Aerospace Propulsion and Combustion

III B. Tech VI semester (Autonomous IARE R-16)

BY

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Course Outcomes

COs	Course Outcome
CO1	Gain knowledge about power plants and aircraft engines performance
CO2	Assess the importance of various types engine components used in the aircraft
CO3	Obtain an insight in the concept of propellers, inlets and various nozzles in aircraft
CO4	Assess the significance of combustion inside the engines and its performance
CO5	Estimate the flammability limits, premixed flames and their significance in the combustion zone

UNIT - I
ELEMENTS OF AIRCRAFT PROPULSION

Course Learning Outcomes

CLOs	Course Learning Outcome
CLO1	Apply knowledge and understand the essential facts, concepts and principles of thermodynamics.
CLO2	Understand the basic function of all aircraft engine components and how they work.
CLO3	Analyze classification of aircraft propulsion.
CLO4	Demonstrate different type's aircraft engine operating principle.

ELEMENTS OF AIRCRAFT 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.

PROPULSION

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THRUST

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 mean by air breathing engine?

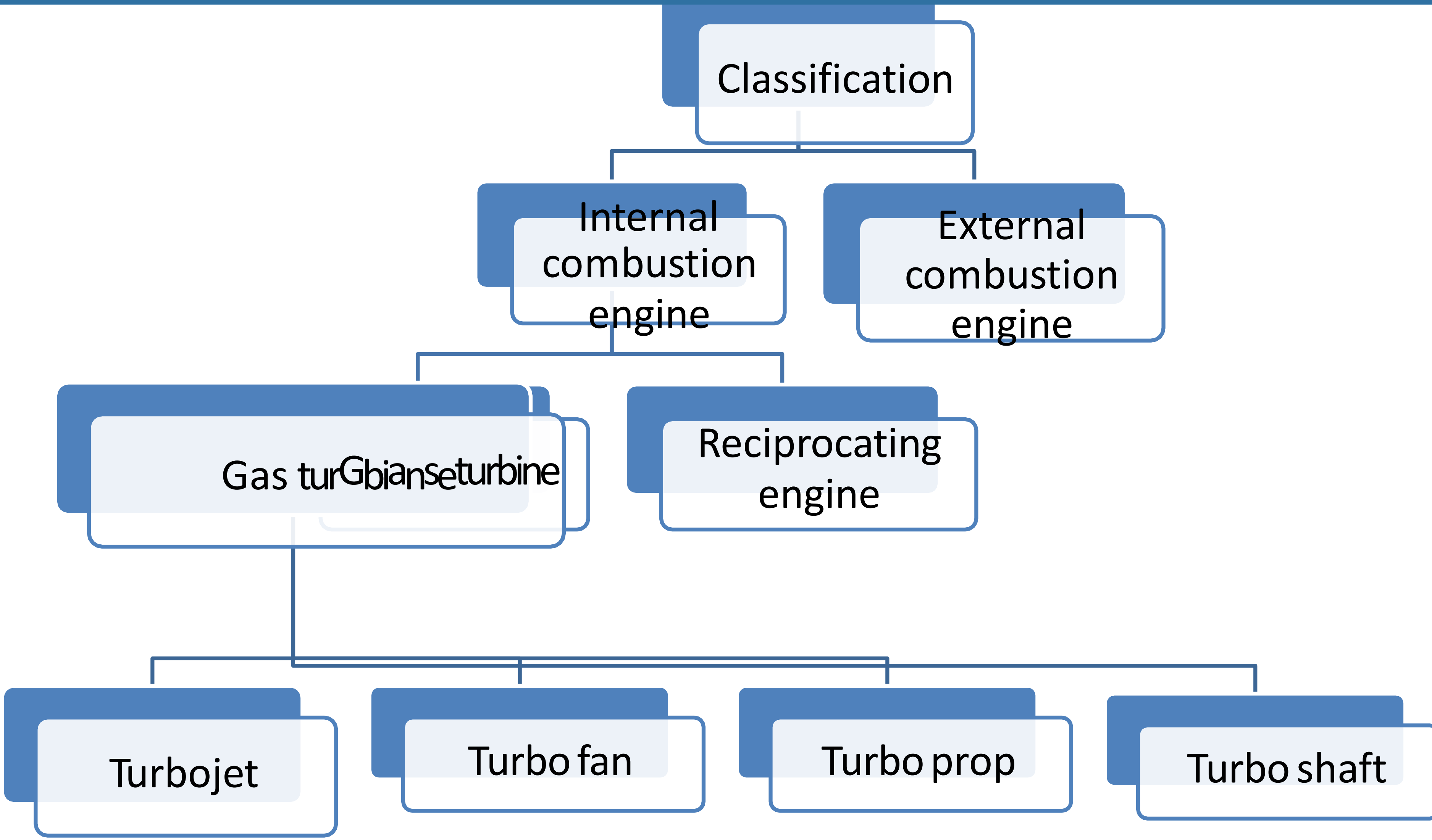
A propulsive unit or engine uses atmospheric air as oxidizer for the combustion process are called air breathing engines.

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.

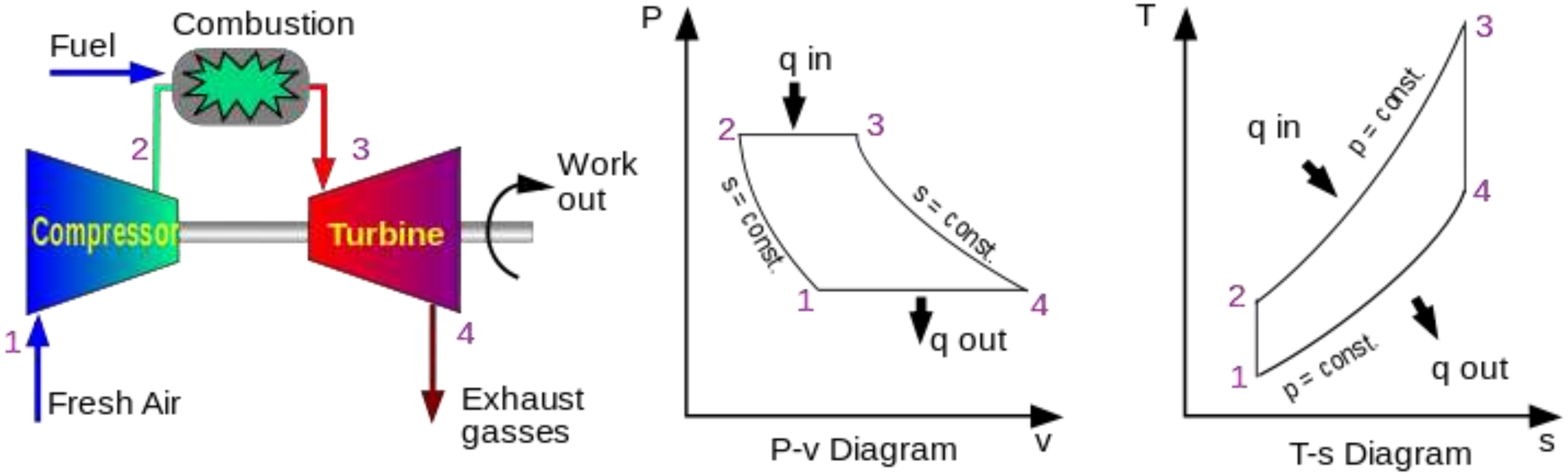
e.g Rocket engine


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

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



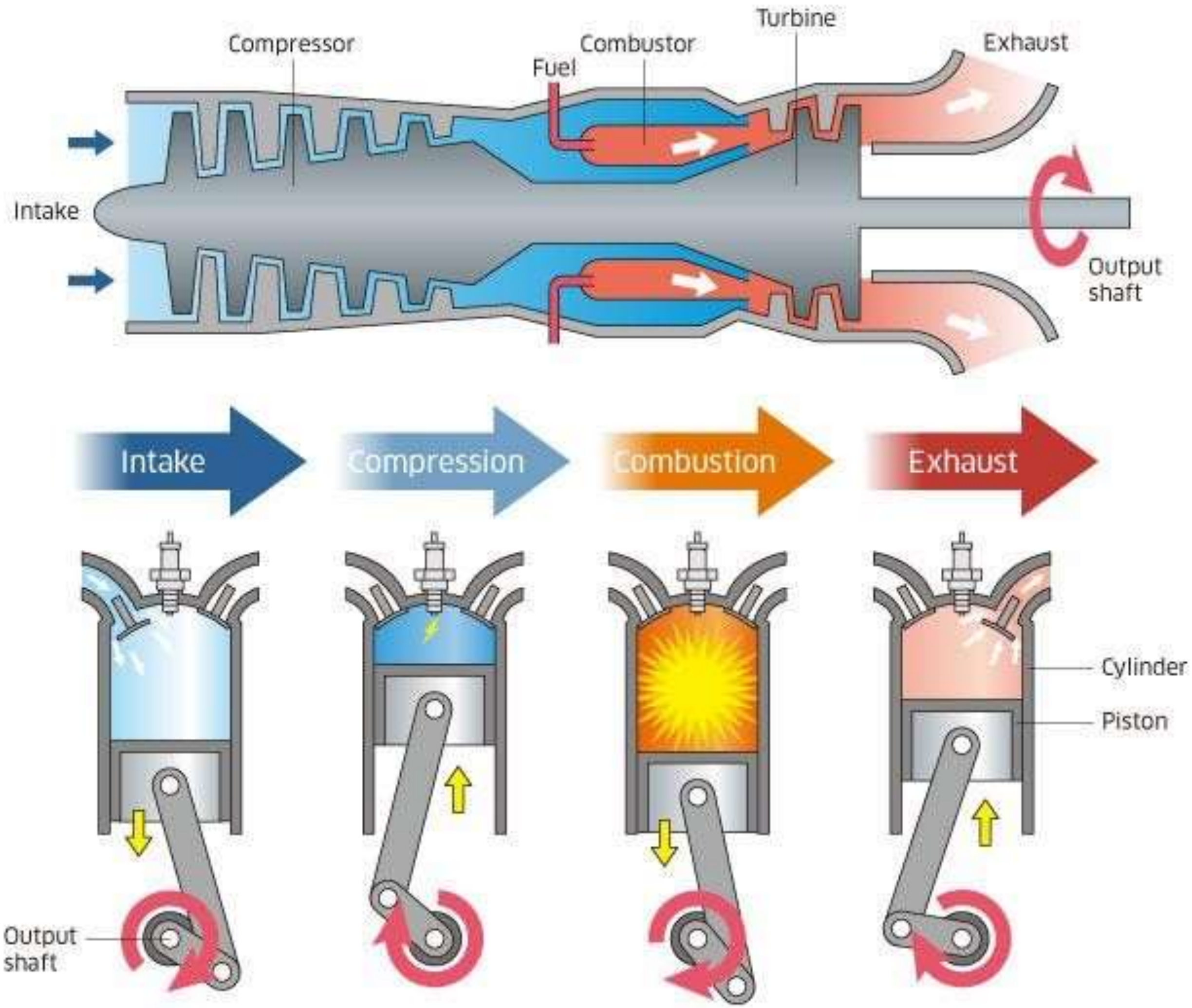
Ideal Brayton cycle:

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.

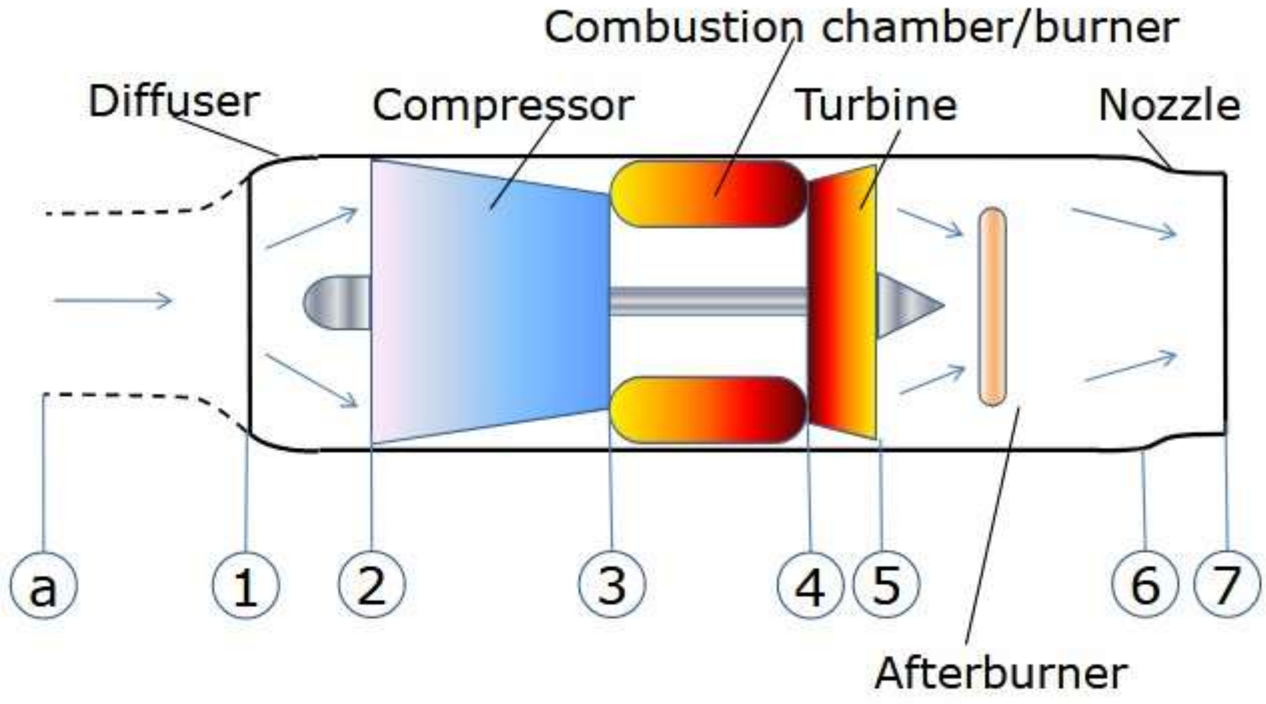
3-4 isentropic process – the heated, pressurized air then gives up its energy, expanding through a turbine (or series of turbines). Some of the work extracted by the turbine is used to drive the compressor.

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


Compressor
Fuel
Combustor
Turbine
Exhaust
Intake
Output
shaft
Intake
Compression
Combustion
Exhaust
Cylinder
Piston
Output
shaft

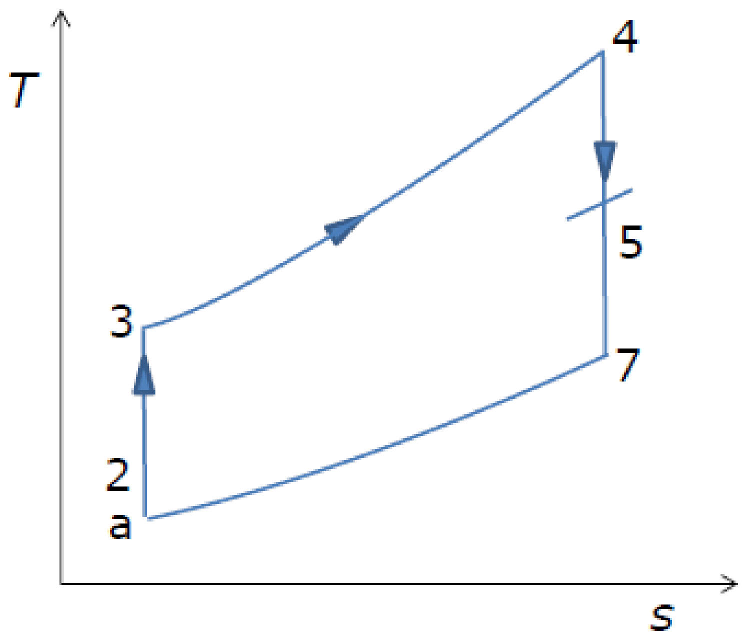
- Gas turbine engines operate on Brayton cycles.
- Ideal Brayton cycle is a closed cycle, whereas gas turbines operate in the open cycle mode.
- Ideal cycle assumes that there are no irreversibilities in the processes, air behaves like an ideal gas with constant specific heats, and that there are no 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

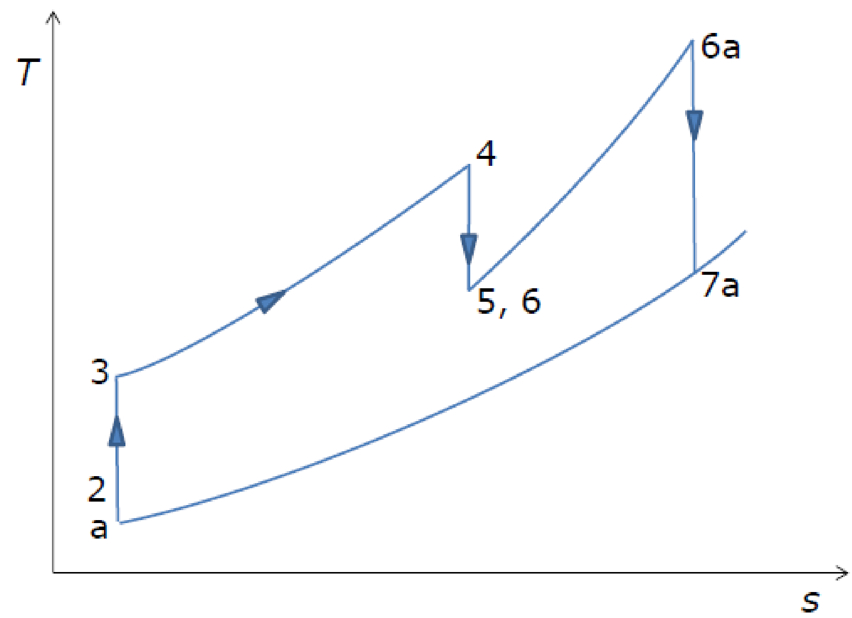


The different processes in a turbojet cycle are the following:

- 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)

