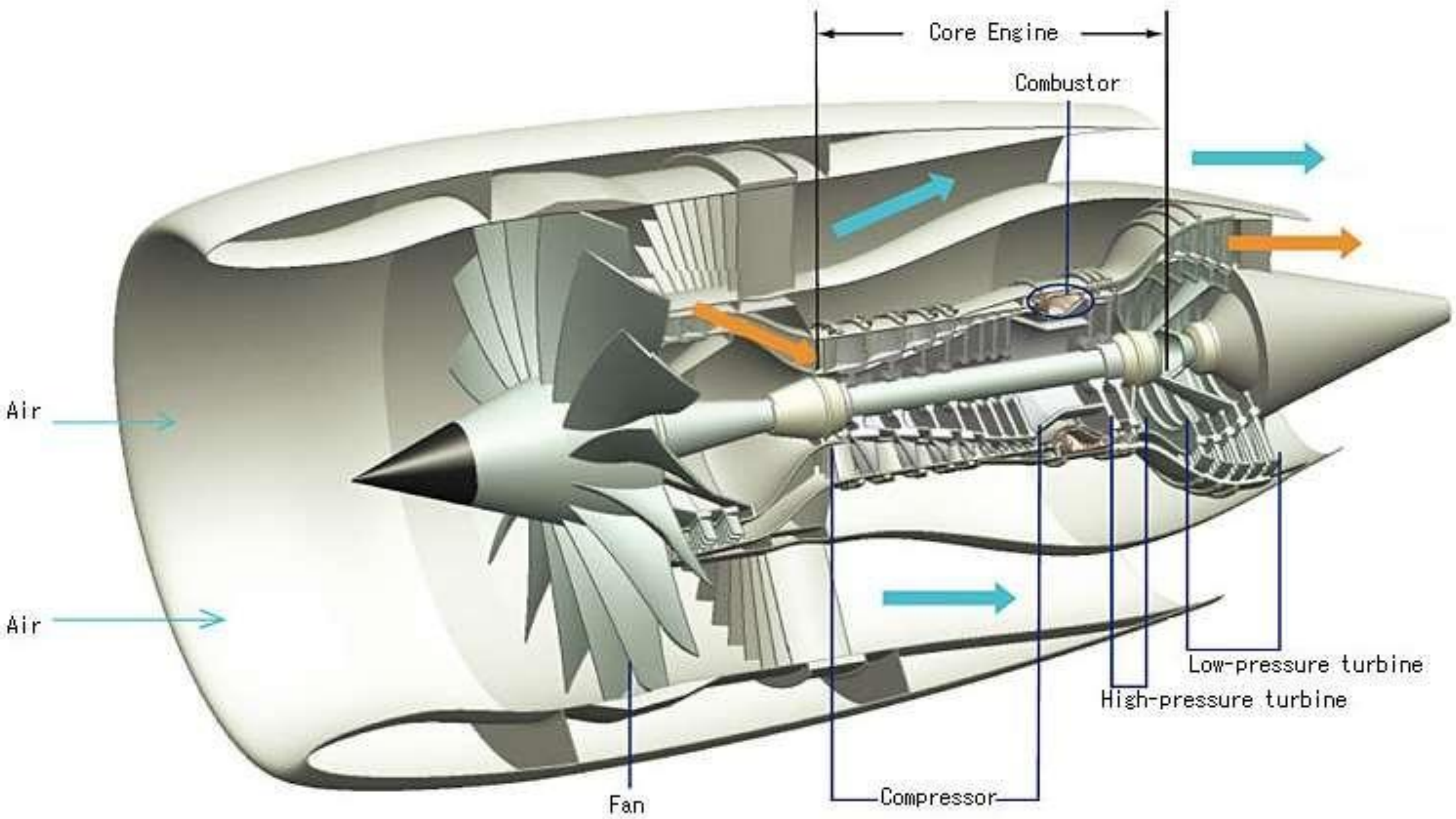
- Since the air-fuel ratio in gas turbine engines are much greater than the stoichiometric values, there is sufficient amount of air available for combustion at the turbine exit.

- There are no rotating components like a turbine in the afterburner, the temperatures can be taken to much higher values than that at turbine entry.

Variations in P, T, V and F



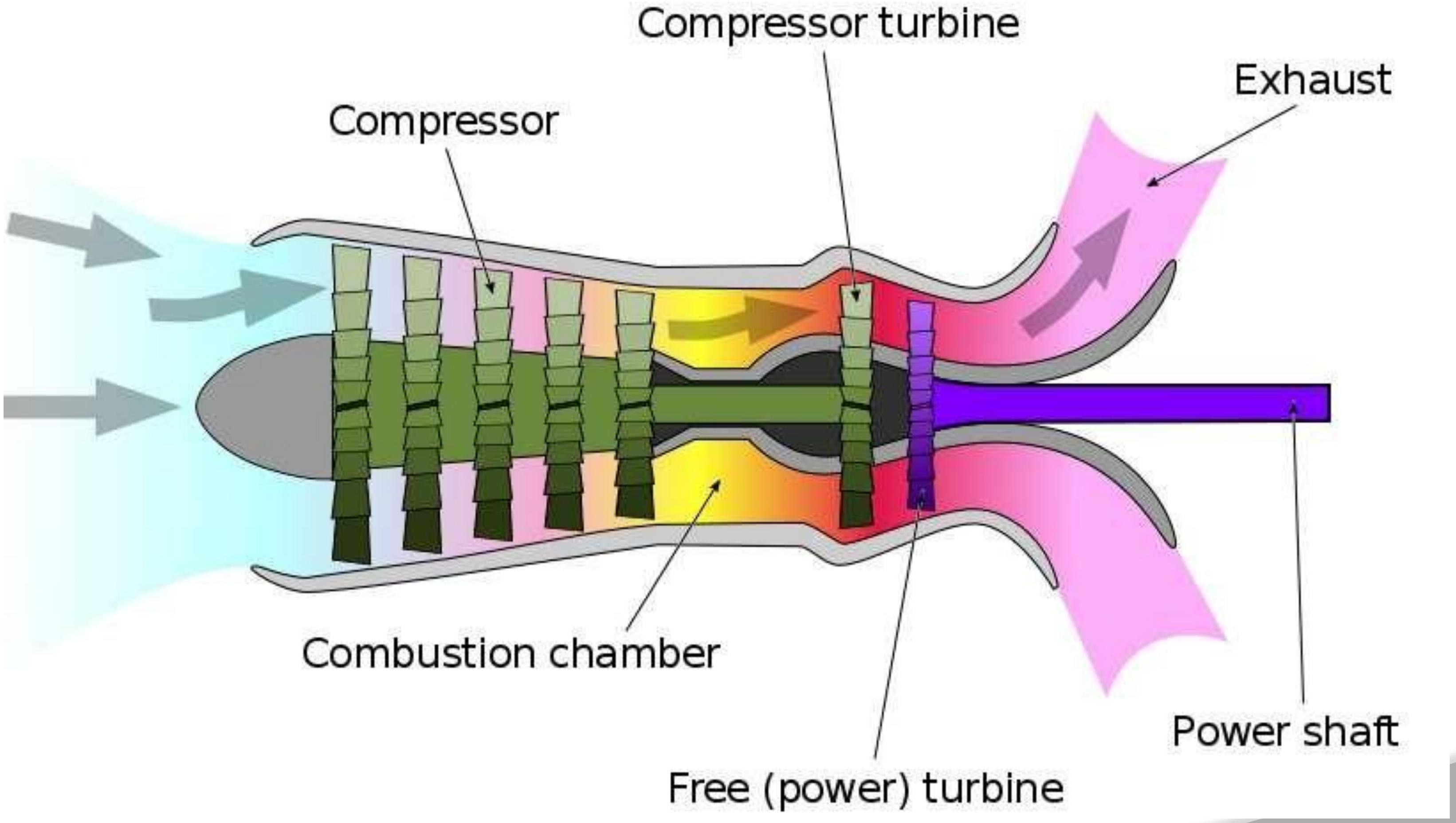
Turbo fan engine



Bypass ratio

The bypass ratio (BPR) of a turbofan engine is the ratio between the mass flow rate of the bypass stream to the mass flow rate entering the core. A 10:1 bypass ratio, for example, means that 10 kg of air passes through the bypass duct for every 1 kg of air passing through the core.

Turboshaft Engine

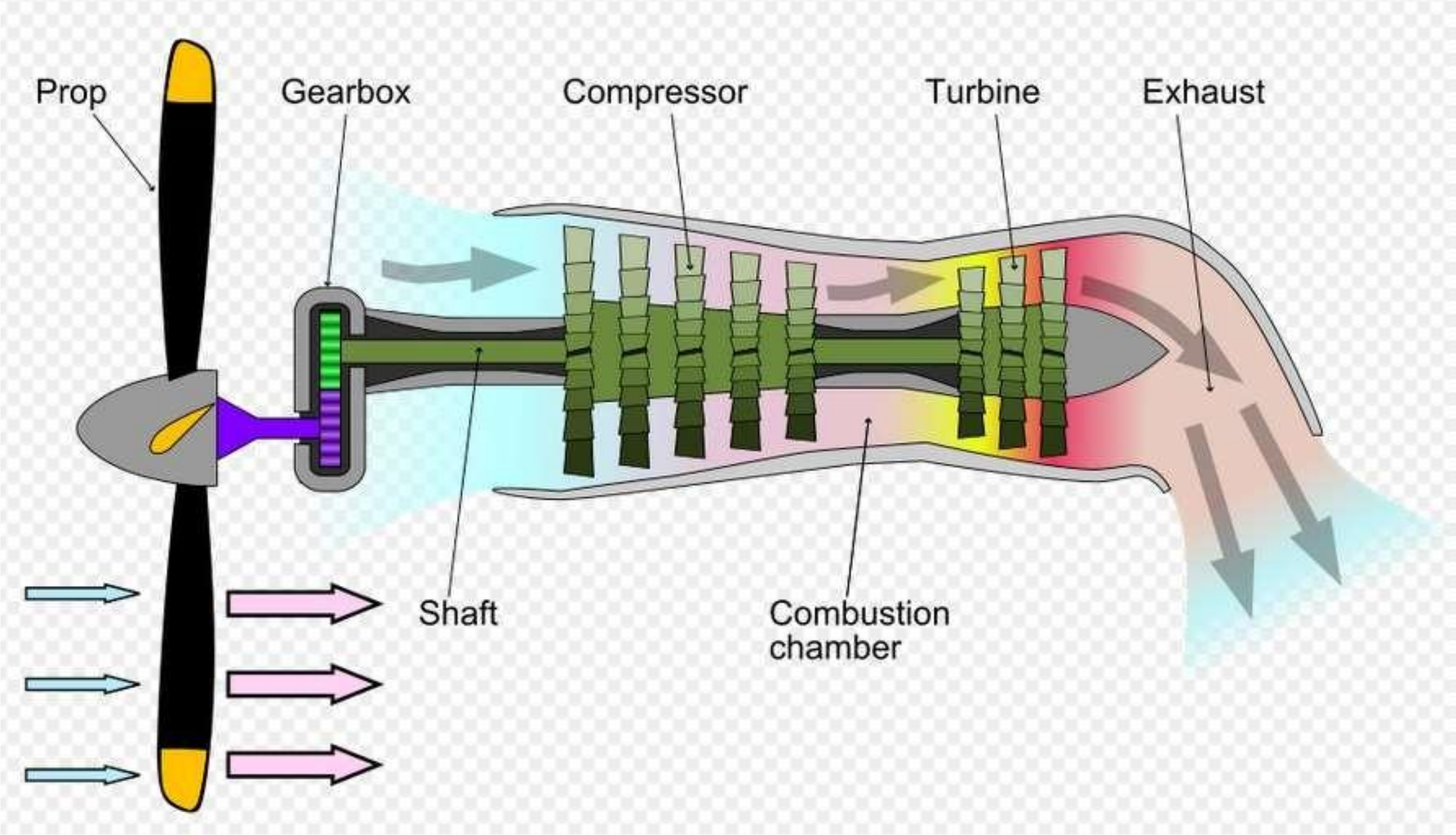


- A turboshaft engine is a form of gas turbine that is optimized to produce shaft power rather than jet thrust.
- They are even more similar to turboprops, with only minor differences.
- Turboshaft engines are commonly used in applications that require a sustained high power output, high reliability, small size, and light weight. These include helicopters, auxiliary power units, boats and ships, tanks, hovercraft, and stationary equipment.
- A turboshaft engine may be made up of two major parts assemblies: the ‘gas generator’ and the ‘power section’.

- The gas generator consists of the compressor, combustion chambers with igniter and fuel nozzles, and one or more stages of turbine. The power section consists of additional stages of turbines, a gear reduction system, and the shaft output. The gas generator creates the hot expanding gases to drive the power section. Depending on the design, the engine accessories may be driven either by the gas generator or by the power section

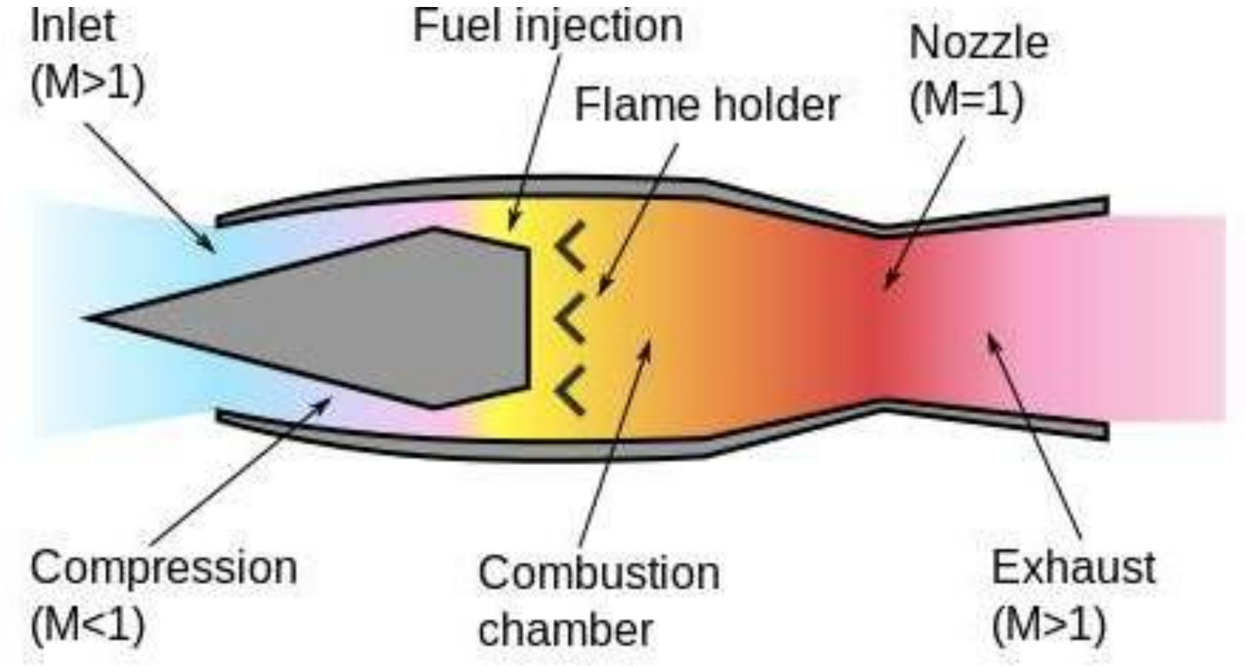
- In most designs, the gas generator and power section are mechanically separate so they can each rotate at different speeds appropriate for the conditions, referred to as a 'free power turbine‘.

Turboprop


Prop
Gearbox
Compressor
Turbine
Exhaust
Shaft
Combustion chamber

- A turboprop engine is a turbine engine that drives an aircraft propeller.
- In its simplest form a turboprop consists of an intake, compressor, combustor, turbine, and a propelling nozzle. Air is drawn into the intake and compressed by the compressor. Fuel is then added to the compressed air in the combustor, where the fuel-air mixture then combusts.
- The hot combustion gases expand through the turbine. Some of the power generated by the turbine is used to drive the compressor. The rest is transmitted through the reduction gearing to the propeller. Further expansion of the gases occurs in the propelling nozzle, where the gases exhaust to atmospheric pressure. The propelling nozzle provides a relatively small proportion of the thrust generated by a turboprop.

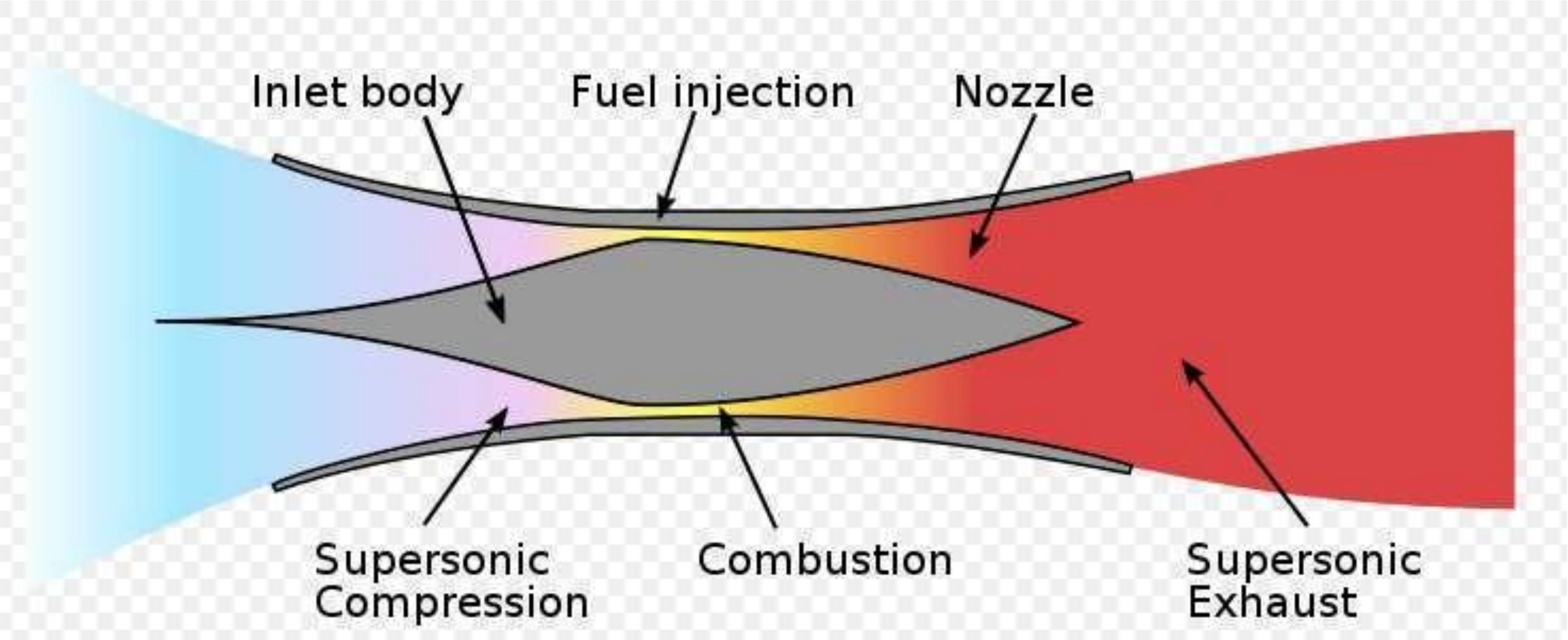
Ramjet


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

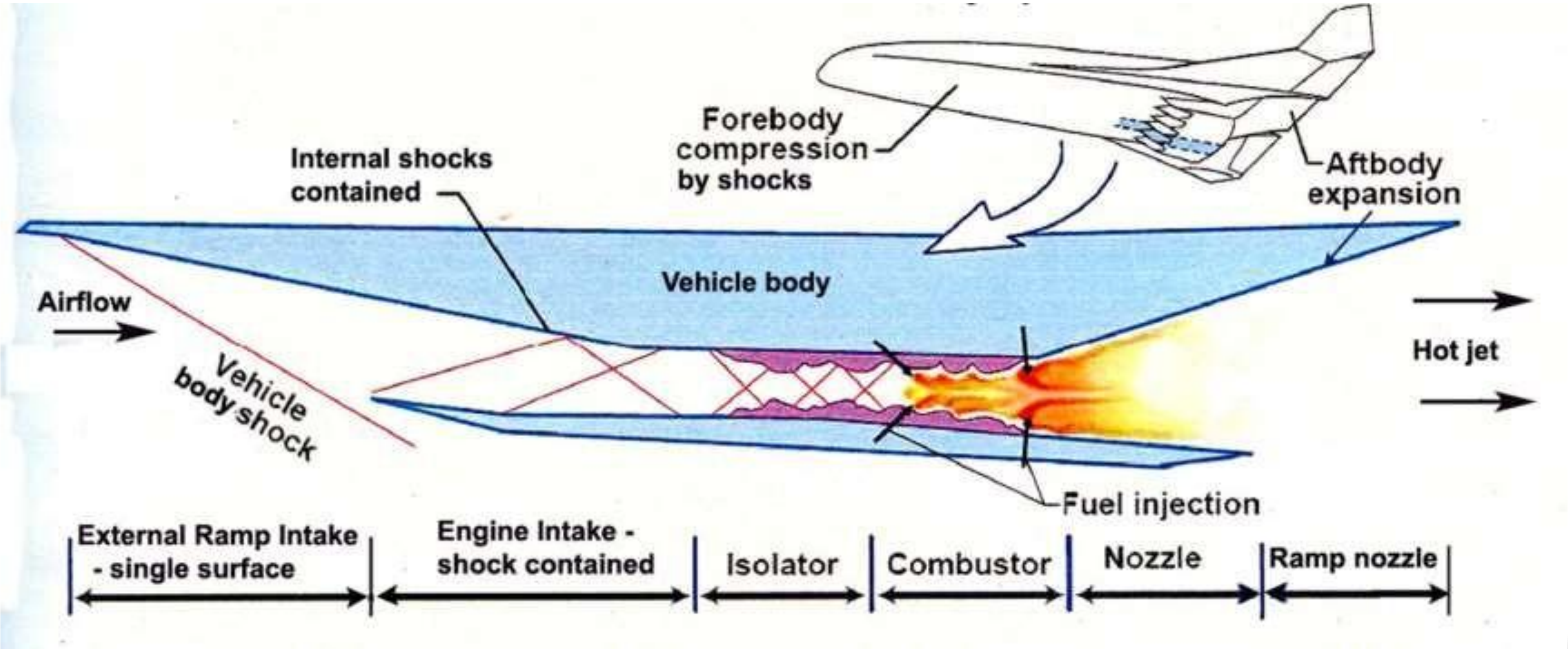
- A ramjet is a form of air-breathing jet engine that uses the engine's forward motion to compress incoming air without a compressor.
- The ramjet engine produces power by increasing the momentum of the working fluid, i.e. air.
- In contrast to the other air-breathing engines, the working cycle is done without compressor and turbine, and also without any need for enclosed combustion.
- Ramjet engine is mechanically the least complicated air-breathing jet engine for thrust production for flying vehicles.
- Ramjets apply compression to the air by ram compression at very high speeds (M>2.0).

- All the compression is done in the diffusing (ram) process.
- This restricts the use of ramjet to only supersonic speeds. No Take off, Landing possible.
- After the diffusion in Intake, fuel is injected into the stream in the combustion zone.
- The high temperature and high pressure gas is expanded through a nozzle, to a supersonic speed at the exit.
- At very high Mach numbers (>5.0) the shocks in the intake produce large losses that restricts the actual performance of the engine.

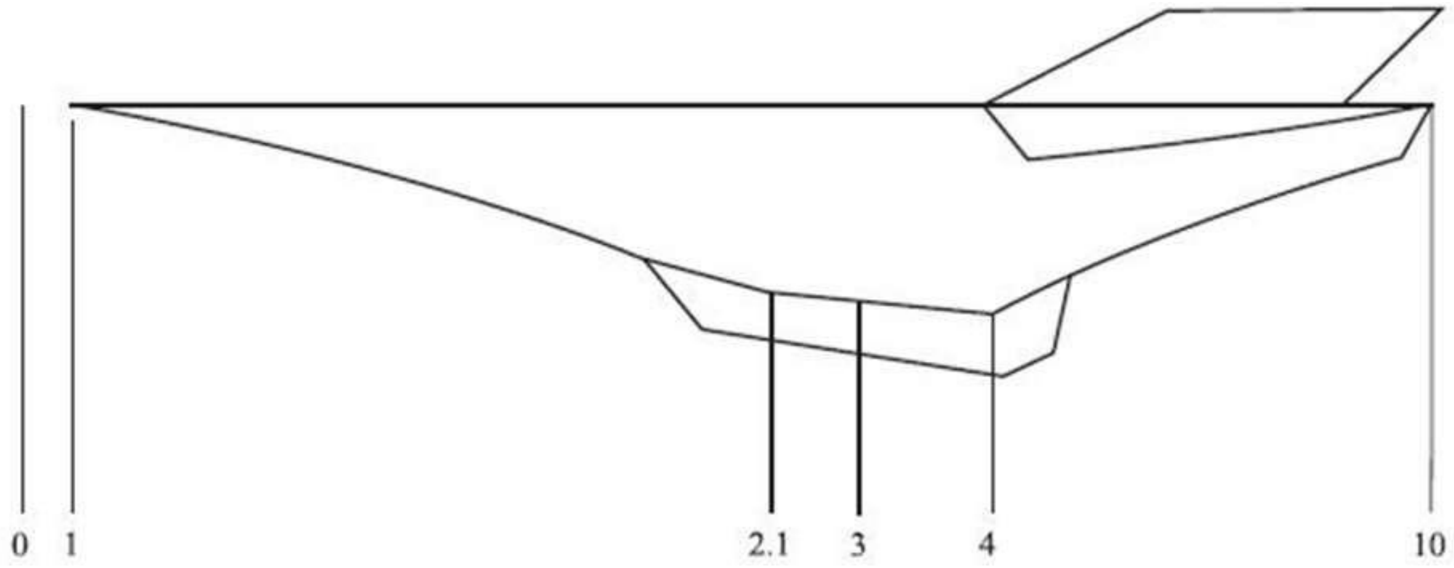
Scramjet


Inlet body
Fuel injection
Nozzle
Supersonic Compression
Combustion
Supersonic Exhaust

Scramjet


Forebody compression by shocks
Internal shocks contained
Aftbody expansion
Vehicle body
Airflow
Hot jet
Vehicle body shock
Fuel injection
External Ramp Intake - single surface
Engine Intake - shock contained
Isolator
Combustor
Nozzle
Ramp nozzle

Scramjet



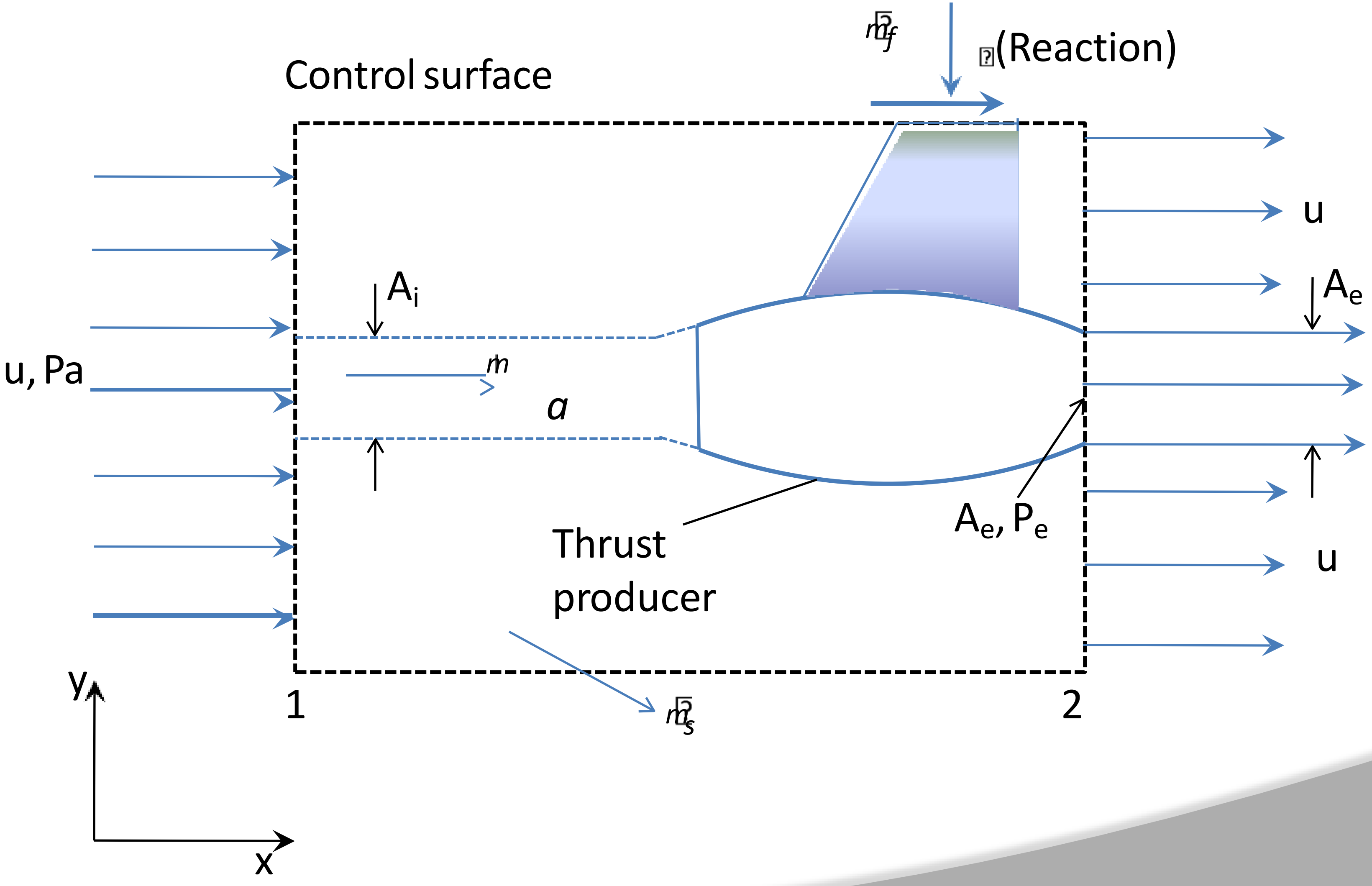
The scramjet engine belongs to the family of Brayton cycles, which consist of two adiabatic and two constant-pressure processes.

- **Station 0** represents the free-stream condition.
- **Station 1** represents the beginning of the compression process. Hypersonic shock-wave angles are small, resulting in long compression ramps (or spikes if an axisymmetric configuration is used) that, in many of the suggested configurations, begin at the vehicle's leading edge. Additional compression takes place inside the inlet duct.
- **Station 2.1** represents the entrance into the isolator section. The role of the isolator is to separate the inlet from the adverse effects of a pressure rise that is due to combustion in the combustion chamber. The presence of a shock train in the isolator further compresses the air before arriving at the combustion chamber.