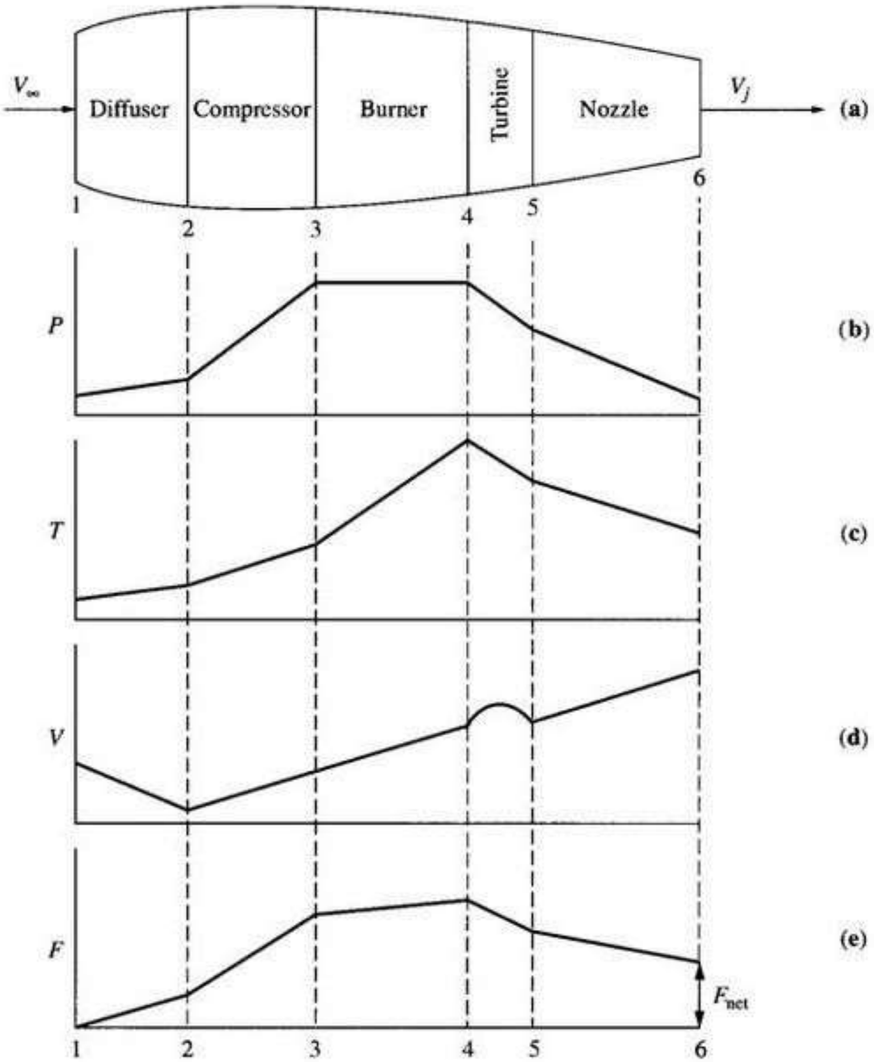


Ideal turbojet cycle with afterburning

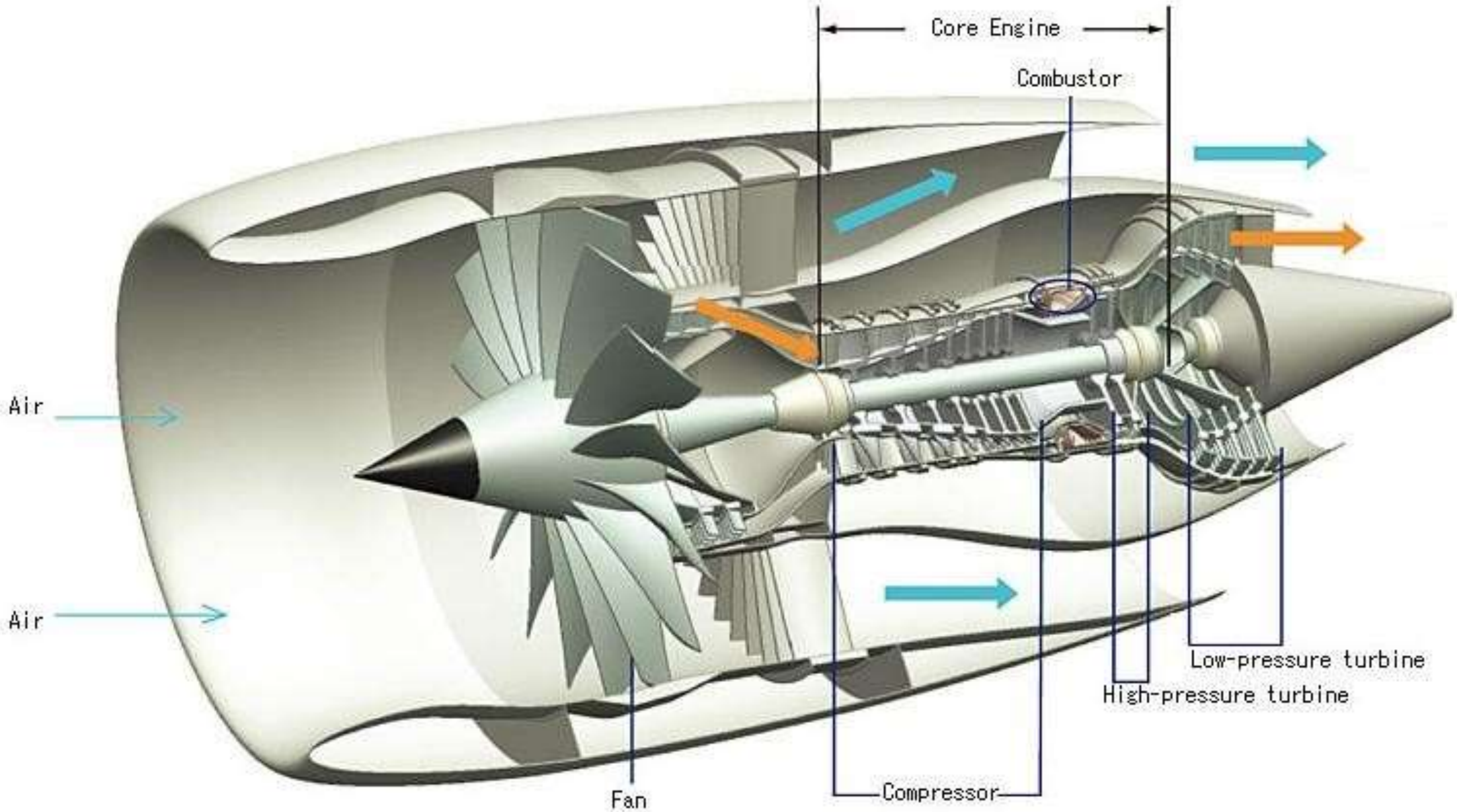
- Afterburning: used when the aircraft needs a substantial increment in thrust. For eg. to accelerate to and cruise at supersonic speeds.

- 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.

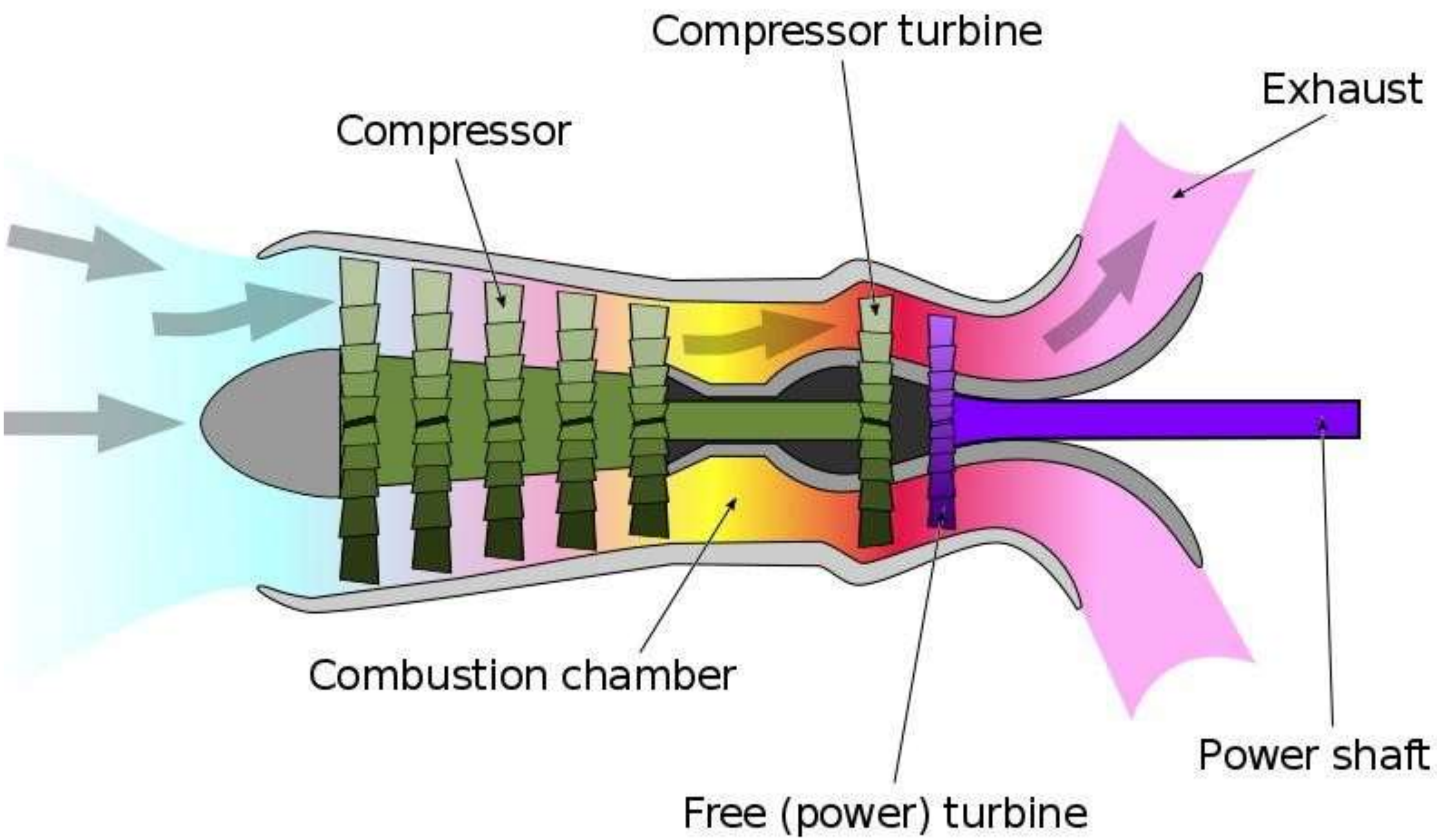


V_∞
Diffuser
Compressor
Burner
Turbine
Nozzle
V_j
(a)
1
2
3
4
5
6
P
(b)
T
(c)
V
(d)
F
(e)
F_{net}
1
2
3
4
5
6


Core Engine
Combustor
Air
Air
Low-pressure turbine
High-pressure turbine
Fan
Compressor

Bypass ratio

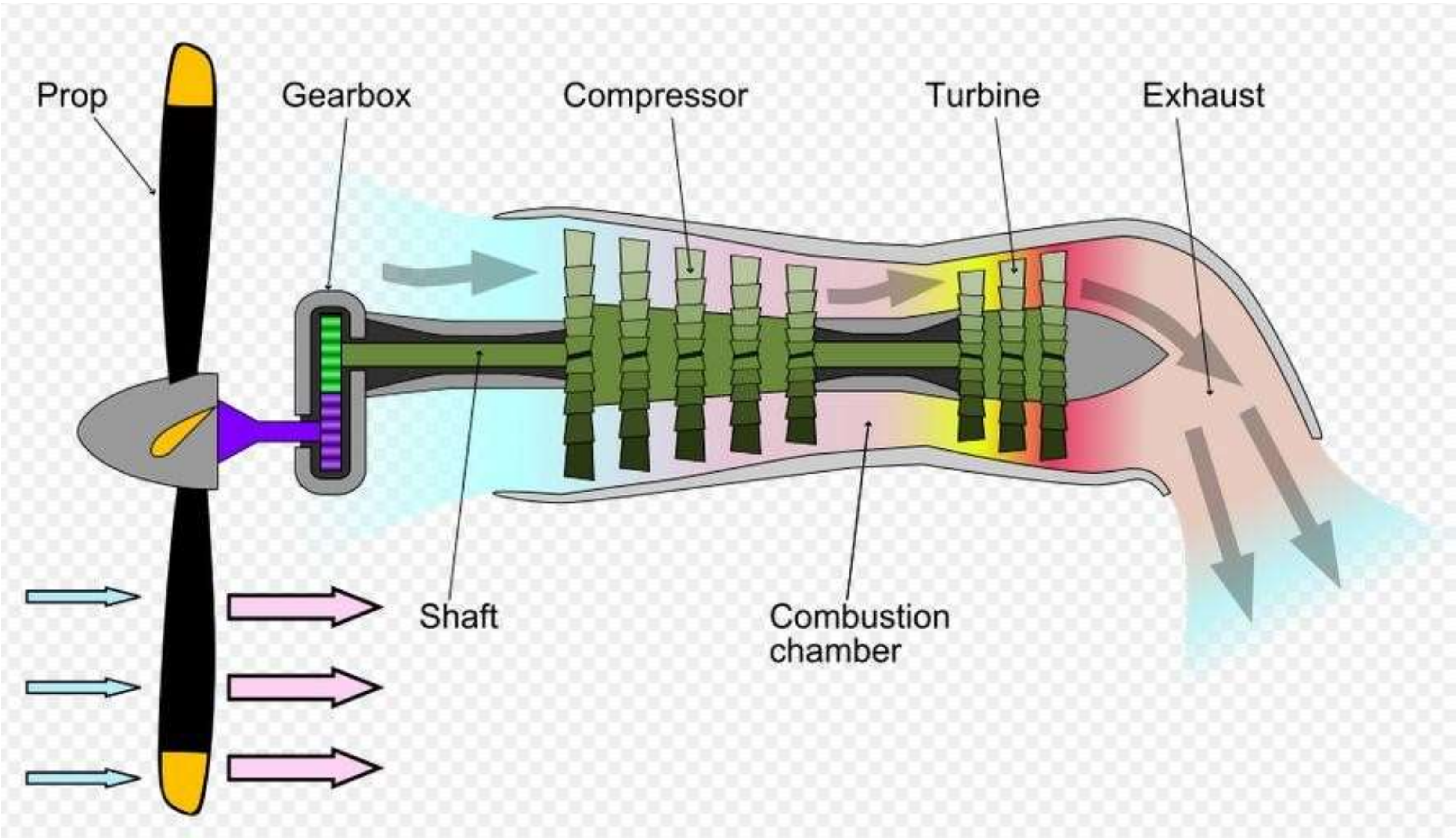
The bypass ratio (BPR) of a turbofan engine is the ratio between the mass flow rate of the bypass stream to the mass flow rate entering the core. A 10:1 bypass ratio, for example, means that 10 kg of air passes through the bypass duct for every 1 kg of air passing through the core.


Compressor turbine
Exhaust
Compressor
Combustion chamber
Power shaft
Free (power) turbine

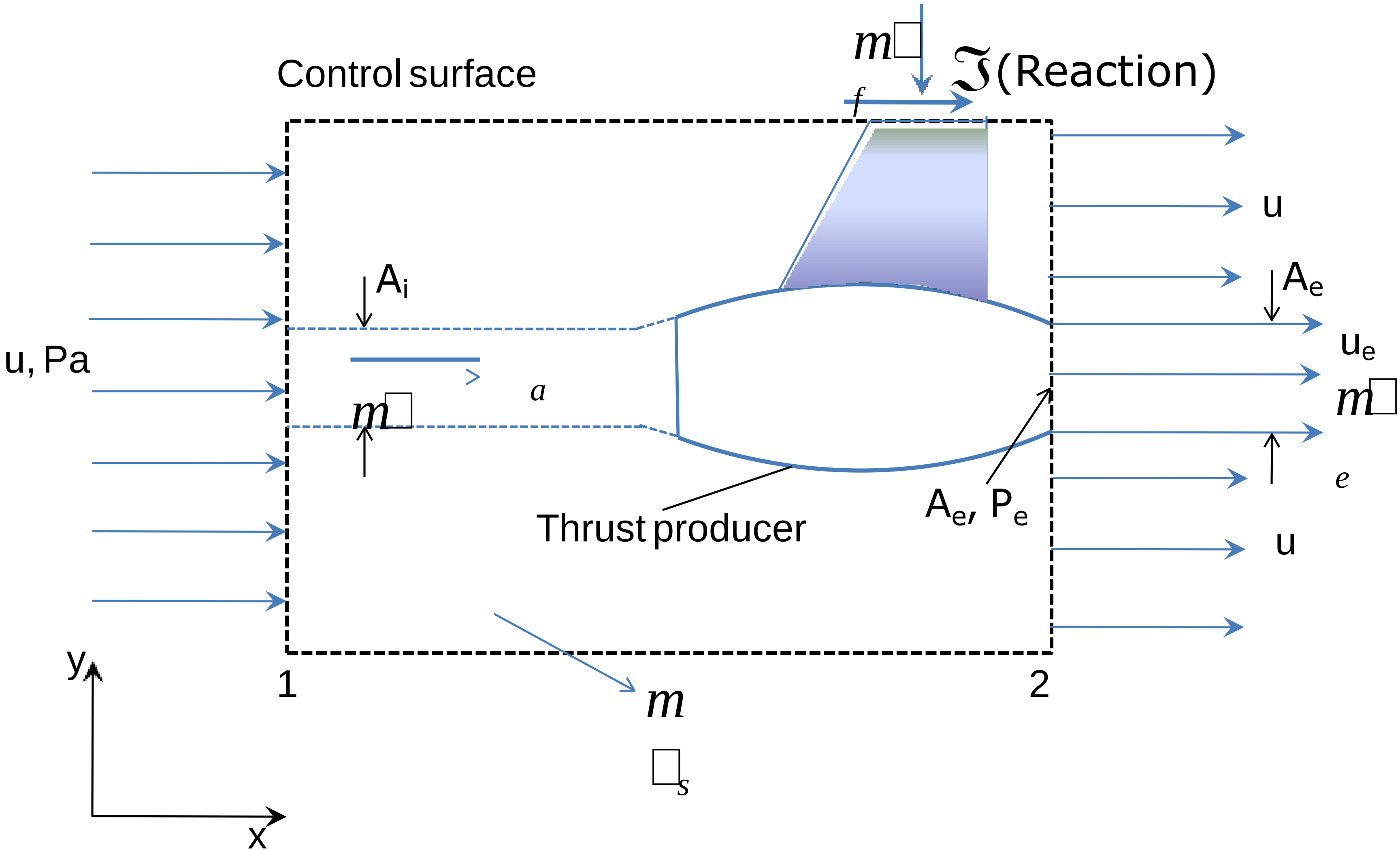
- A turboshaft engine is a form of gas turbine that is optimized to produce shaft power rather than jet thrust.
- They are even more similar to turboprops, with only minor differences.
- Turboshaft engines are commonly used in applications that require a sustained high power output, high reliability, 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 ignitors 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'.


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.



Control surface
ℑ(Reaction)
$\dot{m}_f$
u, Pa
A_i
$\dot{m}_a$
u
A_e
u_e
$\dot{m}_e$
u
Thrust producer
A_e, P_e
$\dot{m}_s$
1
2
y
x

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.

$$\dot{m}_e u_e - \dot{m}_a u = -(P_e - P_a)A_e + \tau$$

$$\therefore \tau = \dot{m}_a\left[(1+f)u_e - u\right] + (P_e - P_a)A_e$$

- 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

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

- Isentropic efficiency, ηd of the diffuser is

$$\eta_d = \frac{h_{02s} - h_a}{h_{0a} - h_a} \cong \frac{T_{02s} - T_a}{T_{0a} - T_a}$$

- Stagnation pressure ratio or pressure recovery is the ratio of the outlet stagnation pressure to the inlet stagnation pressure

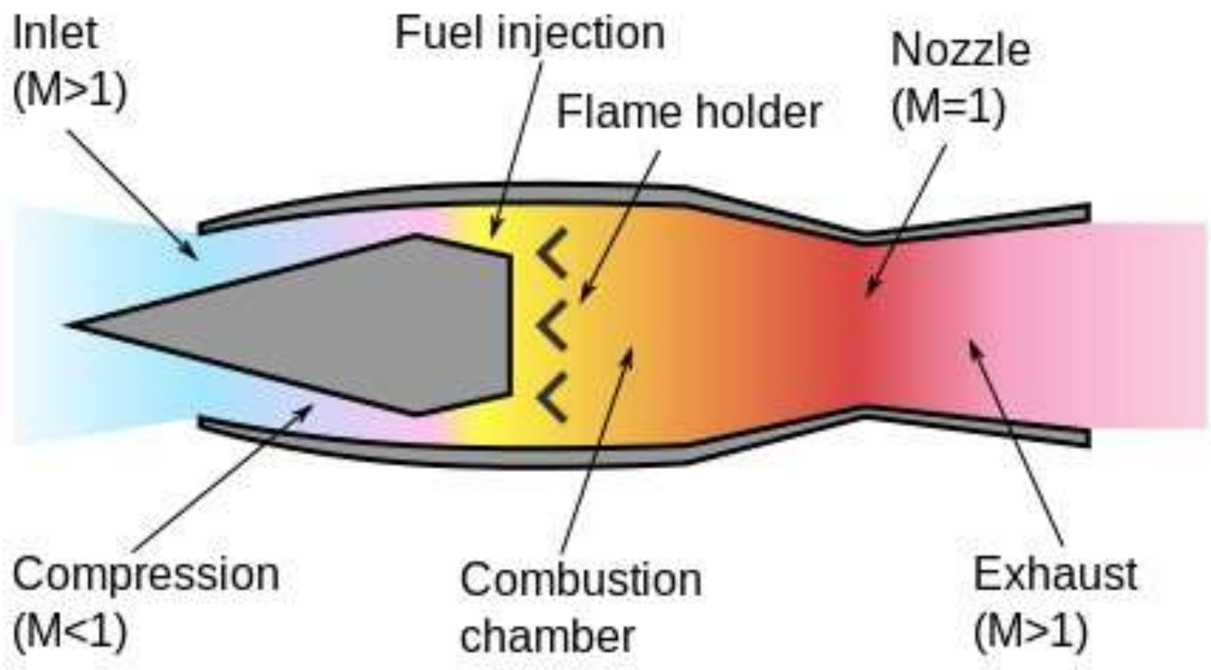
$$\pi_d = P_{02} / P_{0a}$$

UNIT - II
COMPONENTS OF JET ENGINES

Course Learning Outcomes

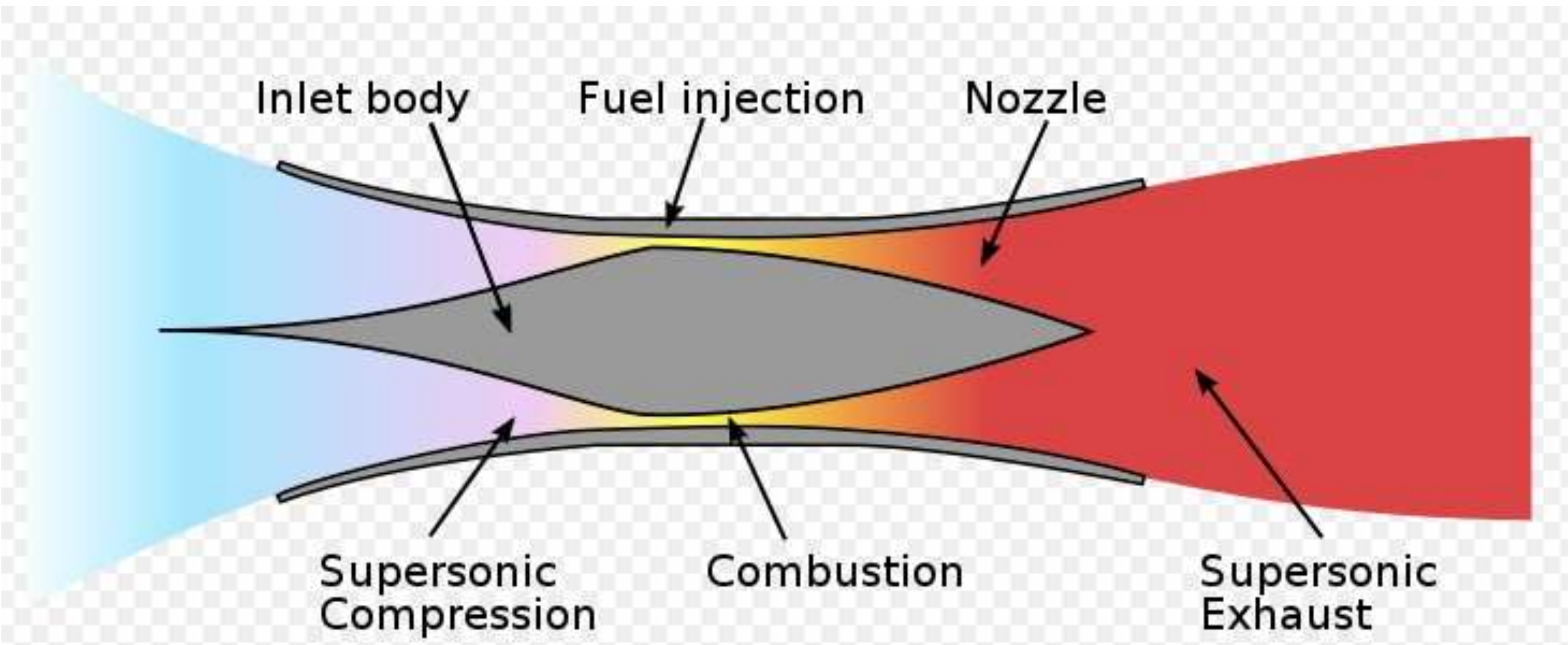
CLOs	Course Learning Outcome
CLO1	Understand step by step procedure of engine parametric cycle analysis.
CLO2	Describe principle of operation of axial and centrifugal compressor.
CLO3	Understand different design of compressor and limitations of each method.
CLO4	Analyze performance characteristics of axial and centrifugal turbines.