- Thermodynamically the isolator is not a desirable component, because it is a source of additional pressure losses, increases the engine cooling loads, and adds to the engine weight. However, operationally it is needed to include a shock train that adjusts such that it fulfils the role just described.
- **Station 3** is the combustion chamber entrance. Unlike the turbojet engine cycle, in which the air compression ratio is controlled by the compressor settings, in a fixed-geometry scramjet the pressure at the combustion chamber entrance varies over a large range.
- **Station 4** is the combustion chamber exit and the beginning of expansion.

- **Station 10** is the exit from the nozzle; because of the large expansion ratios the entire aft part of the vehicle may be part of the engine nozzle.

The Thrust Equation



The following assumptions are made:

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

$$\dot{m}_{\mathrm{e}} u_{\mathrm{e}} - \dot{m}_{\mathrm{a}} u = -(P_{\mathrm{e}} - P_{\mathrm{a}}) A_{\mathrm{e}} + \tau$$

$$\therefore \tau = \dot{m}_{\mathrm{a}} [(1+f)\, u_{\mathrm{e}} - u] + (P_{\mathrm{e}} - P_{\mathrm{a}}) A_{\mathrm{e}}$$

Engine Performance Parameters

- The engine performance is described by different efficiency definitions, thrust and the fuel consumption.
- The efficiency definitions that we shall now be discussing are applicable to an engine with a single propellant stream (turbojets or ramjets).
- For other types of jet engines (turbofan, turboprop) the equations need to be appropriately modified.

Propulsion efficiency: The ratio of thrust power to the rate of production of propellant kinetic energy.

$$\eta_P = \frac{\Im u}{\dot{m}_a\left[(1+f)(u_e^2/2) - u^2/2\right]}$$

If we assume that f«1 and the pressure thrust term is negligible

$$\eta_P = \frac{(u_e - u)u}{u_e^2/2 - u^2/2} = \frac{2u/u_e}{1+u/u_e}$$

Thermal efficiency: The ratio of the rate of production of propellant kinetic energy to the total energy consumption rate

$$\eta_{th} = \frac{\dot{m}_a\left[(1+f)(u_e^2/2) - u^2/2\right]}{\dot{m}_f Q_R} = \frac{\left[(1+f)(u_e^2/2) - u^2/2\right]}{f Q_R}$$

For a turboprop or turboshaft engine, the output is largely shaft power. In this case,

$$\eta_{th} = \frac{P_s}{\dot{m}_f Q_R}$$

Overall efficiency: The product of thermal efficiency and propulsion efficiency.

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

In the case of aircraft that generate thrust using propellers

$$\eta_o = \eta_{pr} \eta_{th}$$

Thrust specific fuel consumption, TSFC

$$TSFC = \frac{\dot{m}_f}{\Im} \approx \frac{\dot{m}_f}{\dot{m}_a\left[(1+f)u_e - u\right]}$$

For turbine engines that produce shaft power, brake specific fuel consumption, BSFC

$$BSFC = \frac{\dot{m}_f}{P_s}$$

UNIT - II

AIRCRAFT ENGINE INLETS, EXHAUST NOZZLES, COMBUSTORS AND AFTERBURNERS

UNIT - II

CLO's	Course Learning Outcomes
CLO4	Describe operational modes of subsonic inlets and parameters influencing it.
CLO5	Understand different types of combustion chamber and functions of all the components.
CLO6	Describe supersonic inlets, starting problem in it and their operating modes.

Introduction

While the Gas Generator, composed of the compressor, combustor and turbine, is the heart of any gas turbine engine, the overall performance of the propulsion system is strongly influenced by the inlet and the nozzle.

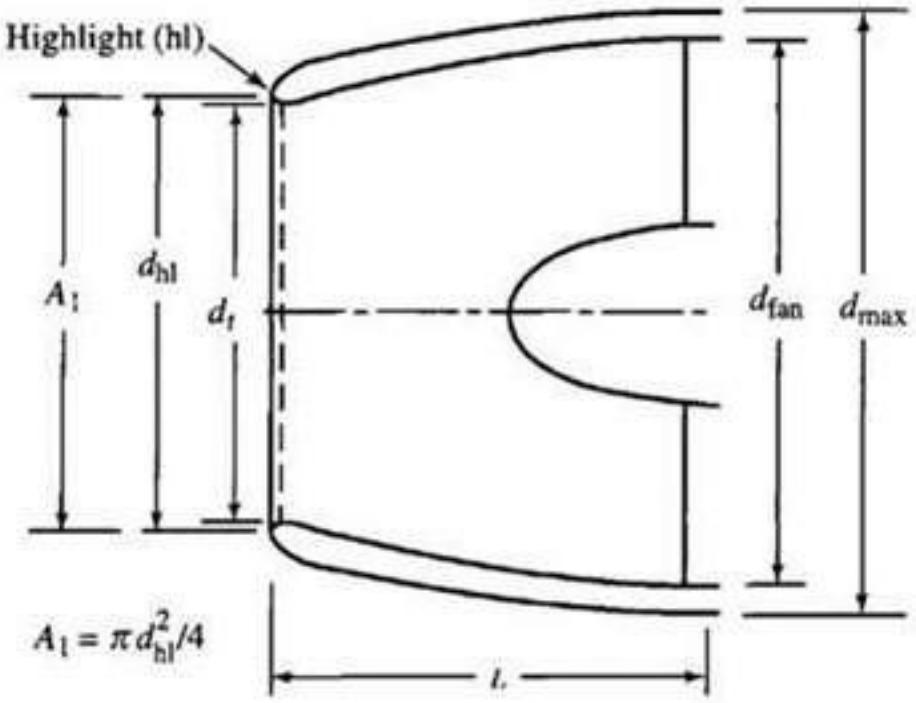
This is especially true for high Mach flight, when a major portion of the overall temperature and pressure rise of the cycle are in the inlet, and a correspondingly large part of the expansion in the nozzle. So it is important to understand how these components function and how they limit the performance of the propulsion system.

Subsonic Inlets

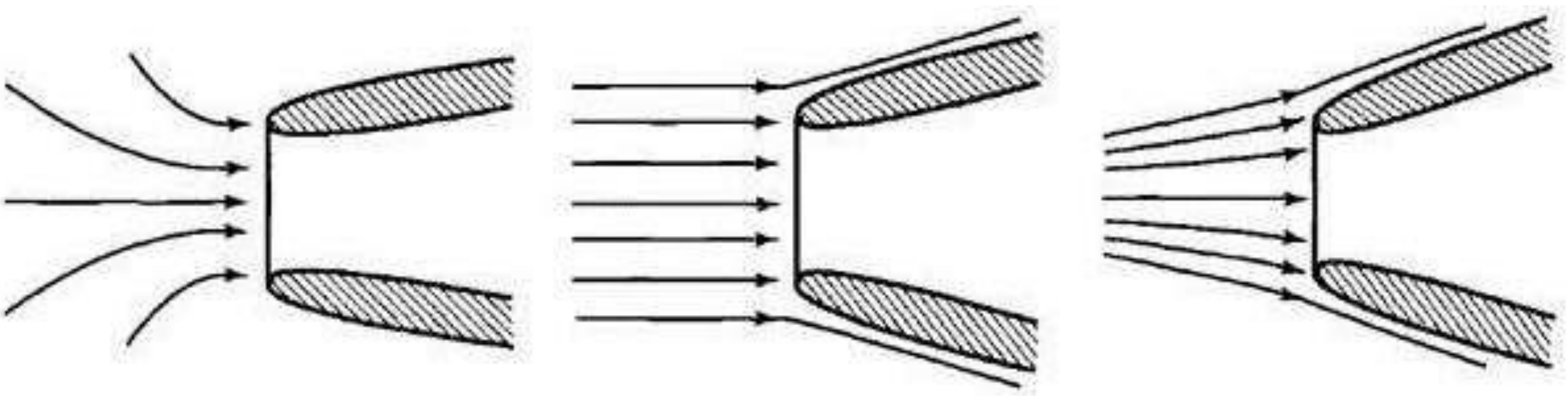
The subsonic inlet must satisfy two basic requirements:

a) Diffusion of the free-stream flow to the compressor inlet condition at cruise.

b) Acceleration of static air to the compressor inlet condition at take-off



Subsonic Inlet Nomenclature



a) Static operation (or insufficient air for engine)

b) Low-speed operation (correct engine air)

c) High-speed operation (more air than the engine needs)

The operating conditions of an inlet depend on the flight velocity and mass flow demanded by the engine

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

1. Inlet total pressure ratio and drag at cruise
2. Engine location on wing or fuselage (avoidance of foreign-object damage, inlet flow up-wash and downwash, exhaust gas reinjection, ground clearance)
3. Aircraft attitude envelope (angle of attack, yaw angle, cross-wind takeoff)
4. Inlet total pressure ratio and distortion levels required for engine operation
5. Engine-out wind milling airflow and drag (nacelle and engine)
6. Integration of diffuser and fan flow path contour

7. Integration of external nacelle contour with thrust reverser and accessories
8. Flow field interaction between nacelle and wing
9. Noise suppression requirements

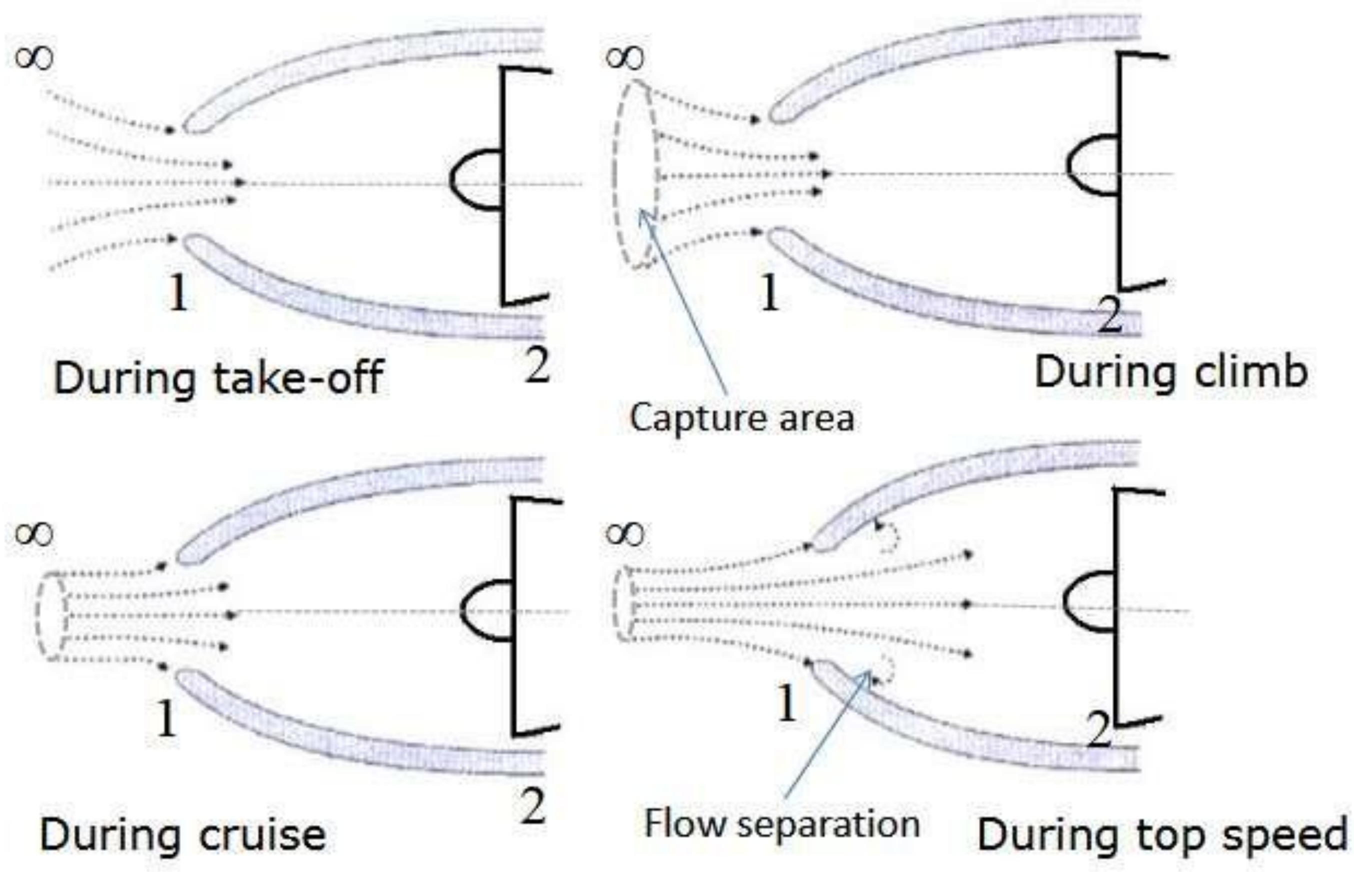
Intake Performance

- Intake operation varies tremendously over the operating range of the engine. During take-off the engine requires high mass flow, but is operating at a lower speed.

- A typical fixed geometry intake may have problems delivering this mass flow.

- The intake design must ensure that under these extreme operating conditions too the intake performance is not drastically affected.

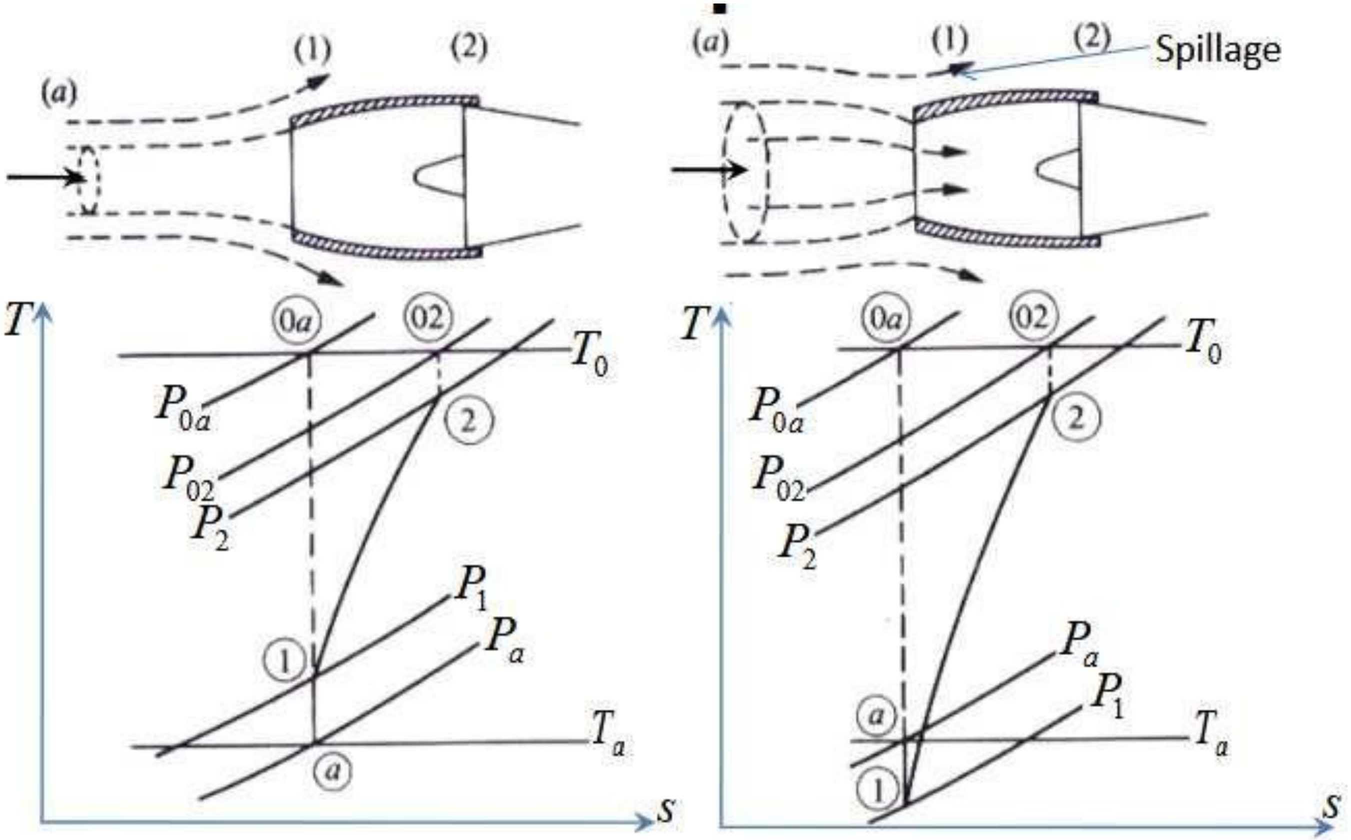
Subsonic Intake Performance



Intake during Take-Off, Climb, Cruise and Top Speed

Subsonic Intake Performance

- The compression in a subsonic intake consists of two components:–Pre-entry compression or external compression– Internal compression or the compression in the diffuser
- Pre-entry compression is always isentropic, whereas internal compression is not.
- However trying to maximize pre-entry compression may result in boundary layer separation within the internal compression.
- Designers try to optimize between external and internal compression.



High speed/low mass flow

Low speed/high mass flow

Temperature and Entropy profile during Top Speed and Take-Off

Flow separation can get initiated at three possible locations

- External to the intake on the nacelle
- Within the diffuser internal surface
- On the centre body or the hub

Separation on the nacelle would lead to increase in overall drag of the aircraft

Separation within the diffuser geometry may lead to higher stagnation pressure losses and therefore lower diffuser efficiency.

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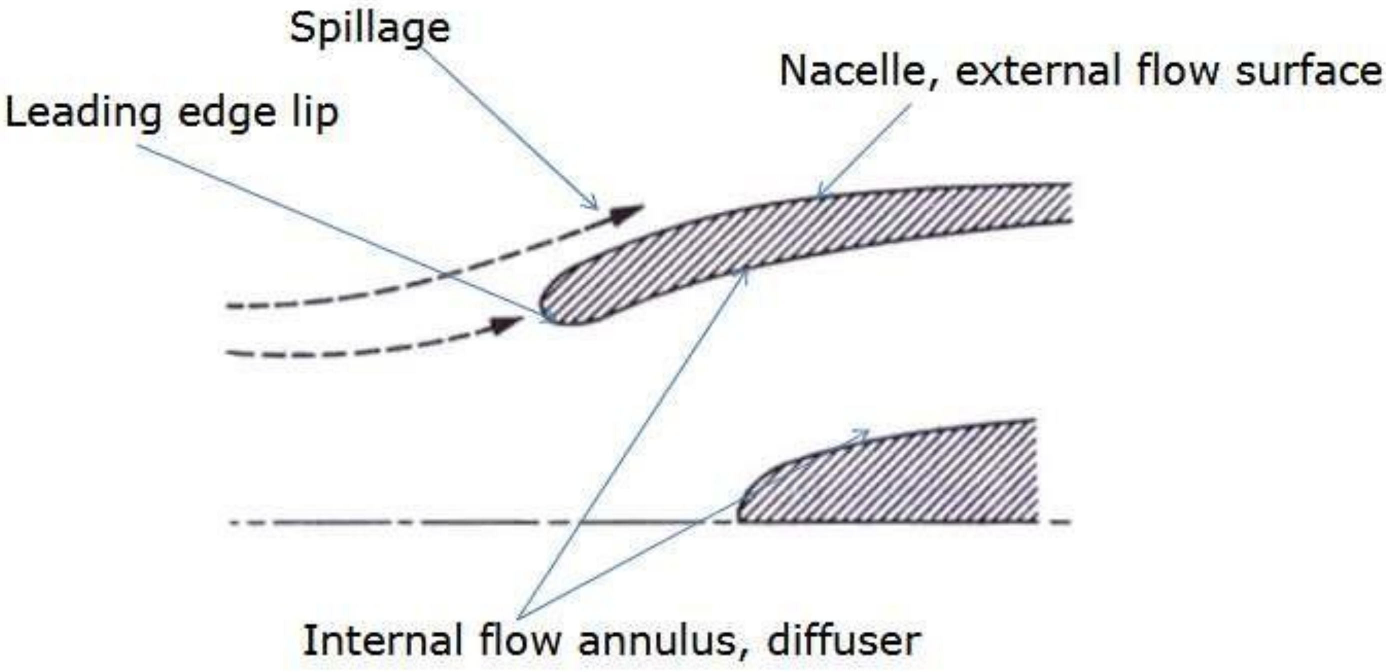


Flow Separation at Nacelle

Spillage:

- Occurs when the incoming stream tube (capture area) is different from the intake entry area
- Leads to increased drag
- May also lead to separation on the cowl

External deceleration

- Devoid of losses (Pre-entry compression)
- Sensitive to operating condition

Trade-off between external and internal deceleration

Performance of intakes

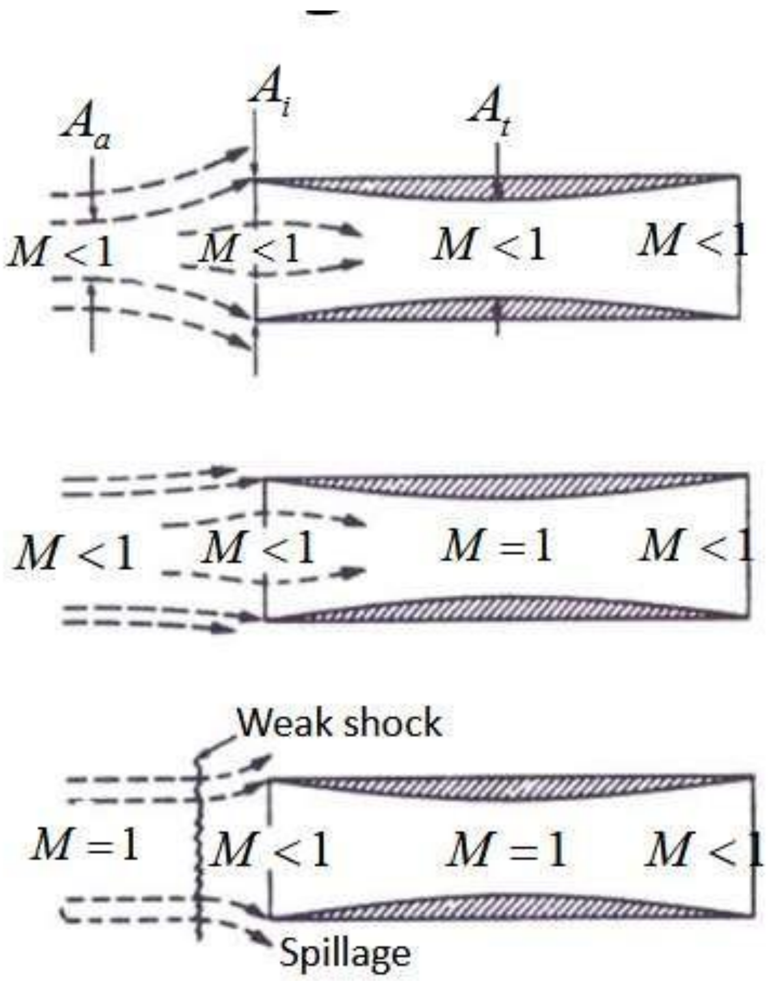
Subsonic as well as supersonic are evaluated using the following:

- Isentropic efficiency
- Stagnation pressure ratio or pressure recovery
- Distortion coefficient

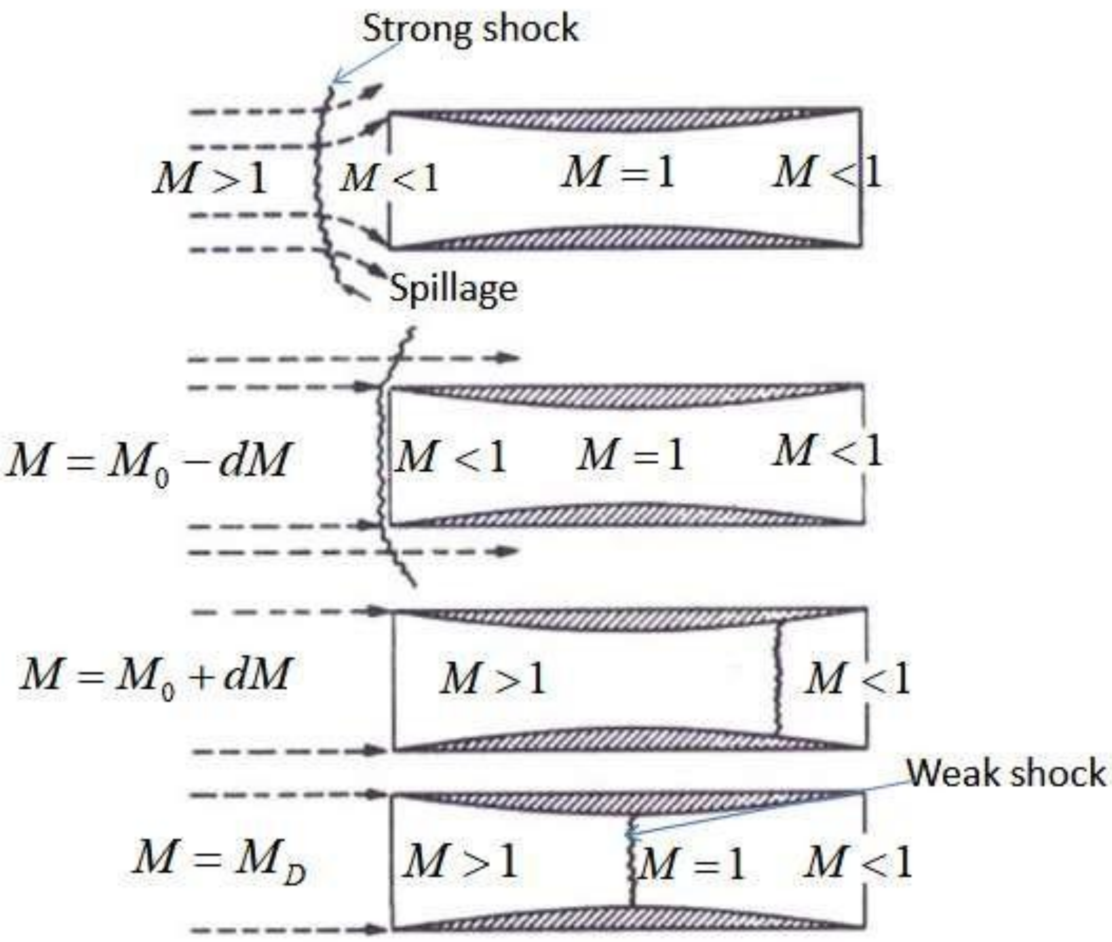
Supersonic diffusers are characterised by the presence of shocks.

- However before the intake operates in a supersonic flow, it must pass through the subsonic flow regime.
- In some types of supersonic intakes, establishing a shock system with minimal losses is not easy.
- The process of establishing a stable shock system is referred to as Starting of an intake.

Starting Of An Intake



Starting Of An Intake



Supersonic Intake Performance

- External compression intakes complete the supersonic diffusion outside the covered portion of the intake.
- These intakes usually have one or more oblique shocks followed by a normal shock.
- Depending upon the location of these shocks, the intake may operate in subcritical, critical or supercritical modes

Mode Of Operation

Subcritical:

- At Mach numbers below the design value
- The normal shock occurs ahead of the cowl lip
- High external drag due to spillage

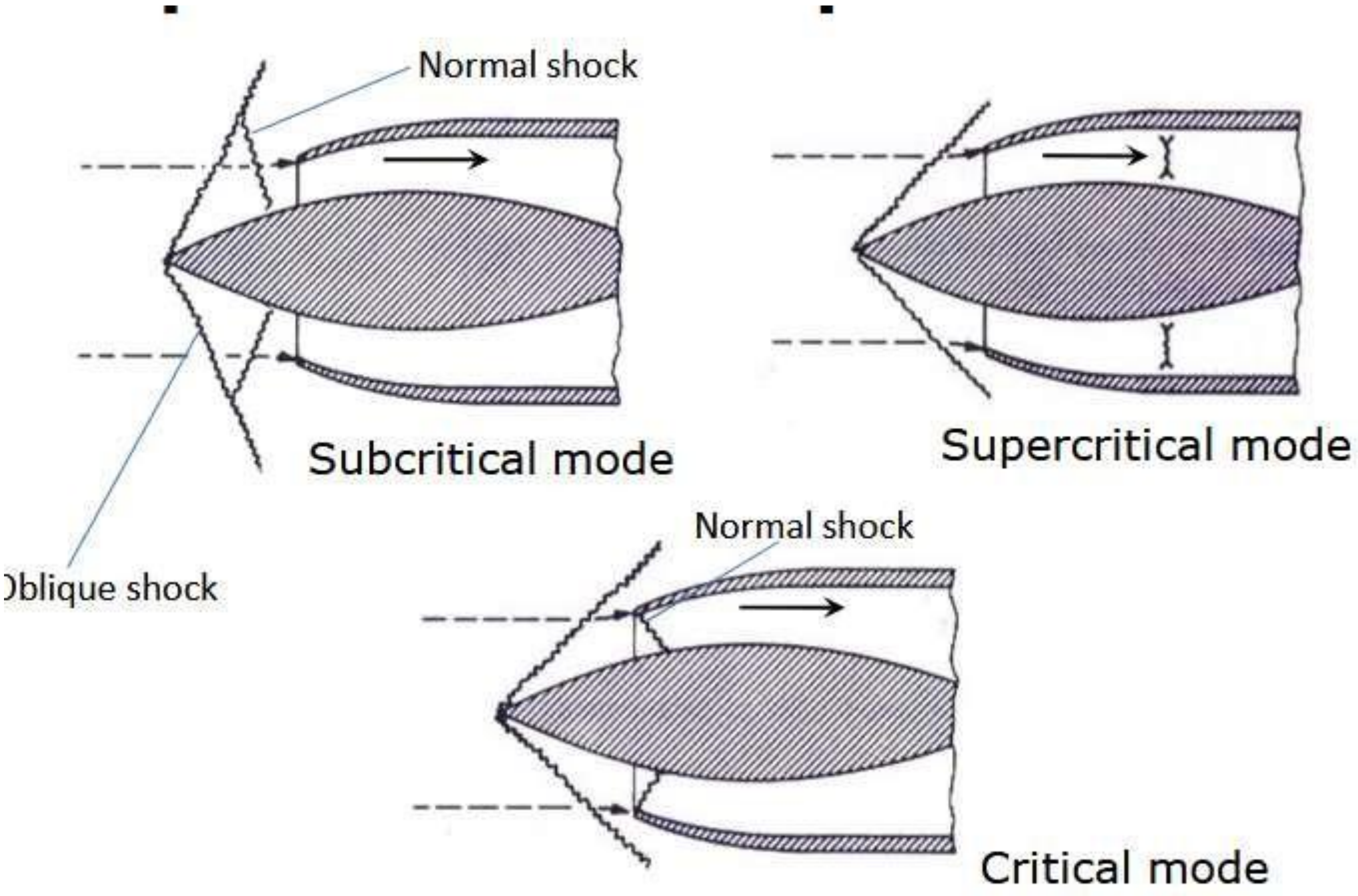
Supercritical:

- Occurs at same mass flow as critical mode
- Higher losses as the normal shock occurs in a region of higher Mach number

Critical:

- Design point operation
- The normal shock is located exactly at the cowl lip

Mode Of Operation



Mode Of Operation

- Total pressure losses are highest in the case of a diffuser with a single normal shock
- A number of oblique shocks followed by a normal shock would lead to lower total pressure losses
- Oblique shocks are generated using steps in the centre body
- A diffuser with a smoothly contoured centre body may have infinite oblique shocks: Isentropic external diffuser

Combustion Chamber

- Combustion systems of aircraft gas turbine engines largely encompass the main burners (also called burners or combustors) and afterburners (also called augmenters or re-heaters)

- The thermal energy of the air/fuel mixture (reactants) flowing through an air-breathing engine is increased by the combustion process. The fuel must be vaporized and mixed with the air before this chemical reaction can occur. Once this is done, the combustion process can occur and thus increase the thermal energy of the mixture (products of combustion)

The following properties of the combustion chambers are desired

- Complete combustion
- Low total pressure loss
- Stability of combustion process
- Proper temperature distribution at exit with no "hot spots"
- Short length and small cross section
- Freedom from flameout
- Re-lightability
- Operation over a wide range of mass flow rates, pressures, and temperatures

Combustion Stability

- The ability of the combustion process to sustain itself in a continuous manner is called *combustion stability.*

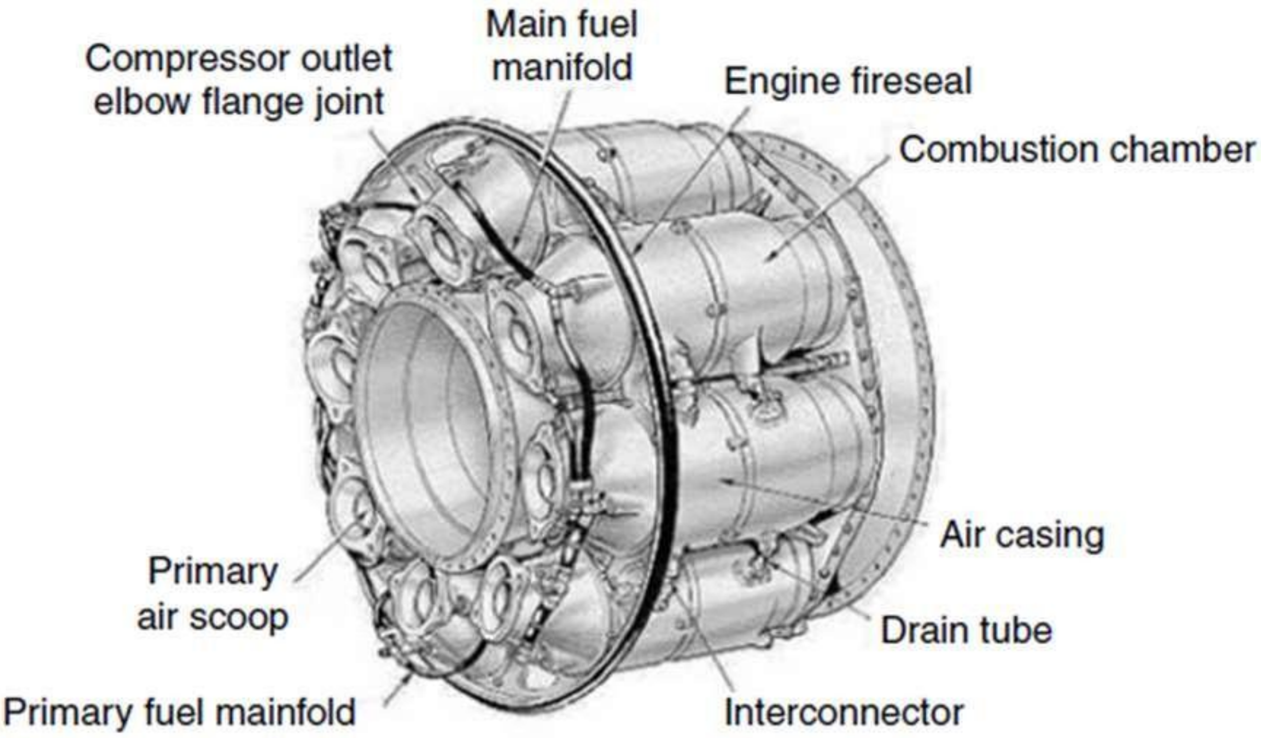
- Stable, efficient combustion can be upset by the fuel/air mixture becoming too lean or too rich such that the temperatures and reaction rates drop below the level necessary to effectively heat and vaporize the incoming fuel and air.

Types Of Chambers

- Turbine engine burners have undergone continuing development over the past 50 years, resulting in the evolution of a variety of basic combustor configurations.

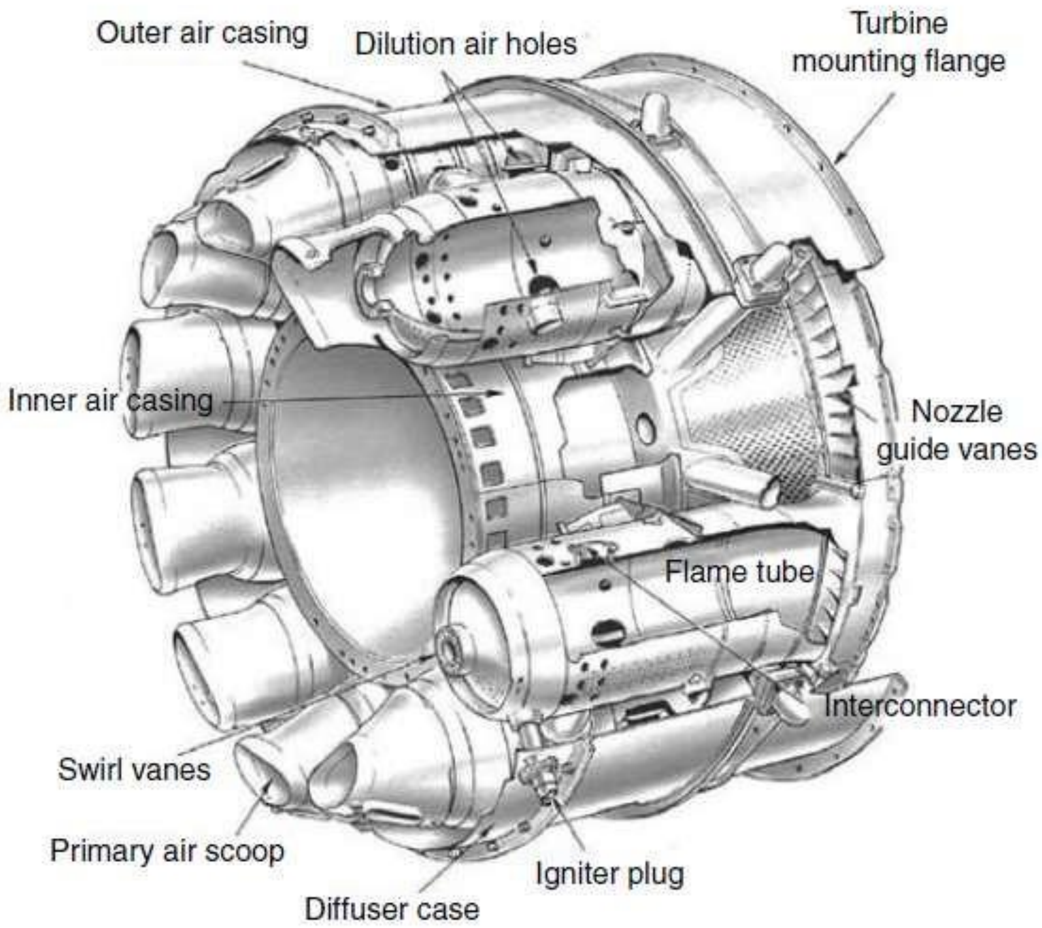
- Contemporary main burner systems may be broadly classified into one of the three types schematically illustrated in figure below: can, can annular, or annular.

CAN

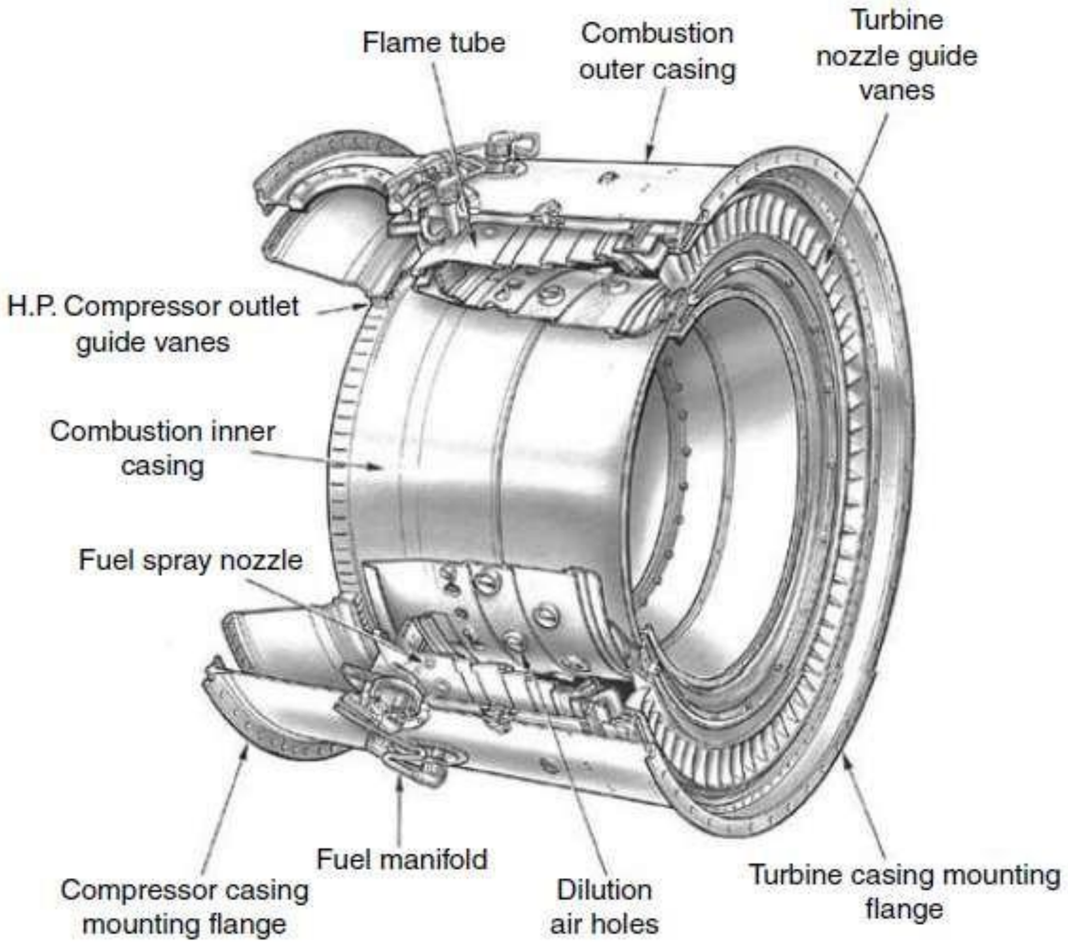


Mutiple can

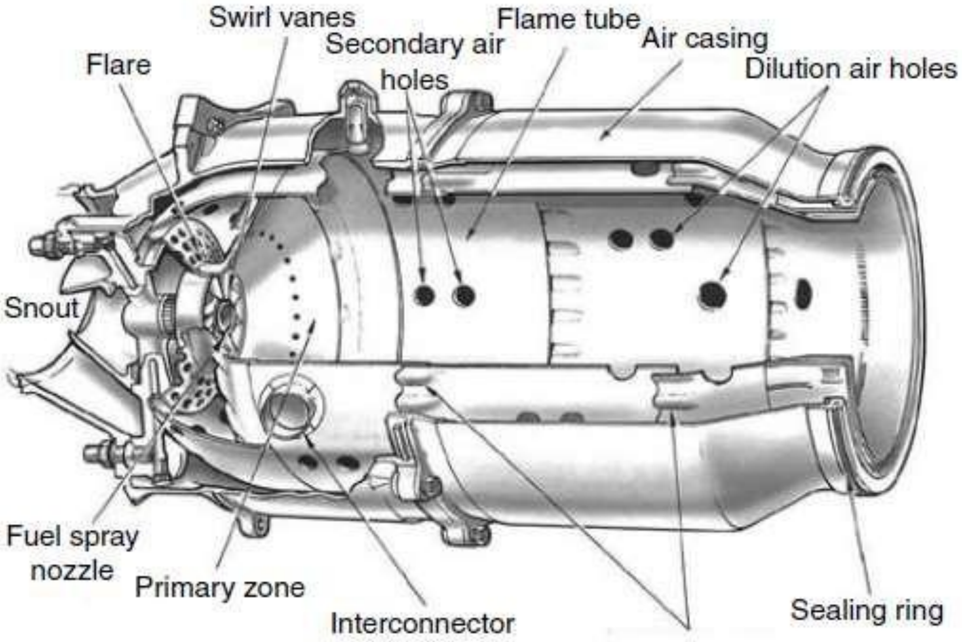
Cannular

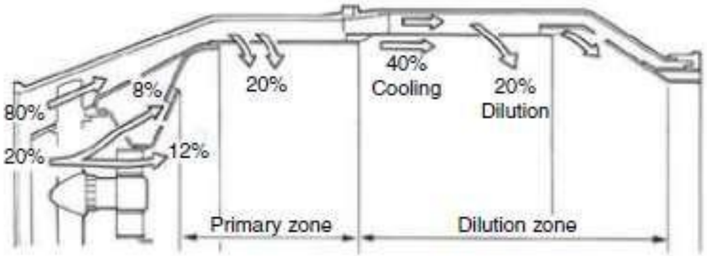

Outer air casing
Dilution air holes
Turbine mounting flange
Inner air casing
Nozzle guide vanes
Flame tube
Interconnector
Swirl vanes
Primary air scoop
Igniter plug
Diffuser case

Annular


Flame tube
Combustion outer casing
Turbine nozzle guide vanes
H.P. Compressor outlet guide vanes
Combustion inner casing
Fuel spray nozzle
Compressor casing mounting flange
Fuel manifold
Dilution air holes
Turbine casing mounting flange

Cut Section Of Can Type Combustor





Combustion Chamber Performance

A combustion chamber must be capable of allowing fuel to burn efficiently over a wide range of operating conditions without incurring a large pressure loss. So, the combustion chamber performance can be evaluated by some conditions or performance as follows:

1. Pressure loss
2. Combustion efficiency
3. Combustion stability
4. Combustion intensity

Pressure Losses

The sources of pressure drop or loss are either cold or hot losses. Cold losses arise from sudden expansion, wall friction, turbulent dissipation, and mixing. Cold losses can be measured by flowing air without fuel through all the slots, holes, orifices, and so on.

$$\text{Pressure loss factor (PLF)} = \frac{\Delta P_0}{m^2/\left(2\rho_1 A_{\mathrm{m}}^2\right)} = \overset{\text{Cold}}{\overline{K_1}} + \overset{\text{Hot}}{\overline{K_2\left(\frac{T_{02}}{T_{01}} - 1\right)}}$$

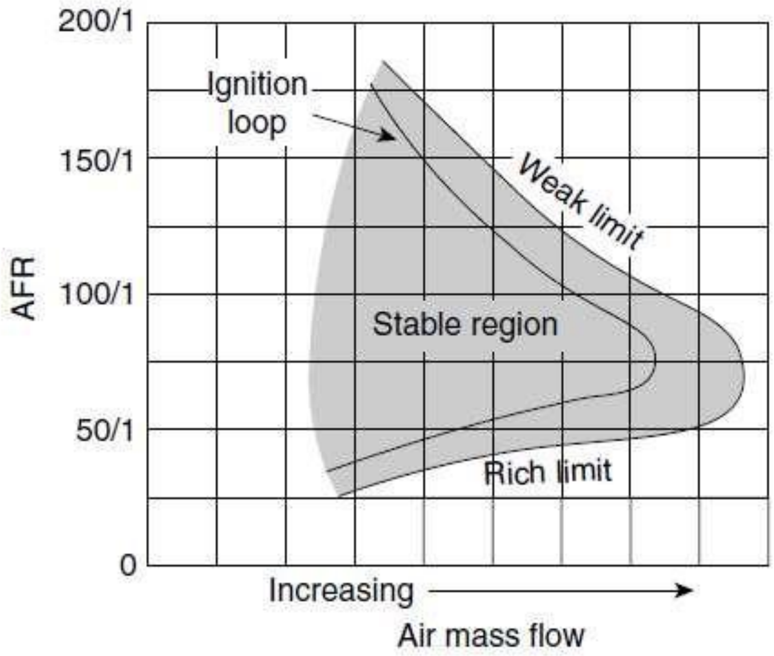
Combustion Efficiency

The main objective of the combustor is to transfer all the energy of the fuel to the gas stream. In practice this will not occur for many reasons; for example, some of the fuel may not find oxygen for combustion in the very short time available. Therefore, it is necessary to define the efficiency of the combustion.

$$\eta_b = f(\text{airflow rate})^{-1}\left(\frac{1}{\text{evaporation rate}} + \frac{1}{\text{mixing rate}} + \frac{1}{\text{reaction rate}}\right)^{-1}$$

Combustion Stability

Combustion stability means smooth burning and the ability of the flame to remain alight over a wide operating range. For any particular type of combustion chamber there is both a rich and a weak limit to the AFR beyond which the flame is extinguished.



Combustion Intensity

The heat released by a combustion chamber is dependent on the volume of the combustion area. So, the combustion area must be increased. Therefore, the average velocities increasing in the combustor will make efficient burning more and more difficult.

$$\mathrm{CI} = \frac{\text{heat release rate}}{\text{combustion volume} \times \text{pressure}} \quad \mathrm{kW/(m^3 atm)}$$

The exhaust nozzles may be classified as

1. Convergent or C-D types
2. Axisymmetric or two-dimensional types
3. Fixed geometry or variable geometry types.

The simplest form is the fixed geometry convergent type, as no moving parts and control mechanisms

are needed. It is found in subsonic commercial aircraft

Requirements

1. Be matched to other engine components for all engine-operating conditions
2. Provide the optimum expansion ratio
3. Have minimum losses at design and off-design conditions
4. Permit afterburner operation (if available) without affecting main engine operation
5. Allow for cooling of walls if necessary
6. Provide reversed thrust when necessary
7. Suppress jet noise and infrared radiation (IR) if desired
8. Provide necessary manoeuvring for military aircraft fitted with thrust vectoring systems
9. Do all the above with minimal cost, weight, and boat tail drag while meeting life and reliability goals.

The mass flow rate m may be determined in terms of the local area from the relation

$$\dot{m} = \rho u A = \left(\frac{P}{RT}\right)\left(M\sqrt{\gamma RT}\right)A = (MA)\left(\frac{P}{P_0}\right)P_0\frac{\sqrt{\gamma}}{\sqrt{RT}}\sqrt{\frac{T_0}{T_0}}$$

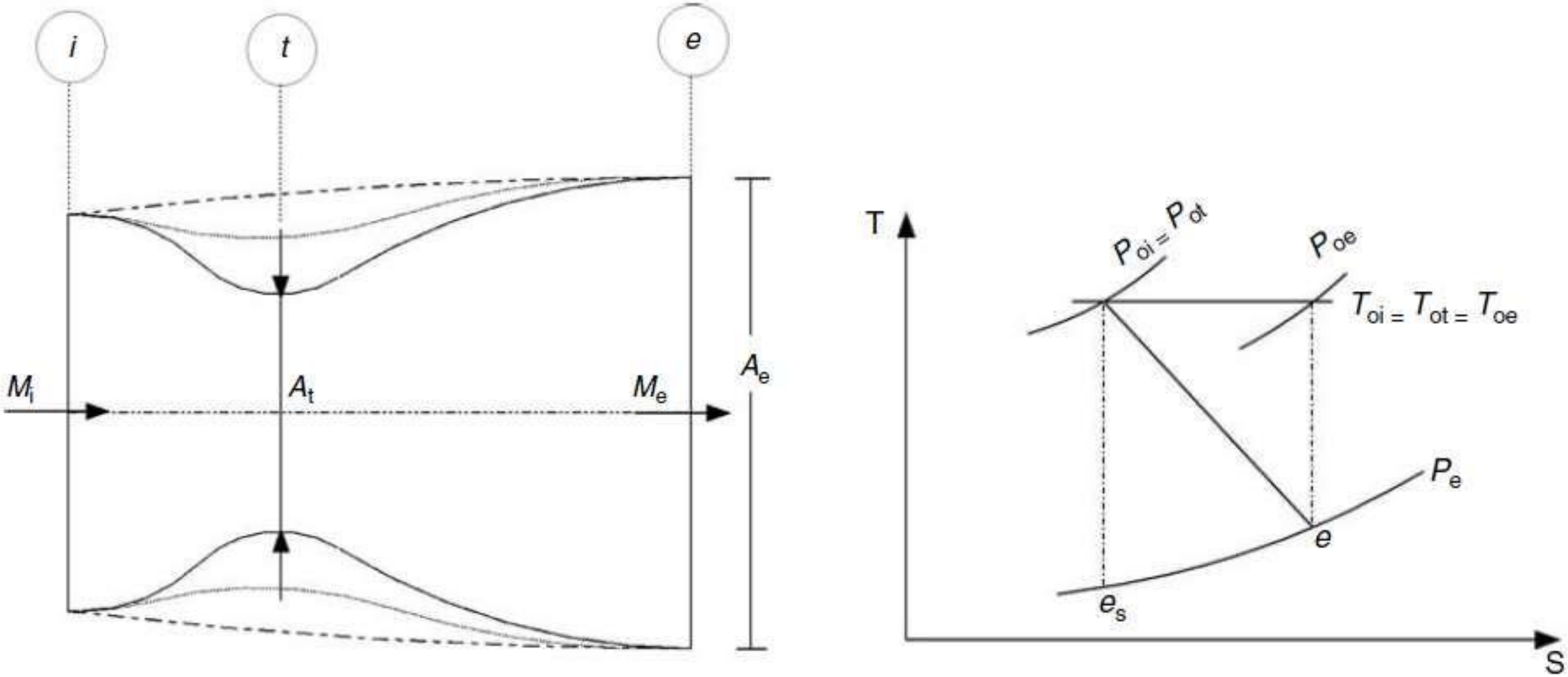
$$\dot{m} = \frac{\sqrt{\gamma}P_0}{\sqrt{T_0 R}}MA\frac{\left\{1+((\gamma-1)/2)\,M^2\right\}^{1/2}}{\left\{1+((\gamma-1)/2)\,M^2\right\}^{\gamma/(\gamma-1)}}$$

$$\dot{m} = \frac{AP_0}{\sqrt{T_0}}\sqrt{\frac{\gamma}{R}}\frac{M}{\left\{1+((\gamma-1)/2)\,M^2\right\}^{(\gamma+1)/2(\gamma-1)}}$$

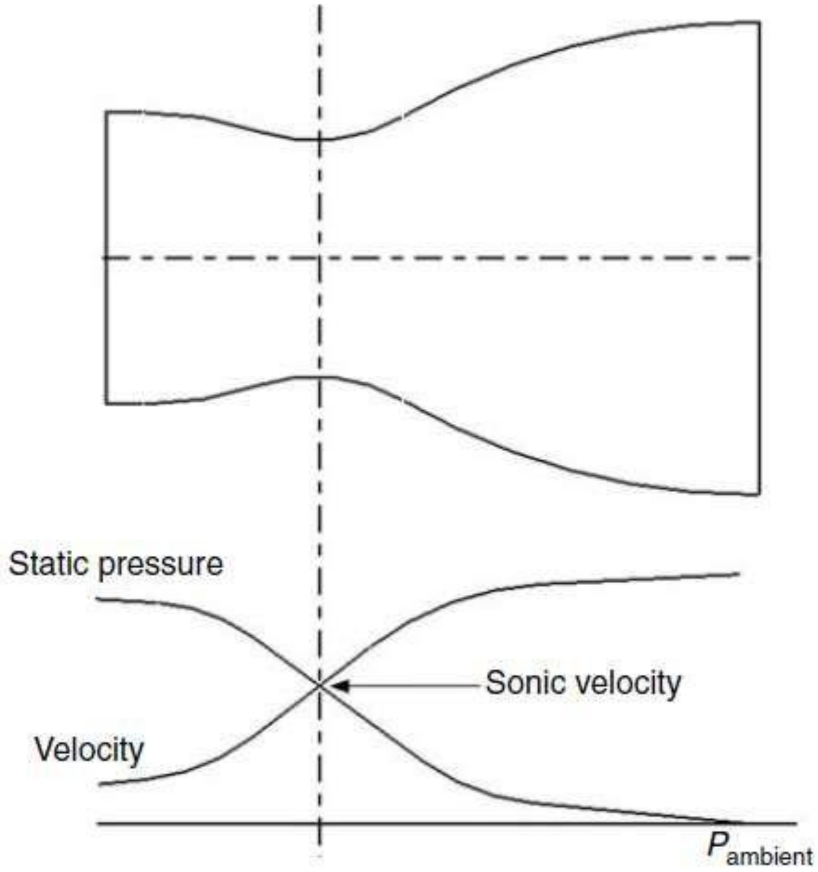
Convergent–Divergent Nozzle

- The flow in nozzles can be assumed adiabatic as the heat transfer per unit mass of fluid is much smaller than the difference in enthalpy between inlet and exit.
- Figure illustrates a C-D nozzle together with its T-S diagram. Three states are identified, namely, inlet (i), throat (t), and exit (e).
- Flow is assumed to be isentropic from the inlet section up to the throat, thus $P_{0i} = P_{0t}$, $T_{0i} = T_{0t}$ .Flow from the throat and up to the exit is assumed to be adiabatic but irreversible due to possible boundary layer separation, thus $P_{0t} > P_{0e}$ but $T_{0t} = T_{0e}$.
- Gases expand to the static pressure Pe with an adiabatic efficiency η_n in the range from 0.95 to 0.98.

Flow Through C-D Nozzle



Velocity And Static Pressure Distribution In Convergent–Divergent Nozzle

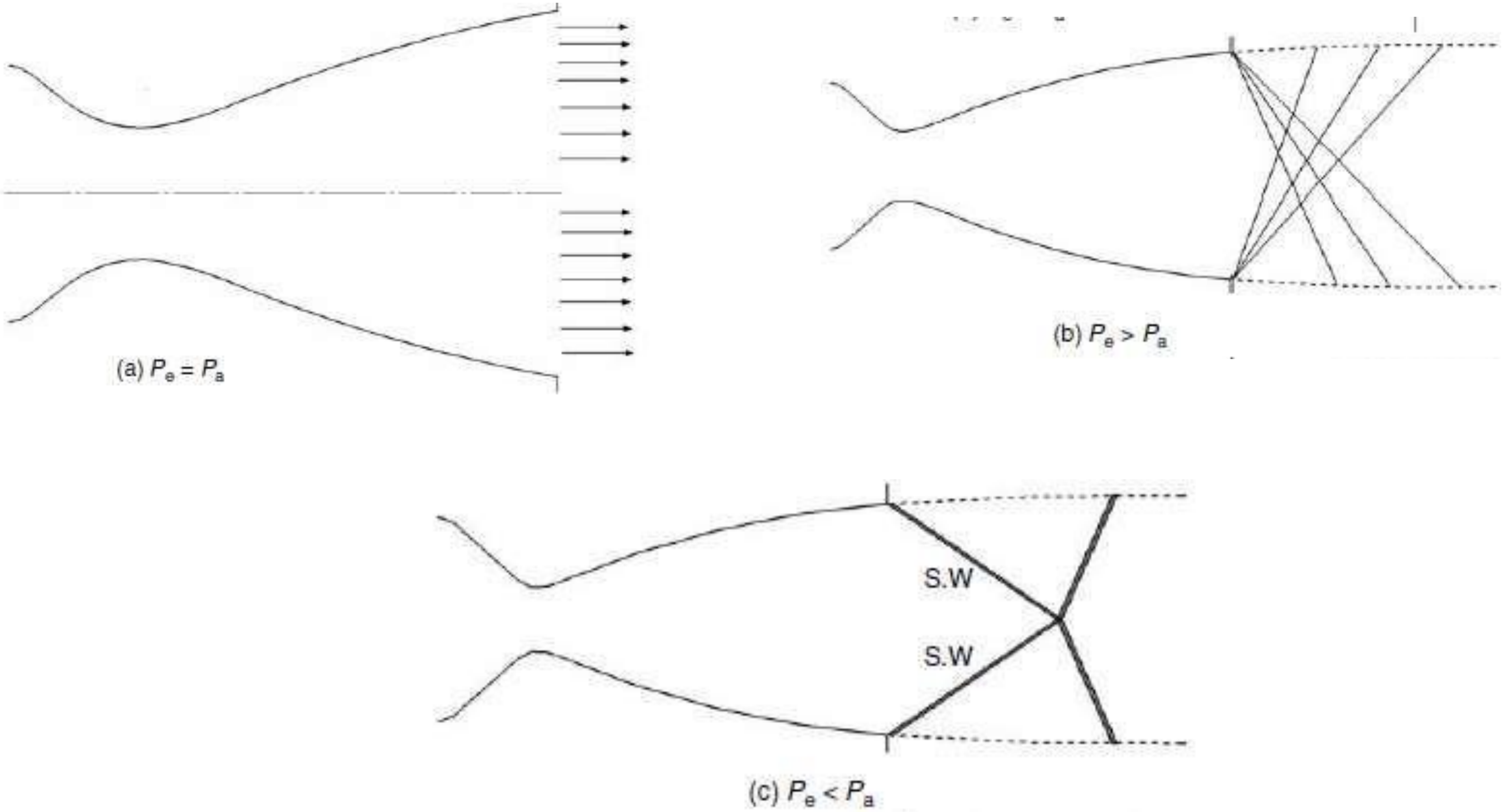


$$\frac{A^*}{A_e} = \frac{P_{0e}M_e}{P^*_{0i}}\left[\frac{(\gamma+1)/2}{1+((\gamma-1)/2)M_e^2}\right]^{(\gamma+1)/2(\gamma-1)}$$

$$\dot{m} = \frac{A^*P^*_{0i}}{\sqrt{T^*_{0i}}}\sqrt{\frac{\gamma}{R}}\frac{1}{((\gamma+1)/2)^{(\gamma+1)/2(\gamma-1)}}$$

Convergent Nozzle

The convergent nozzle is very similar to the convergent part of the previously discussed C-D nozzle. However, the main difference is that the flow in the C-D nozzle is assumed isentropic, while the flow in the convergent nozzle is assumed only adiabatic.