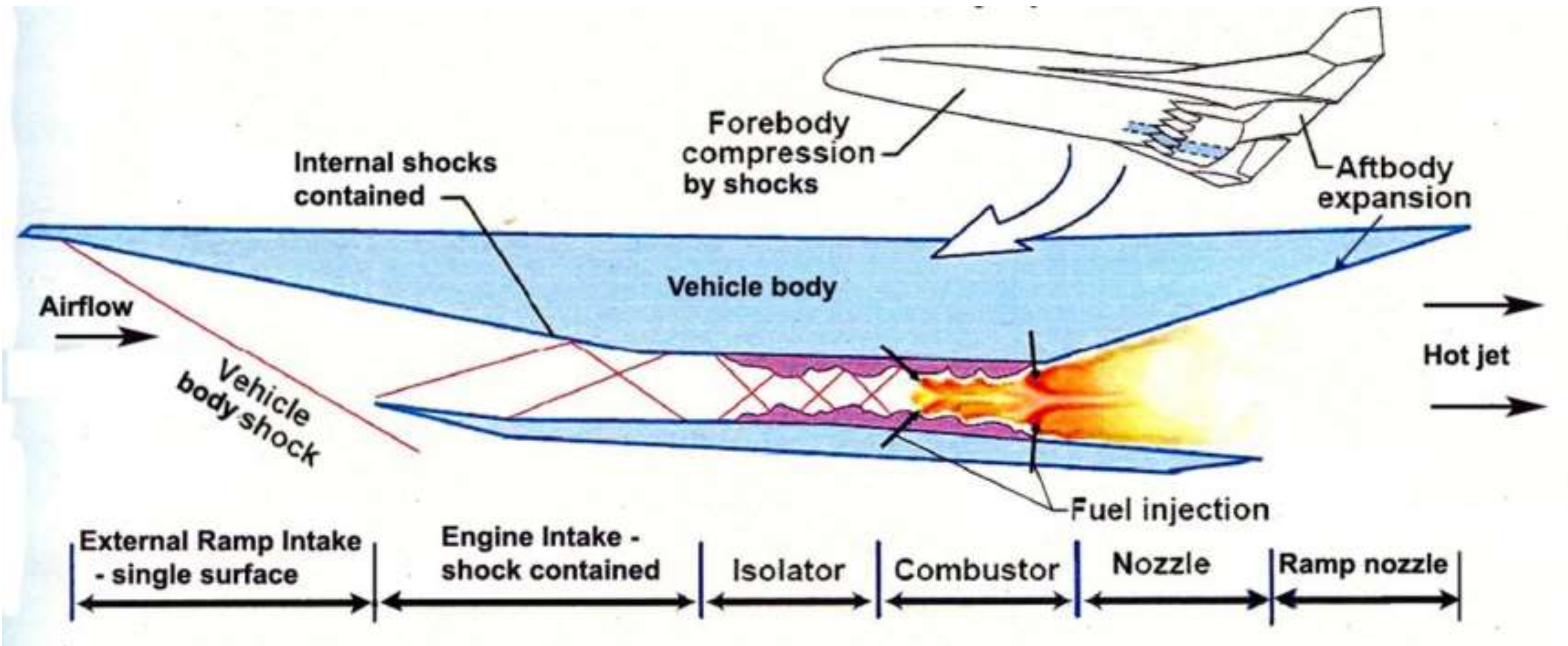
While the Gas Generator, composed of the compressor, combustor and turbine, is the heart of any gas turbine engine, the overall performance of the propulsion system is strongly influenced by the inlet and the nozzle. This is especially true for high Mo flight, when a major portion of the overall Temperature 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.

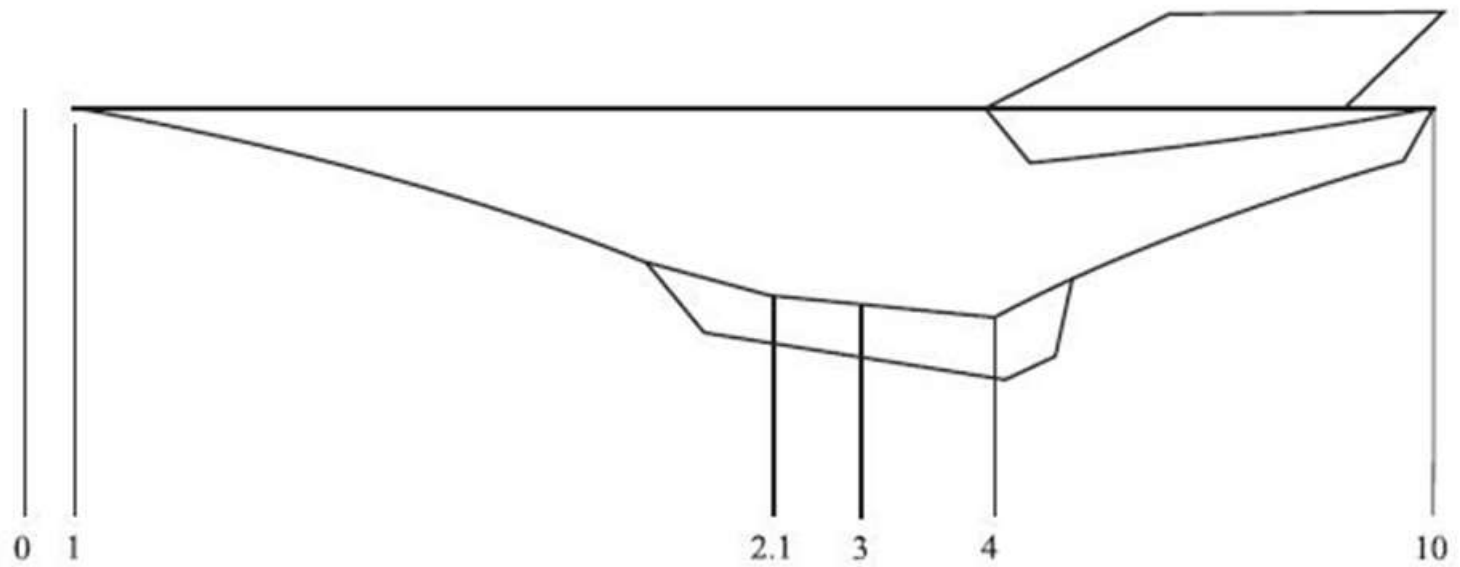

Inlet
(M>1)
Fuel injection
Flame holder
Nozzle
(M=1)
Compression
(M<1)
Combustion
chamber
Exhaust
(M>1)

- A ramjet is a form of airbreathing 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.


Inlet body
Fuel injection
Nozzle
Supersonic Compression
Combustion
Supersonic Exhaust


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



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 fulfills 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.

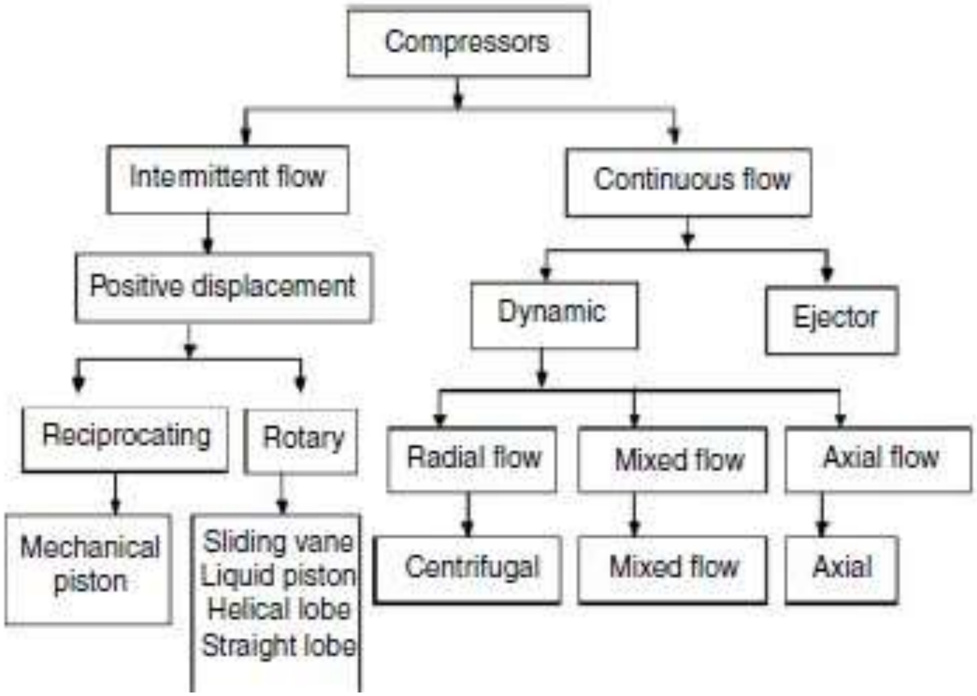
- Combustion systems of aircraft gas turbine engines largely encompass the main burners (also called burners or combustors) and afterburners (also called augmenters or re heaters)

- 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
- Relightability
- Operation over a wide range of mass flow rates, pressures, and temperatures

Classification of compressors.



CENTRIFUGAL COMPRESSORS

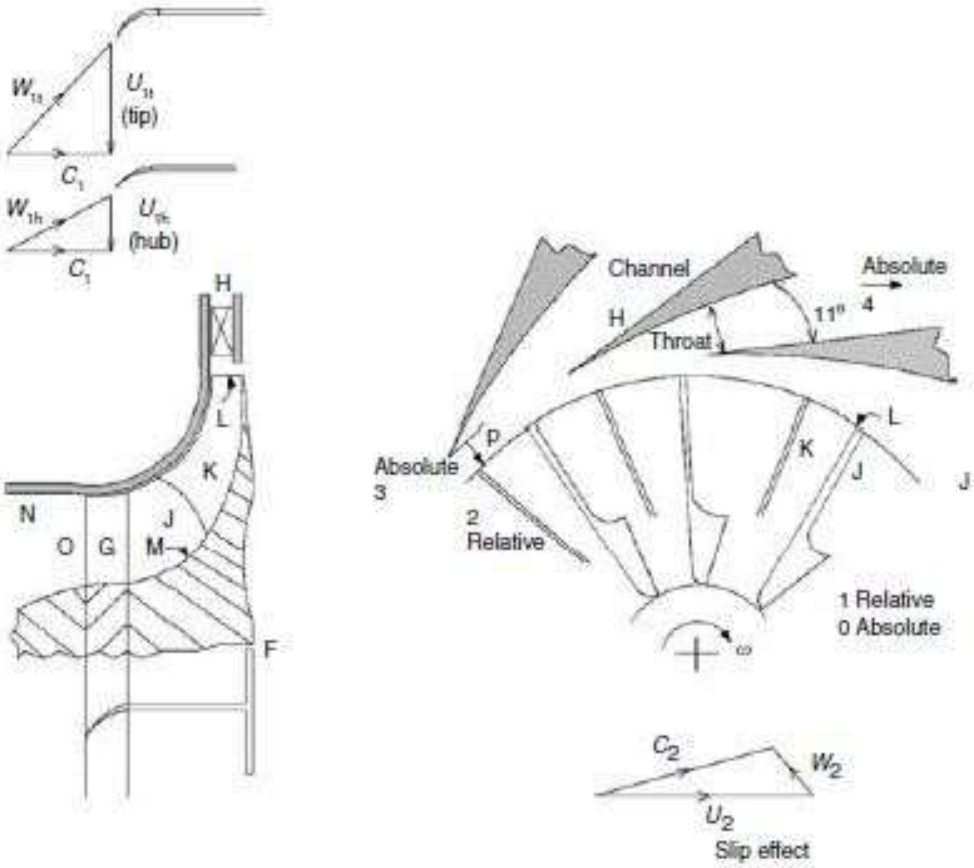


Figure illustrates a typical centrifugal compressor . Nomenclature for a single-sided compressor with channel-type diffuser is given here.

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

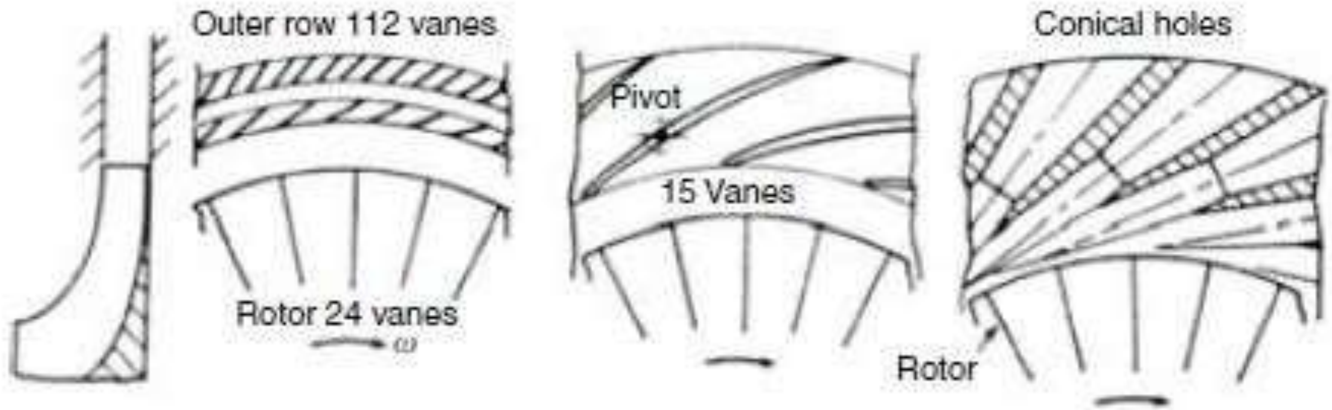
static pressure. The impeller is a complicated diffuser compared with the conventional straight conical diffuser as the passage is doubly curved first in the axial plane and then in the radial plane

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.

TYPES OF DIFFUSERS



(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

Centrifugal compressors may be classified as

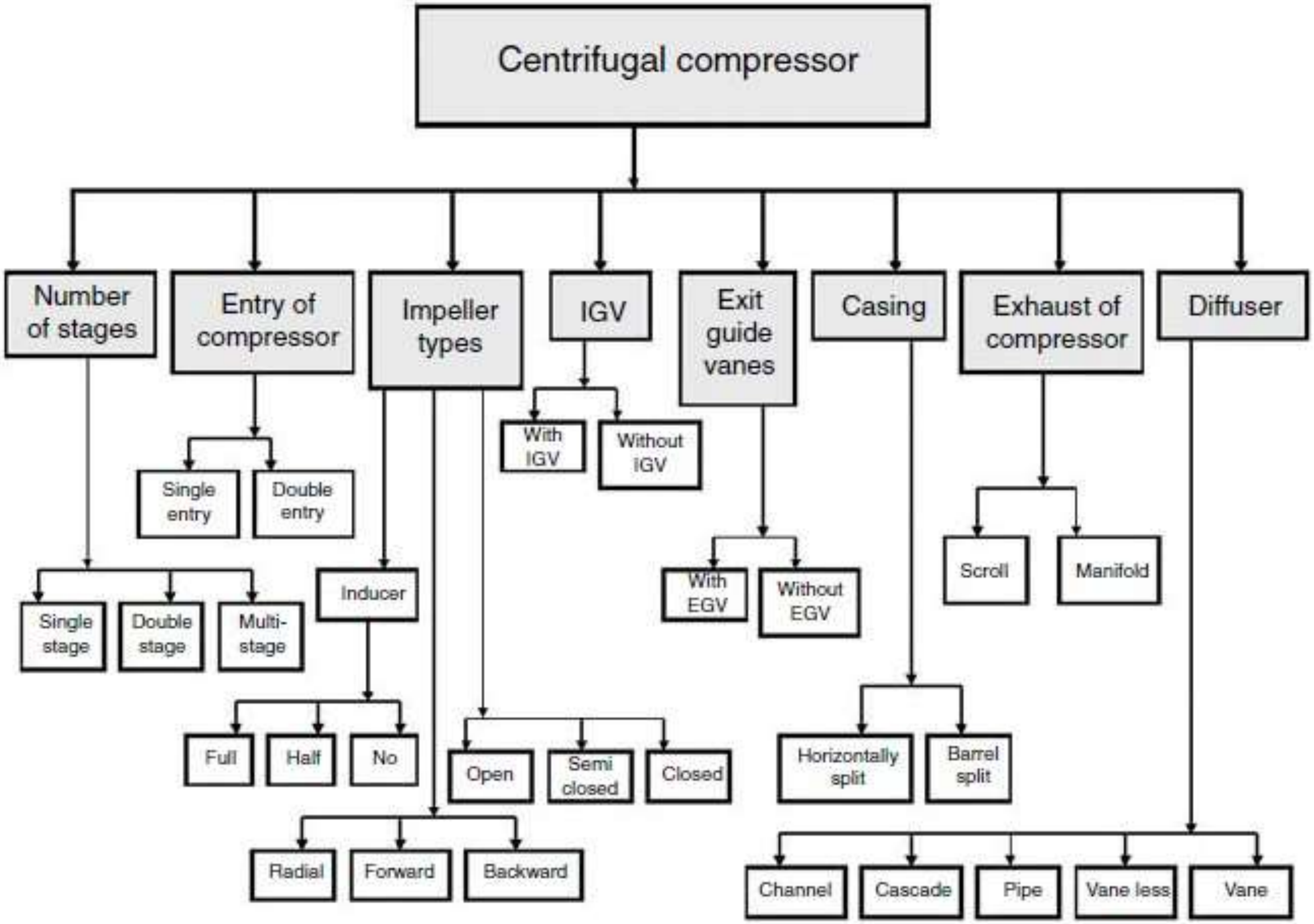
- **Single or multiple stages**: For aero engines, centrifugal compressors have either single

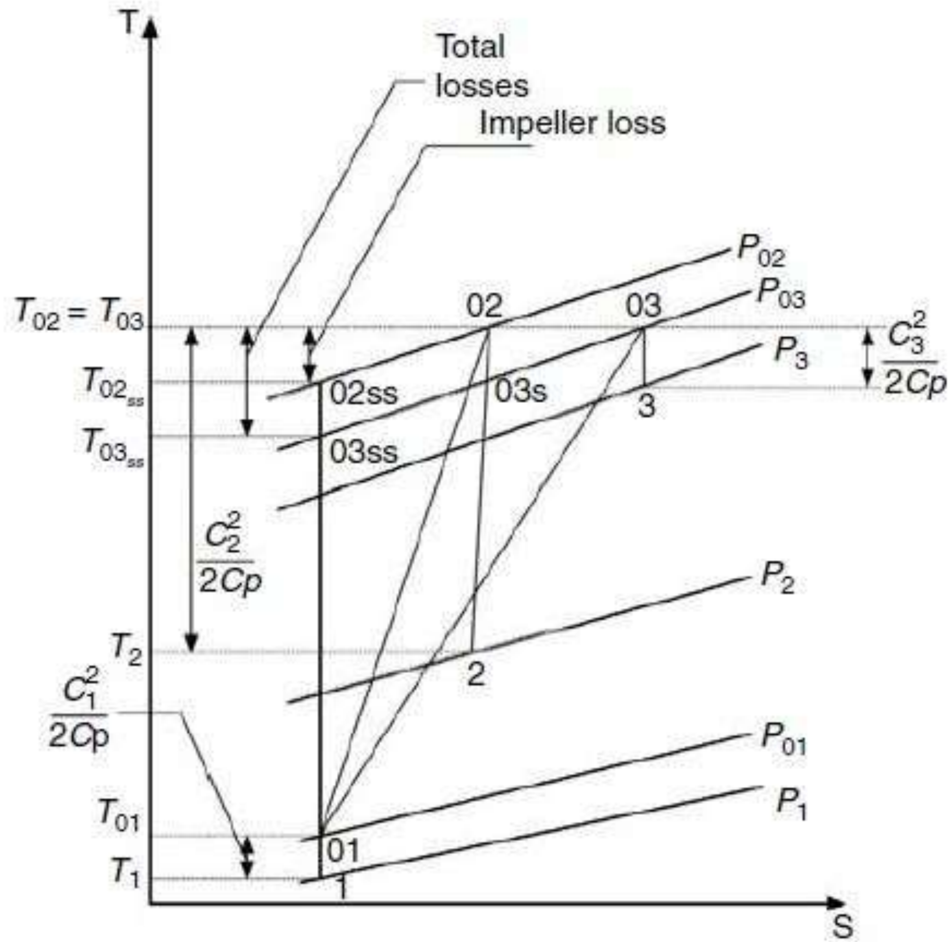
stage or double (two or tandem) stages. In some cases two(or double) stages are mounted on the same shaft, handling the fluid in series to boost its final pressure. In industrial gas turbines such as pipeline compressors, there are up to five stages.

- **Single entry (or single-face) or double (dual) entry (or double-face):**

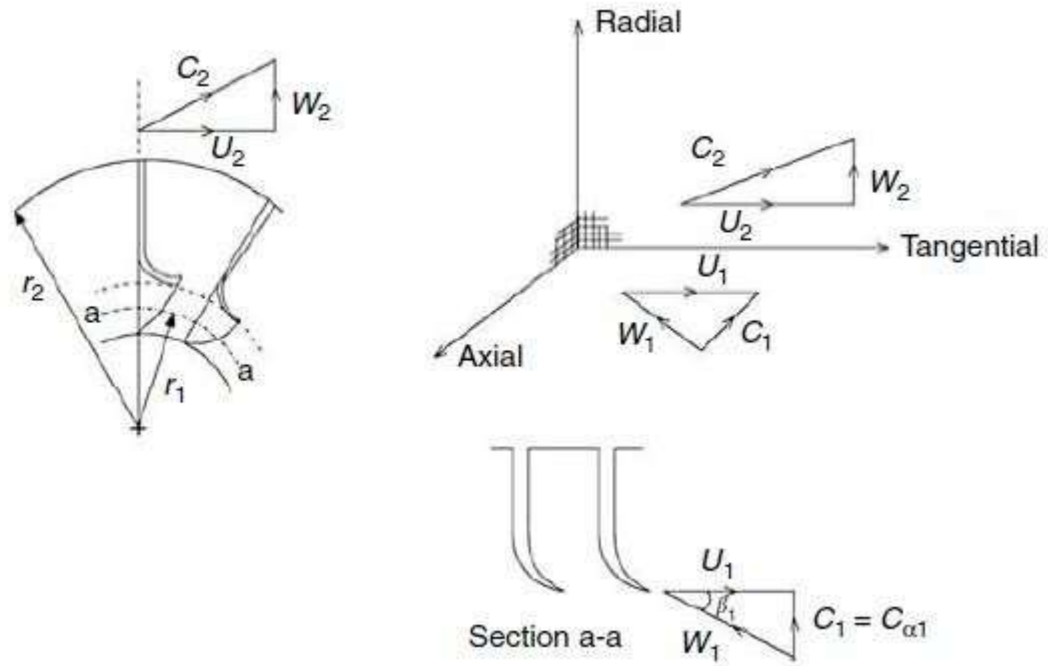
The principal differences between the single entry and dual entry are the size of the impeller and the ducting arrangement. The single-entry impeller permits ducting directly to the inducer vanes, as opposed to the more complicated ducting needed to reach the rear side of the dual-entry type. Although 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.


Centrifugal compressor
Number of stages
Entry of compressor
Impeller types
IGV
Exit guide vanes
Casing
Exhaust of compressor
Diffuser
With IGV
Without IGV
Single entry
Double entry
Single stage
Double stage
Multi-stage
Inducer
Full
Half
No
Open
Semi closed
Closed
Radial
Forward
Backward
With EGV
Without EGV
Scroll
Manifold
Horizontally split
Barrel split
Channel
Cascade
Pipe
Vane less
Vane



T–S diagram for a centrifugal compressor

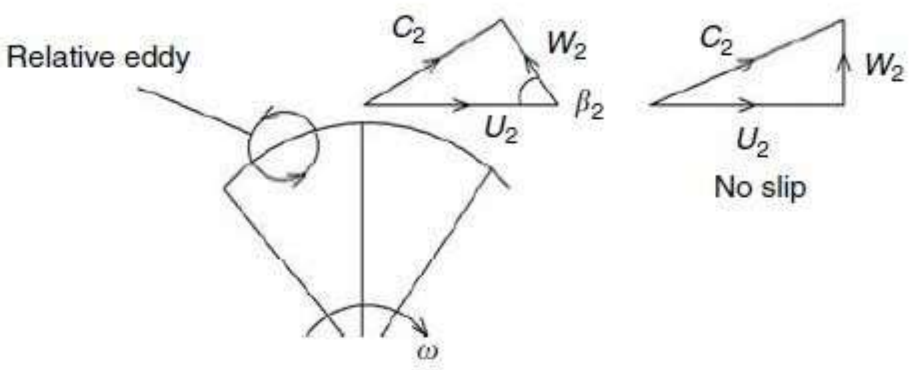


Layout and velocity triangles

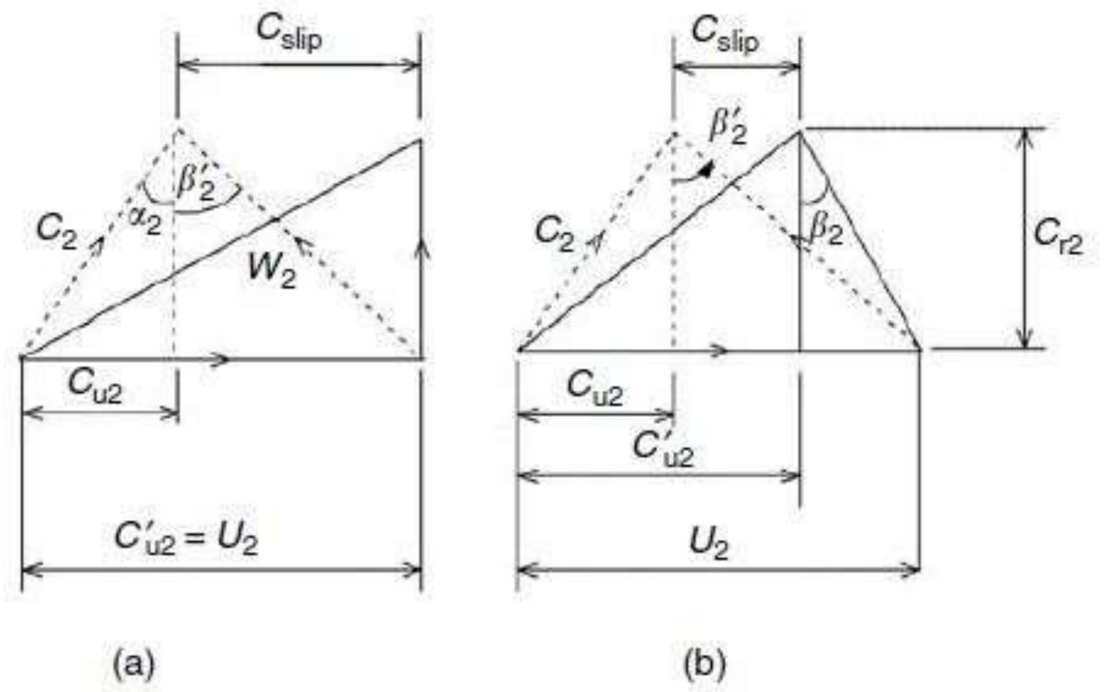
Inlet and outlet velocity triangles at space.

SLIP FACTOR

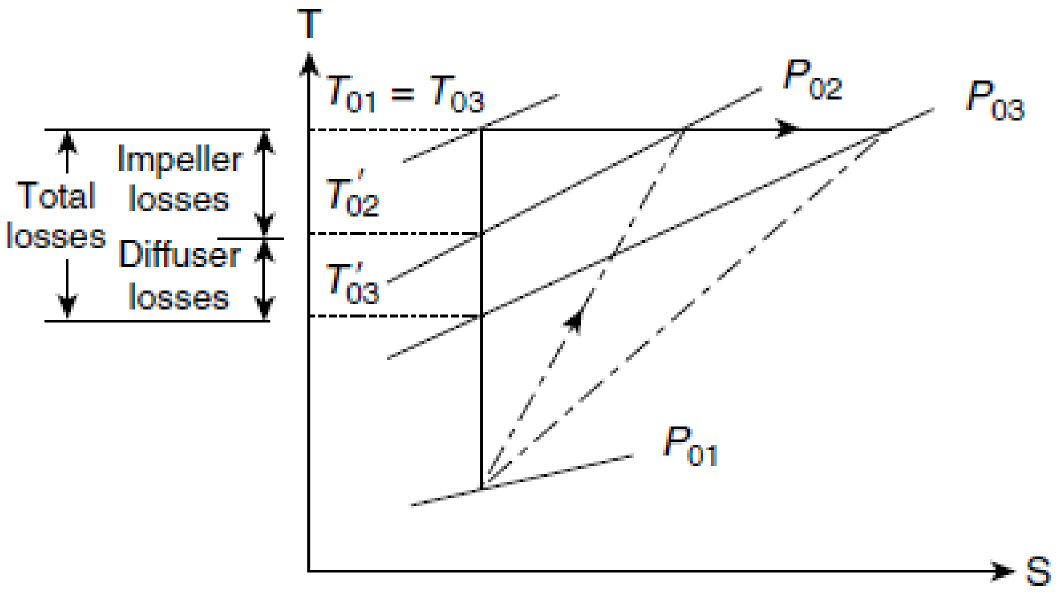
If the flow at impeller discharge is perfectly guided by the impeller blades then the tangential component of the absolute velocity (swirl velocity) is equal to the rotational velocity (Cu2 = U2) in the case of radial type impeller. In practice, the flow cannot be perfectly guided by a finite number of blades and it is said to slip. Thus, at the impeller outlet, the swirl velocity is less than the impeller rotational speed (Cu2 < U2).



Slip due to relative eddies



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



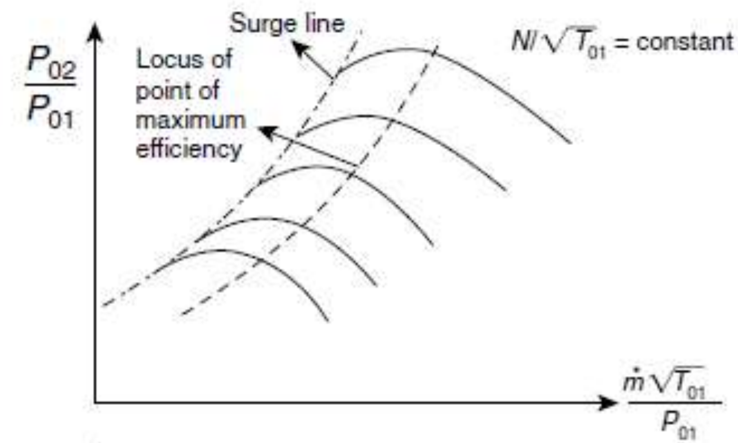
Impeller and diffuser losses

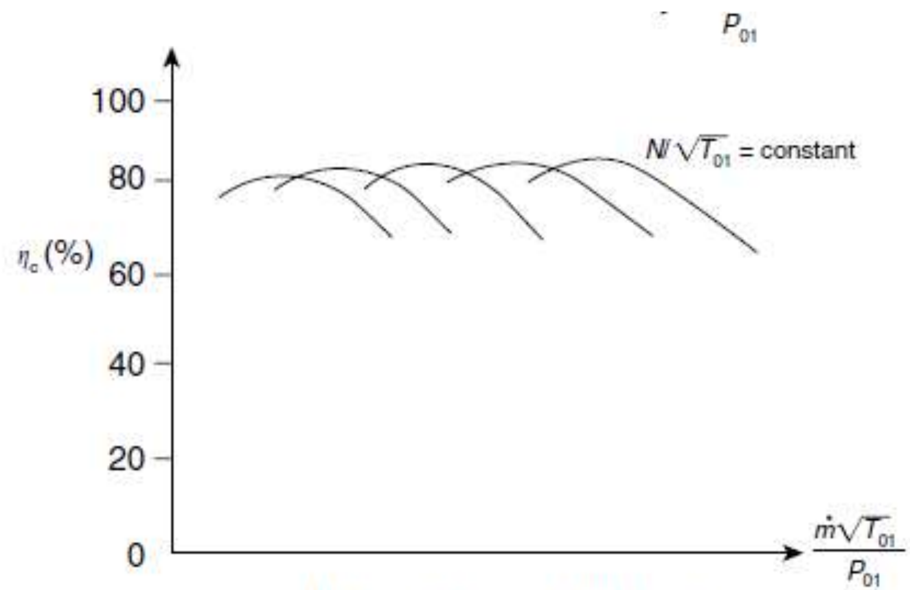
PERFORMANCE

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

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

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

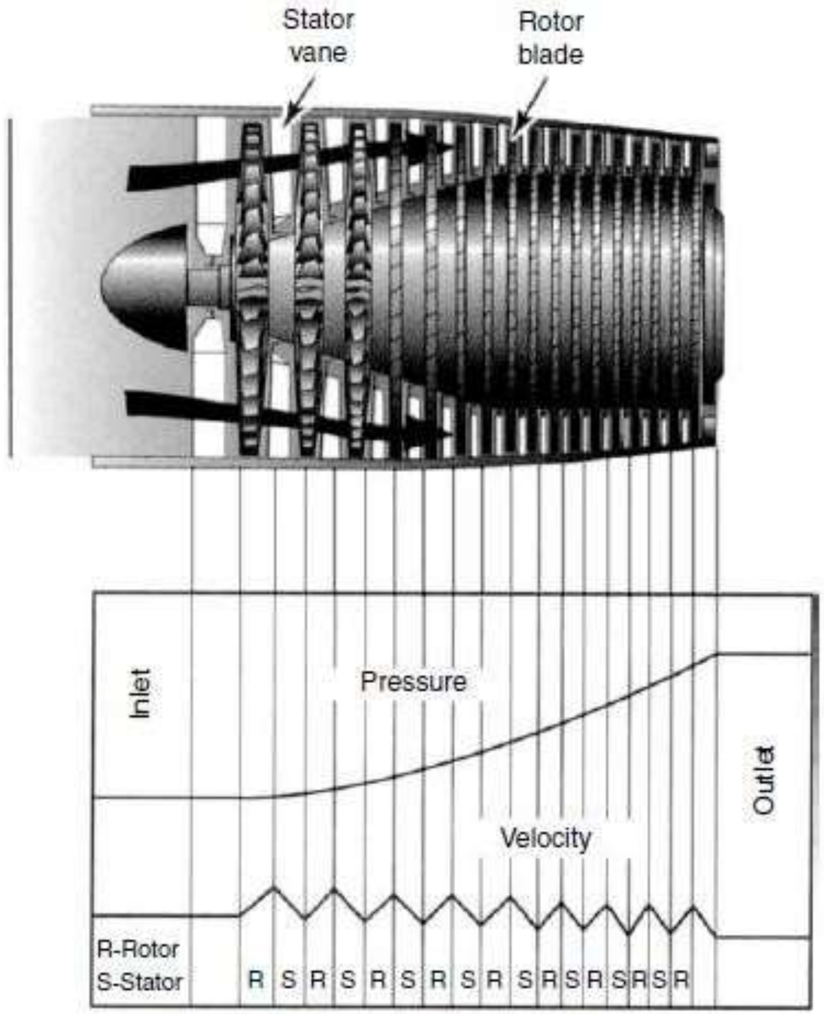




Centrifugal compressor characteristics

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 OVERTHE 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 as shown in Figure 13.3.

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 [2]), 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.

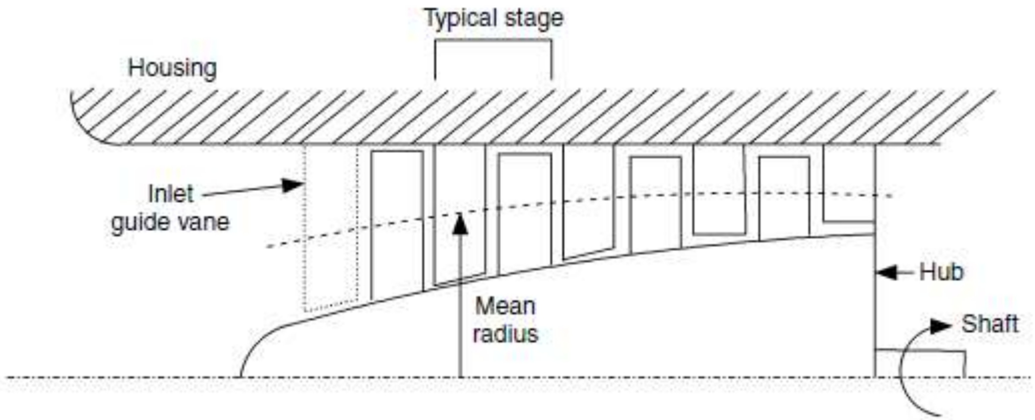
4. May use cascade experiment research in developing compressor.

5. Somewhat higher efficiency at high pressure ratio (perhaps 4%–5% higher than centrifugal compressor).

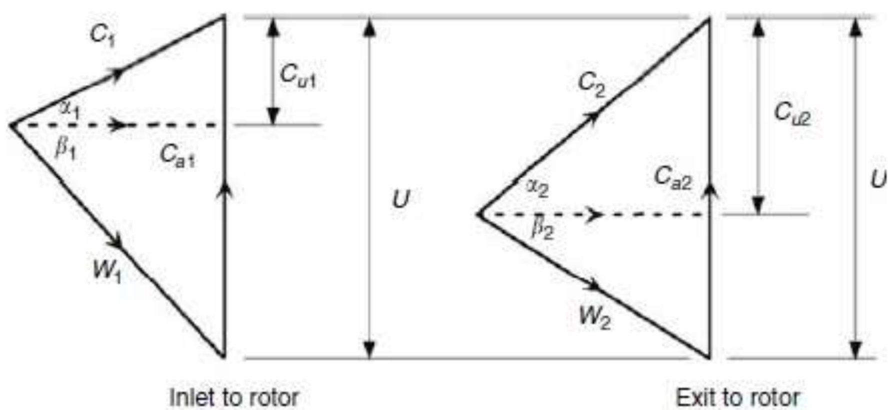
6. Higher maximum pressure ratio, which was about 17 in the 1960s and achieved up to 45 for the present transonic compressors

ADVANTAGES OF CENTRIFUGAL-FLOW COMPRESSOR OVER THEAXIAL-FLOW COMPRESSOR

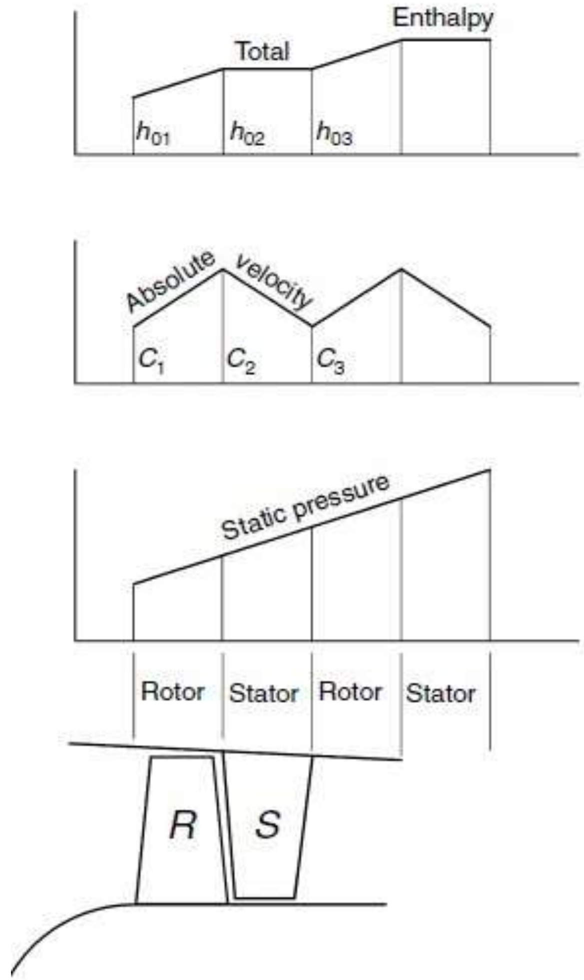
1. Higher stage pressure ratio (5:1 or even 10:1).
2. Simplicity and ruggedness of construction.
3. Shorter length for the same overall pressure ratio.
4. Generally less severe stall characteristics.
5. Less drop in performance with the adherence of dust to blades.
6. Cheaper to manufacture for equal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



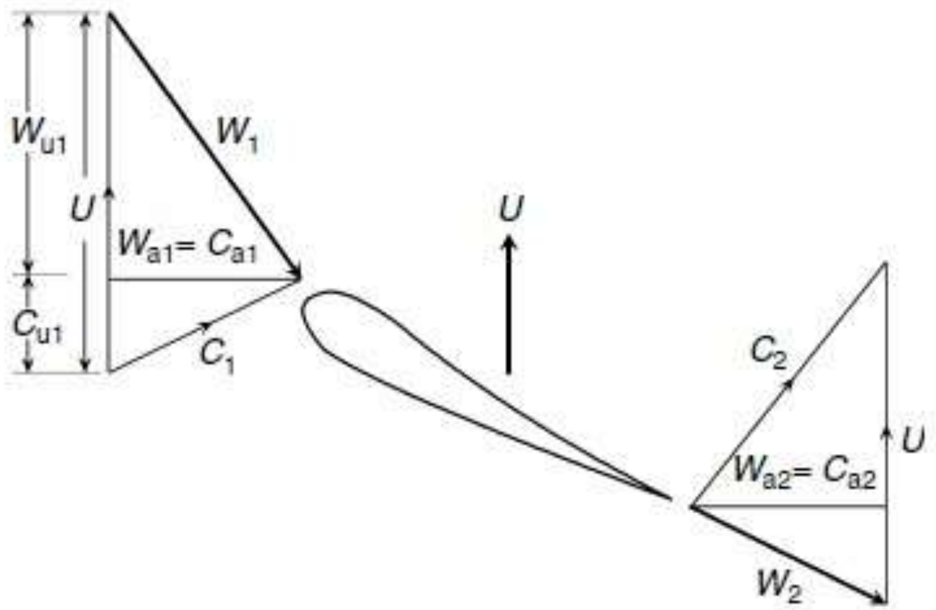
Variation of mean radius for a constant casing multistage axial compressor.