(a) $P_e = P_a$

(b) $P_e > P_a$

(c) $P_e < P_a$

Behaviour of convergent–divergent nozzle:

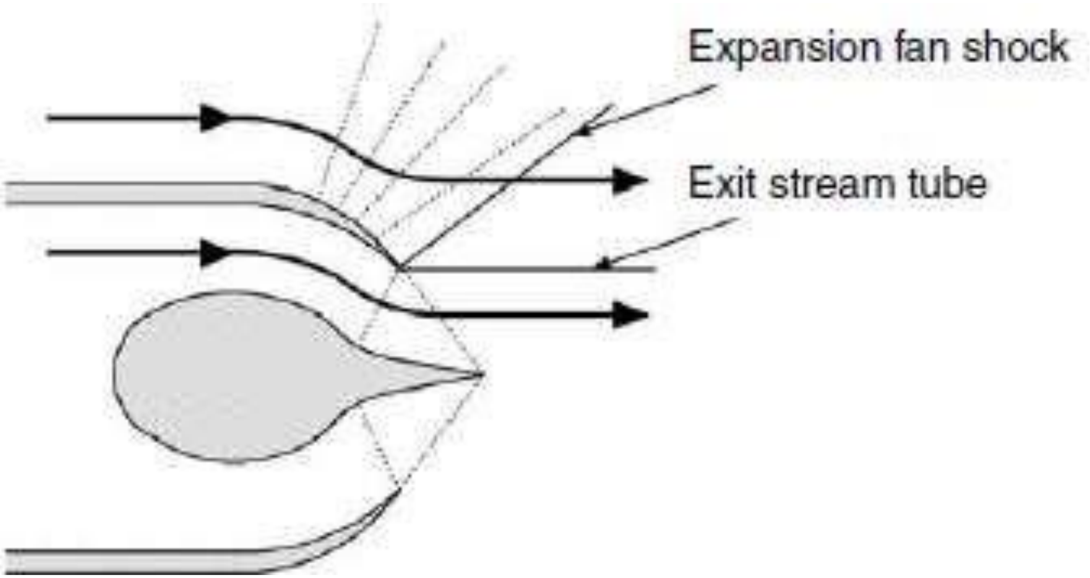
(a) Design condition (b) Exit pressure exceeds ambient pressure

(c)Ambient pressure exceeds exit pressure.

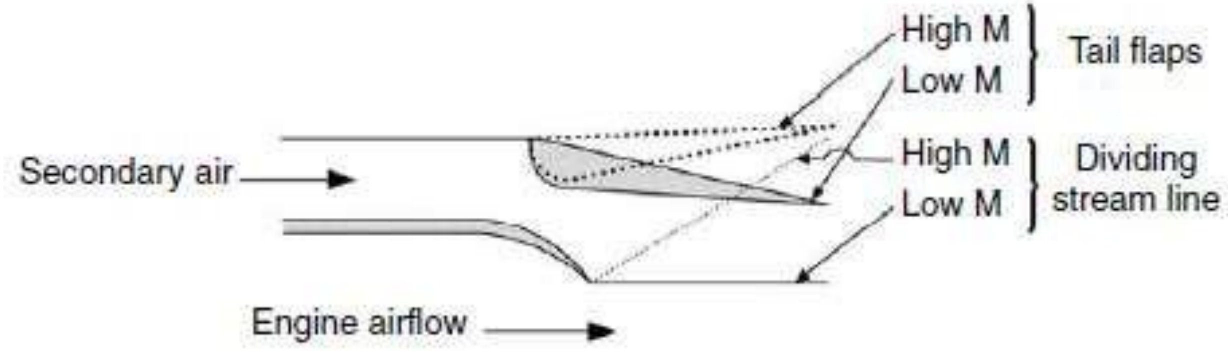
Variable Geometry Nozzles

Variable area nozzle, which is sometimes identified as adjustable nozzle, is necessary for engines fitted with afterburners. Generally, as the nozzle is reduced in area, the turbine inlet temperature increases and the exhaust velocity and thrust increase. Three methods are available, namely:

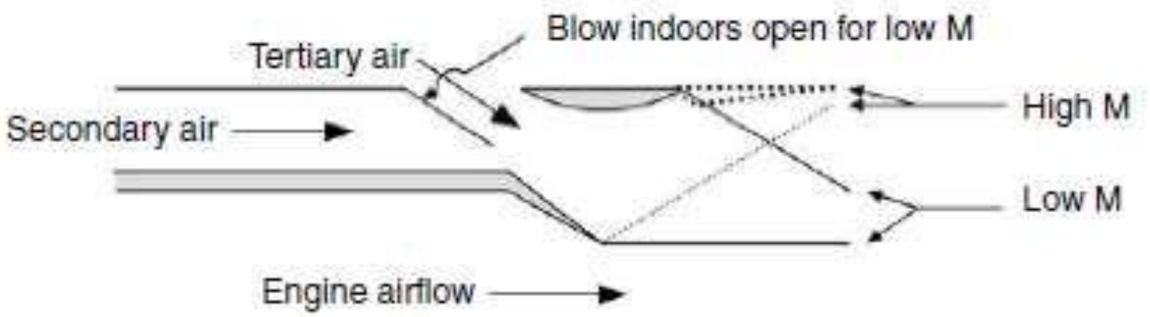
1. Central plug at nozzle outlet
2. Ejector type nozzle
3. IRIS nozzle



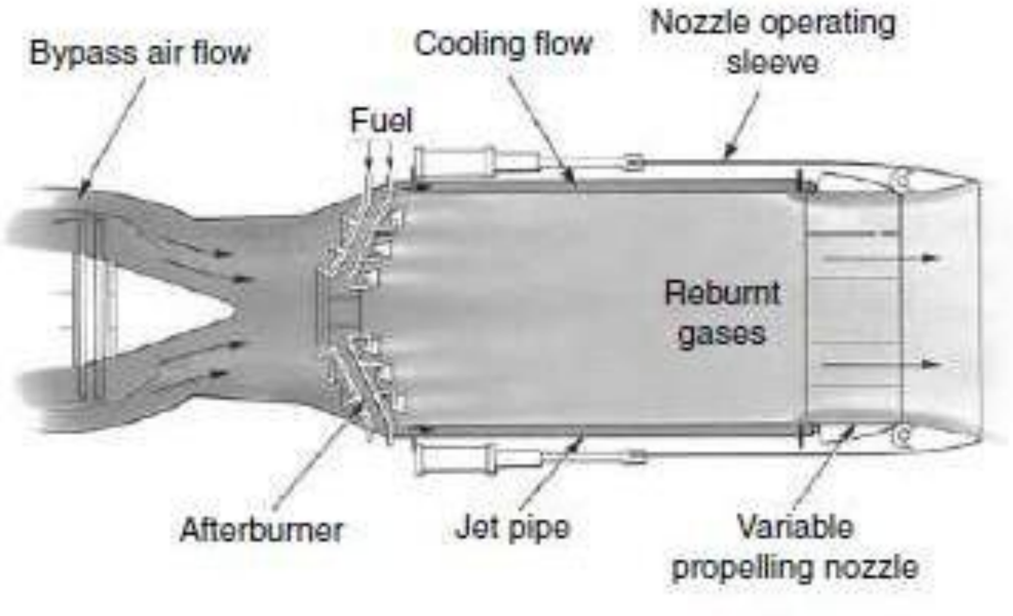
Plug nozzle at design point



Variable geometry ejector nozzle



Ejector nozzle with blow-in doors for tertiary air



Variable geometry nozzle for afterburning engine

Thrust Reversal

- Stopping an aircraft after landing is not an easy problem due to the increases in its gross weight, wing loadings and landing speeds.
- The amount of force required for stopping an aircraft at a given distance after touchdown increases with the gross weight of the aircraft and the square of the landing speed.
- The size of modern transport aircraft, which results in higher wing loadings and increased landing speeds, makes the use of wheel brakes alone unsatisfactory for routine operations.
- Moreover, in the cases of wet, icy, or snow-covered runways, the efficiency of aircraft brakes may be reduced by the loss of adhesion between aircraft tire and the runway .

Thrust Reversal

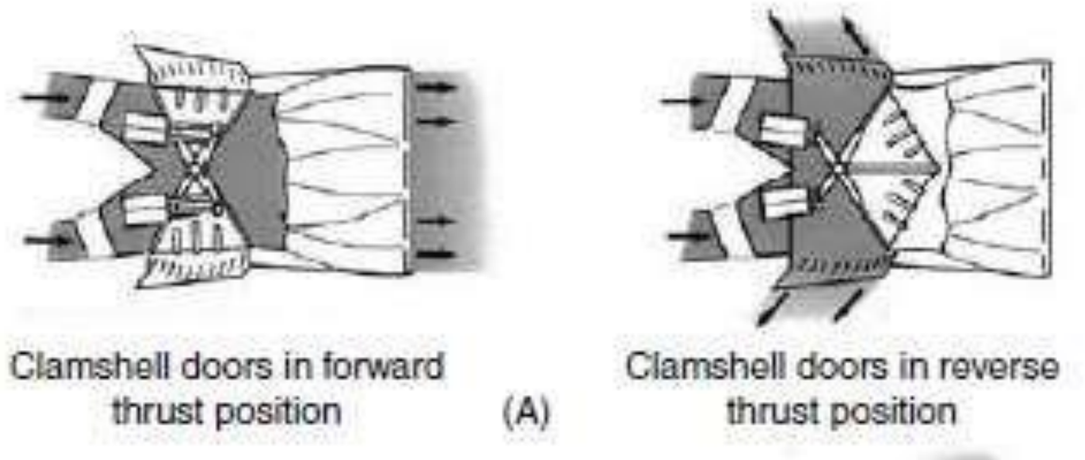
- Thus, for present large aircraft and under such runway conditions, there is a need for additional methods for augmenting the stopping power provided by the brakes to bring the aircraft to rest within the required distance

A good thrust reverser must fulfil the following conditions

1. Must not affect the engine operation whether the thrust reverser is applied or stowed
2. Withstand high temperature if it is used in the turbine exhaust
3. Mechanically strong
4. Relatively light in weight
5. When stowed should be streamlined into the engine nacelle and should not add appreciably to the frontal area of the engine
6. Reliable and fail safe
7. Cause few increased maintenance problems
8. Provide at least 50% of the full forward thrust

Classification Of Thrust Reverser Systems

Clamshell door system—sometimes identified as pre-exist thrust reverser is a pneumatically operated system. When reverse thrust is applied, the doors rotate to uncover the ducts and close the normal gas stream. Sometimes clamshell doors are employed together with cascade vanes

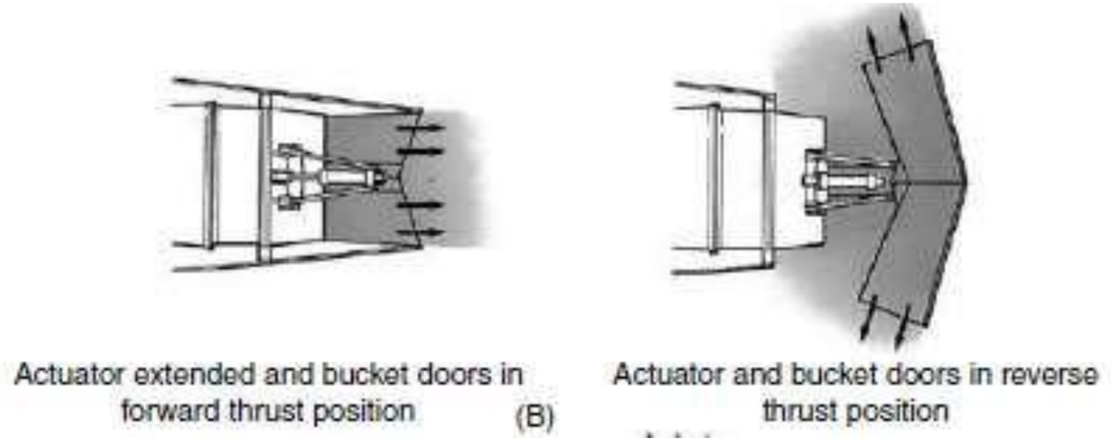

Clamshell doors in forward thrust position

(A)

Clamshell doors in reverse thrust position

Thrust Reversal System

The bucket target system-is hydraulically actuated and uses bucket-type doors to reverse the hot gas stream. Sometimes it is identified as post exit or target thrust reverser .In the forward (stowed) thrust mode, the thrust reverser doors form the convergent–divergent final nozzle for the engine. When the thrust reverser is applied, the reverser automatically opens to form a “clamshell” approximately three-fourth to one nozzle diameter to the rear of the engine exhaust nozzle.



Thrust Reversal System

Nozzle Coefficients

Nozzle performance is ordinarily evaluated by two dimensionless coefficients: the gross thrust coefficient and the discharge or flow coefficient.

Gross thrust coefficient. The gross thrust coefficient C_{fg} is the ratio of the actual gross thrust Fg actual to the ideal gross thrust $F_{gideal.}$

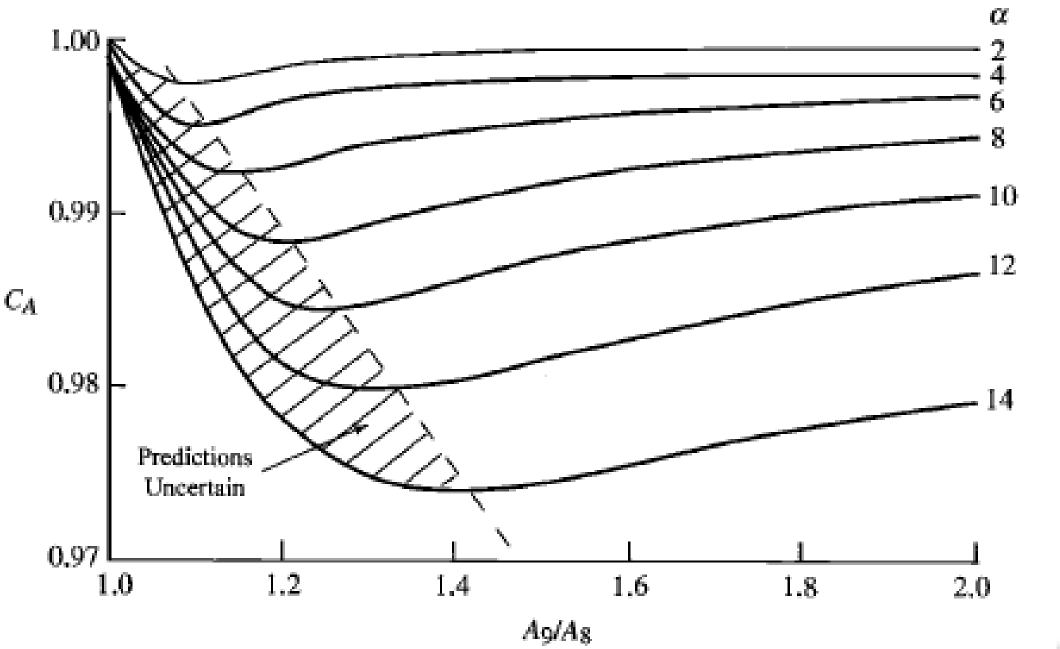
$$C_{fg} = \frac{F_{g\ \text{actual}}}{F_{g\ \text{ideal}}}$$

The following basic losses is accounted for:

1) Thrust loss due to exhaust velocity vector angularity.
2) Thrust loss due to the reduction in velocity magnitude caused by friction in the boundary layers.
3) Thrust loss due to loss of mass flow between stations 7 and 9 from leakage through the nozzle walls.
4) Thrust loss due to flow non-uniformities

Nozzle Performance

- Many nozzle coefficients simplify to algebraic expressions or become unity for the special case of *one-dimensional adiabatic flow.* This is a useful limit for understanding each coefficient and for preliminary analysis of nozzle performance using engine cycle performance data.



Nozzle Performance Graph

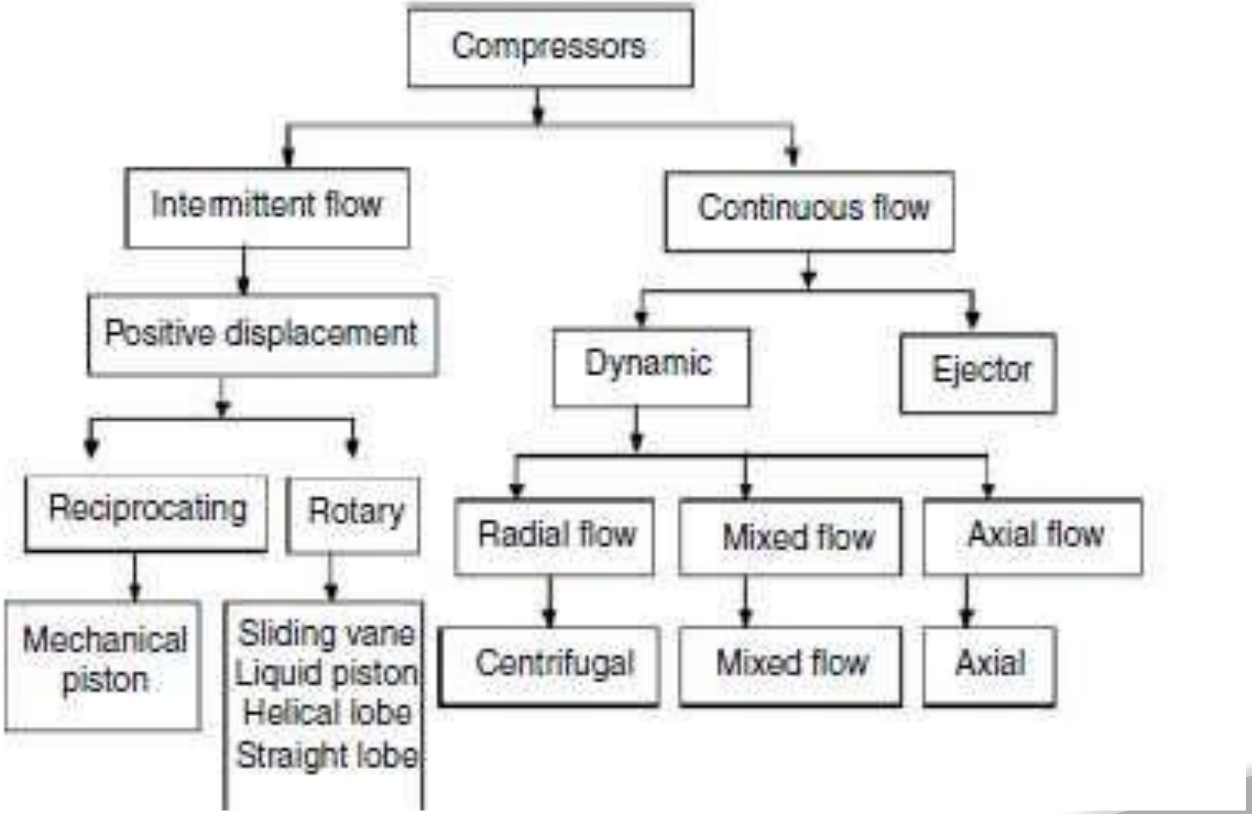
UNIT– III

AXIAL FLOW COMPRESSORS AND TURBINES

UNIT - III

CLO’s	Course Learning Outcomes
CLO7	Understand different design of compressor and limitations of each method.
CLO8	Describe principle of operation of centrifugal and axial flow turbine.
CLO9	Analyze performance characteristics of axial and centrifugal compressor.

Classification of compressors:



Centrifugal Compressors

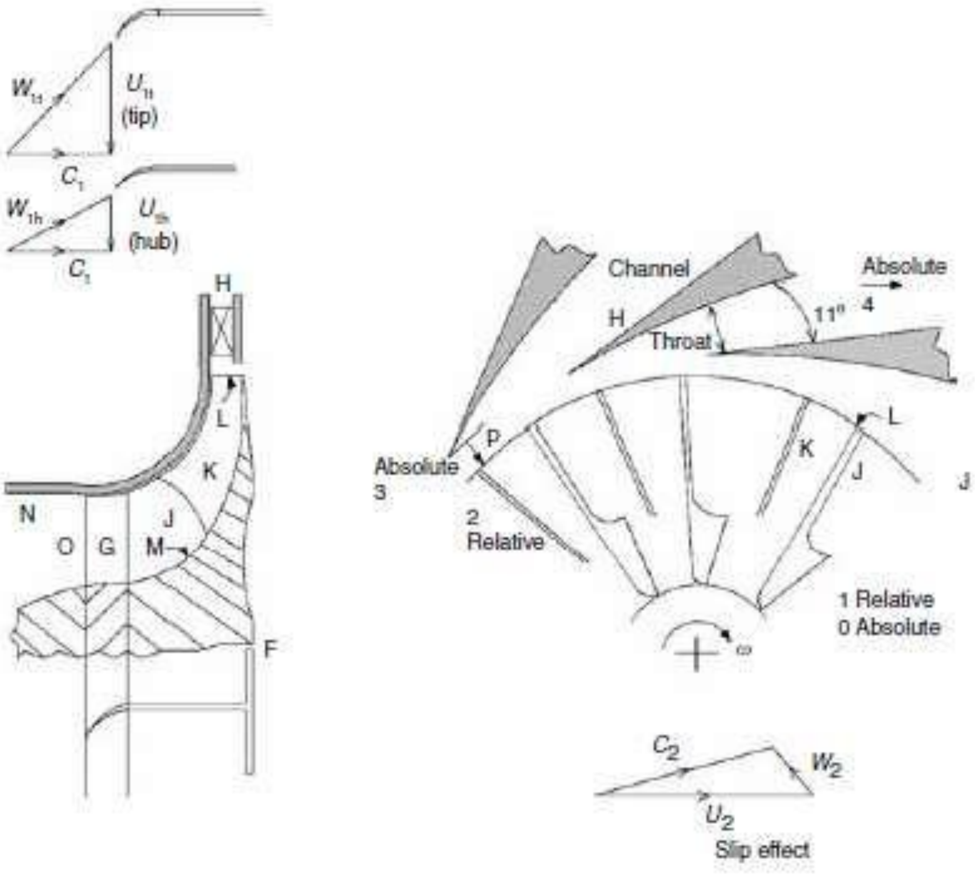


Figure illustrates a typical centrifugal compressor . Nomenclature for a single-sided compressor with channel-type diffuser is given here.

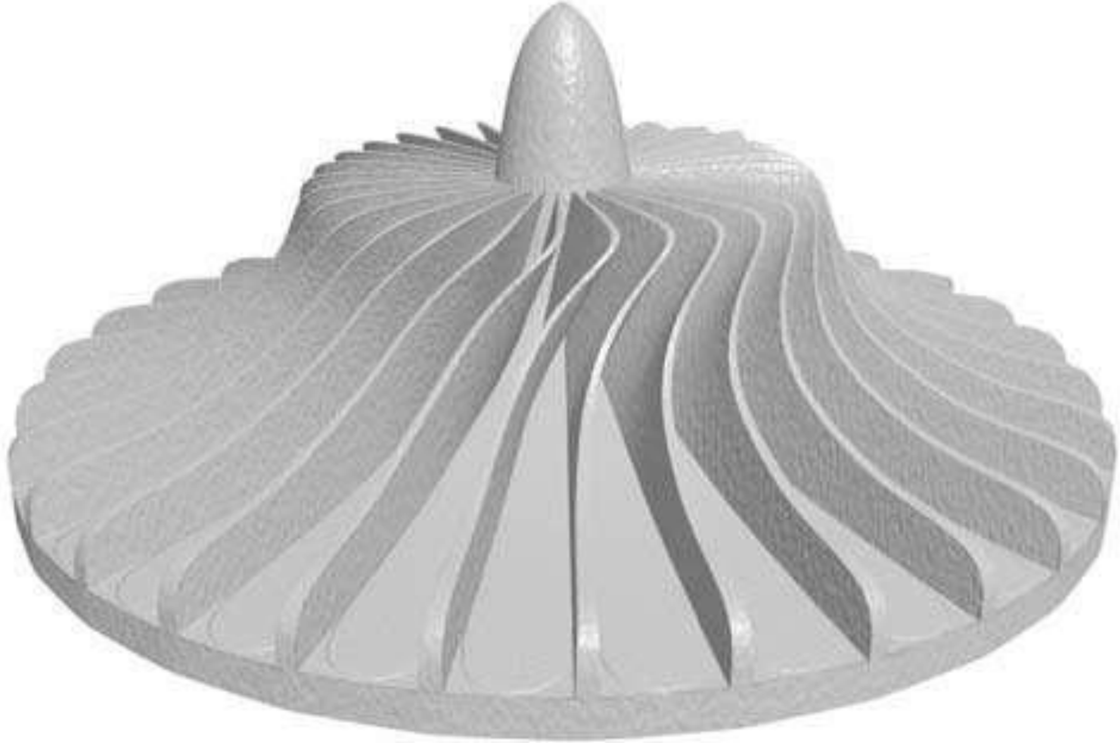
F: Impeller G: Inducer (rotating guide vane)
H: Diffuser J: Impeller vane
K: Half vane (or splitter) L: Impeller tip
M: Impeller hub N: Shroud (casing)
O: Impeller eye P: Vaneless gap

Impeller

- Impeller scoops in the working fluid (air/gas). Air is drawn at the center or eye of the impeller, then accelerated through the fast spinning speed of the impeller and finally thrown out at the tip. The forces exerted on the air are centripetal.
- At the eye (inlet), the vanes are curved to induce the flow: this axial portion is called the inducer or rotating guide vane and may be integral with separated from the main impeller.
- The divergence (increasing cross-sectional area) of these passages diffuses (slows) the flow to a lower relative velocity and higher static pressure.

The impeller is a complicated diffuser compared with the conventional straight conical diffuser as the passage is doubly curved first in the axial plane and then in the radial plane

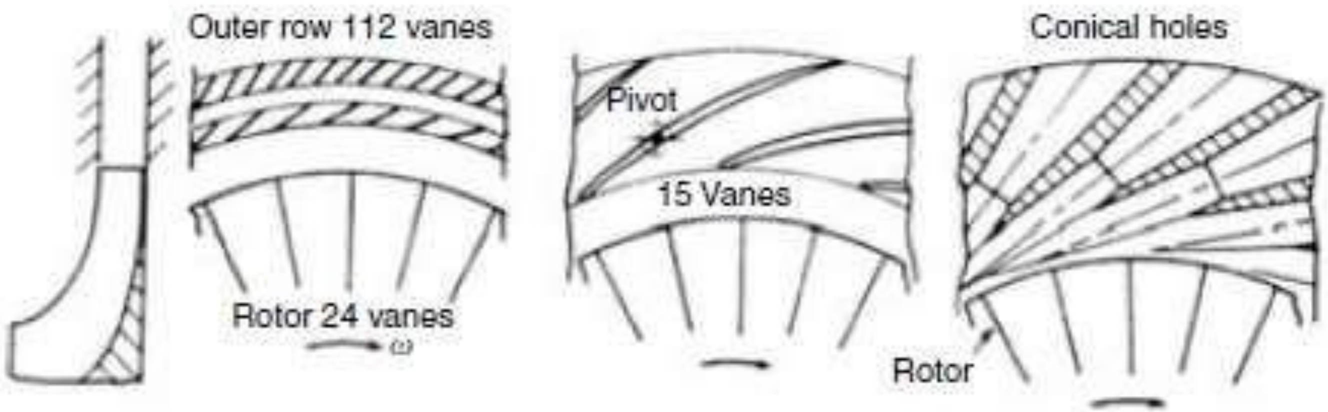
Impeller Shape



Diffuser

- The impeller blades sling the air radially outward where it is once again collected (at higher pressure)before it enters the diffuser. The diffuser represents a part of the fixed structure of the compressor.
- It discharges air from the compressor impeller with a high absolute velocity and the role of diffusion is to reduce the kinetic energy, thereby increasing the static pressure.
- The diffuser is either a vaneless passage or a vaneless passage followed by a vaned section. The vaned diffuser represents a large group including the vanes together with the cascade, channel, and pipe types.

Different Types Of Diffuser



(a) Vaneless (b) Cascade (2 rows) (c) Vaned diffuser (d) Pipe diffuser

Scroll Or Manifold

- The final element of centrifugal compressors is either a manifold or a scroll.
- Centrifugal compressors with manifolds are used when the compressor is a part of a gas generator—in either a gas turbine or an aero engine—and thus the compressor is followed by a combustion chamber.
- In this case, the diffuser is bolted to the manifold, and often the entire assembly is referred to as the diffuser.
- The working fluid leaving the stators is collected in a spiral casing surrounding the diffuser called a volute or scroll.

Classification Of Centrifugal Compressors

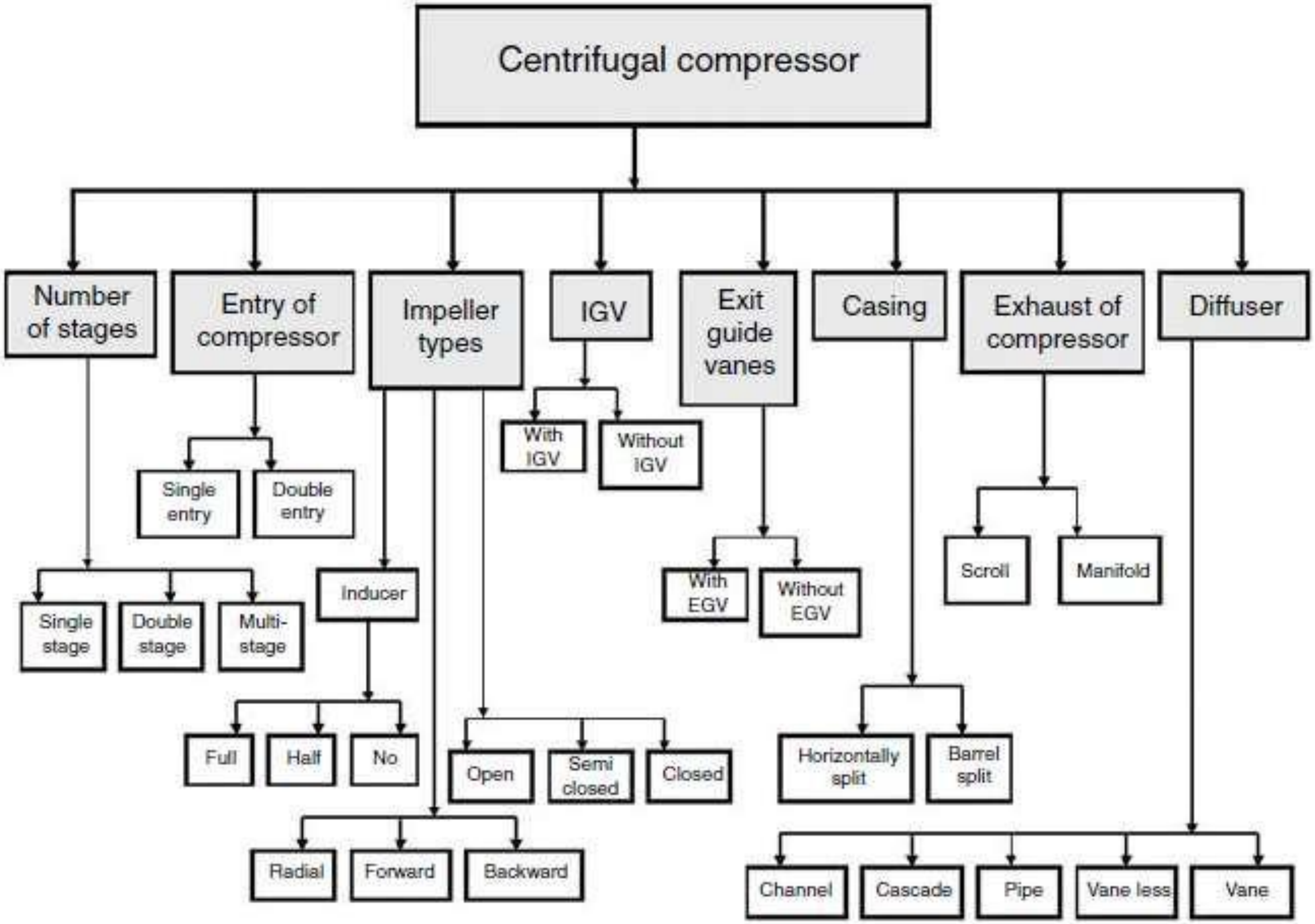
Centrifugal compressors may be classified as

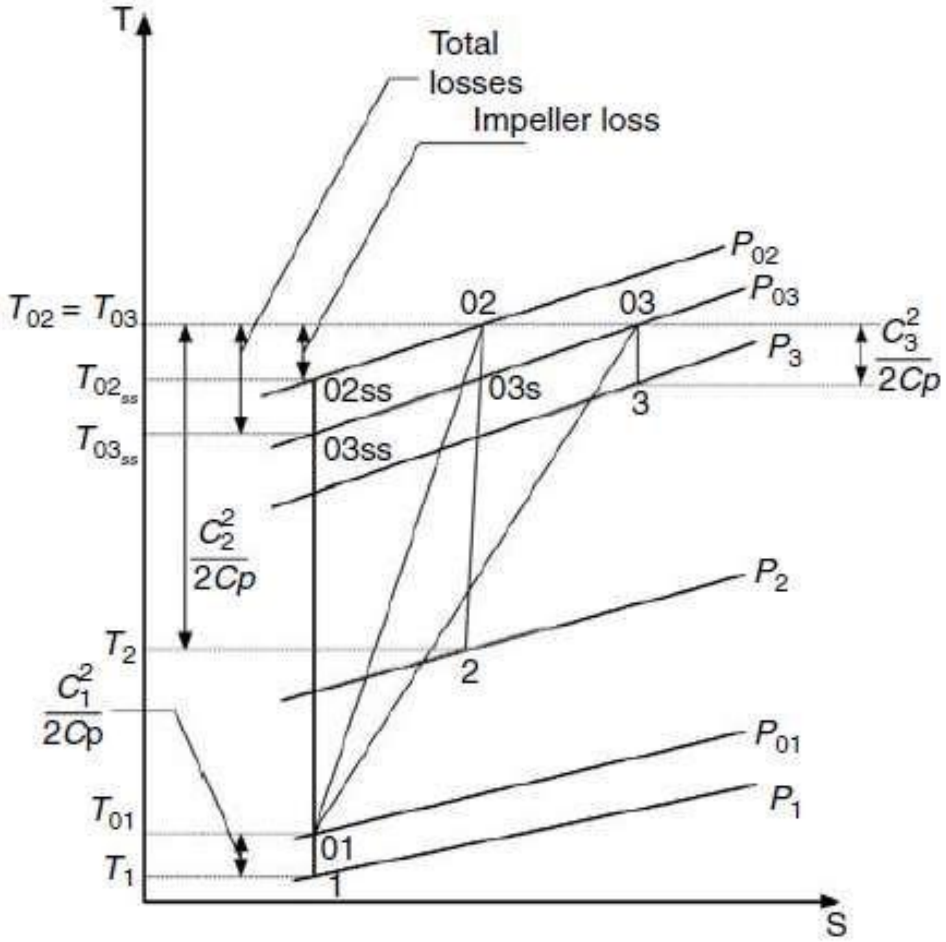
- **Single or multiple stages:**
  - For aero engines, centrifugal compressors have either single stage or double (two or tandem) stages.
  - In some cases two(or double) stages are mounted on the same shaft, handling the fluid in series to boost its final pressure.
  - In industrial gas turbines such as pipeline compressors, there are up to five stages.

- **Single entry (or single-face) or double (dual) entry (or double-face):**
  - The principal differences between the single entry and dual entry are the size of the impeller and the ducting arrangement.
  - The single-entry impeller permits ducting directly to the inducer vanes, as opposed to the more complicated ducting needed to reach the rear side of the dual-entry type.
  - Although slightly more efficient in receiving air, single-entry impellers must be of greater diameter to provide sufficient air.
  - Dual-entry impellers are smaller in diameter and rotate at higher speeds to ensure sufficient airflow.

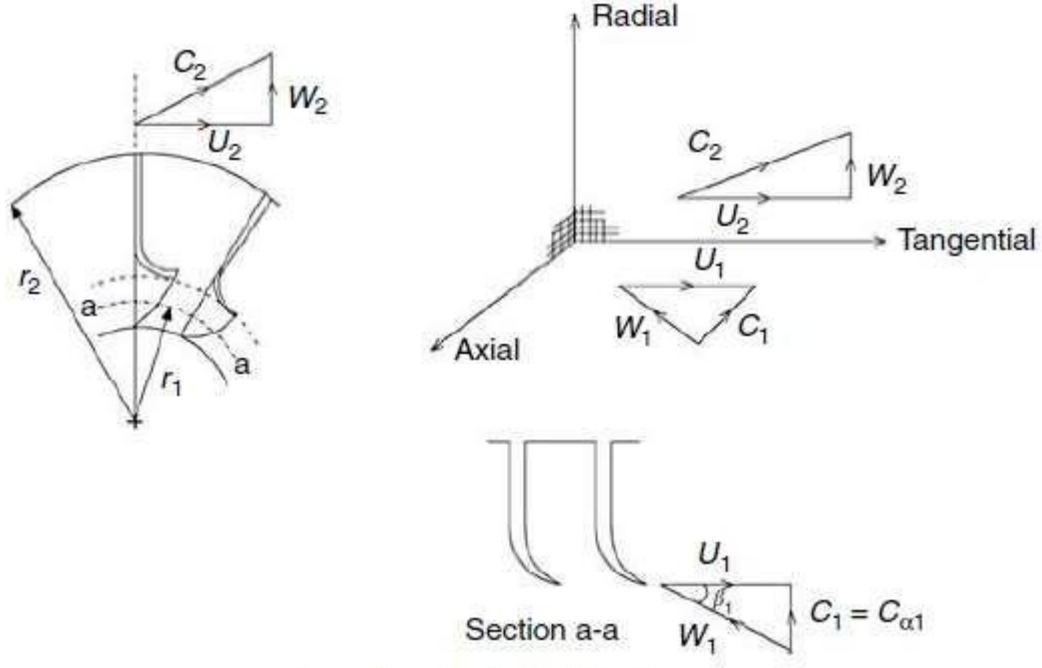
- **Shrouded or unshrouded impeller:**
  - Unshrouded impeller means that there is a clearance between the ends of the impeller vanes and a stationary shroud, while shrouded impeller means that there is a rotating shroud fixed to the impeller vanes.
  - Shrouding reduces the losses due to leakage of air from the pressure side to the suction side of the blade.
- The impeller may have a non-, semi-, and full-inducer.
- For full-inducer impeller, the impeller vanes are continued around into the axial direction and the compressor resembles an axial compressor at inlet.
- The impeller vanes may be radial at outlet or they may be inclined backward or forward, thus identified as backward-leaning (commonly now described as backswept) or forward leaning compressor.

Classification Of Centrifugal Compressors





T–S diagram for a centrifugal compressor

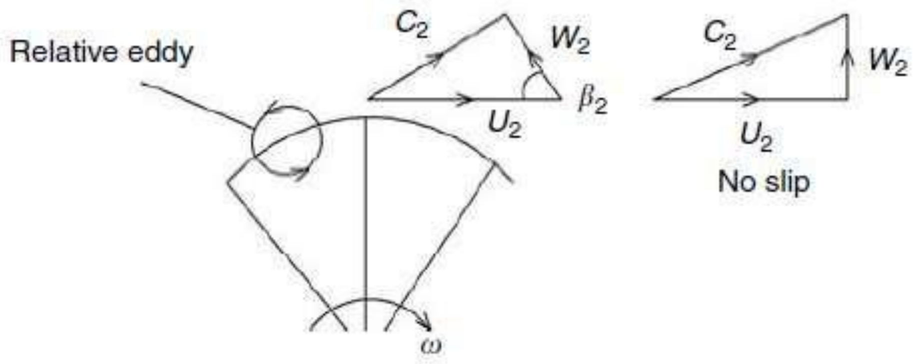


Layout and velocity triangles

Inlet and outlet velocity triangles at space

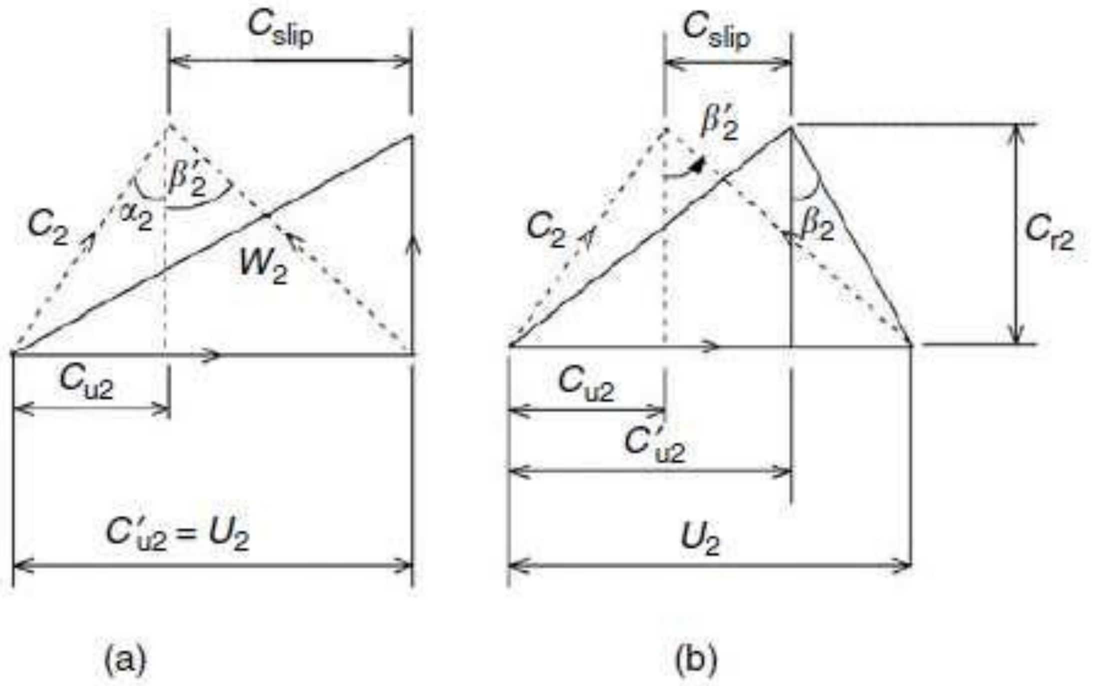
Slip Factor σ

If the flow at impeller discharge is perfectly guided by the impeller blades then the tangential component of the absolute velocity (swirl velocity) is equal to the rotational velocity ($Cu_2=U_2$) in the case of radial type impeller. In practice, the flow cannot be perfectly guided by a finite number of blades and it is said to slip. Thus, at the impeller outlet, the swirl velocity is less than the impeller rotational speed ($Cu_2 < U_2$).

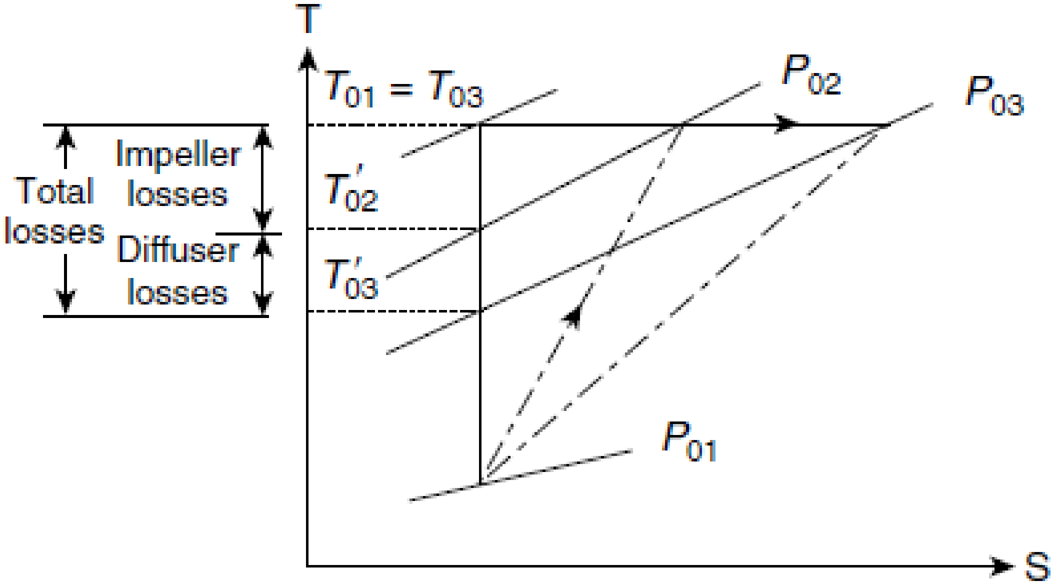


Slip due to relative eddies

Slip due to relative eddies



Slip for (a) radial and (b) backward impellers



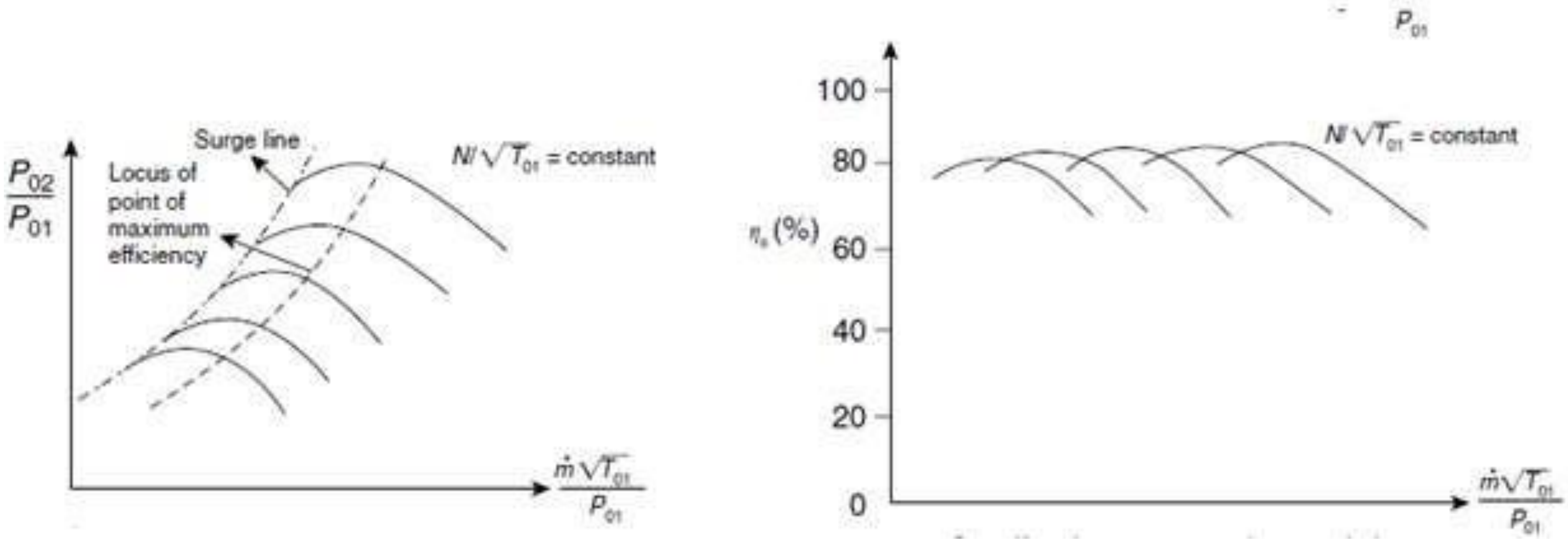
Impeller and diffuser losses

Performance Of A Centrifugal Compressor

For any compressor, the stagnation pressure at outlet and the overall compressor efficiency are dependent on other physical properties as follows.

$$P_{02},\ \eta_c = f\,(\dot{m},\ P_{01},\ T_{01},\ \gamma,\ R,\ D,\ N,\ \text{and } \upsilon)$$

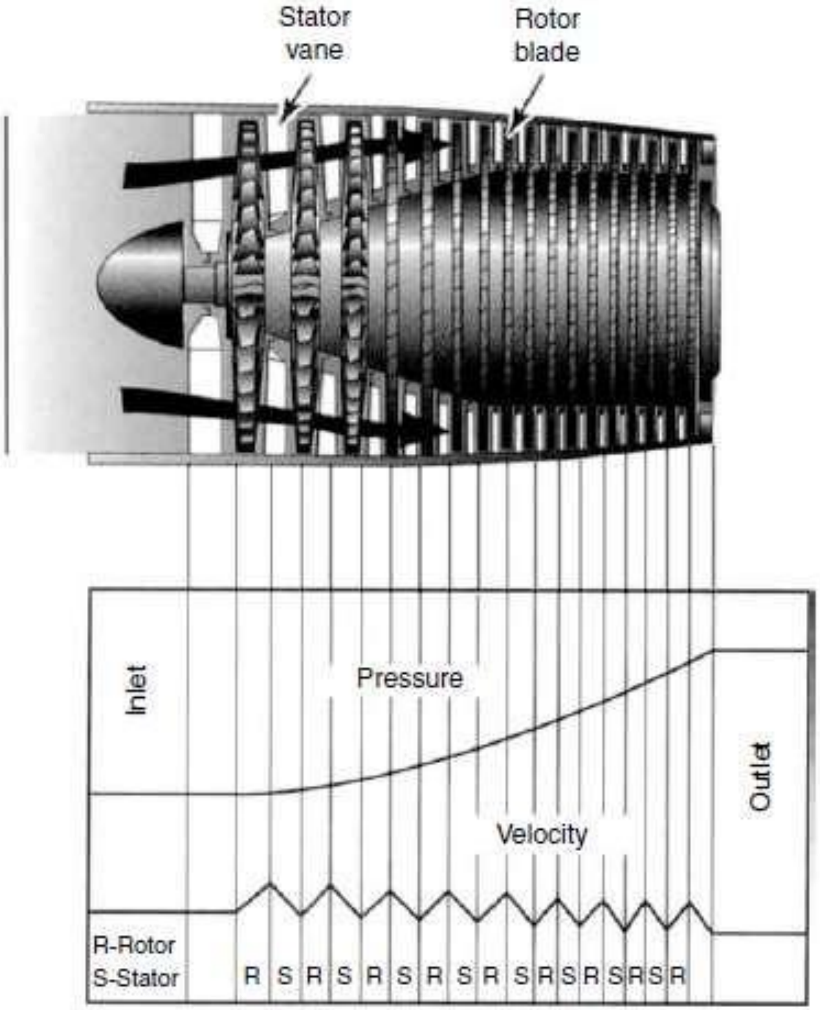
$$\frac{P_{02}}{P_{01}}, \eta_C = f\left(\frac{\dot{m}\sqrt{\gamma R T_{01}}}{P_{01} D^2}, \frac{ND}{\sqrt{RT_{01}}} \frac{ND^2}{\upsilon}, \gamma\right).$$



Centrifugal Compressor Characteristics

Axial-Flow Compressor

- In axial compressors, the air flows mainly parallel to the rotational axis. Axial-flow compressors have large mass flow capacity, high reliability, and high efficiency, but have a smaller pressure rise per stage (1.1:1 to 1.4:1) than centrifugal compressors (4:1 to 5:1).
- However, it is easy to link together several stages and produce a multistage axial compressor having pressure ratios up to 40:1 in recent compressors.
- Axial compressors are widely used in gas turbines, notably jet engines, wind tunnels, air blowers, and blast furnaces. Engines using an axial compressor are known as axial flow engines; for example, axial flow turbofan.



Layout of an axial compressor

- The axial compressor is built up of a series of stagers, each consisting of a disc of rotor blades followed by a ring of stator vanes. The axial compressor is generally composed of four main elements: **front frame, casing with inlet (stator) vanes, rotor with rotor blades, and rear frame.**
- The front frame in turbojet engines is a ring-shaped single piece lightweight structure made up of aluminum alloy or steel, usually cast and then machined. It is composed of an outer ring, an inner hub, and 6–8 streamlined supporting struts.
- If the compressor is a part of the turbofan engine, then this front frame is replaced by a row of inlet guide vanes (IGVs).

- The compressor casing is a tube-like construction split lengthwise to facilitate engine assembly and maintenance. To retain the stator blades and variable IGVs in modern engines, the inner surfaces of the casing are machined with circumferential T-section grooves.
- The final ring of the stator blades (vanes) may be called outlet guide vanes (OGVs), as they guide the flow to the axial direction to suit the compressor outlet. After installing the rotor, both casing halves are bolted together through longitudinal flanges. The compressor casing is made up of lightweight titanium.
- The rotor blades (similar to the stator ones)have aerofoil section similar to the aircraft wing, but they are highly twisted from root to tip to obtain the optimum angle-of-attack to the flow everywhere along the blade length.

Advantages Of The Axial-flow Compressor Over The Centrifugal Compressor

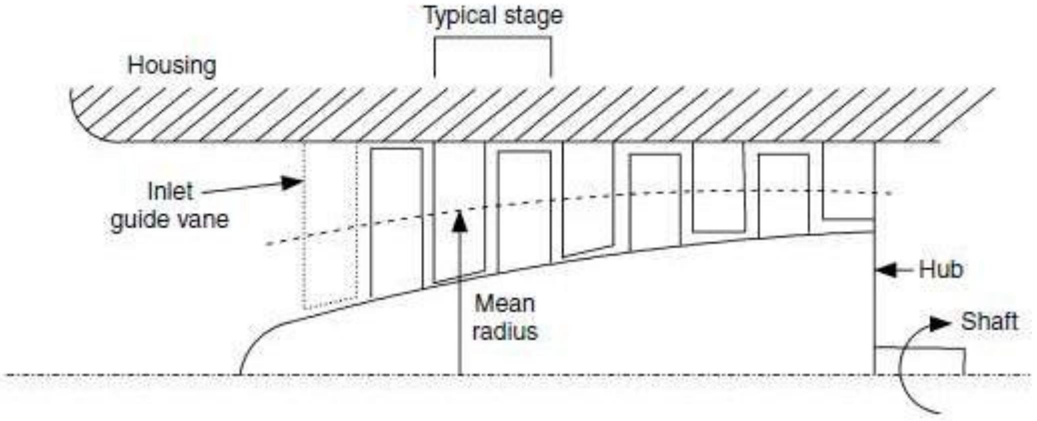
1. Smaller frontal area for a given mass rate of flow (may be 1/2 or 1/3); thus, the aerodynamic drag or nacelle housing the engine is smaller.
2. Much greater mass flow rates (e.g., present day axial compressors); have mass flow rates up to 200 kg/s (up to 900 kg/s for high-bypass ratio turbofan engines), while centrifugal compressors have mass flow rates less than 100 kg/s.
3. Flow direction of discharge is more suitable for mitigating; thus, suitable for large engines.
4. May use cascade experiment research in developing compressor.
5. Somewhat higher efficiency at high pressure ratio (perhaps 4%–5% higher than centrifugal compressor).

Advantages Of The Axial-flow Compressor Over The Centrifugal Compressor

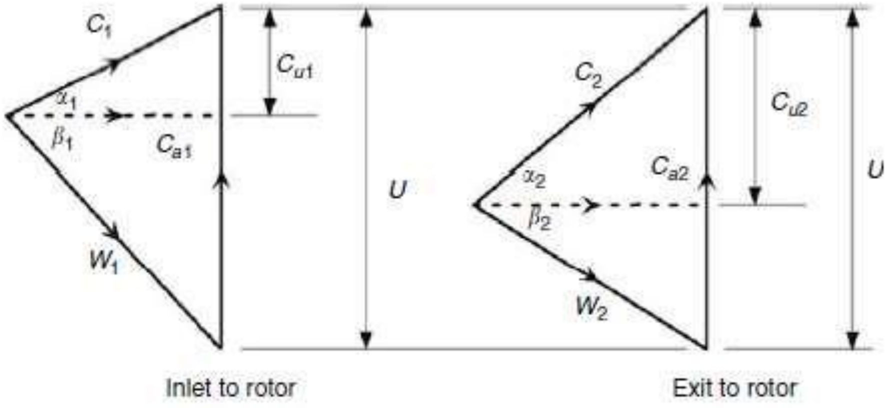
6. Higher maximum pressure ratio, which was about 17 in the 1960s and achieved up to 45 for the present transonic compressors

Advantages Of Centrifugal-flow Compressor Over The Axial-Flow Compressor

1. Higher stage pressure ratio (5:1 or even 10:1).
2. Simplicity and ruggedness of construction.
3. Shorter length for the same overall pressure ratio.
4. Generally less severe stall characteristics.
5. Less drop in performance with the adherence of dust to blades.
6. Cheaper to manufacture for equal pressure ratio.
7. Flow direction of discharge air is convenient for the installation of an intercooler and/or heat exchanger in gas turbines.

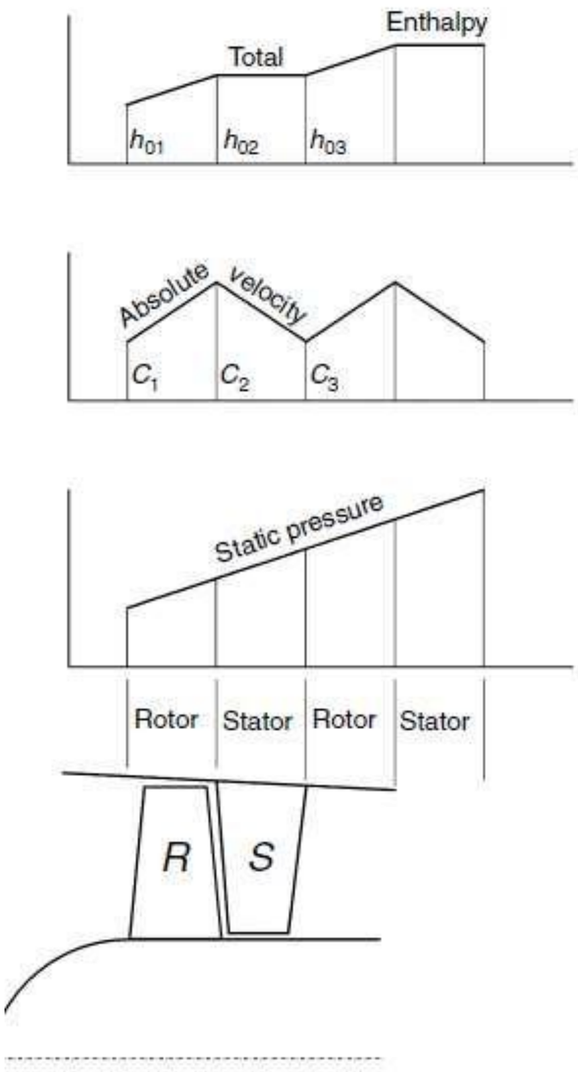


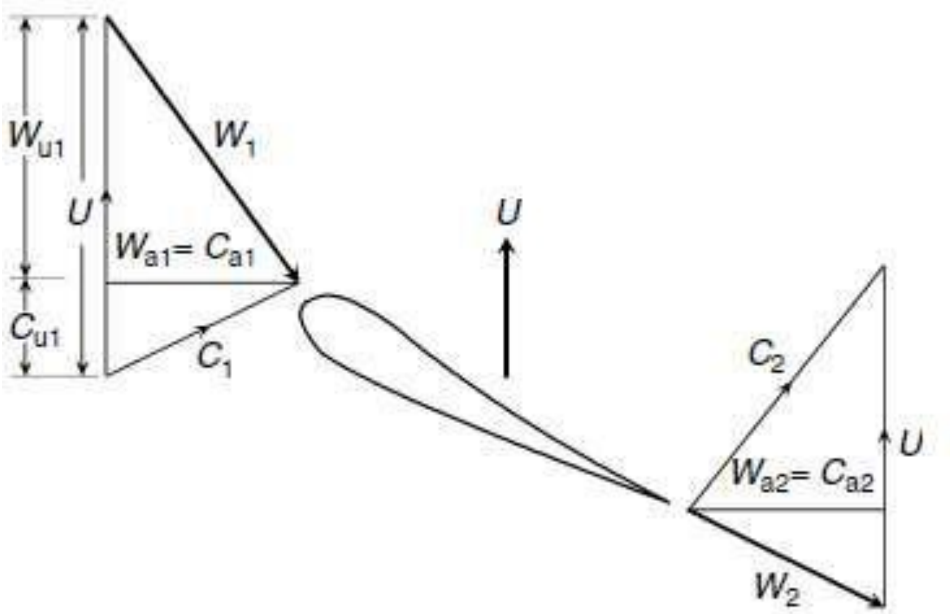
Variation of mean radius for a constant casing multistage axial compressor.



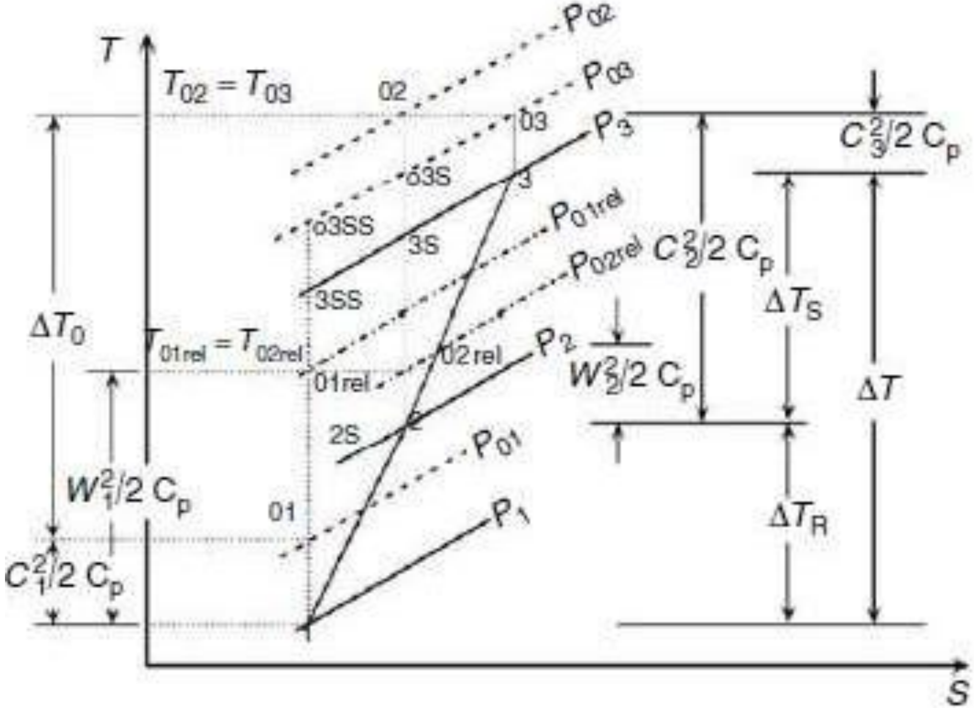
Velocity triangles at inlet and outlet of a constant mean radius

Variations Of The Enthalpy, Absolute Velocity, And Static Pressure Across Two Successive Stages.