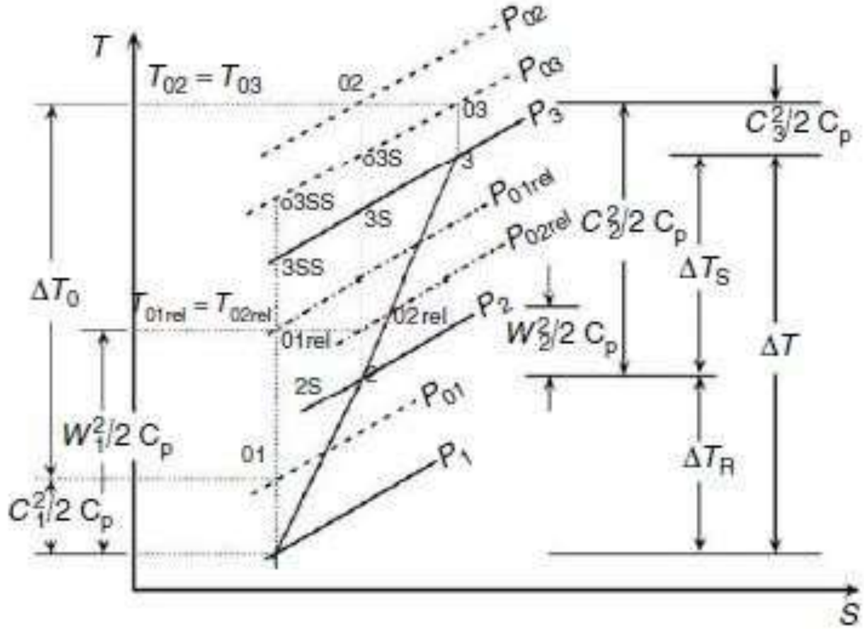
Velocity triangles at inlet and outlet of a constant mean radius.



Variations of the enthalpy, absolute velocity, and static pressure across two successive stages.



Detailed velocity triangles.



Temperature–entropy diagram for a compressor stage.

BASIC DESIGN PARAMETERS

The pressure ratio per stage is expressed by the relation

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

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

1. High blade speed (U)
2. High axial velocity (Ca)
3. High fluid deflection (β1 − β2) in the rotor blade

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,

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

DEGREE OF REACTION

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

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. Nonuniform 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,

$$\therefore C_{\mathrm{a}}\frac{\mathrm{d}C_{\mathrm{a}}}{\mathrm{d}r}+C_{\mathrm{u}}\frac{\mathrm{d}C_{\mathrm{u}}}{\mathrm{d}r}+\frac{C_{\mathrm{u}}^{2}}{r}=0$$

2. Assuming constant axial velocity at all radii,

$$\therefore C_{\mathrm{u}}\frac{\mathrm{d}C_{\mathrm{u}}}{\mathrm{d}r}+\frac{C_{\mathrm{u}}^{2}}{r}=0$$

$$\frac{\mathrm{d}C_{\mathrm{u}}}{C_{\mathrm{u}}}=-\frac{\mathrm{d}r}{r}$$

Integrating, we get $rC_{\mathrm{u}}=\text{constant}$

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

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

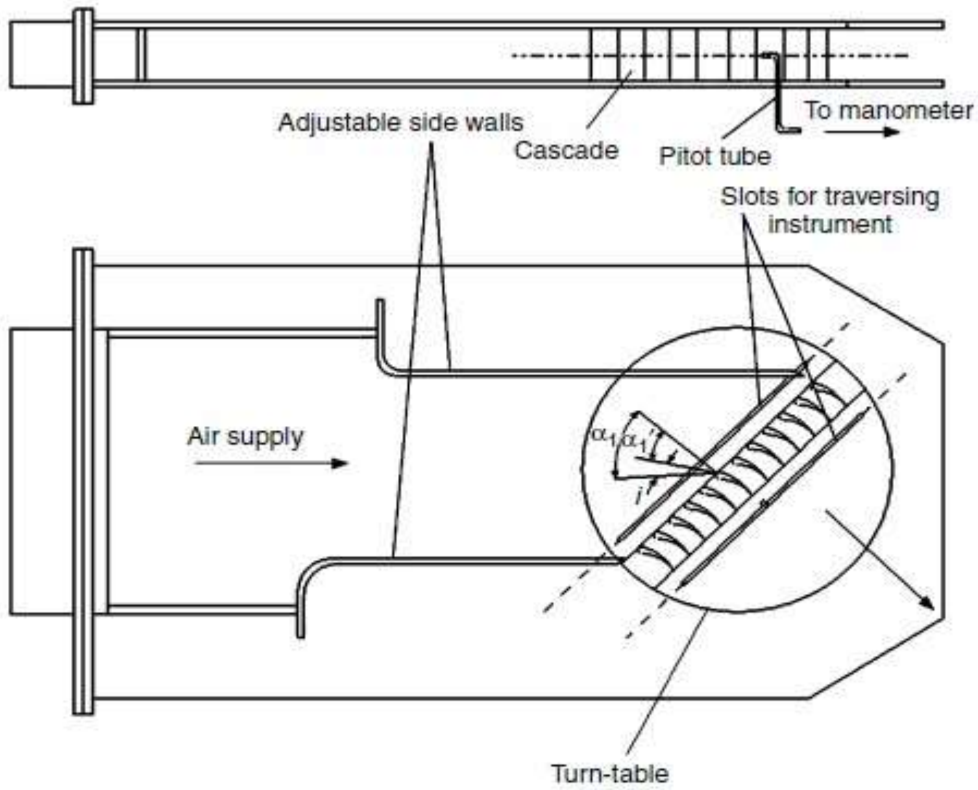
9. Rig testing

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.

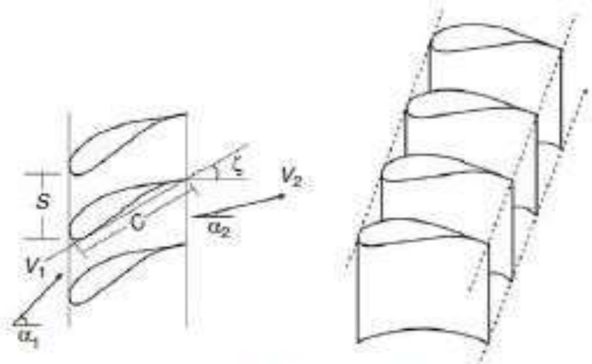
- The first objective is to turn air through the required angles ($\beta_1 - \beta_2$) for rotor and ($\alpha_2 - \alpha_3$) for stator, with the angle ($\beta_1 - \beta_2$) as maximum as possible to maximize the stage pressure ratio.
- The second objective is to achieve the diffusing process with optimum 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.



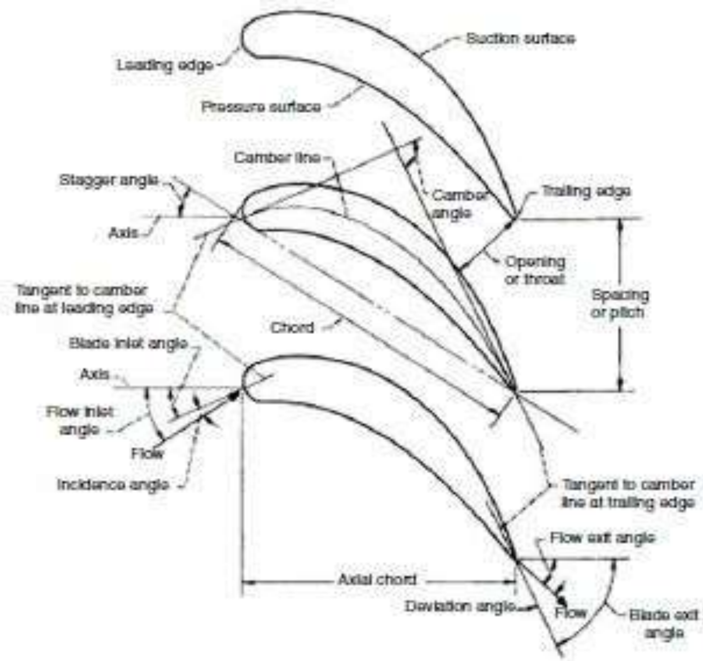
- The height and length of cascade in a cascade wind tunnel, made as large as the available air supply, will allow eliminating the interference effects due to the tunnel walls. Boundary layer suction on the walls is frequently applied to prevent contraction of air stream
- Cascade is mounted on a turn-table so that its angular direction with respect to the inflow duct ($\alpha 1$) 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.

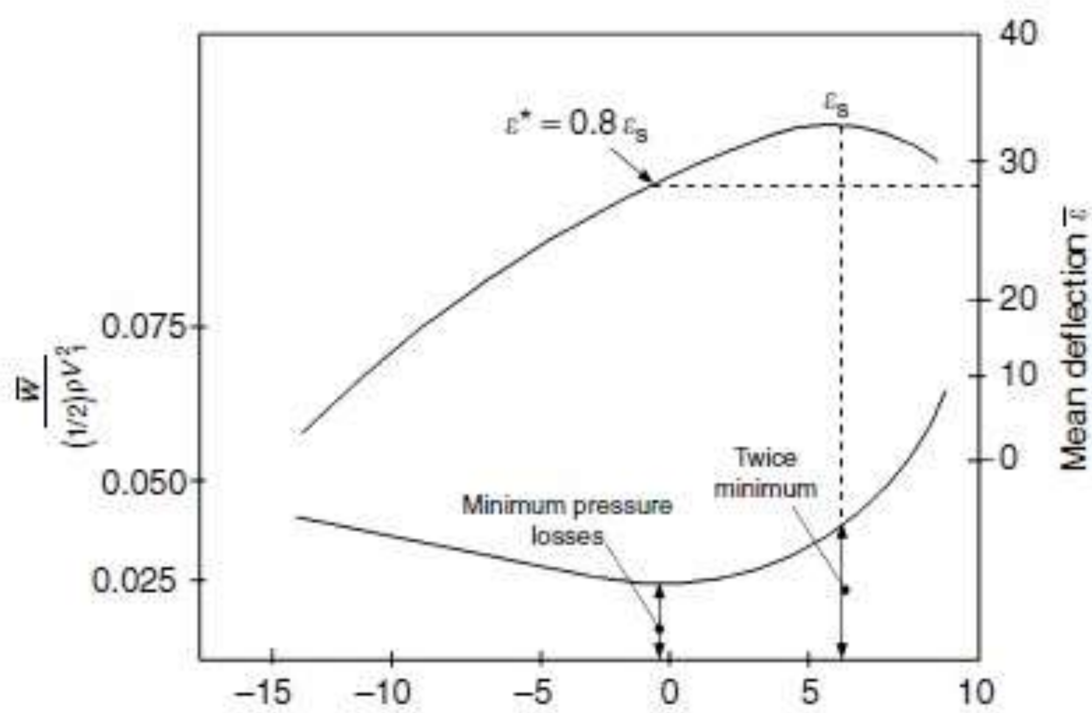
(a)



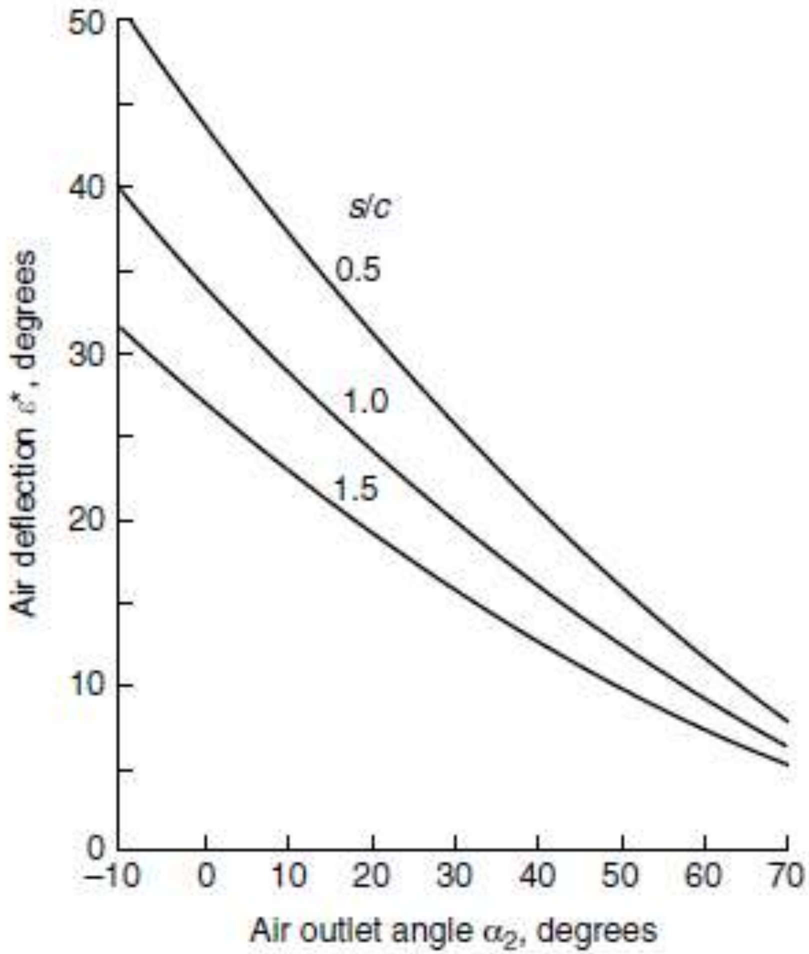
Geometry of rectilinear cascade

(b)





Mean deflection and mean stagnation pressure loss curves.



Cascade nominal deflection versus outlet angle

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

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

The blade row efficiency is defined as

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

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

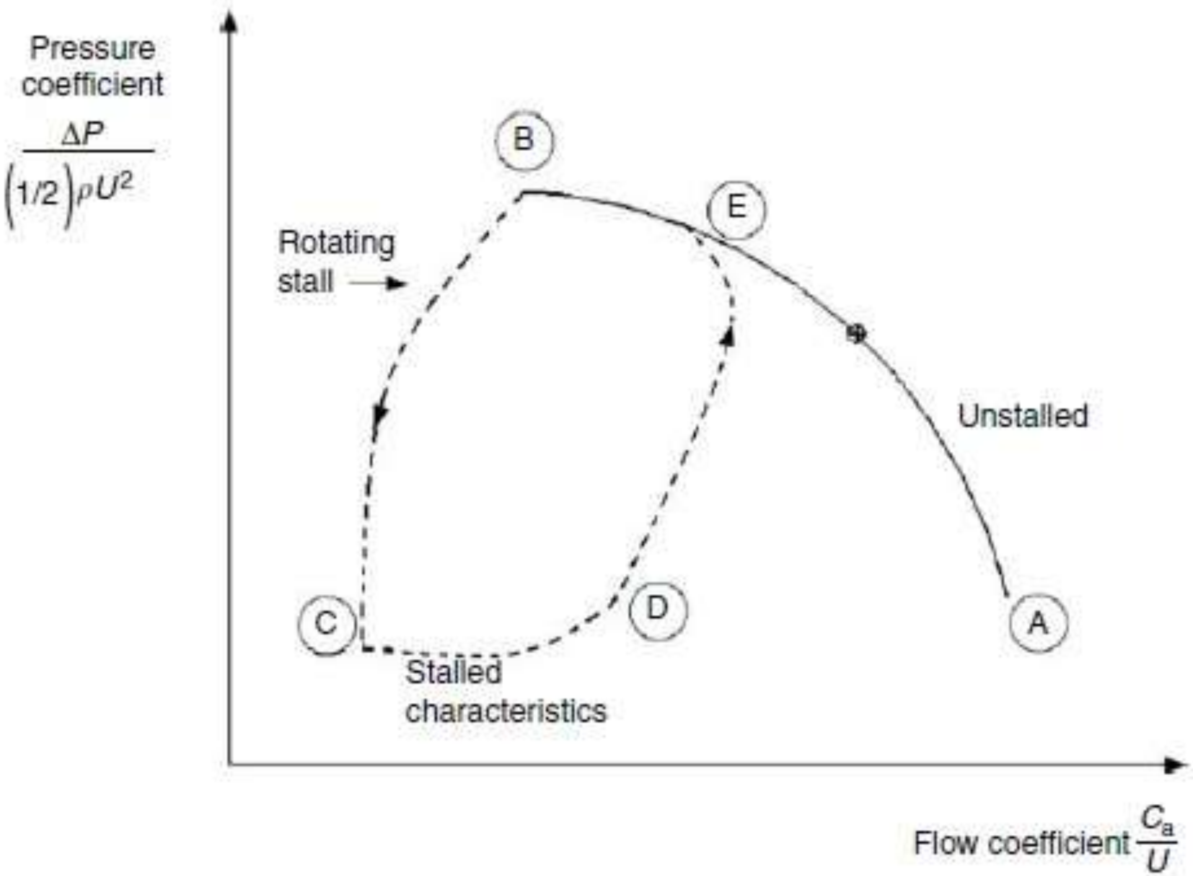
$$\eta_b = 1 - \frac{\overline{w}/(\rho V_1^2/2)}{\Delta P_{th}/(\rho V_1^2/2)}$$

STALL AND SURGE

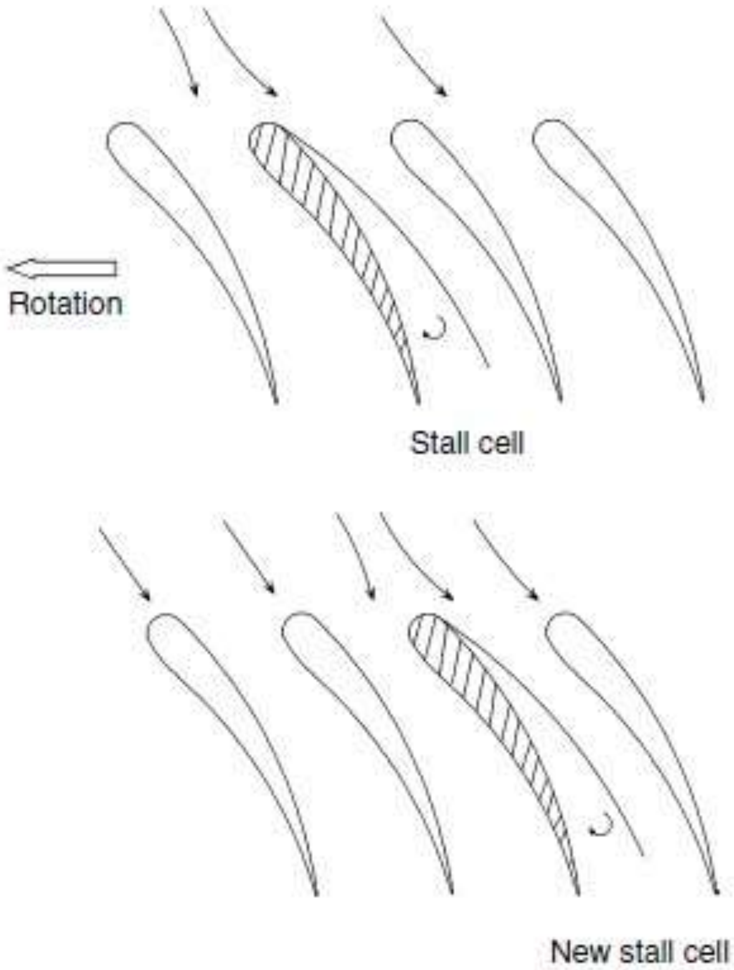
Stall is a situation of abnormal airflow through a single stage or multiple stages of the compressor, while the whole compressor stall, known as compressor surge, results in a loss of engine power. This power failure may only be momentary or may shut down the engine completely causing a flameout.

The following factors can induce compressor stall

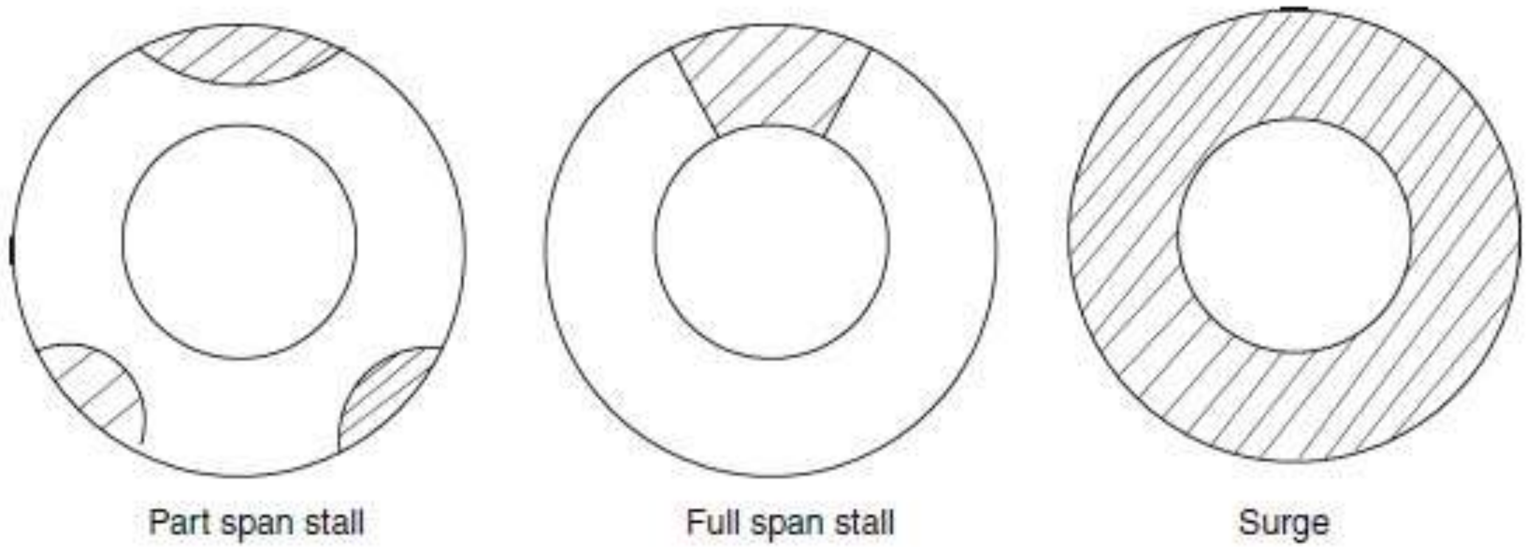
- Engine over-speed
- Engine operation outside specified engineering parameters
- Turbulent or disrupted airflow to the engine intake
- Contaminated or damaged engine components.



Performance characteristics for unstalled and stalled operation



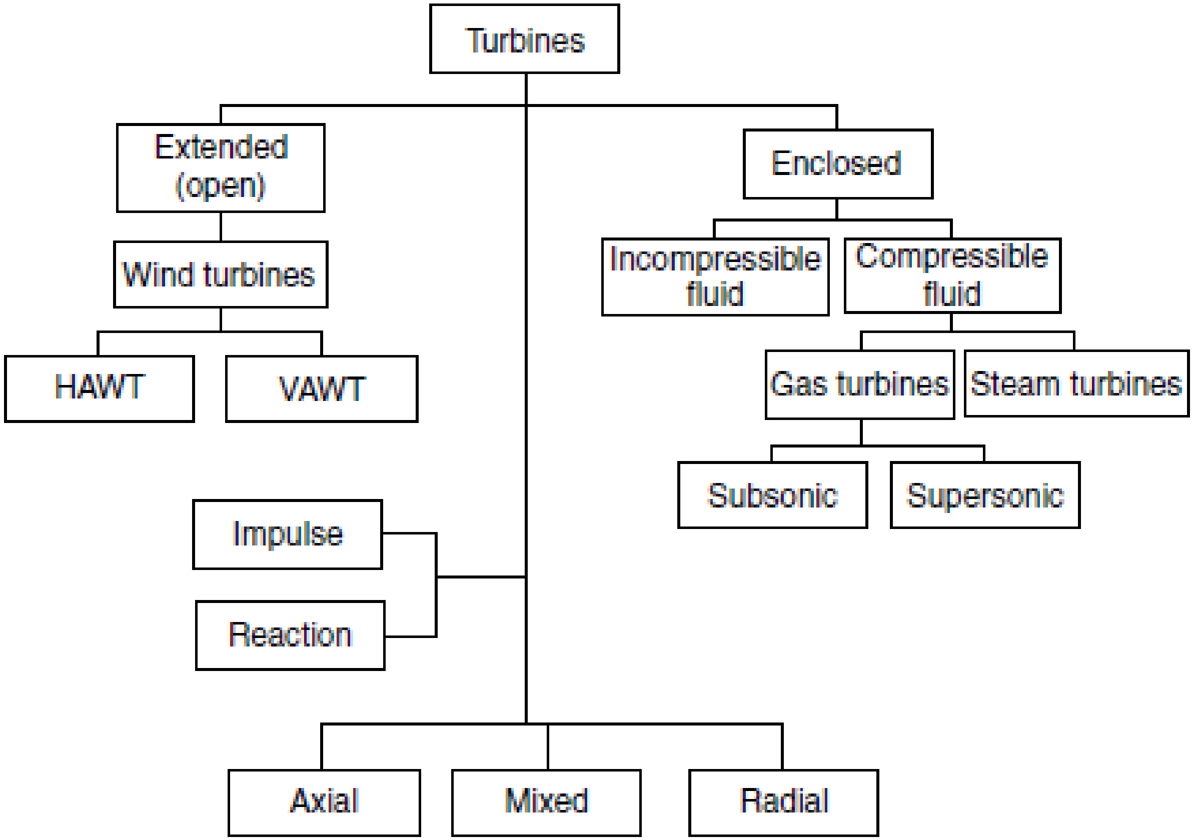
The propagation of rotating stall.



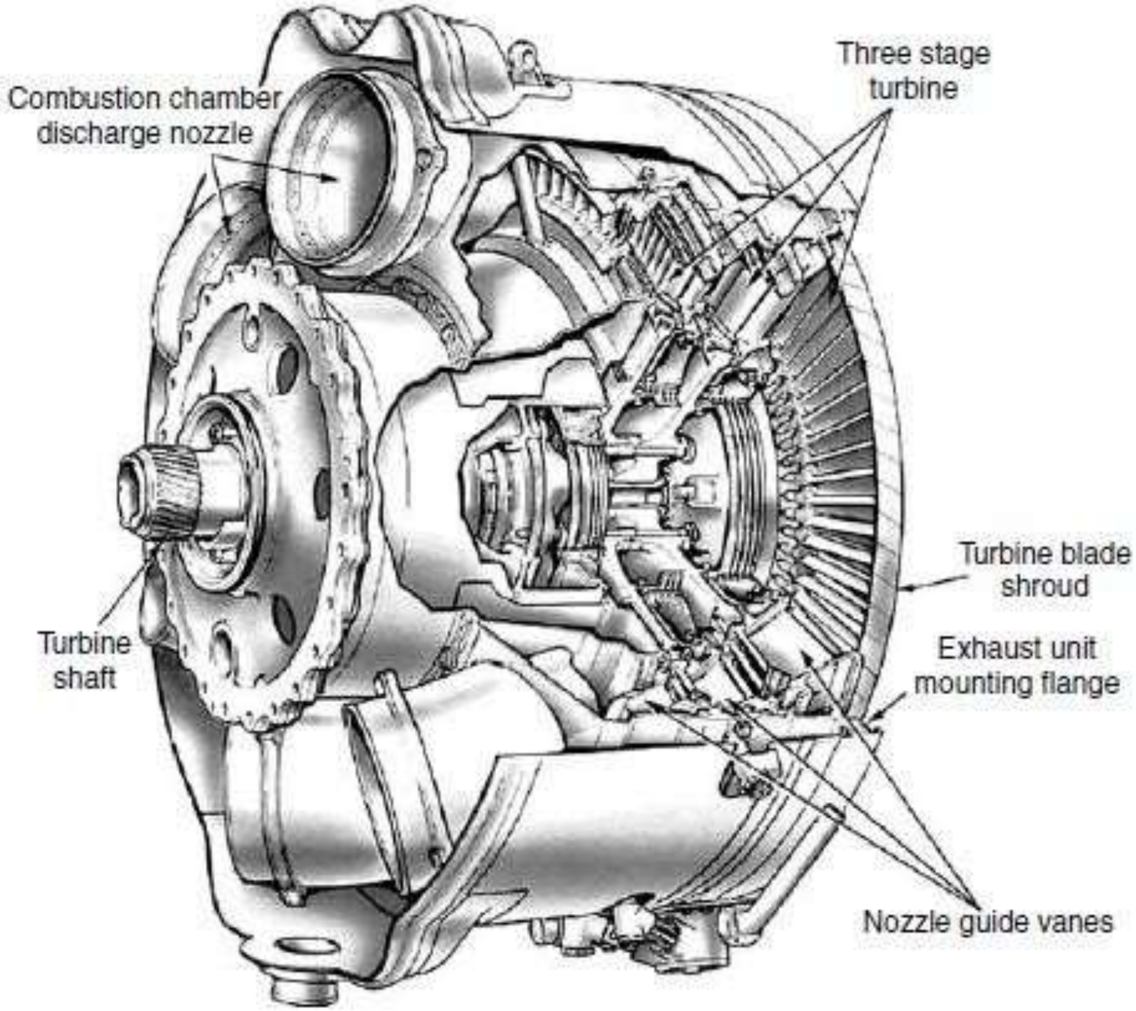
Rotating stall and surge.

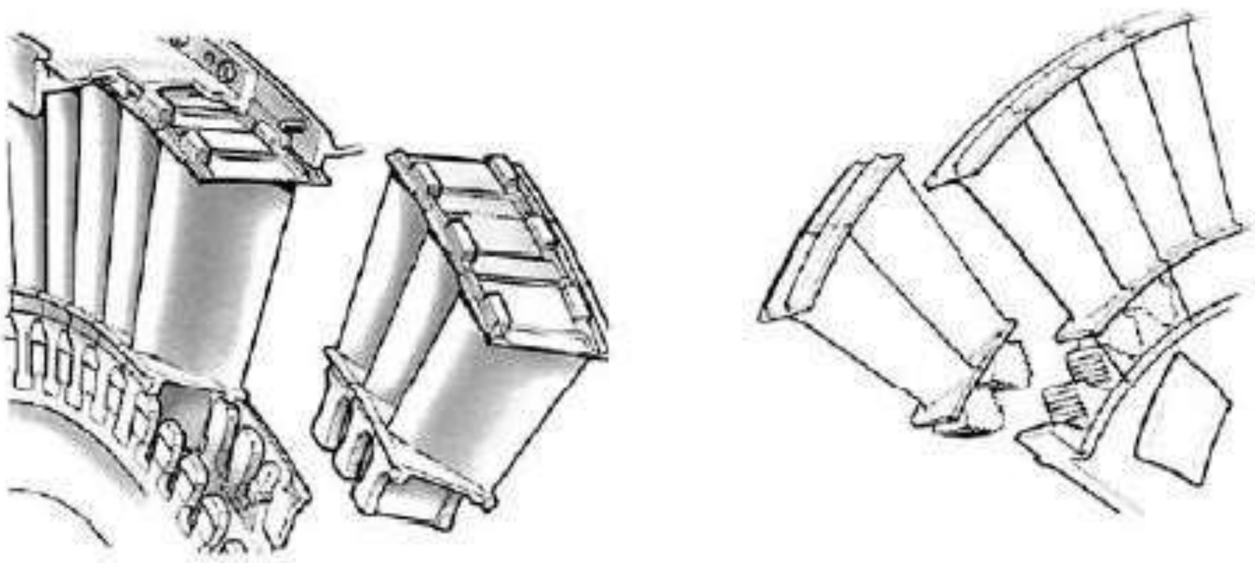
TURBINES

- 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.


Combustion chamber
discharge nozzle
Three stage
turbine
Turbine blade
shroud
Exhaust unit
mounting flange
Turbine
shaft
Nozzle guide vanes

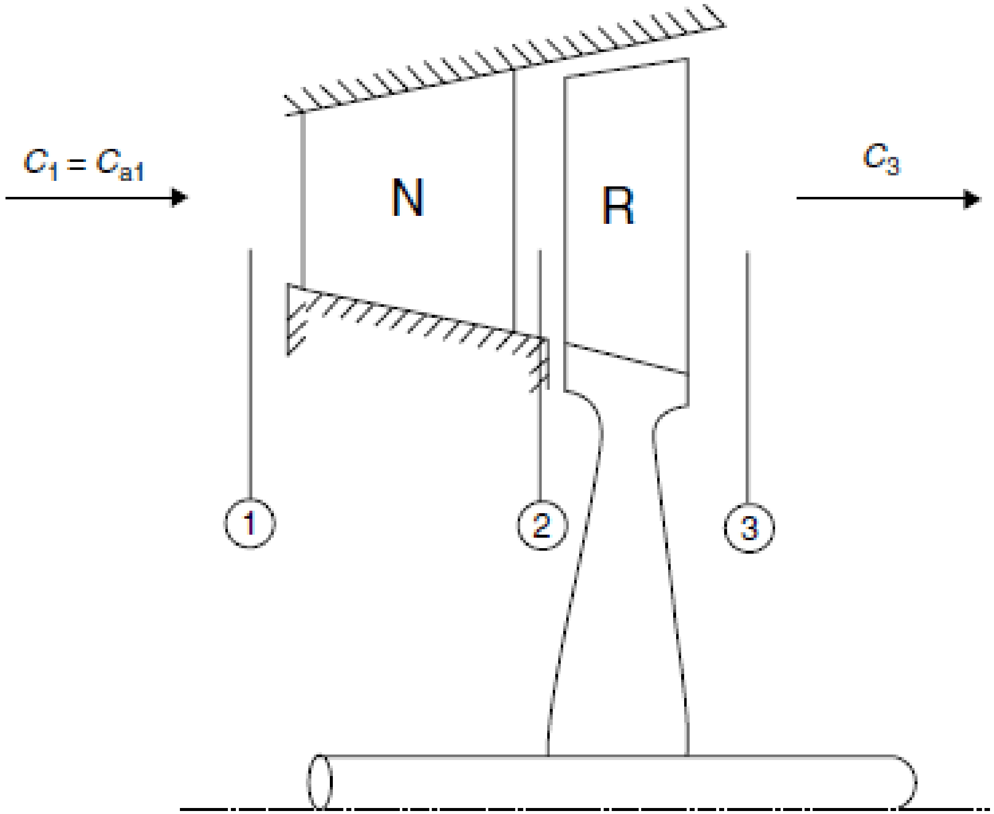


Nozzle guide vanes Rotor blades

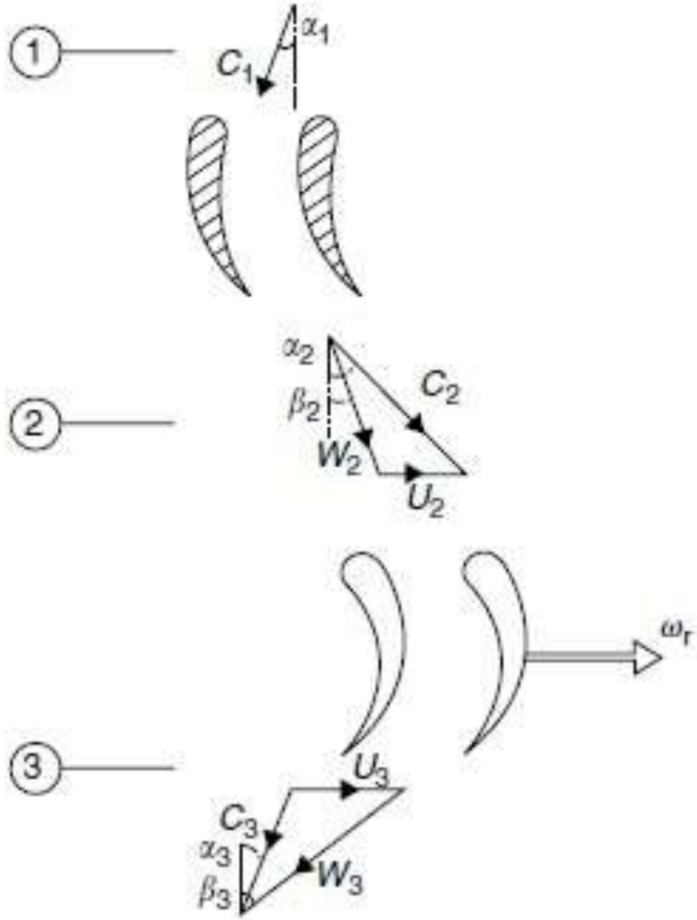
Shrouded blades of both the nozzle and rotor.

VELOCITY TRIANGLES

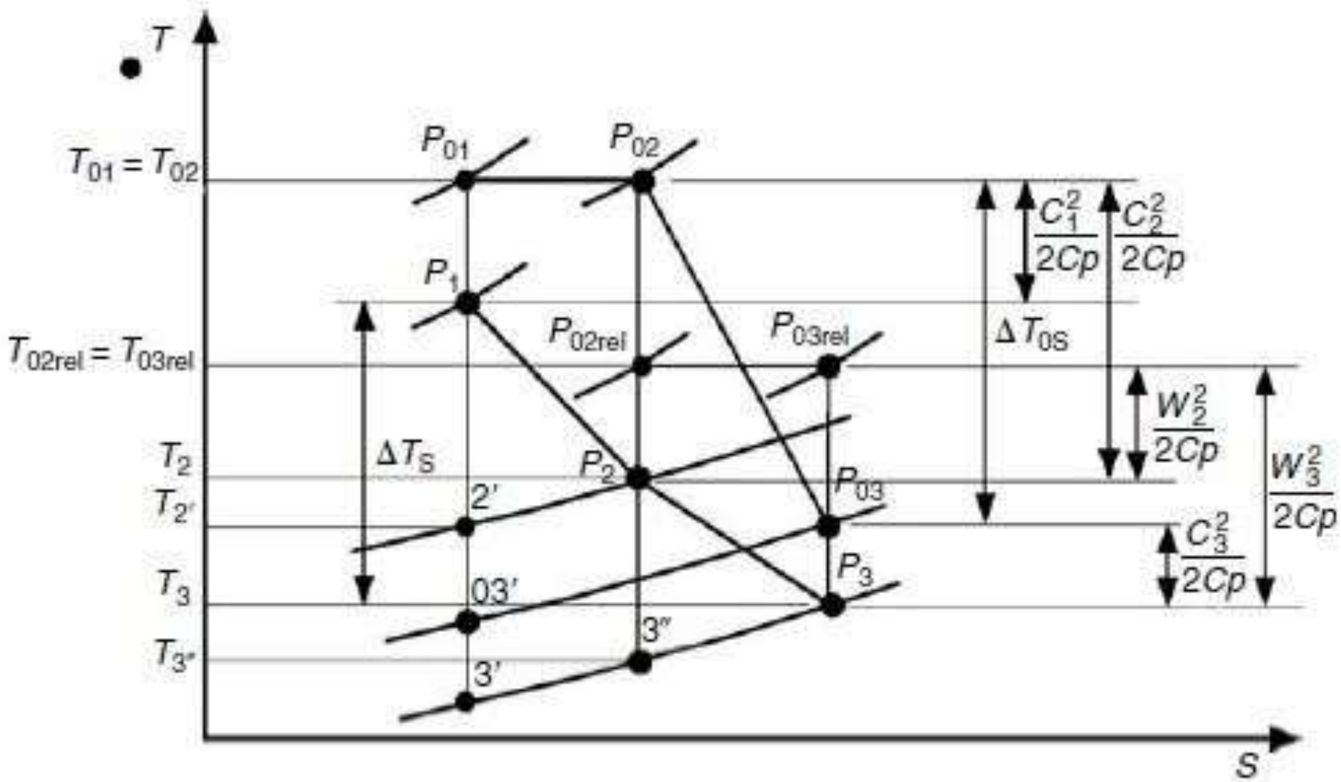
- Single stage of axial turbine is shown in Fig. The gases leaving the combustion chamber approach the stator (or nozzle) with an absolute velocity ($C1$), normally in an axial direction and thus, the absolute angle to the axial direction ($\alpha 1 = 0$).
- The flow leaves the stator passage at a speed ($C2$), where ($C2 > C1$). 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, $P1 > P2$, $P01 > P02$.



Layout of an axial turbine stage.



Velocity triangles.



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

EFFICIENCY, LOSSES AND PRESSURE RATIO

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

The turbine efficiency ηt is related to the stage efficiency ηs 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 (λN) 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 (h01)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): $rC_{u3} = \text{constant} \quad C_{a3} = C_{a2} = \text{constant}$

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

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

NOZZLE DESIGN

An alternative design procedure to free vortex flow is the constant nozzle angle. The appropriateconditions 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 (dh02/dr = 0).

2. Constant nozzle outlet angle (α2) that avoids manufacturing nozzles of varying outlet angle.

Three cases are available for rotor outlet, namely,

1. Constant total conditions at outlet
2. Free vortex at outlet
3. Zero whirl at outlet

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 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.

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

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 (Yk)

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

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

UNIT - III
INLETS, NOZZLES AND PROPERLLER THEROY

Course Learning Outcomes

CLOs	Course Learning Outcome
CLO1	Analyze propeller performance and its types and explain their impact on engine performance.
CLO2	Describe operational modes of subsonic inlets and parameters influencing it.
CLO3	Describe theory of flow in isentropic nozzle and physics behind nozzle operation.
CLO4	Understand different nozzle operating conditions for convergent and divergent nozzle.