Illustrations of velocity triangles



Temperature–entropy diagram for a compressor stage

BASIC DESIGN PARAMETERS

The pressure ratio per stage is expressed by the relation

$$\pi_s = \left[1 + \eta_s \frac{\lambda U C_a}{CpT_{01}} (\tan \beta_1 - \tan \beta_2) \right]^{\gamma/(\gamma-1)}$$

To obtain high pressure ratio per stage it is needed to have

1. High blade speed (U)
2. High axial velocity (C_a)
3. High fluid deflection ($\beta_1 - \beta_2$) in the rotor blade

DESIGN PARAMETERS

The following three design parameters are frequently used in the parametric study of a repeating axial stage:

1. Flow coefficient ϕ
2. Stage loading ψ
3. Degree of reaction

The **flow coefficient** is defined as the ratio between the axial and the rotational speeds

$$\phi = \frac{C_a}{U}$$

The stage loading is defined as the ratio between the total enthalpy rise per stage to the square of the rotational speed,

$$\psi = \frac{\Delta h_0}{U^2} = \frac{W_s}{U^2} = \frac{\Delta C_u}{U}$$

The degree of reaction of a compressor stage has several definitions. For an incompressible flow, the degree of reaction is defined as the ratio of the static pressure rise in the rotor to the static pressure rise in the stage

$$\Lambda = \frac{p_2 - p_1}{p_3 - p_1}$$

Three-Dimensional Flow

In present day turbo machinery, 3D effects are hardly negligible and their incorporation into the design

1. Compressibility and radial density and pressure gradients
2. Radial variation in blade thickness and geometry
3. Presence of finite hub and annulus walls, annulus area changes, flaring, curvature, and rotation
4. Radially varying work input or output
5. Presence of two phase flow (water injection, rain, sand, ice) and coolant injection
6. Radial component of blade force and effects of blade skew, sweep, lean, and twist
7. Leakage flow due to tip clearance and axial gaps
8. Non-uniform inlet flow and presence of upstream and downstream blade rows

Free Vortex Method

Free vortex method is one of the simplest design methods in axial compressors

1. Assuming constant specific work at all radii,

$$\therefore C_a \frac{dC_a}{dr} + C_u \frac{dC_u}{dr} + \frac{C_u^2}{r} = 0$$

2. Assuming constant axial velocity at all radii,

$$\therefore C_u \frac{dC_u}{dr} + \frac{C_u^2}{r} = 0$$

$$\frac{dC_u}{C_u} = -\frac{dr}{r}$$

$$rC_u = \text{constant}$$

Thus, the whirl velocity component of the flow varies inversely with radius, which is known as free vortex.

The design steps

1. Choice of the compressor rotational speed (rpm) and annulus dimensions
2. Determination of the number of stages (assuming efficiency)
3. Calculation of air angles for each stage at the mean section
4. Determination of the variation of air angles from root to tip based on the type of blading (free vortex-exponential–first power)
5. Selection of compressor blading using experimentally cascade data
6. Efficiency check (previously assumed) using cascade data
7. Investigation of compressibility effects
8. Estimation of off-design performance
9. Rig testing

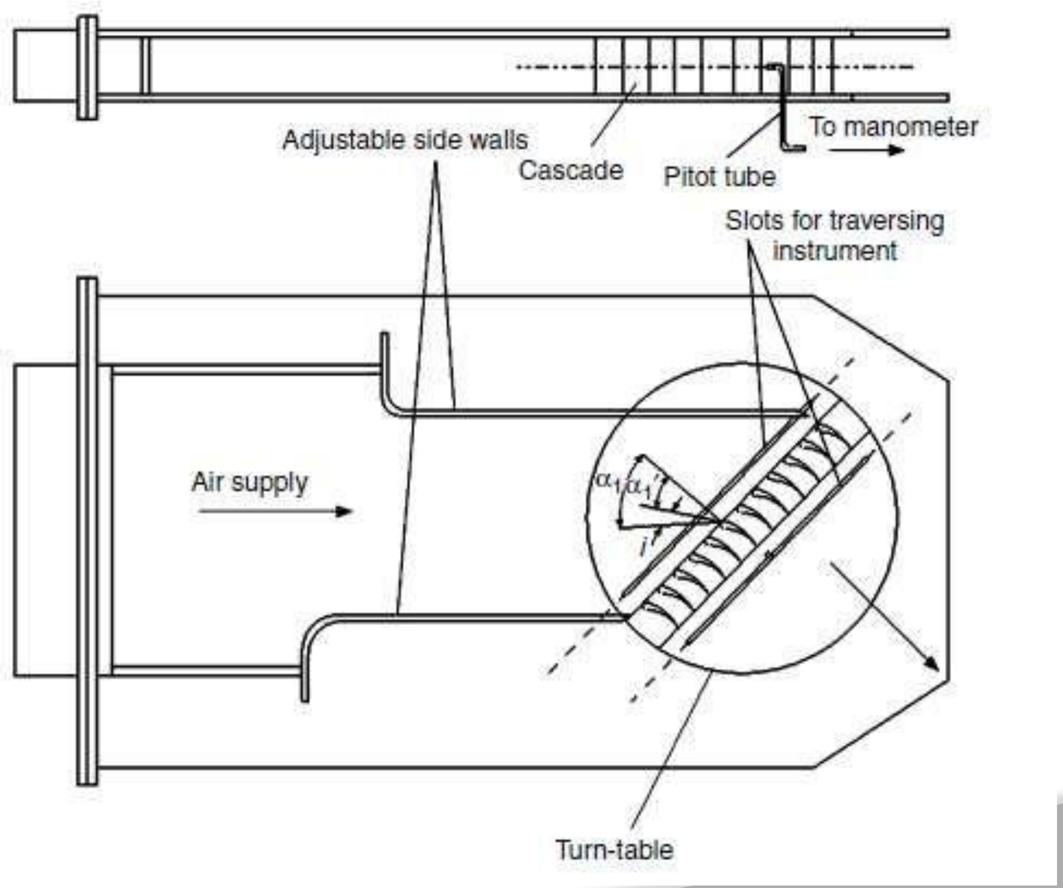
CASCADE MEASUREMENTS

To obtain information with respect to the effect of different blade designs on air flow angles, pressure losses, and energy transfer across blade rows, one must resort to cascade wind tunnels and cascade theory.

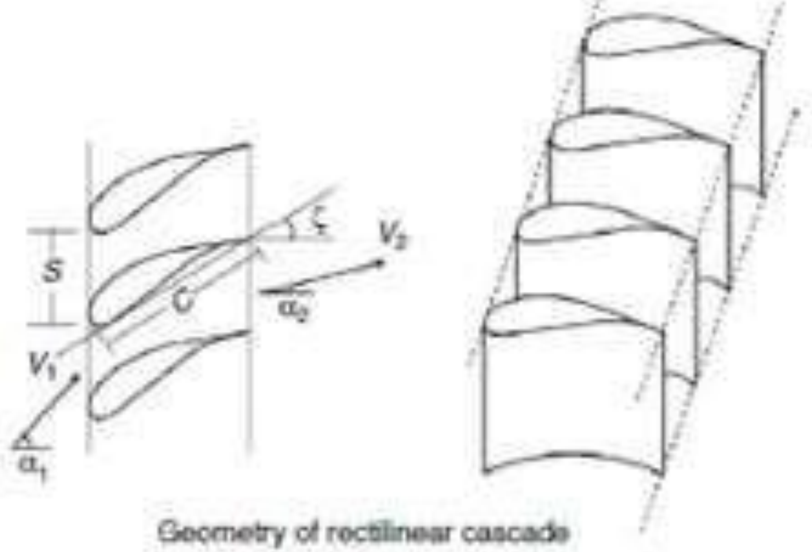
Experimentation is performed to ensure that the blade row satisfies its objectives.

- The first objective is to turn air through the required angles ($\beta_1-\beta_2$) for rotor and ($\alpha_2-\alpha_3$) for stator, with the angle ($\beta_1-\beta 2$) as maximum as possible to maximize the stage pressure ratio.
- The second objective is to achieve the diffusing process with optimum efficiency, that is, with minimum loss of stagnation pressure.
- Experiments are generally performed in a straight wind tunnel rather than in the annular form of tunnel.

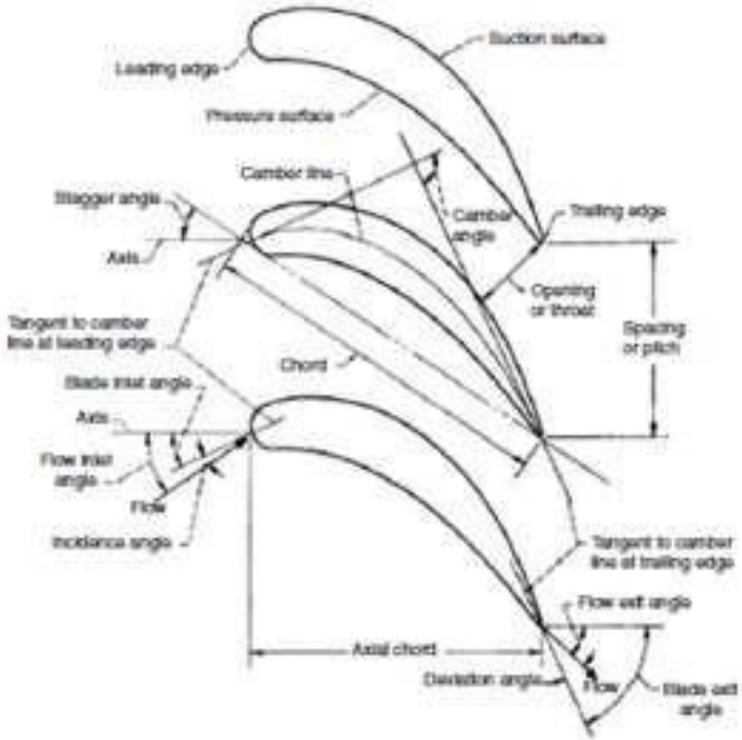
The word cascade denotes a row of identical (geometrically similar) blades equally spaced and parallel to each other aligned to the flow direction as shown in Figure.



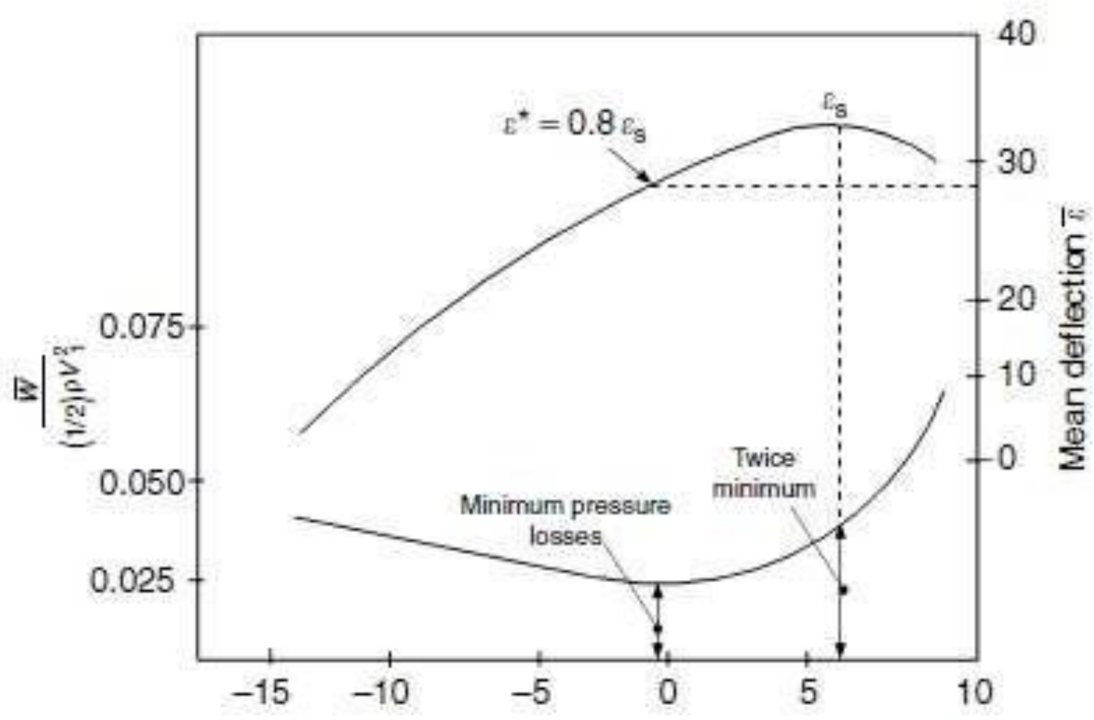
- The height and length of cascade in a cascade wind tunnel, made as large as the available air supply, will allow eliminating the interference effects due to the tunnel walls. Boundary layer suction on the walls is frequently applied to prevent contraction of air stream
- Cascade is mounted on a turn-table so that its angular direction with respect to the inflow duct (α1) can be set in any desired value and thus the incidence angle (i) may be varied.
- Vertical traverses over two planes usually a distance of one blade chord upstream and downstream of the cascade are provided with pitot tubes and yaw meters to measure the pressure and airflow angles.



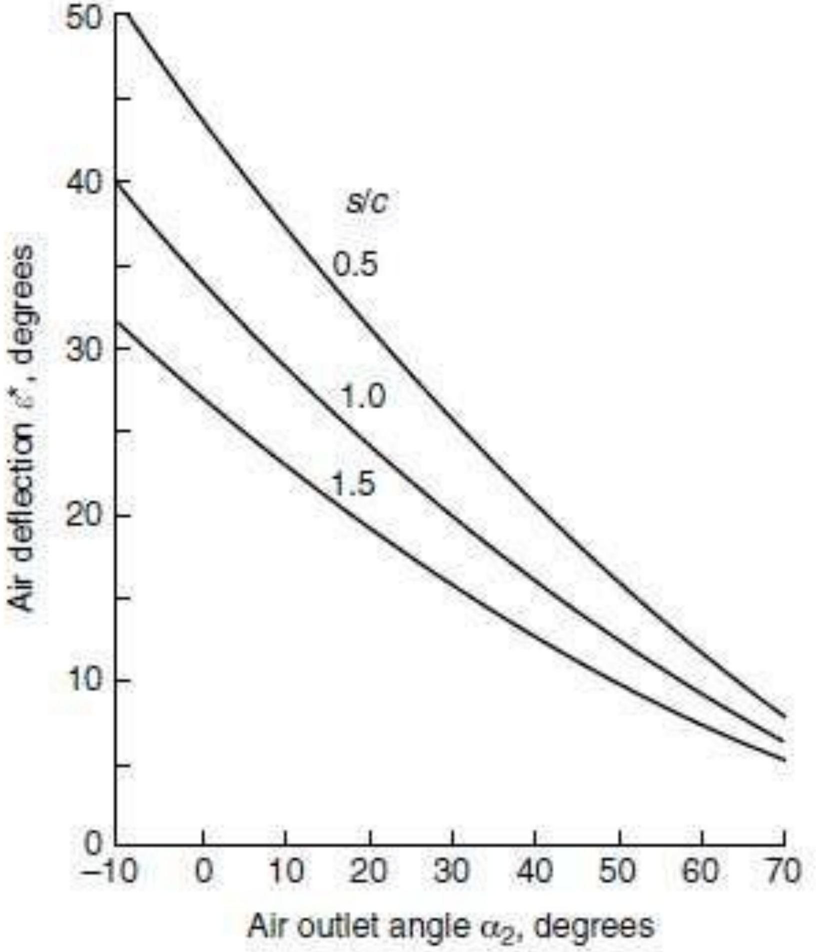
Geometry of rectilinear cascade



Geometry of Rectilinear Cascade



Mean deflection and mean stagnation pressure loss curves



Cascade nominal deflection versus outlet angle

Blade Efficiency and Stage Efficiency

Since the static pressure rise P is given by the relation

$$\Delta P = \frac{1}{2}\rho V_a^2\left(\tan^2\alpha_1 - \tan^2\alpha_2\right) - \overline{w}$$

the ideal or theoretical pressure rise will be

$$\Delta P_{\text{th}} = \frac{1}{2}\rho V_a^2\left(\tan^2\alpha_1 - \tan^2\alpha_2\right)$$

The blade row efficiency is defined as

$$\eta_b = \frac{\text{Actual pressure rise in compressor blade row}}{\text{Theoretical pressure rise in blade row}}$$

$$\eta_b = \frac{\Delta P}{\Delta P_{\text{th}}} = \frac{\Delta P_{\text{th}} - \overline{w}}{\Delta P_{\text{th}}} = 1 - \frac{\overline{w}}{\Delta P_{\text{th}}} = 1 - \frac{\overline{w}/(\rho V_1^2/2)}{\Delta P_{\text{th}}/(\rho V_1^2/2)}$$

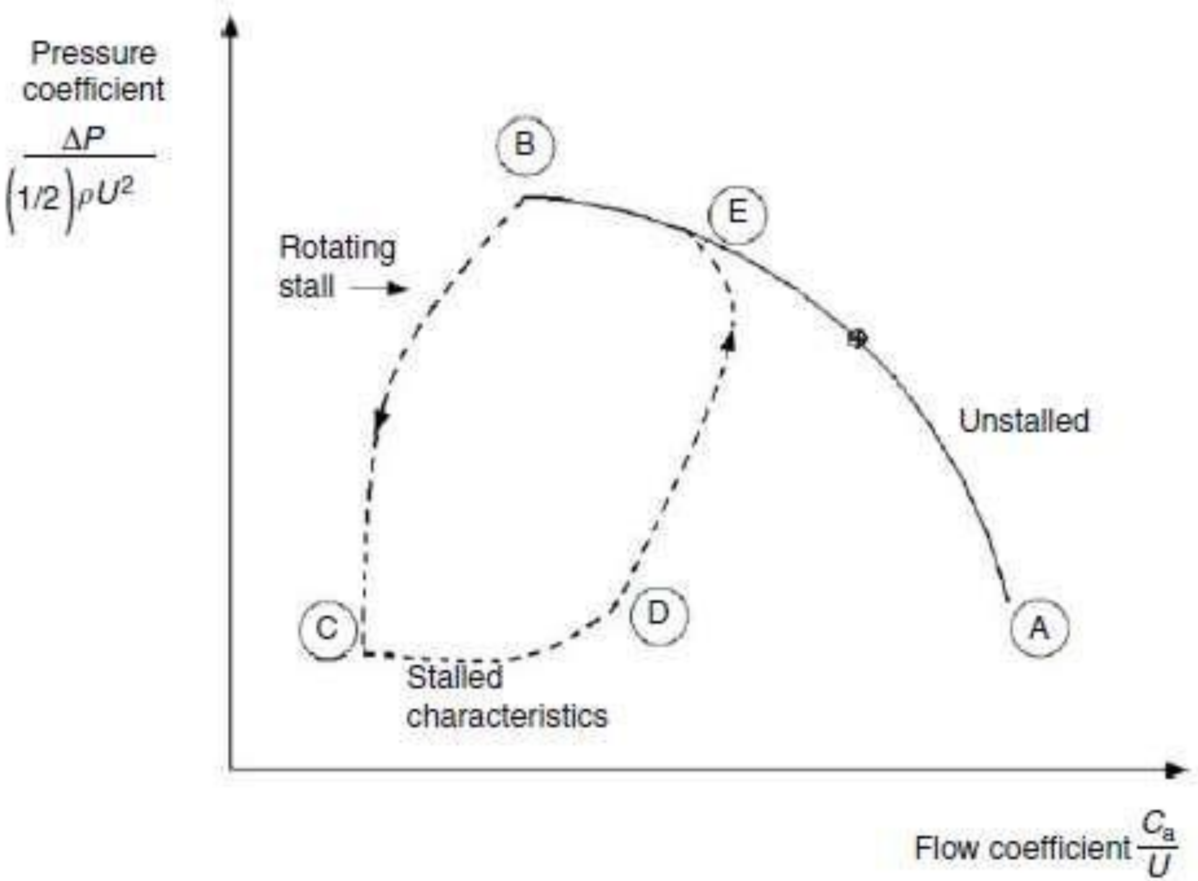
$$\eta_b = 1 - \frac{\overline{w}/(\rho V_1^2/2)}{\Delta P_{\text{th}}/(\rho V_1^2/2)}$$

Stall And Surge

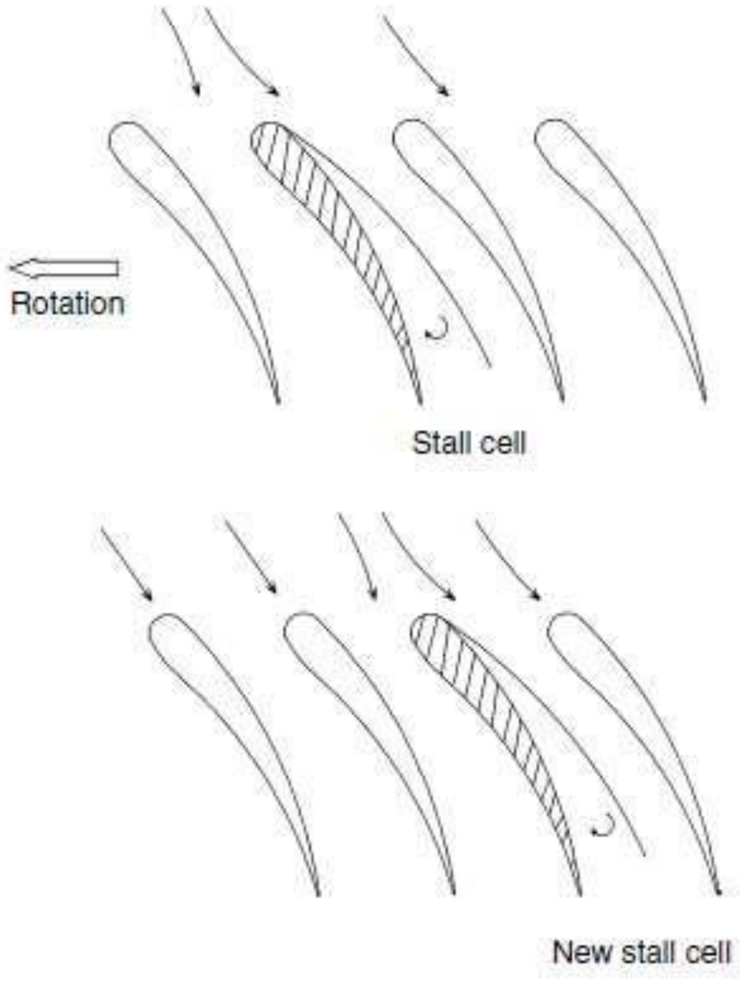
Stall is a situation of abnormal airflow through a single stage or multiple stages of the compressor, while the whole compressor stall, known as compressor surge, results in a loss of engine power. This power failure may only be momentary or may shut down the engine completely causing a flameout.

The following factors can induce compressor stall

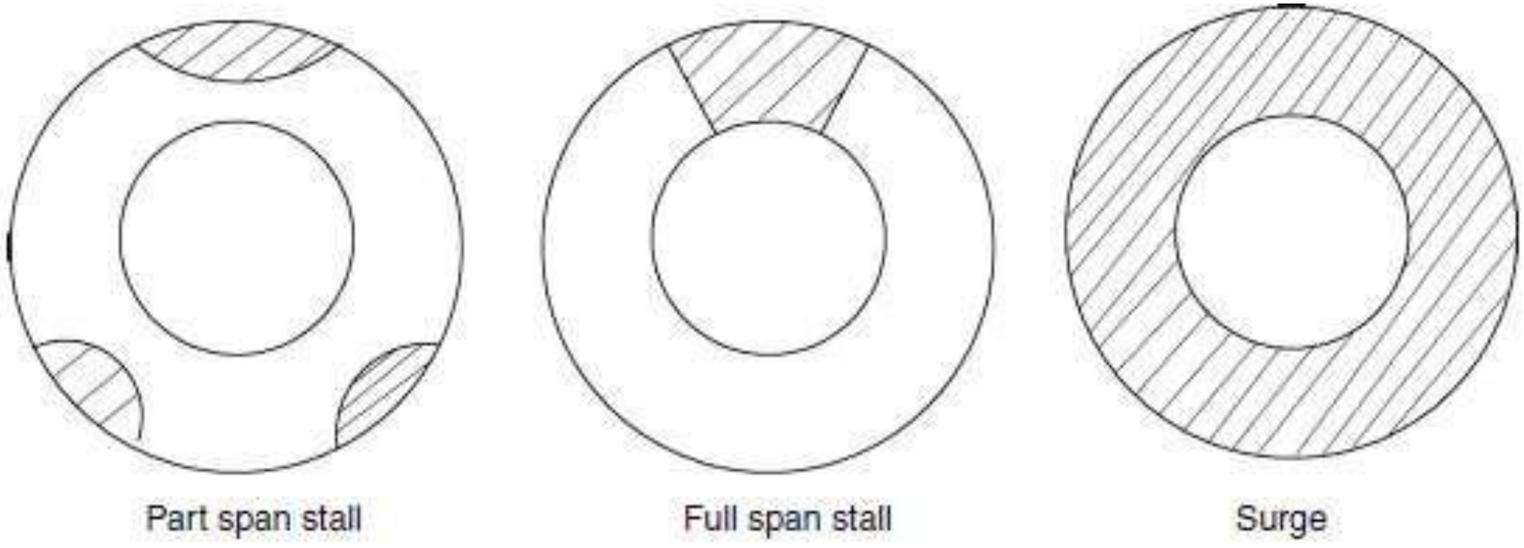
- Engine over-speed
- Engine operation outside specified engineering parameters
- Turbulent or disrupted airflow to the engine intake
- Contaminated or damaged engine components.



Performance characteristics for un-stalled and stalled operation

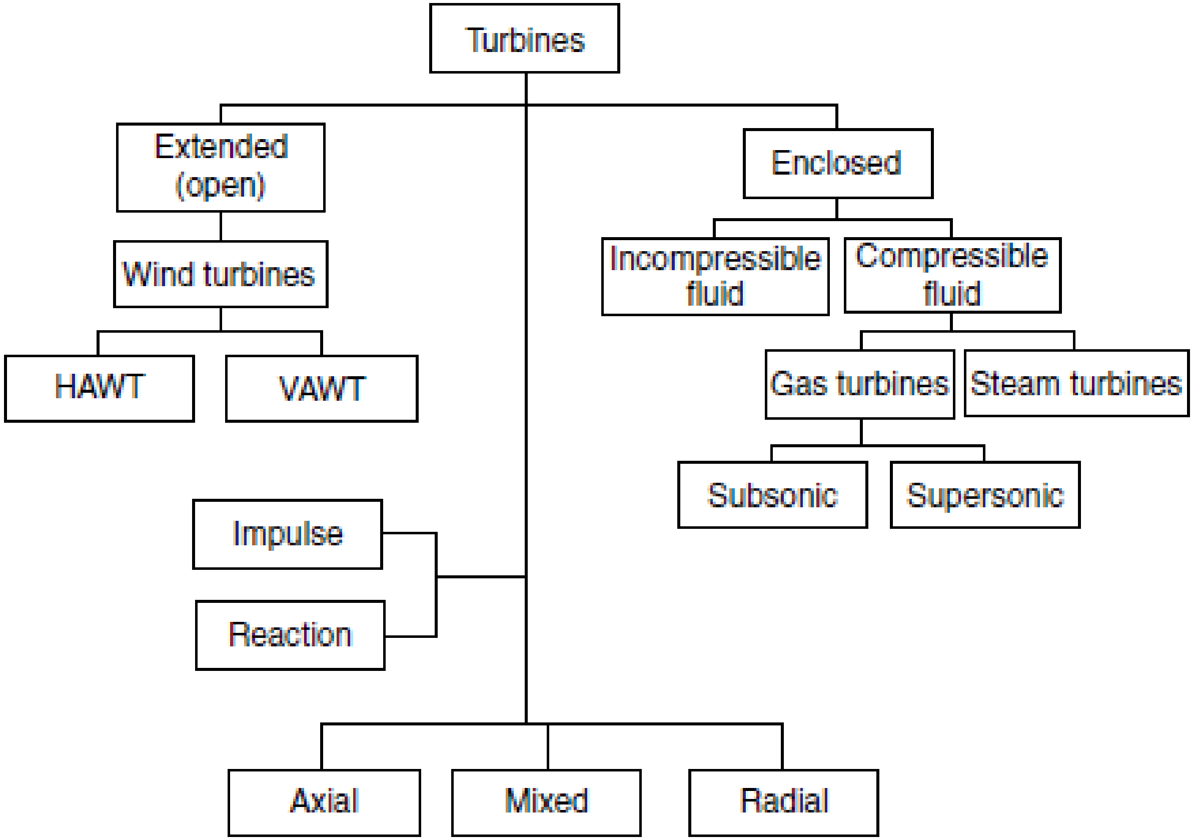


The propagation of rotating stall.