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 Mo 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

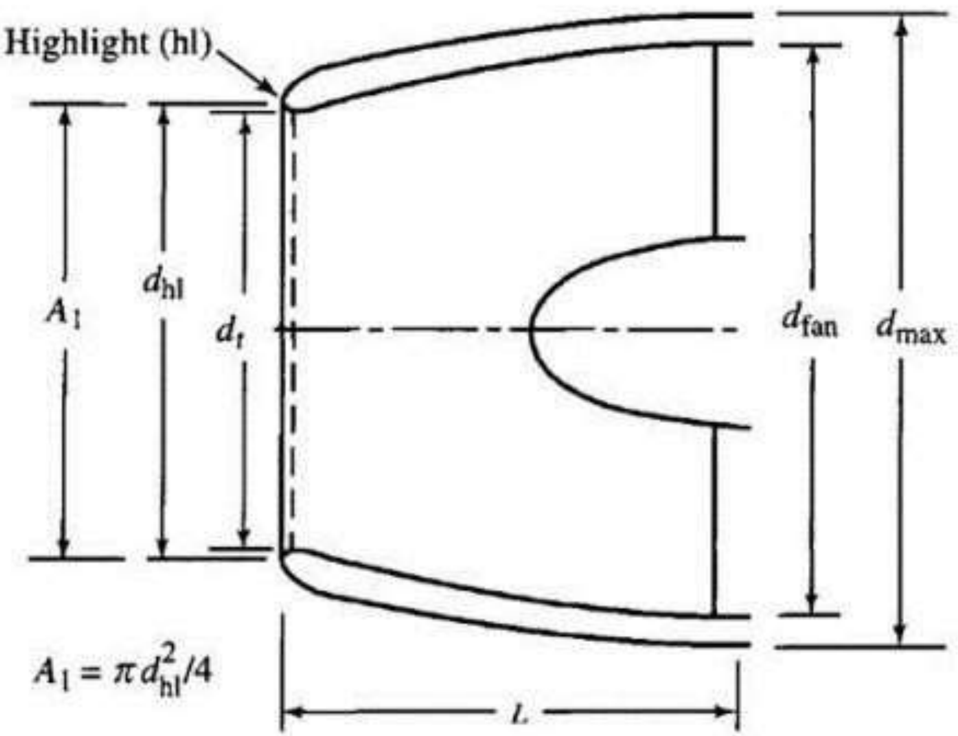
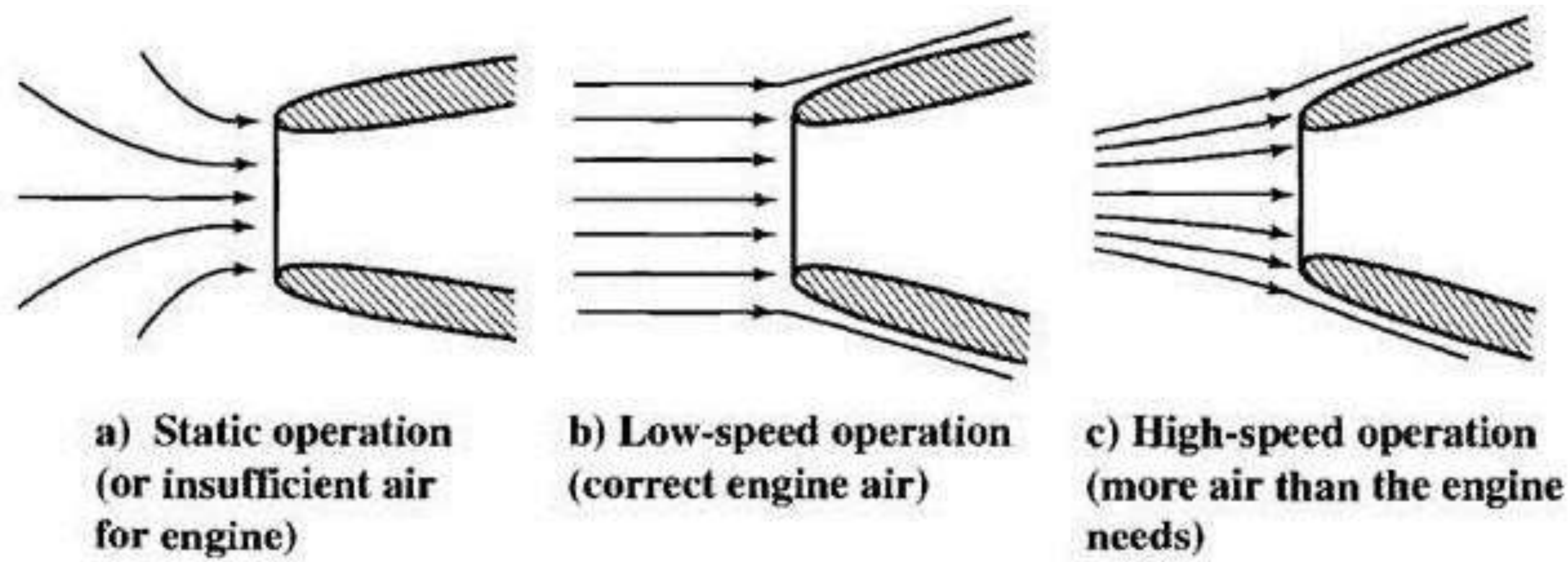


Fig. 10.1 Subsonic inlet nomenclature.

Subsonic inlet nomenclature



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 upwash and downwash, exhaust gas reingestion, 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, pylon, and wing

9) Noise suppression requirements

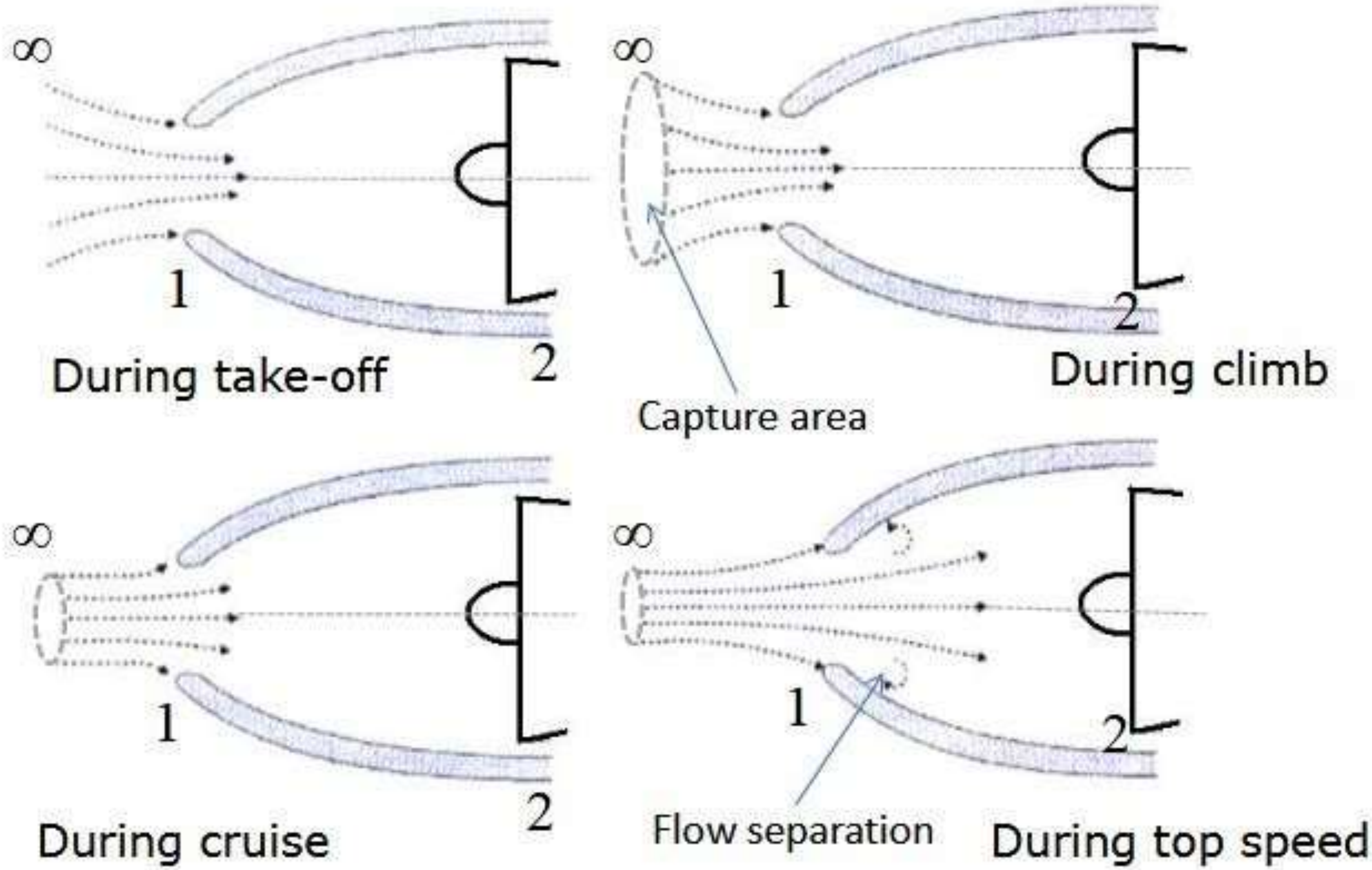
INTAKE PERFORMANCE

- 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.

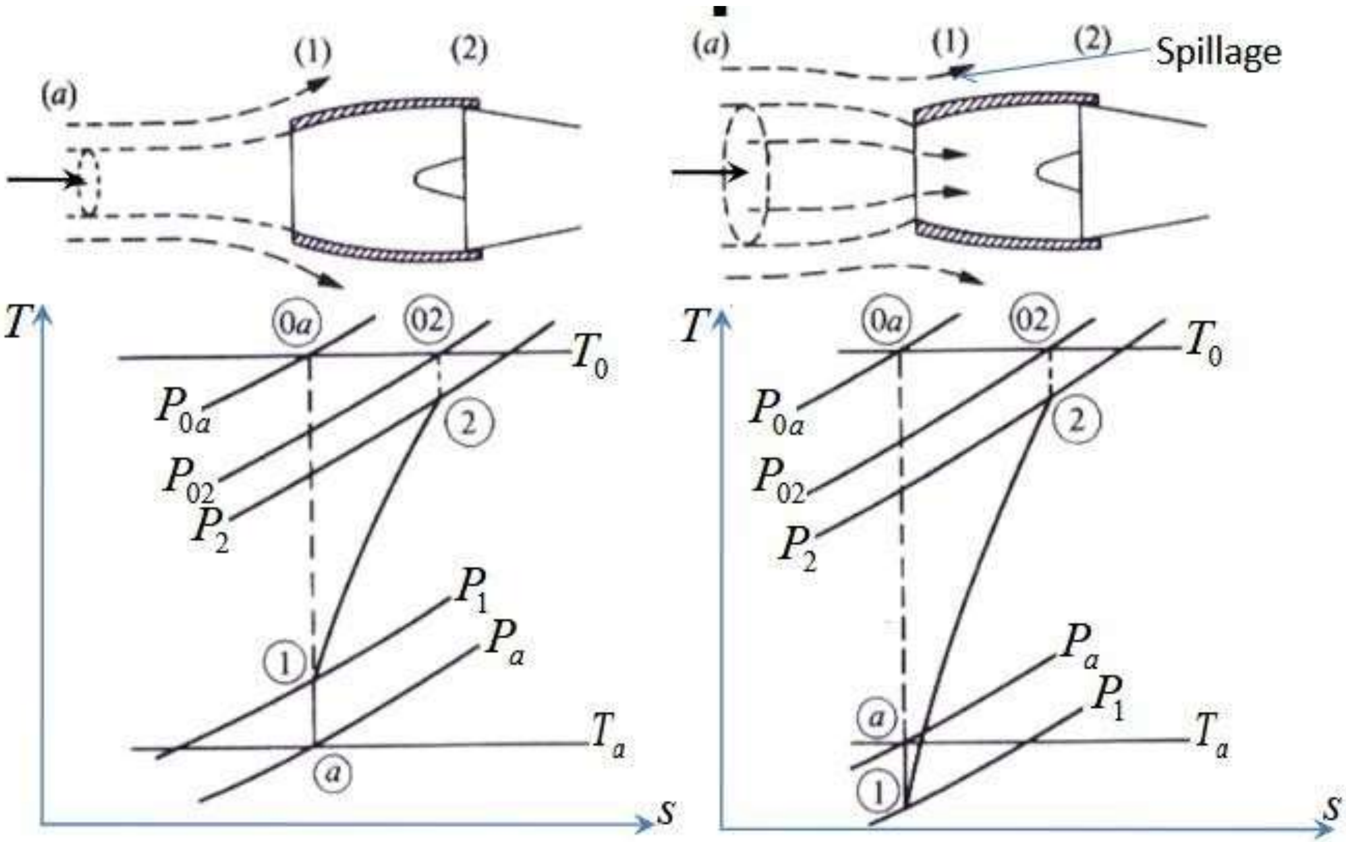
- A typical fixed geometry intake may have problems delivering this mass flow.

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

SUBSONIC INTAKE PERFORMANCE



- 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

Flow separation can get initiated at three possible locations

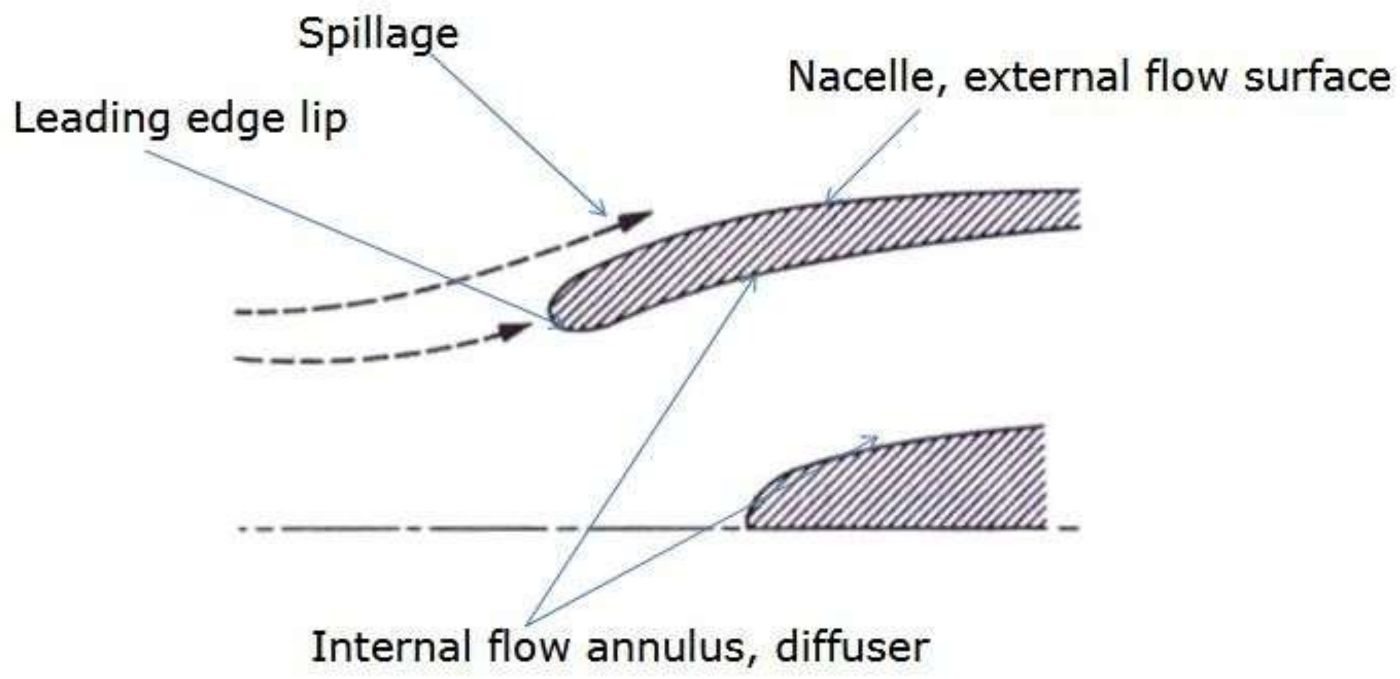
–External to the intake on the nacelle

–Within the diffuser internal surface

–On the center 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.



Possible regions of flow separation

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 (pre-entry compression) devoid of losses

–Sensitive to operating condition

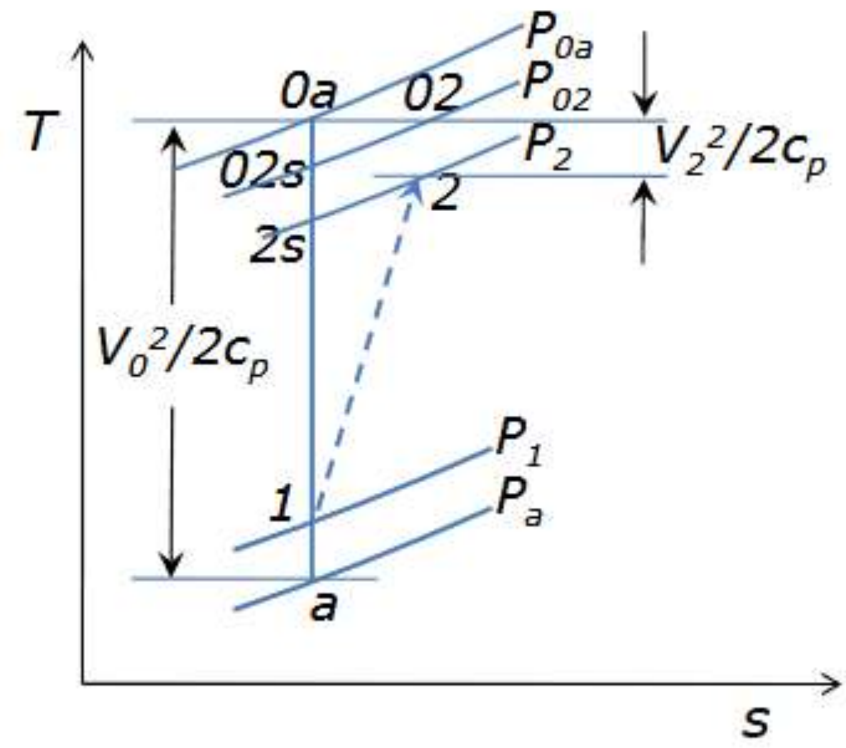
Trade-off between external and internal deceleration

Performance of intakes (subsonic as well as supersonic) are evaluated using the following:

–Isentropic efficiency

–Stagnation pressure ratio or pressure recovery

–Distortion coefficient



Isentropic efficiency

- Isentropic efficiency, ηd of the diffuser is

$$\eta_d = \frac{h_{02s} - h_a}{h_{0a} - h_a} \cong \frac{T_{02s} - T_a}{T_{0a} - T_a}$$

- Stagnation pressure ratio or pressure recovery is the ratio of the outlet stagnation pressure to the inlet stagnation pressure

$$\pi_d = P_{02} / P_{0a}$$

SUPERSONIC INTAKES

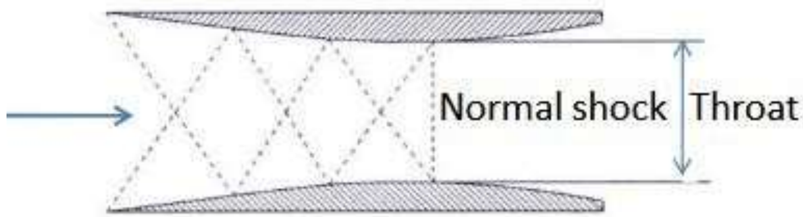
Internal, external or mixed compression

–Depending upon the location of the shocks

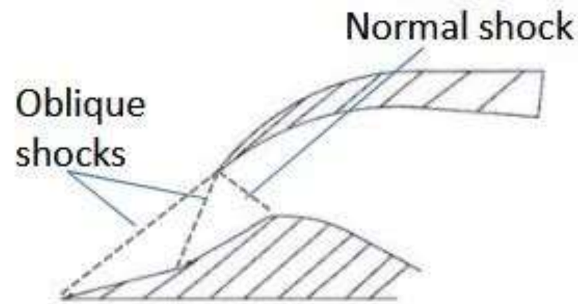
–Internal compression intakes have shocks that are located within the intake geometry

–External compression intakes have shocks located outside the intake

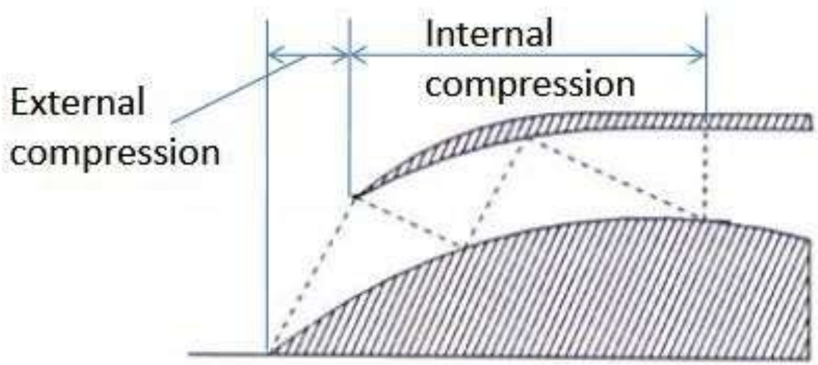
–Mixed compression intakes have shock that are located within as well as outside the intake geometry.