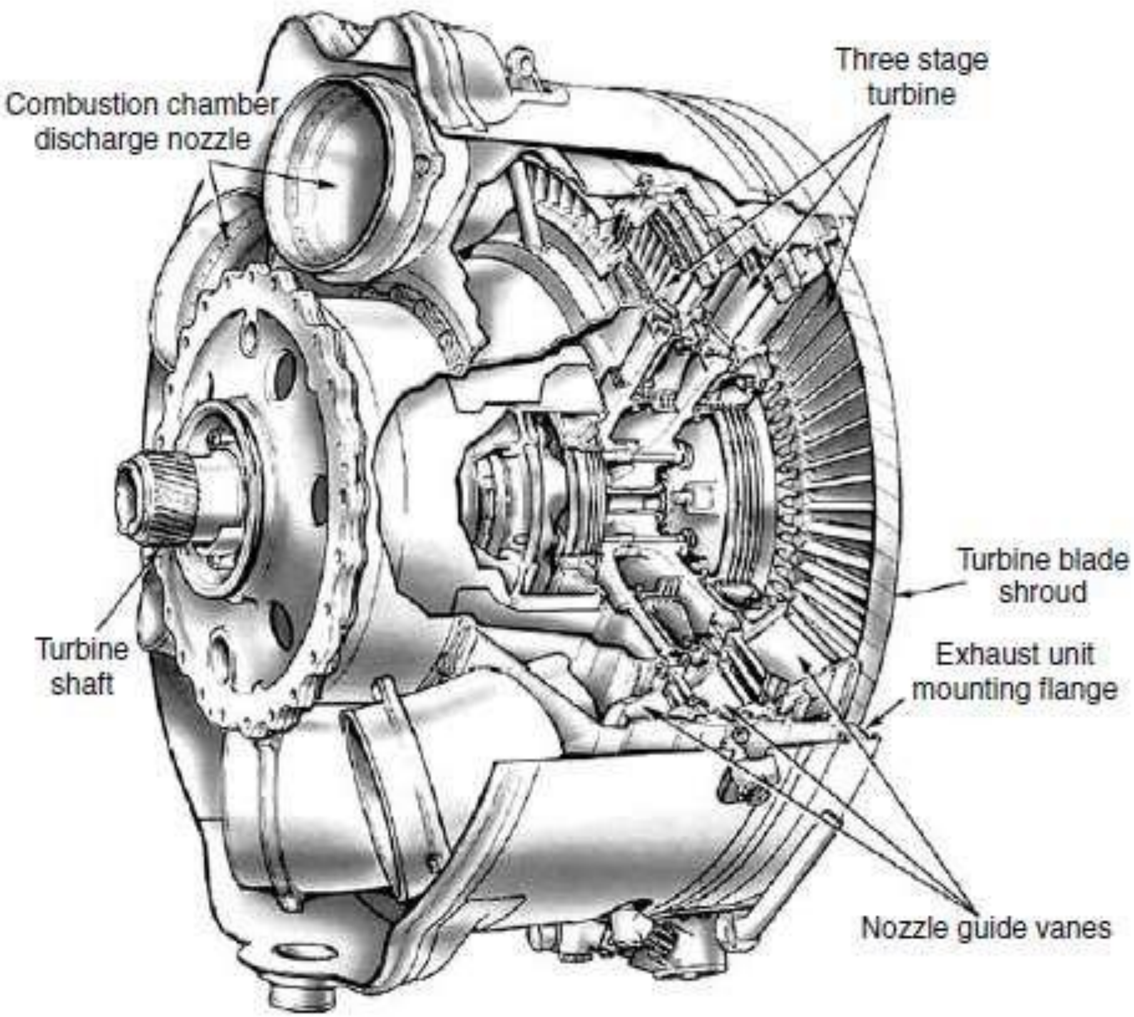
Rotating stall and surge.

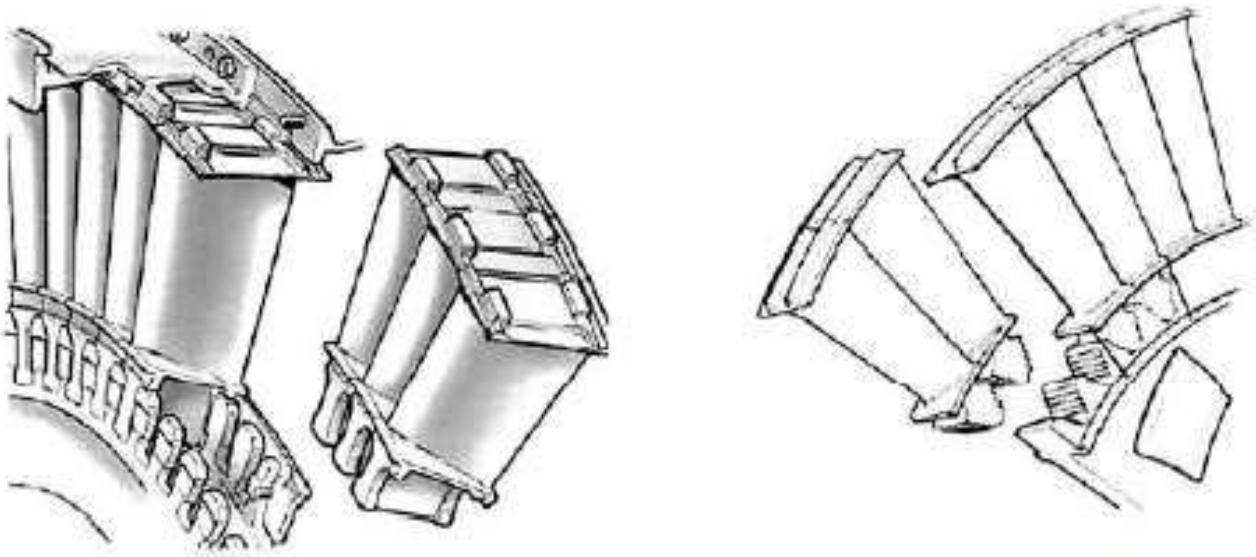
- Turbines may be defined as turbo machines that extract energy from the fluid and convert it into mechanical/electrical energy.
- Turbines may be classified based on the surrounding fluid, whether it is extended or enclosed. Example for extended turbines is the wind turbines, which may be horizontal axis wind turbines (HAWT) or vertical axis wind turbines (VAWT).
- The term gas turbine applies to a turbine with hot gases released from the combustion system as their working fluid. Virtually, all turbines in aircraft engines are of the axial type, regardless of the type of compressor used. Axial turbine is similar to axial compressor operating in reverse.



Classification of turbines.

Turbine Blade Configuration





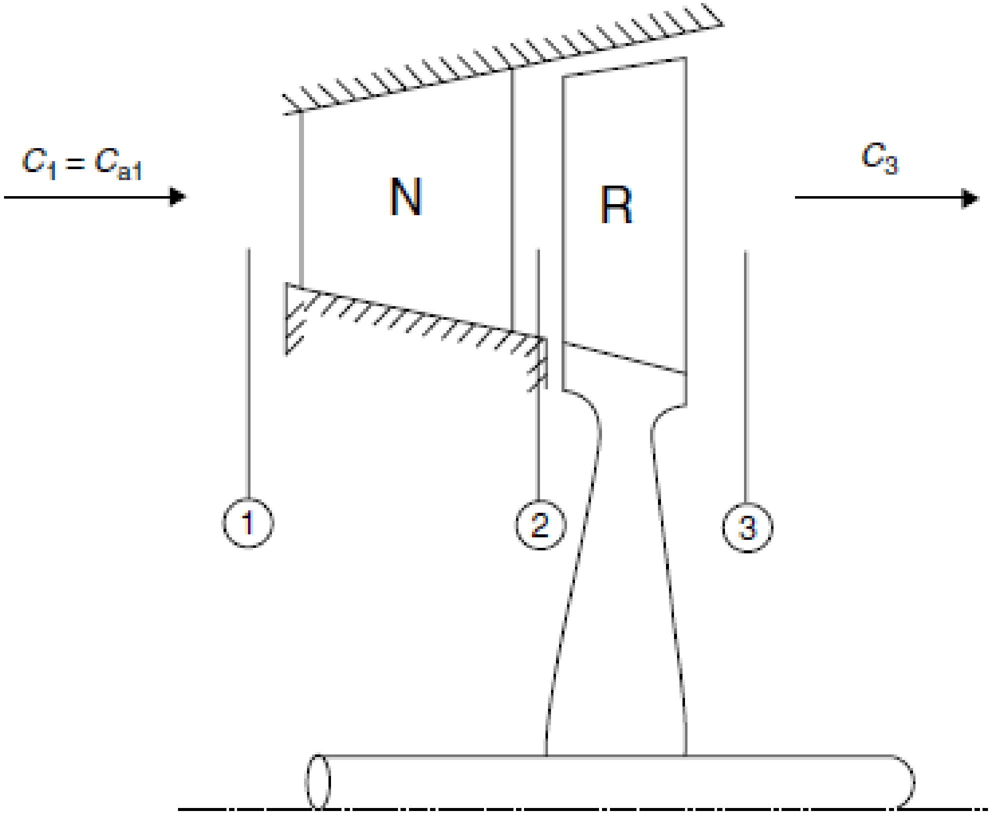
Nozzle guide vanes

Rotor blades

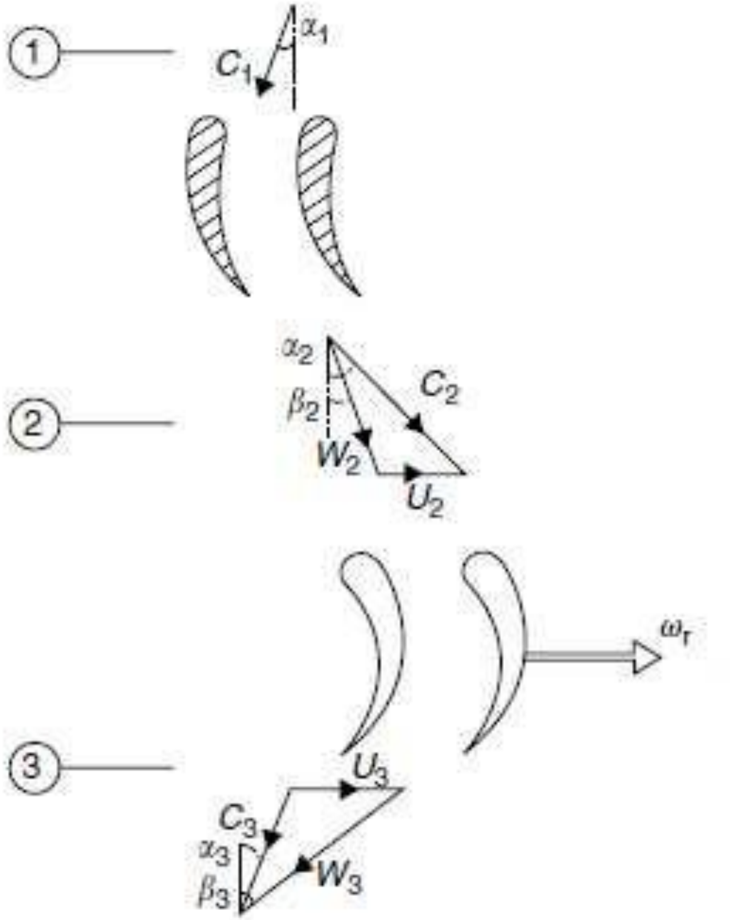
Shrouded blades of both the nozzle and rotor

Velocity Triangles

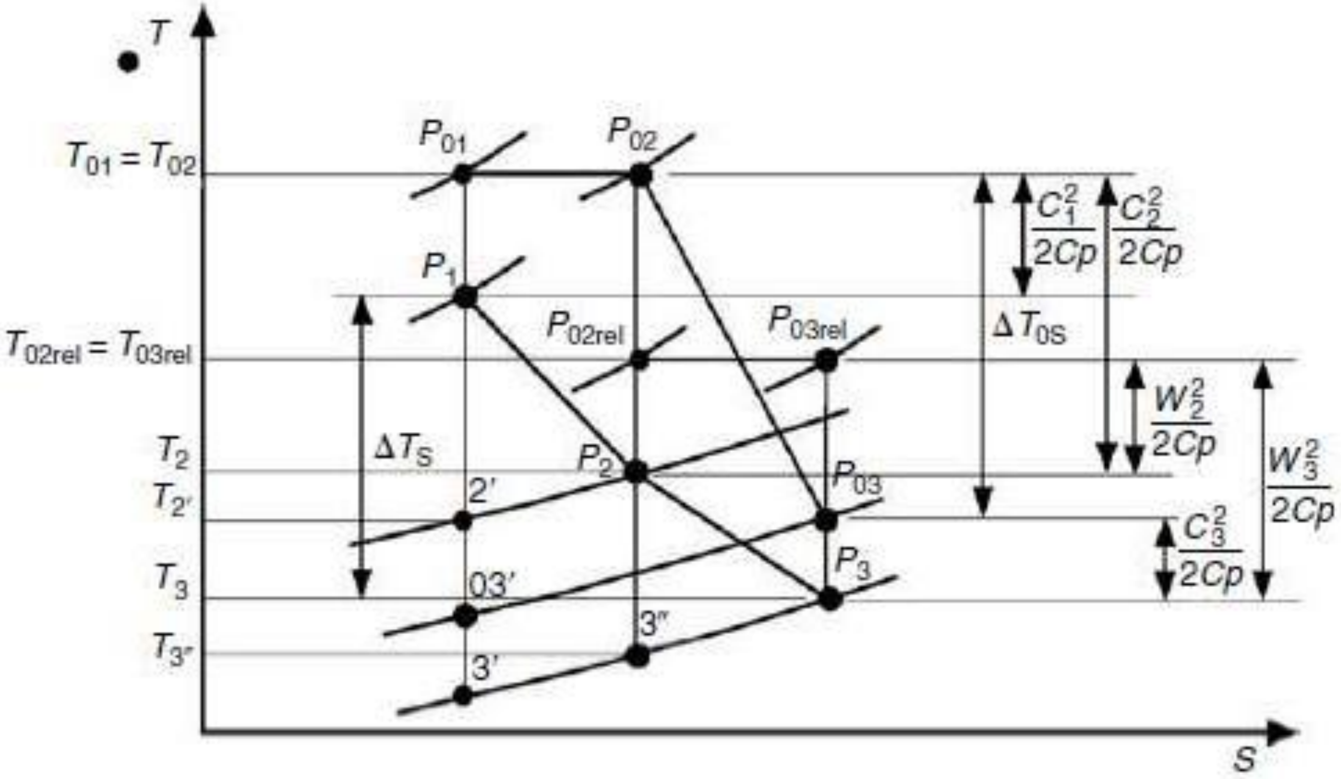
- Single stage of axial turbine is shown in figure next page. The gases leaving the combustion chamber approach the stator (or nozzle) with an absolute velocity (C_1), normally in an axial direction and thus, the absolute angle to the axial direction (α_1=0).
- The flow leaves the stator passage at a speed (C_2), where (C_2>C_1). The static pressure decreases as usual as the gas passes through the nozzle. Moreover, total pressure decrease due to skin friction and other sources of losses will be described later. Thus, $P_1 > P_2$, $P_{01} > P_{02}$.



Layout of an axial turbine stage.



Velocity triangle for turbine



T–S diagram for a reaction turbine stage having equal mean radii at inlet and outlet of rotor

Efficiency, Losses, And Pressure Ratio

The pressure ratio of a stage, πs is given by the relation

$$\pi_s = \frac{P_{01}}{P_{03}} = \frac{1}{(1 - (\Delta T_{0s}/\eta_s T_{01}))^{\gamma/(\gamma-1)}}$$

The turbine efficiency η_t is related to the stage efficiency η_s by the relation

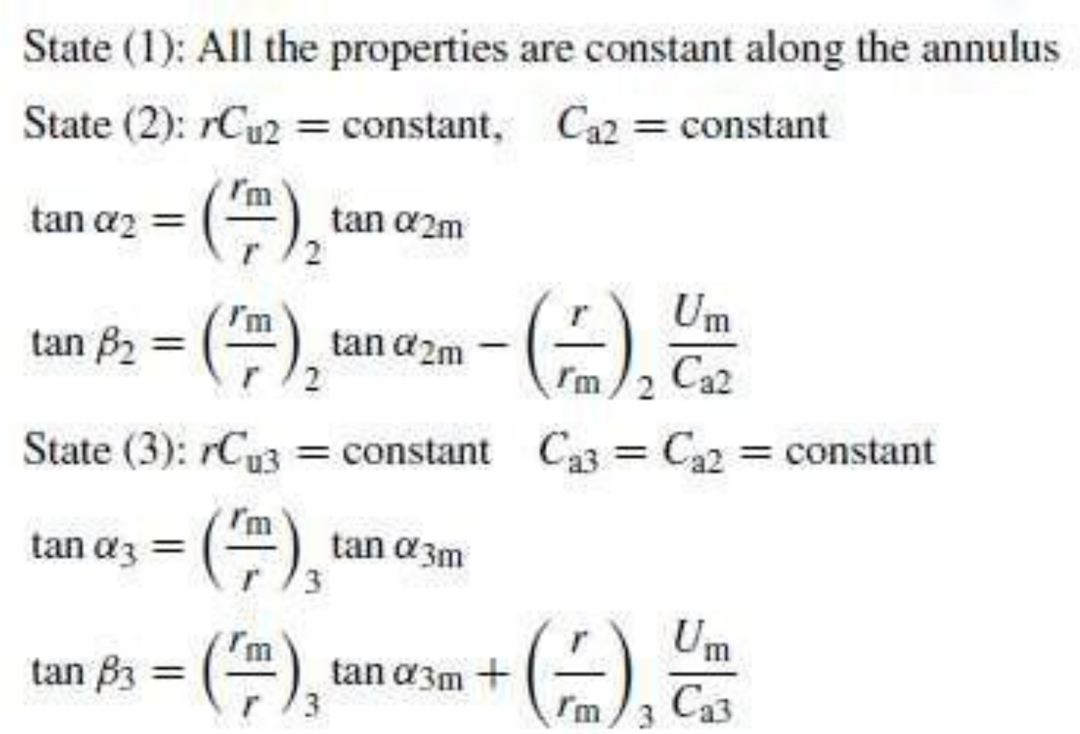
$$\eta_t = \frac{1 - \pi_t^{(\gamma-1)/\gamma}}{1 - \pi_t^{\eta_s(\gamma-1)/\gamma}}$$

Loss coefficient in nozzle: It is expressed either as an enthalpy loss coefficient (λ_N) or as a pressure loss coefficient (YN) [4], where

$$\lambda_N = \frac{T_2 - T_2'}{(C_2^2/2)\,Cp}$$

Free Vortex Design

It is usually assumed that the total enthalpy (h_{01})and entropy are constant at entry to stage, or $dh_{01}/dr = 0$. A free vortex stage will be obtained if $dh_{01}/dr = 0$, and the whirl velocity components at rotor inlet and outlet satisfy the following conditions:



State (1): All the properties are constant along the annulus

State (2): $rC_{u2} = \text{constant}, \quad C_{a2} = \text{constant}$

$$\tan \alpha_2 = \left(\frac{r_m}{r}\right)_2 \tan \alpha_{2m}$$

$$\tan \beta_2 = \left(\frac{r_m}{r}\right)_2 \tan \alpha_{2m} - \left(\frac{r}{r_m}\right)_2 \frac{U_m}{C_{a2}}$$

State (3): $rC_{u3} = \text{constant} \quad C_{a3} = C_{a2} = \text{constant}$

$$\tan \alpha_3 = \left(\frac{r_m}{r}\right)_3 \tan \alpha_{3m}$$

$$\tan \beta_3 = \left(\frac{r_m}{r}\right)_3 \tan \alpha_{3m} + \left(\frac{r}{r_m}\right)_3 \frac{U_m}{C_{a3}}$$

Constant Nozzle Angle Design (A_2)

An alternative design procedure to free vortex flow is the constant nozzle angle. The appropriate conditions that also provide radial equilibrium are

1. Uniform flow in the annulus space between the nozzles and rotor blades, which is satisfied when the outlet flow to nozzle is uniform ($dh_{02}/dr = 0$).
2. Constant nozzle outlet angle (α_2) that avoids manufacturing nozzles of varying outlet angle.

Three cases are available for rotor outlet, namely,

a) Constant total conditions at outlet

b) Free vortex at outlet

c) Zero whirl at outlet

Main Design Steps

The following steps are used in preliminary analysis:

1. **The number of stages** (n) is first determined by assuming the temperature drop per stage.
2. **Aerodynamic design:** It may be subdivided into mean line design and variations along the blade span.
3. **Blade profile selection:** The blade profile and the number of blades for both stator and rotor are determined.
4. **Structural analysis:** It includes mechanical design for blades and discs, rotor dynamic analysis, and modal analysis. Normally, both aerodynamic and mechanical designs are closely connected and there is considerable iteration between them.

5. **Cooling:** The different methods of cooling are examined. In most cases, combinations of these methods are adopted to satisfy an adequate lifetime. Structural and cooling analyses determine the material of both stator and rotor blades.
6. **Check stage efficiency**.
7. **Off-design.**
8. **Rig testing**

Losses And Efficiency

Losses are dependent on several parameters including the blade geometry, incidence angle, Reynolds number, the ratios s/c, h/c.tmax/c, Mach number, and turbulence level. There are three major components of loss, namely, the profile loss, annulus and secondary flow losses, and tip clearance loss. The overall blade loss coefficient is identified as either Y or λ, which is equal to the sum of these three types.

Profile Loss (Y_p)

The profile loss is the loss due to skin friction on the area of the blade surface. It depends on several factors including the area of blade in contact with fluid, the surface finish, and the Reynolds and Mach numbers of the flow through the passage.

$$Y_{\mathrm{p}} = \left\{ Y_{\mathrm{p}(\beta_2=0)} + \left(\frac{\beta_2}{\beta_3}\right)^2 \left[Y_{\mathrm{p}(\beta_2=\beta_3)} - Y_{\mathrm{p}(\beta_2=0)}\right] \left(\frac{t/c}{0.2}\right)^{\beta_2/\beta_3} \right\}$$

Annulus Loss

The profile loss is caused by friction and associated with the boundary layer growth over the inner and outer walls of the annulus. Annulus losses are similar to profile losses as both are caused by friction. However, a fresh boundary layer grows from the leading edge of blade whereas the annulus boundary layer may have its origin some way upstream of the leading edge depending on the details of the annulus itself.

Secondary Flow Loss

Secondary flows are contra rotating vortices that occur due to curvature of the passage and boundary layers. Secondary flows tend to scrub both the end wall and blade boundary layers and redistribute low momentum fluid through the passage.

Tip Clearance Loss (Y_k)

Tip clearance loss occurs in the rotors. Some fluid leaks in the gap between the blade tip and the shroud, and therefore contributes little or no expansion work.

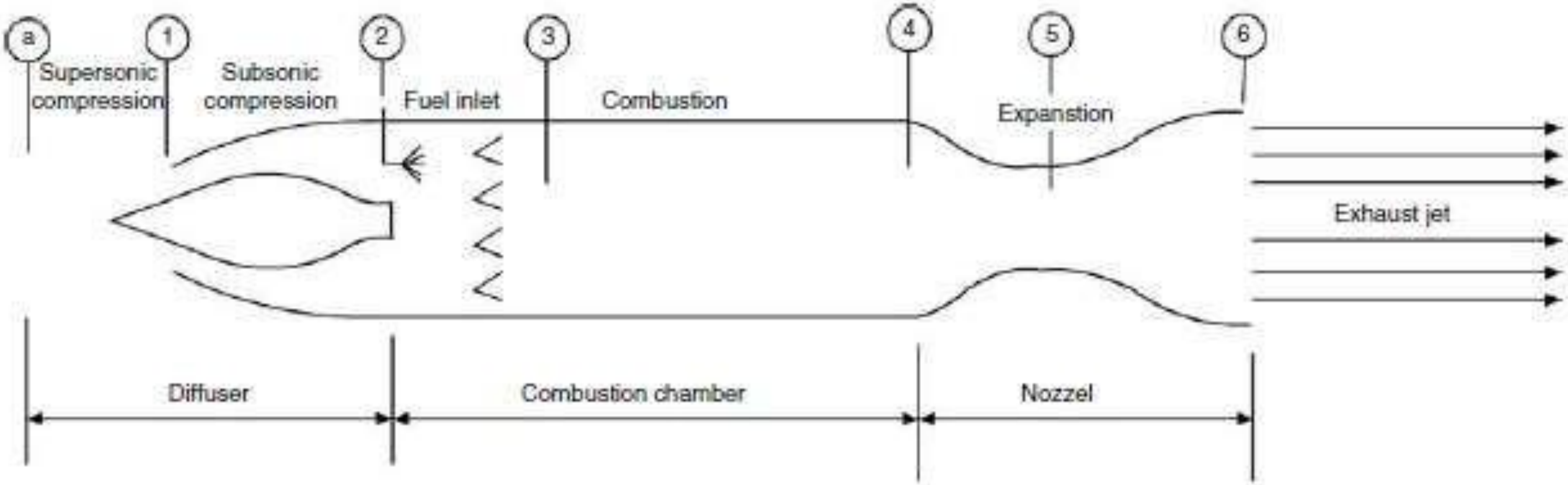
The combined secondary loss and tip leakage are expressed by the relation

$$Y_s + Y_k = \left[\lambda + B\left(\frac{k}{h}\right)\right]\left(\frac{C_L}{s/c}\right)^2\left[\frac{\cos^2\beta_3}{\cos^3\beta_m}\right]$$

Ramjet Engine

- A ramjet has no moving parts, much like a valveless pulsejet but they operate with continuous combustion rather than the series of explosions that give a pulsejet its characteristic noise.
- Ramjet engine may be of the subsonic or supersonic type. Although ramjet can operate at subsonic flight speed, the increasing pressure rise accompanying higher flight speeds renders the ramjet most suitable for supersonic flight.
- The ramjet has been called a flying stovepipe, because it is open at both ends and has only fuel nozzles in the middle.
- A straight stovepipe would not work, however; a ramjet must have a properly shaped inlet-diffusion section to produce low-velocity, high-pressure air at the combustion section and it must also have a properly shaped exhaust nozzle.

Ramjet Engine



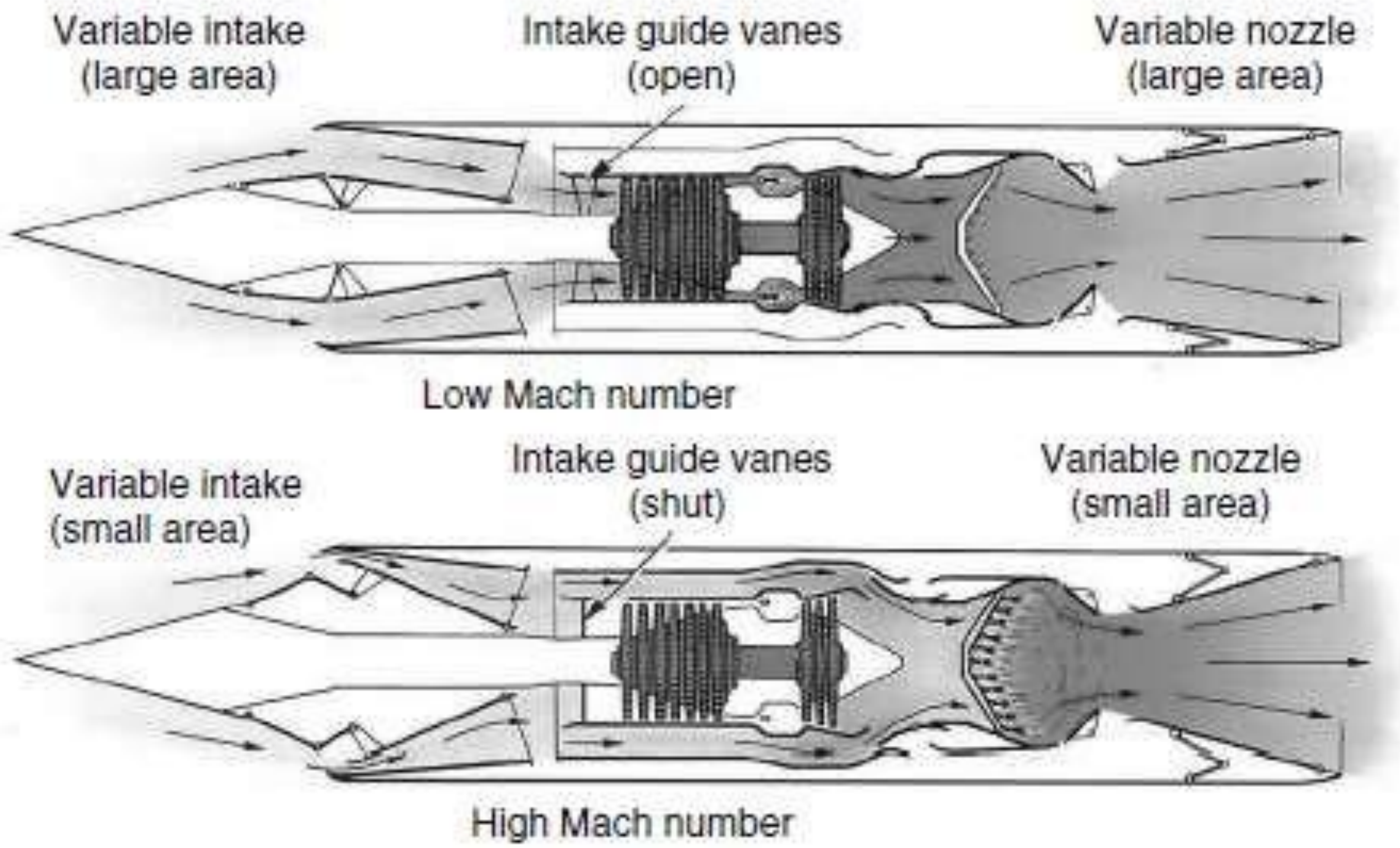
The relation between total and static conditions (temperature and pressure) at the inlet and outlet of the engine, states (a) and (6 or e) are

$$\left.\begin{aligned} \frac{T_{0a}}{T_a} &= 1 + \frac{\gamma_a - 1}{2} M^2 = \frac{T_{02}}{T_a} \\ \frac{T_{0e}}{T_e} &= \frac{T_{06}}{T_6} = 1 + \frac{\gamma_6 - 1}{2} M_e^2 = \frac{T_{04}}{T_e} \end{aligned}\right\}$$

$$\left.\begin{aligned} \frac{P_{0a}}{P_a} &= \left(1 + \frac{\gamma_a - 1}{2} M^2\right)^{\frac{\gamma_a}{\gamma_a - 1}} \\ \frac{P_{06}}{P_e} &= \left(1 + \frac{\gamma_6 - 1}{2} M_e^2\right)^{\frac{\gamma_6}{\gamma_6 - 1}} \end{aligned}\right\}$$

$$u_e = \frac{a_e}{a} u = \sqrt{\frac{\gamma_6 R T_e}{\gamma_a R T_a}}\, u \qquad \therefore f = \frac{(C_{p_4} T_{04} / C_{p_2} T_{0a}) - 1}{(Q_R / C_{p_2} T_{0a}) - (C_{p_4} T_{04} / C_{p_2} T_{0a})}$$

Turboramjet



A turbo/ramjet engine.

UNIT– IV

SOLID-PROPELLANT ROCKET MOTORS

UNIT - IV

CLO's	Course Learning Outcomes
CLO10	Appreciate the different propellant feed system options for both chemical and electric propulsion systems, and their similarities/differences.
CLO11	Demonstrate the salient features of solid propellants rockets and estimate the grain configuration designs suitable for different missions.
CLO12	Identify the applications of standard and reverse hybrid systems with an overview of its limitations.

INTRODUCTION

- Solid propellant motors are the simplest of all rocket designs. They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust.
- When ignited, a solid propellant burns from the center out towards the sides of the casing.
- The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust.
- Unlike liquid propellant engines, solid propellant motors cannot be shut down.
- Once ignited, they will burn until all the propellant is exhausted.

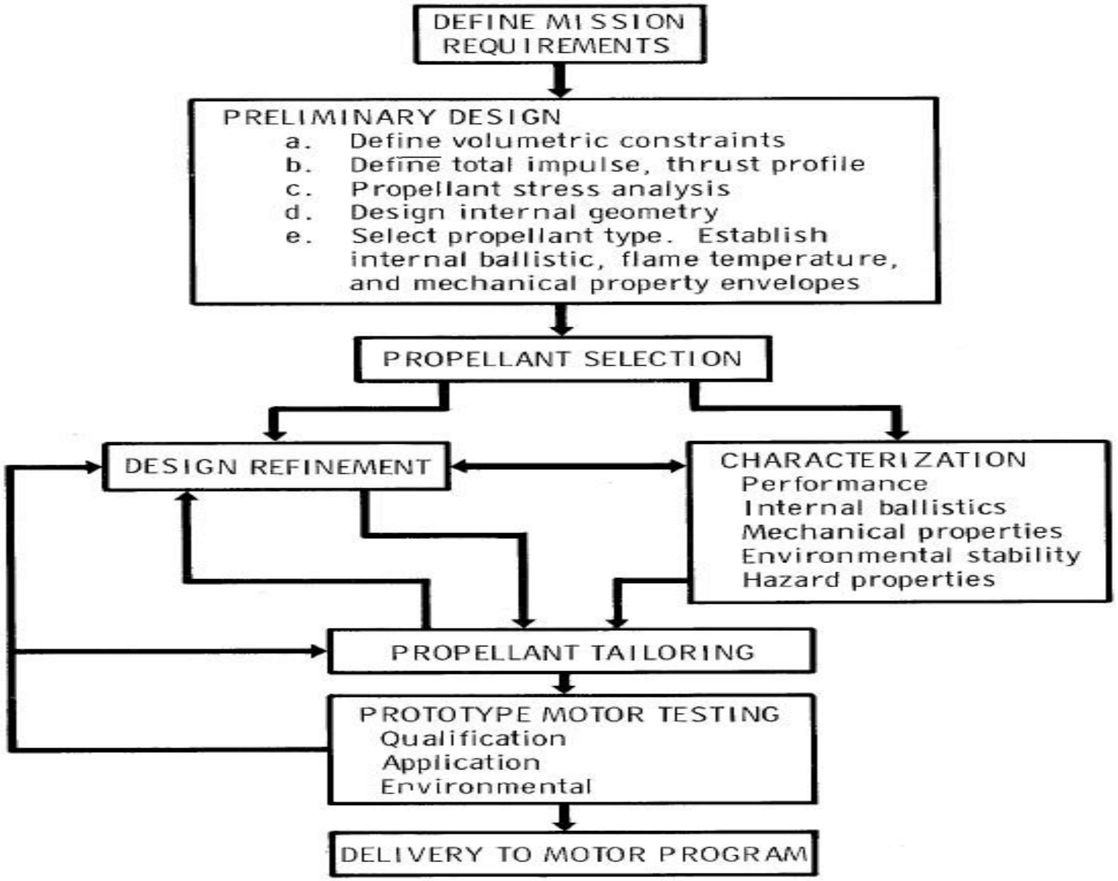


Figure 3.1 propellant selection

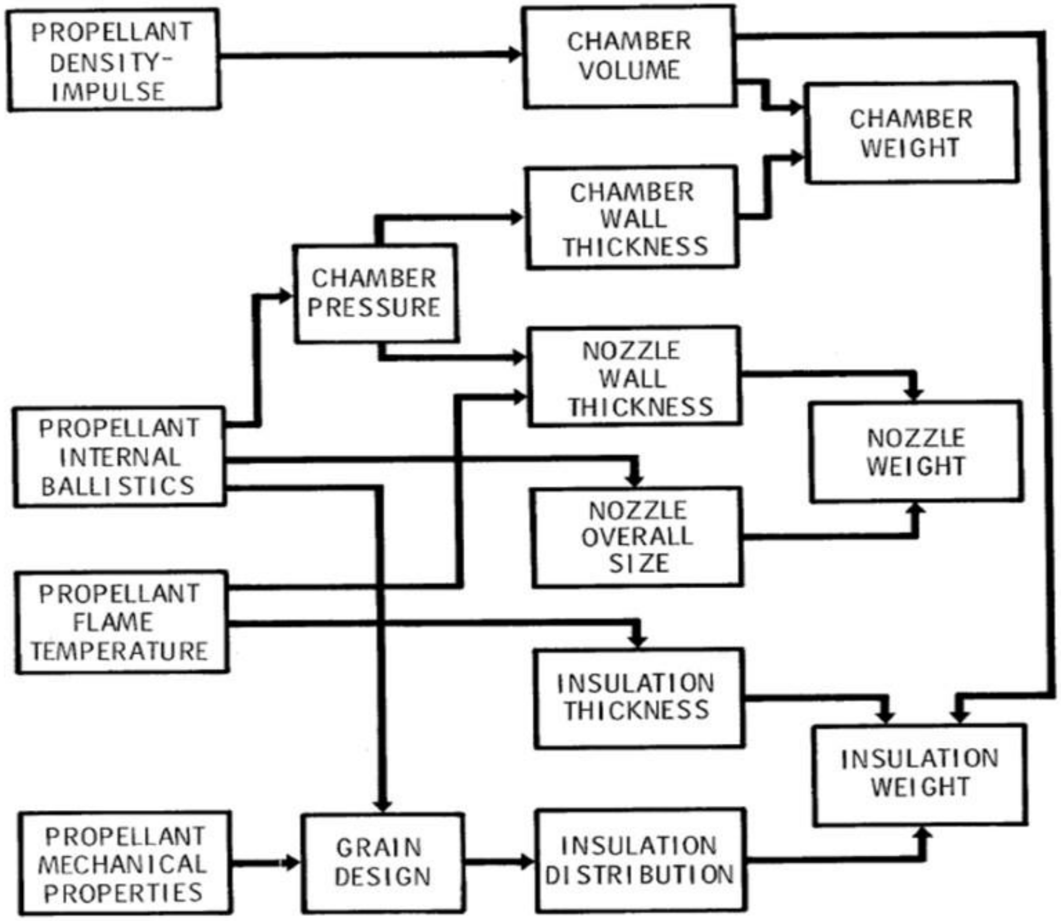


Figure 3.2 influence of propellant properties

Propellant grain design considerations:

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

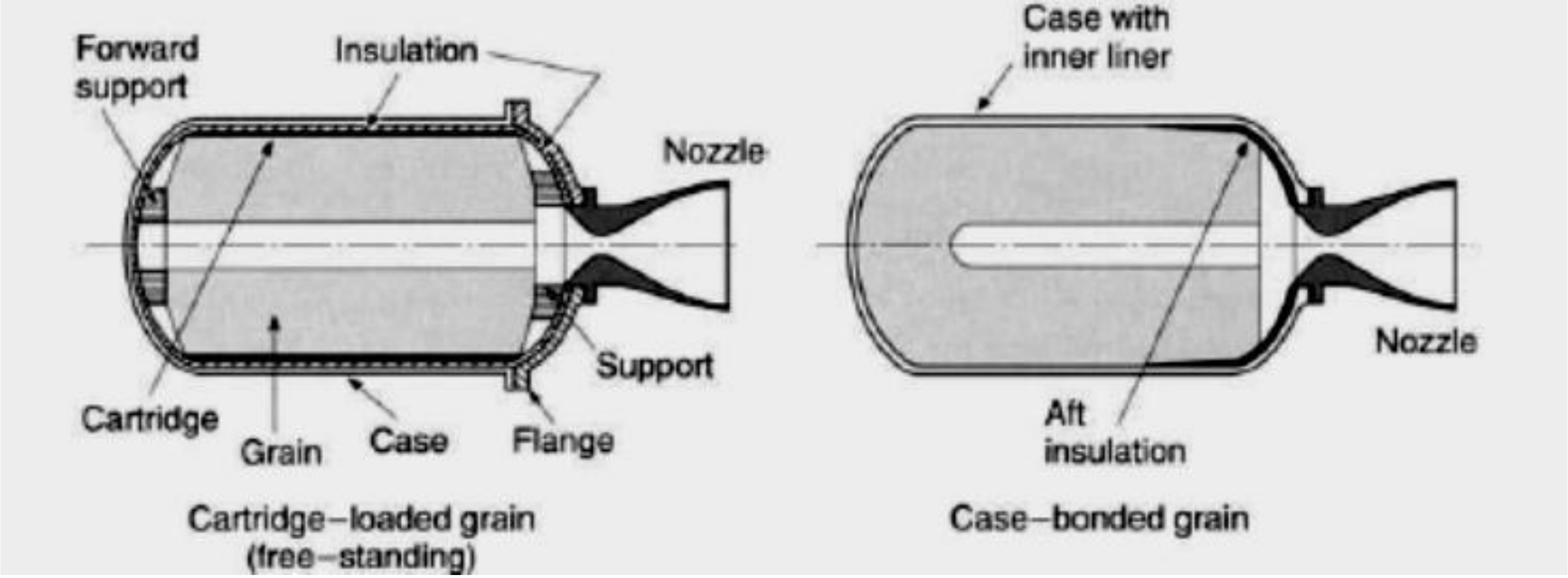


Figure 3.3 Simplified schematic diagrams of a free-standing (or cartridge-loaded) and a case-bonded grain.

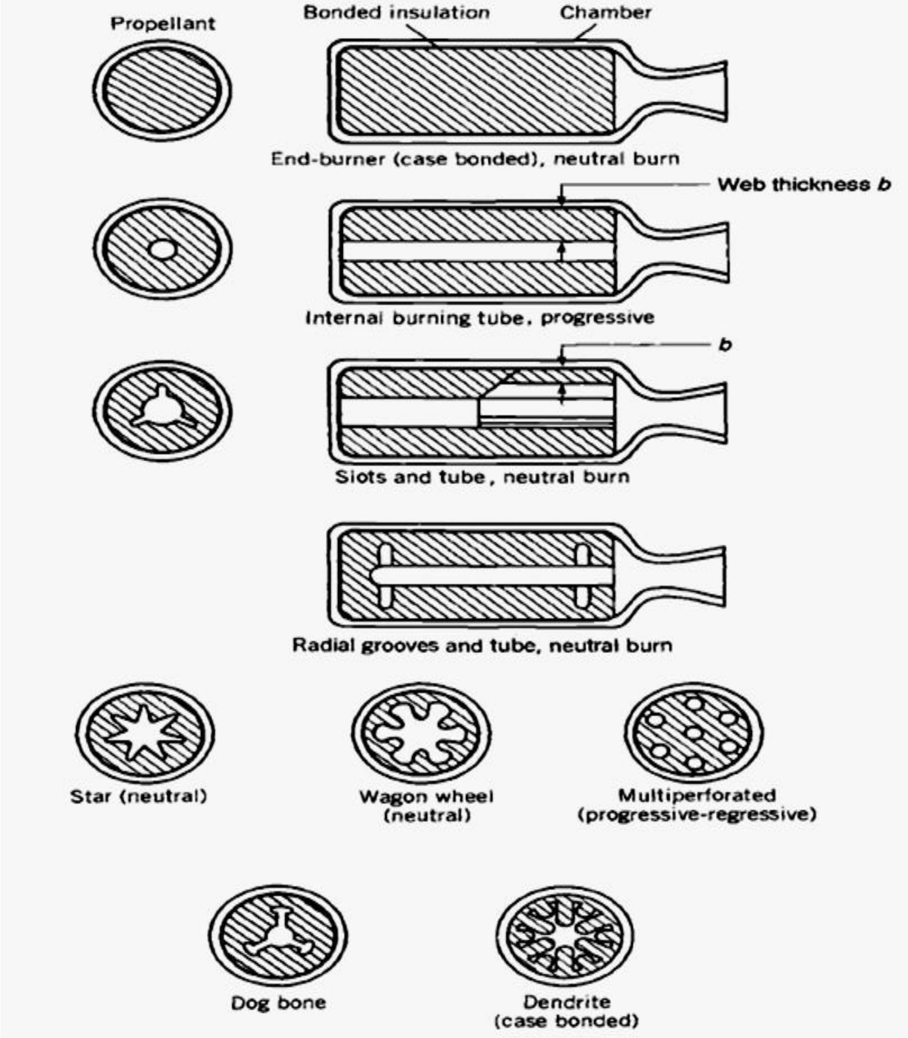


Fig. 3.4 Grain design configurations

UNIT– V

LIQUID PROPELLANT ROCKET ENGINES: PROPELLANT TYPES

UNIT - V

CLO’s	Course Learning Outcomes
CLO13	Discuss the various feed systems and injectors for liquid propellants rockets and associated heat transfer problems.
CLO14	Appreciate the different propellant feed system options for both chemical and electric propulsion systems, and their similarities/differences.
CLO15	Discuss the various feed systems and injectors for liquid propellants rockets and associated heat transfer problems.

INTRODUCTION

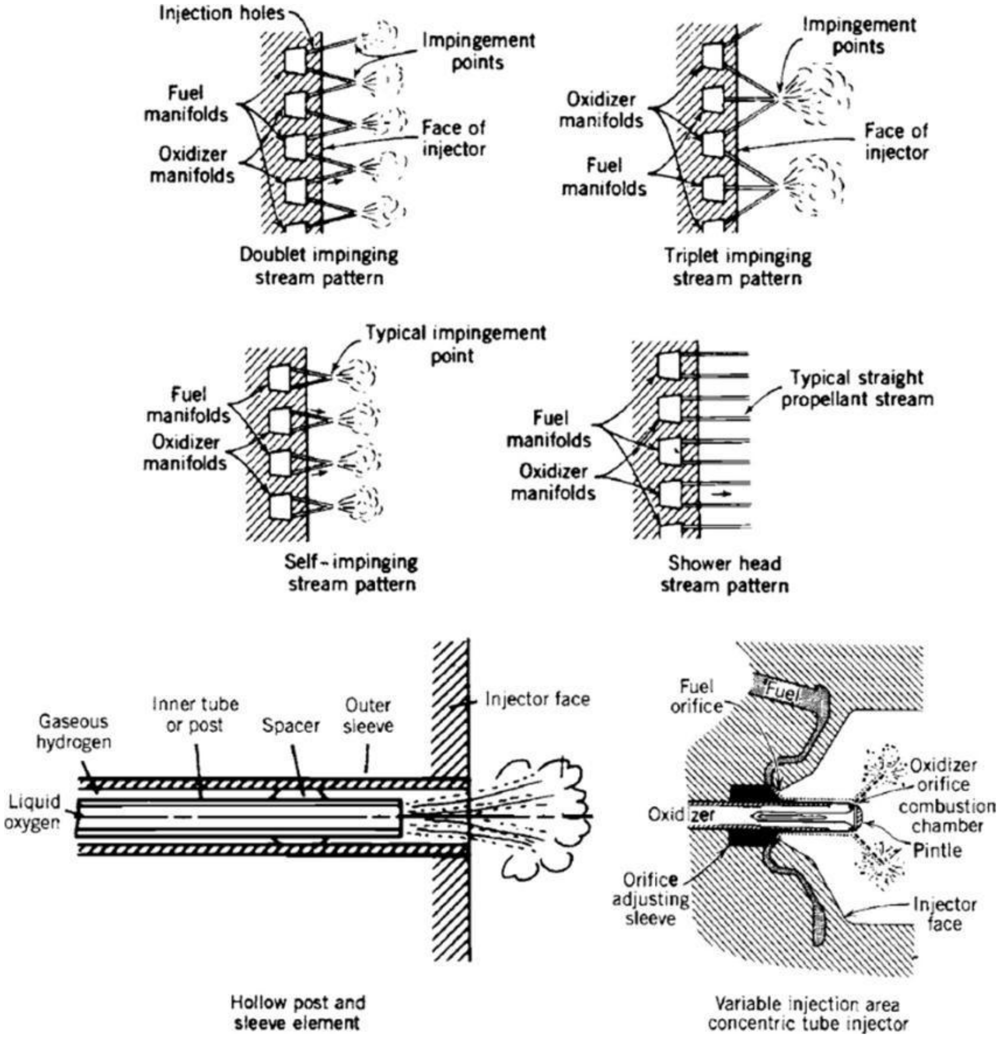
The propellants, which are the working substance of rocket engines, constitute the fluid that undergoes chemical and thermodynamic changes. The term liquid propellant embraces all the various liquids used and may be one of the following:

1. Oxidizer (liquid oxygen, nitric acid, etc.)
2. Fuel (gasoline, alcohol, liquid hydrogen, etc.).
3. Chemical compound or mixture of oxidizer and fuel ingredients, capable of self-decomposition.
4. Any of the above, but with a gelling agent.

SELECTION OF LIQUID PROPELLANTS

- Mission Definition
- Affordability (Cost)
- System Performance
- Survivability (Safety)
- Reliability
- Controllability
- Maintainability
- Geometric Constraints
- Prior Related Experience
- Operability
- Producibility
- Schedule
- Environmental Acceptability
- Reusability

INJECTOR



Doublet impinging stream pattern

Triplet impinging stream pattern

Self-impinging stream pattern

Shower head stream pattern

Hollow post and sleeve element

Variable injection area concentric tube injector

Figure 4.1 Schematic Diagrams Of Several Injector Types.

Combustion Instability In Liquid Propellant Rockets

•The main effect of increasing the oxidizer mass flux in a hybrid rocket is the occurrence of problems such as flame holding and combustion instabilities (usually in combination with lower combustion efficiency), eventually followed by complete blow-off of the flame.

•Combustion instability and low combustion efficiency have been observed at mass flux levels above 650 kg/s·m2 in N2O-paraffin hybrid rockets at both laboratory scale and full-scale items such as the Peregrine engine.

•Blow-off is expected above a certain upper mass flux level which is inferred by the analysis conducted in several literature references on the flooding limit of hybrid rocket engines.

- However, it is still unclear how to quantify this upper mass flux level and what are the critical design factors influencing it, with different references even providing contradictory information in this respect.

- Furthermore, this upper mass flux limit is usually studied by means of analytical or semi-empirical models and little experimental data are available.

COMBUSTION MECHANISM IN HYBRID PROPELLANT ROCKETS

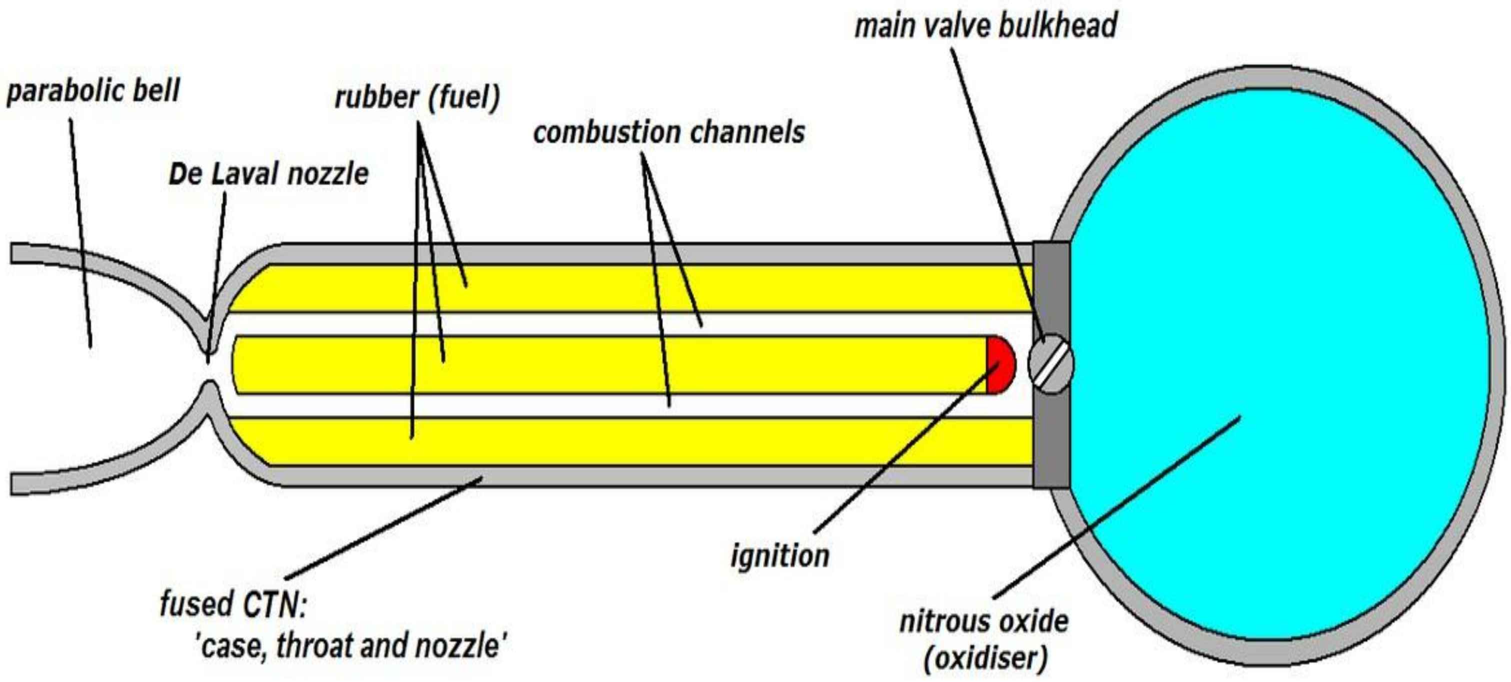
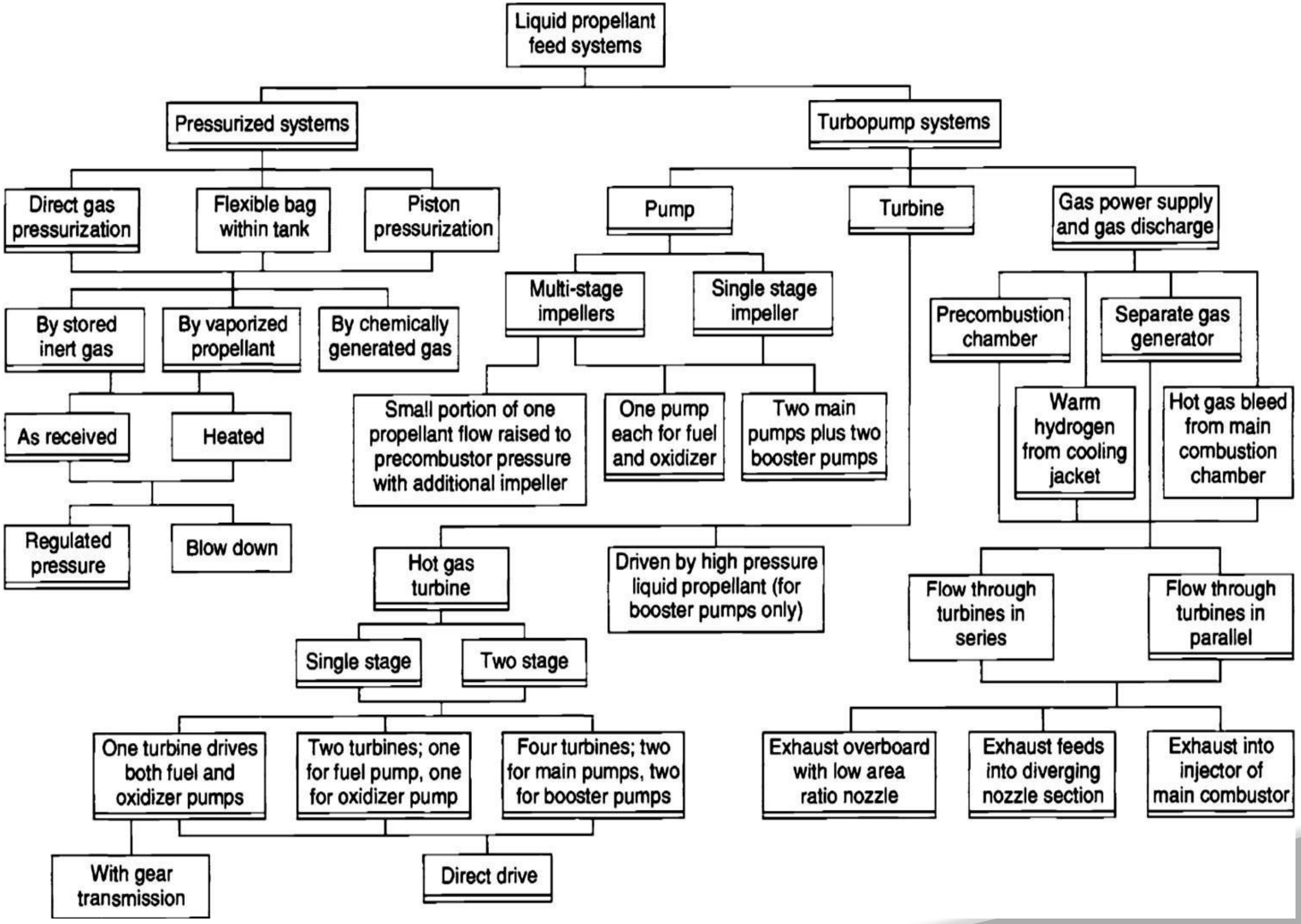


Figure 4.2 hybrid propellant

PROPELLANT FEED SYSTEMS



Liquid propellant feed systems

Pressurized systems

Turbopump systems

Direct gas pressurization

Flexible bag within tank

Piston pressurization

Pump

Turbine

Gas power supply and gas discharge

Multi-stage impellers

Single stage impeller

Precombustion chamber

Separate gas generator

By stored inert gas

By vaporized propellant

By chemically generated gas

Warm hydrogen from cooling jacket

Hot gas bleed from main combustion chamber

As received

Heated

Small portion of one propellant flow raised to precombustor pressure with additional impeller

One pump each for fuel and oxidizer

Two main pumps plus two booster pumps

Regulated pressure

Blow down

Hot gas turbine

Driven by high pressure liquid propellant (for booster pumps only)

Flow through turbines in series

Flow through turbines in parallel

Single stage

Two stage

One turbine drives both fuel and oxidizer pumps

Two turbines; one for fuel pump, one for oxidizer pump

Four turbines; two for main pumps, two for booster pumps

Exhaust overboard with low area ratio nozzle

Exhaust feeds into diverging nozzle section

Exhaust into injector of main combustor

With gear transmission

Direct drive

PROBLEMS ASSOCIATED WITH OPERATION OF CRYOGENIC ENGINES

- The thrust comes from the rapid expansion from liquid to gas with the gas emerging from the motor at very high speed.
- The energy needed to heat the fuels comes from burning them, once they are gasses.
- Cryogenic engines are the highest performing rocket motors. Cryogenic engines are fundamentally different from electric motors because there isn't anything rotating in them. They're essentially reaction engines.
- By 'reaction' I'm referring to Newton's law: "to every action there is an equal and opposite reaction.

Contd…

- The cryogenic (or rocket) engine throws mass in one direction, and the reaction to this is a thrust in the opposite direction.
- Therefore, to get the required mass flow rate, the only option was to cool the propellants down to cryogenic temperatures (below −183 °C *90 K+, −253 °C [20 K]), converting them to liquid form.
- Hence, all cryogenic rocket engines are also, by definition, either liquid-propellant rocket engines or hybrid rocket engines Introduction to hybrid rocket propulsion-standard and reverse hybrid systems-combustion mechanism in hybrid propellant rockets applications and limitations.

INTRODUCTION TO HYBRID ROCKET PROPULSION

- Rocket propulsion concepts in which one component of the propellant is stored in liquid phase while the other is stored in solid phase are called hybrid propulsion systems.
- Such systems most commonly employ a liquid oxidizer and solid fuel.
- Various combinations of solid fuels and liquid oxidizers as well as liquid fuels and solid oxidizers have been experimentally evaluated for use in hybrid rocket motors.

The main advantages of a hybrid rocket propulsion system are:

- Safety during fabrication, storage, or operation without any possibility of explosion or detonation;
- Start-stop-restart capabilities;
- Relatively low system cost;
- Higher specific impulse than solid rocket motors and higher density-specific impulse than liquid bipropellant engines; and
- The ability to smoothly change motor thrust over a wide range on demand.

The disadvantages of hybrid rocket propulsion systems are:

- mixture ratio and, hence, specific impulse will vary somewhat during steady-state operation and throttling;
- lower density-specific impulse than solid propellant systems;
- some fuel sliver must be retained in the combustion chamber at end-of burn, which slightly reduces motor mass fraction; and
- unproven propulsion system feasibility at large scale.

Applications

- Hybrid propulsion is well suited to applications or missions requiring throttling, command shutdown and restart, long-duration missions requiring storable nontoxic propellants, or infrastructure operations (manufacturing and launch) that would benefit from a non-self-deflagrating propulsion system.
- Such applications would include primary boost propulsion for space launch vehicles, upper stages, and satellite maneuvering systems.
- Many early hybrid rocket motor developments were aimed at target missiles and low-cost tactical missile applications.
- Other development efforts focused on high-energy upper-stage motors. In recent years development efforts have concentrated on booster prototypes for space launch applications.

HYBRID ROCKET CONFIGURATION

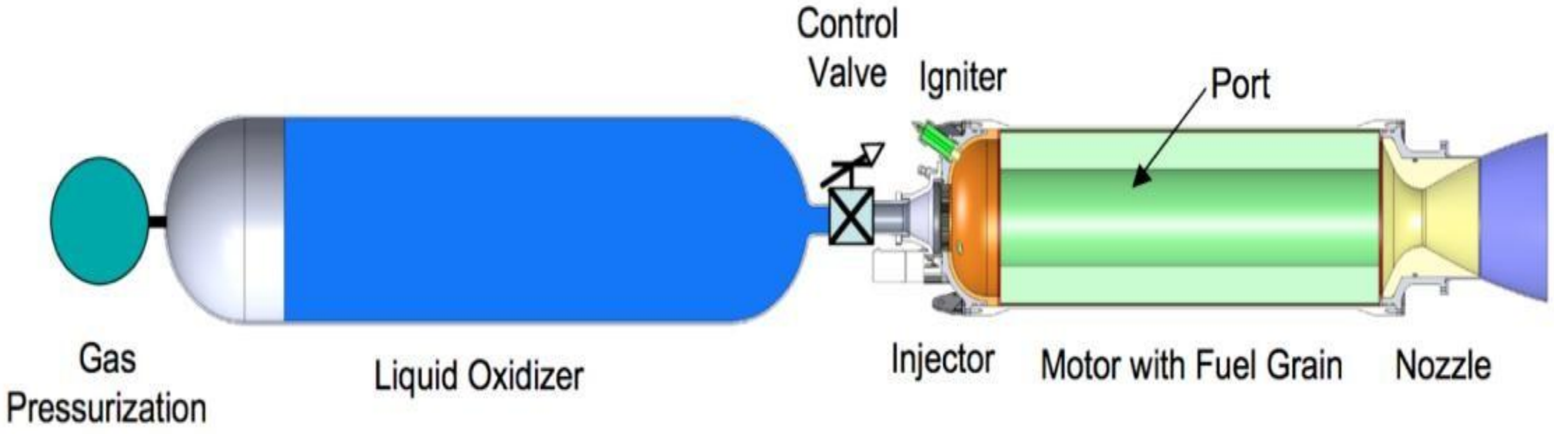


Figure:5.3 Schematic of a hybrid rocket motor

THANK YOU