Internal compression intake



External compression intake



Mixed compression intake

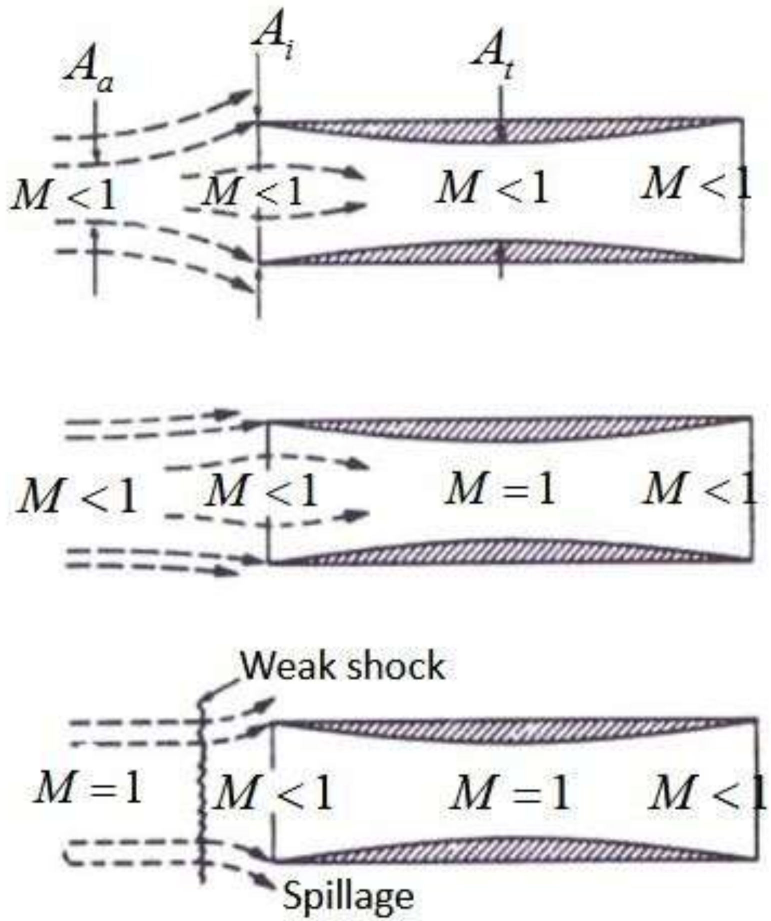
Supersonic diffusers are characterised by the presence of shocks.

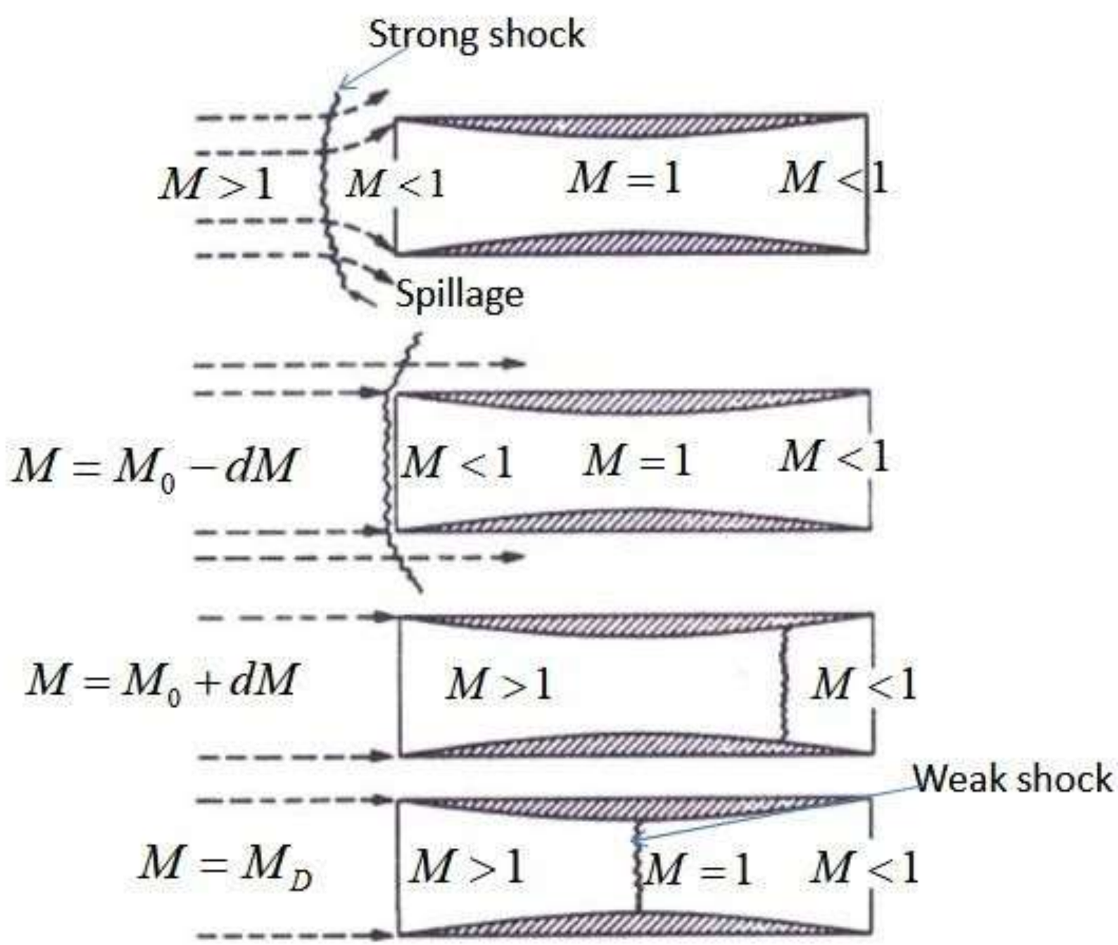
- However before the intake operates in a supersonic flow, it must pass through the subsonic flow regime.
- In some types of supersonic intakes, establishing a shock system with minimal

losses is not easy.

- The process of establishing a stable shock system is referred to as Starting of an intake.

STARTING OF AN INTAKE




Strong shock
$M>1$
$M<1$
$M=1$
$M<1$
Spillage
$M=M_0-dM$
$M<1$
$M=1$
$M<1$
$M=M_0+dM$
$M>1$
$M<1$
Weak shock
$M=M_D$
$M>1$
$M=1$
$M<1$

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

MODES 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:

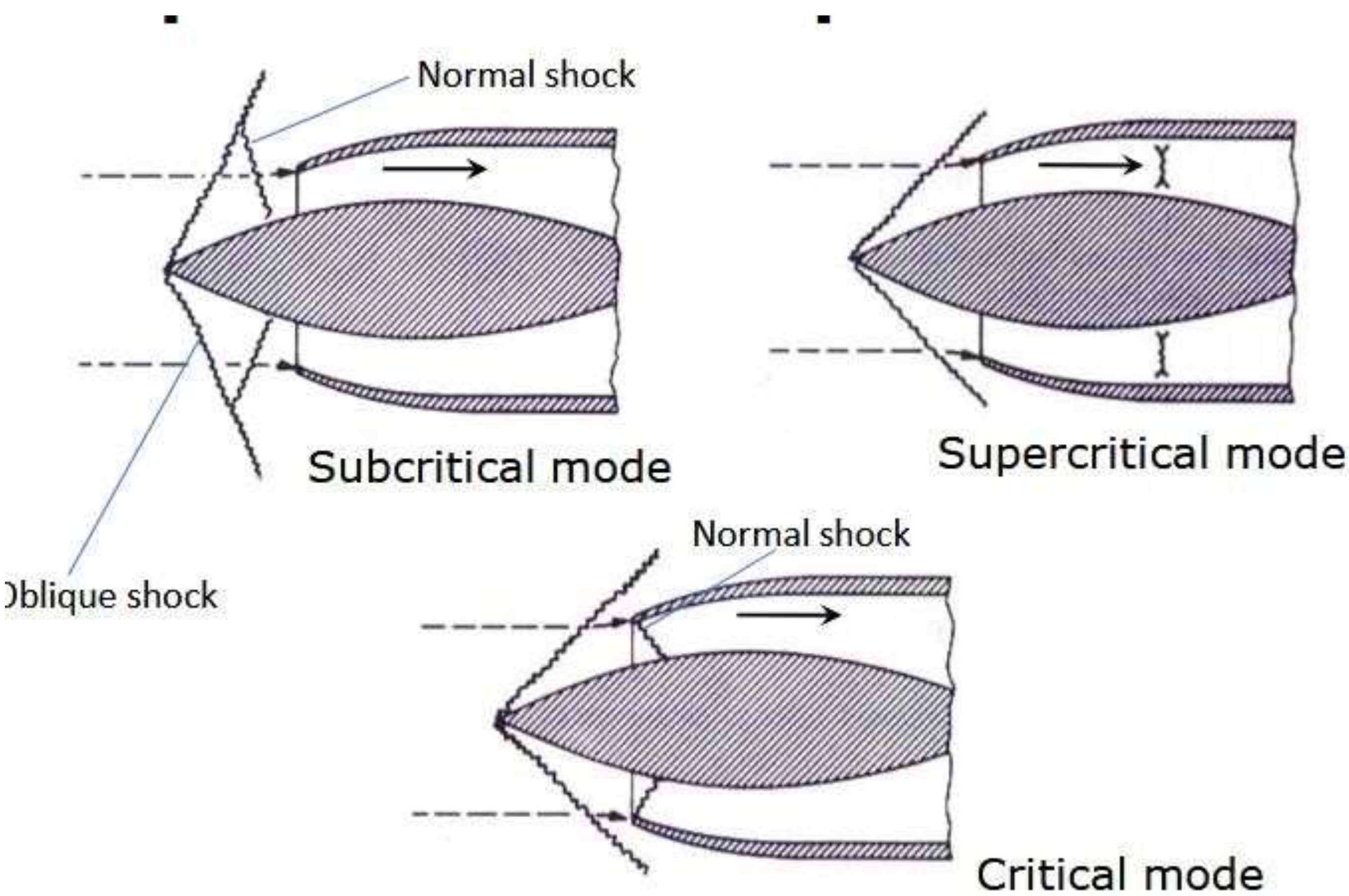
–Occurs at same mass flow as critical mode

–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


Normal shock
Subcritical mode
Oblique shock
Supercritical mode
Normal shock
Critical mode

- 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

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 maneuvering for military aircraft fitted with thrust vectoring systems.

19. 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

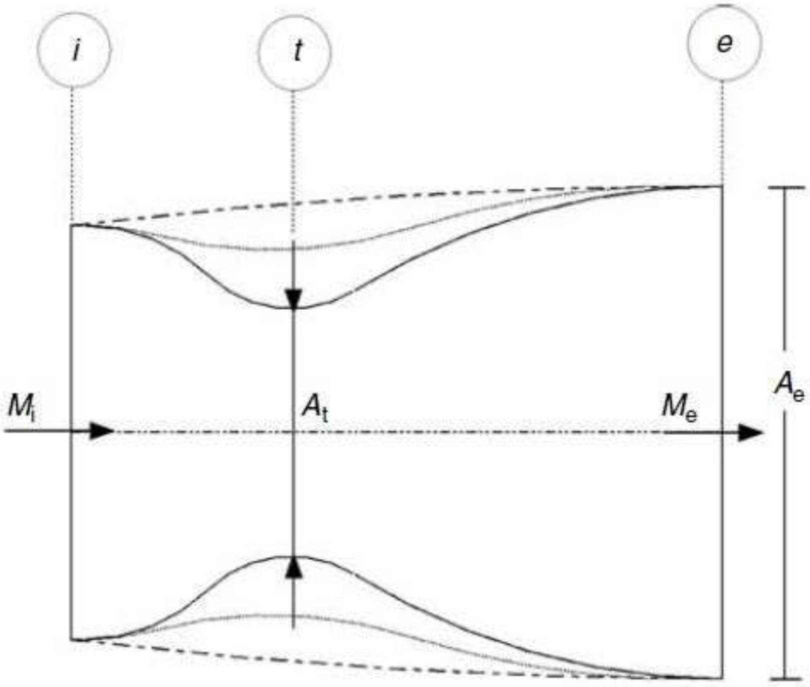
$$\dot{m} = \rho u A = \left(\frac{P}{RT}\right)\left(M\sqrt{\gamma RT}\right)A = (MA)\left(\frac{P}{P_0}\right)P_0\frac{\sqrt{\gamma}}{\sqrt{RT}}\sqrt{\frac{T_0}{T_0}}$$

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

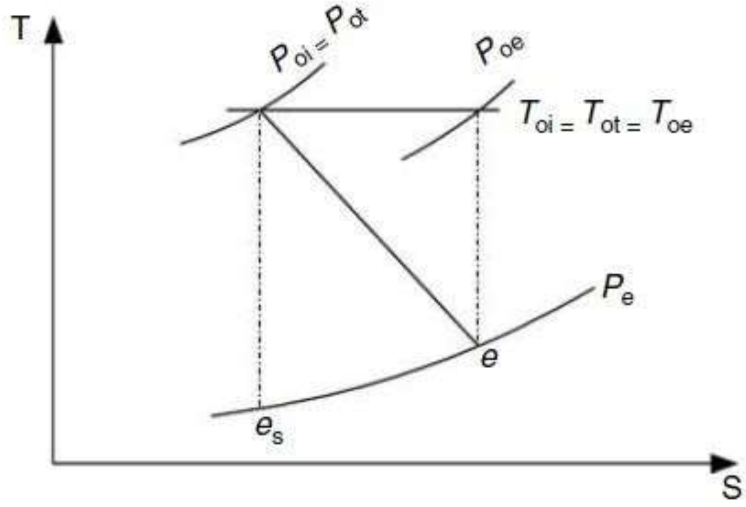
$$\dot{m} = \frac{AP_0}{\sqrt{T_0}}\sqrt{\frac{\gamma}{R}}\frac{M}{\left\{1+((\gamma-1)/2)\,M^2\right\}^{(\gamma+1)/2(\gamma-1)}}$$

C-D NOZZLE

- The flow in nozzles can be assumed adiabatic as the heat transfer per unit mass of fluid is muchsmaller 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 $P0i = P0t$, $T0i = T0t$ .Flow from the throat and up to the exit is assumed to be adiabatic but irreversible due to possible boundary layer separation, thus $P0t > P0e$ but $T0t = T0e$.
- Gases expand to the static pressure Pe with an adiabatic efficiency ηn in the range from 0.95 to 0.98.

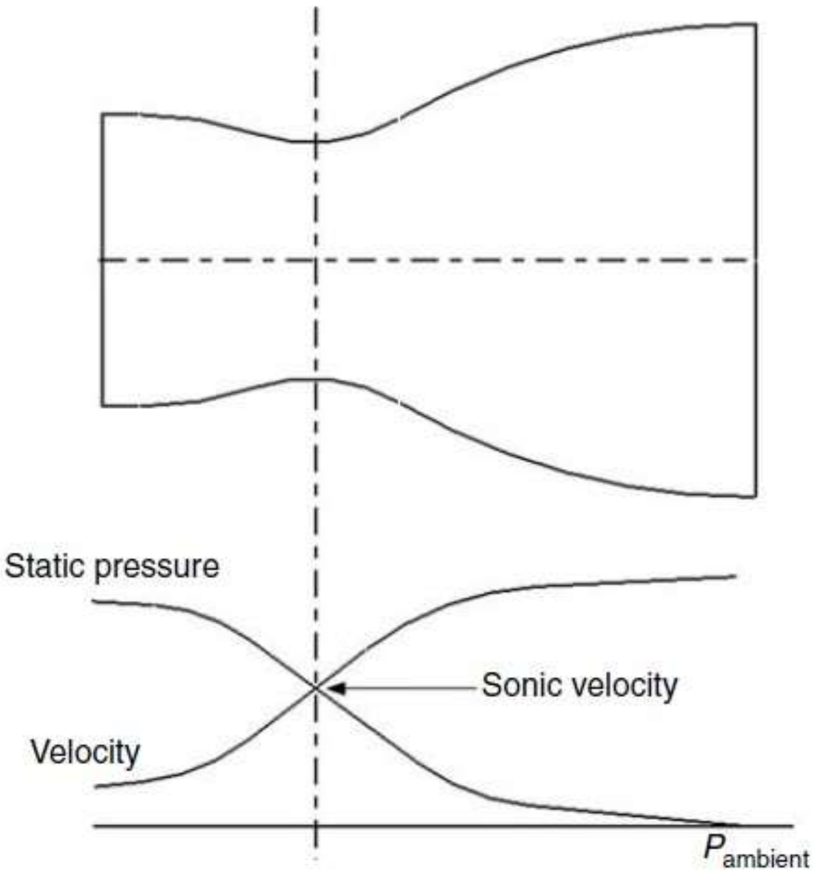


i
t
e
M_i
A_t
M_e
A_e



T
S
$P_{oi} = P_{ot}$
P_{oe}
$T_{oi} = T_{ot} = T_{oe}$
P_e
e
e_s

Velocity and static pressure distribution in convergent– divergent nozzle.

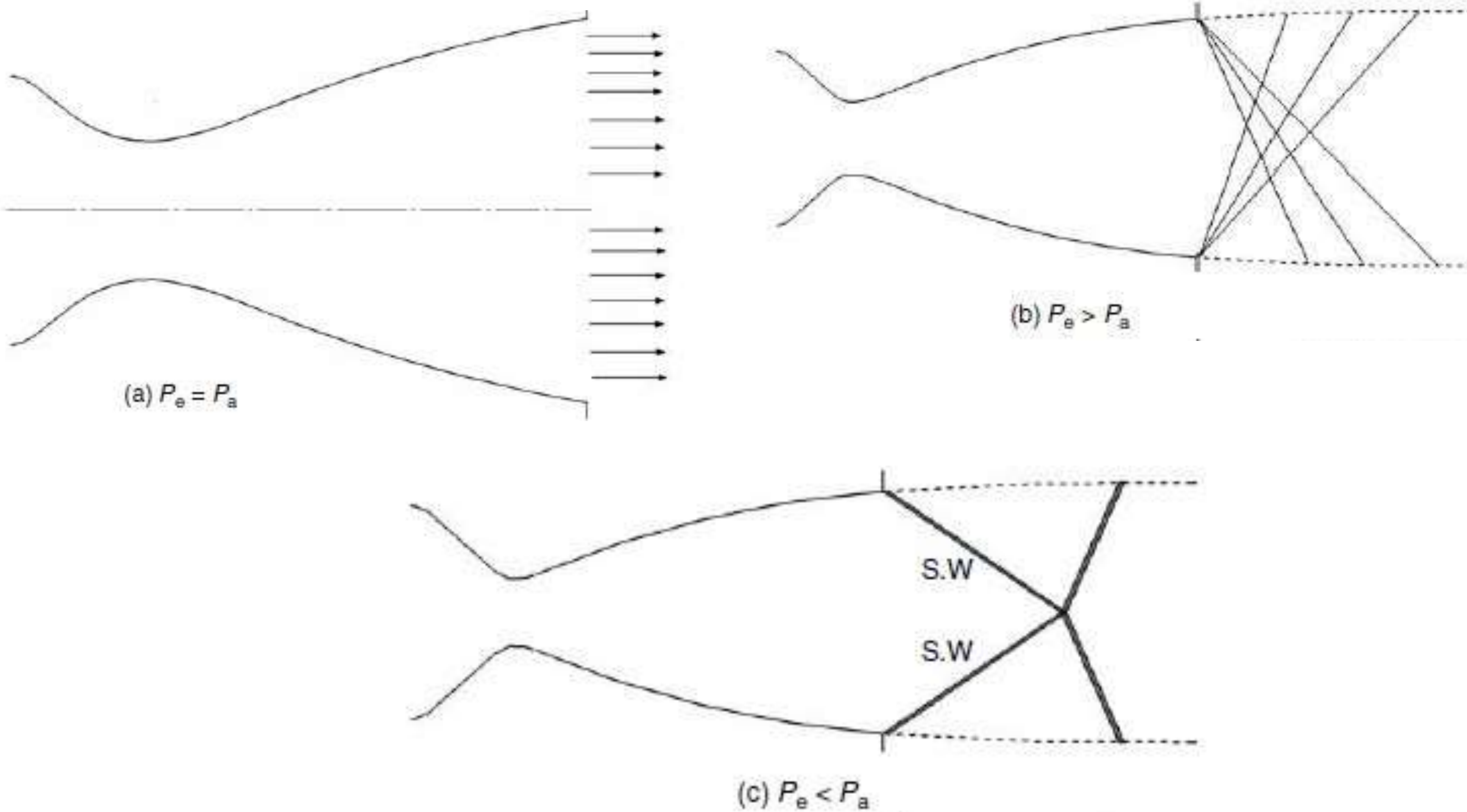


$$\frac{A^*}{A_e} = \frac{P_{0e}M_e}{P^*_{0i}}\left[\frac{(\gamma+1)/2}{1+((\gamma-1)/2)M_e^2}\right]^{(\gamma+1)/2(\gamma-1)}$$

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

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.

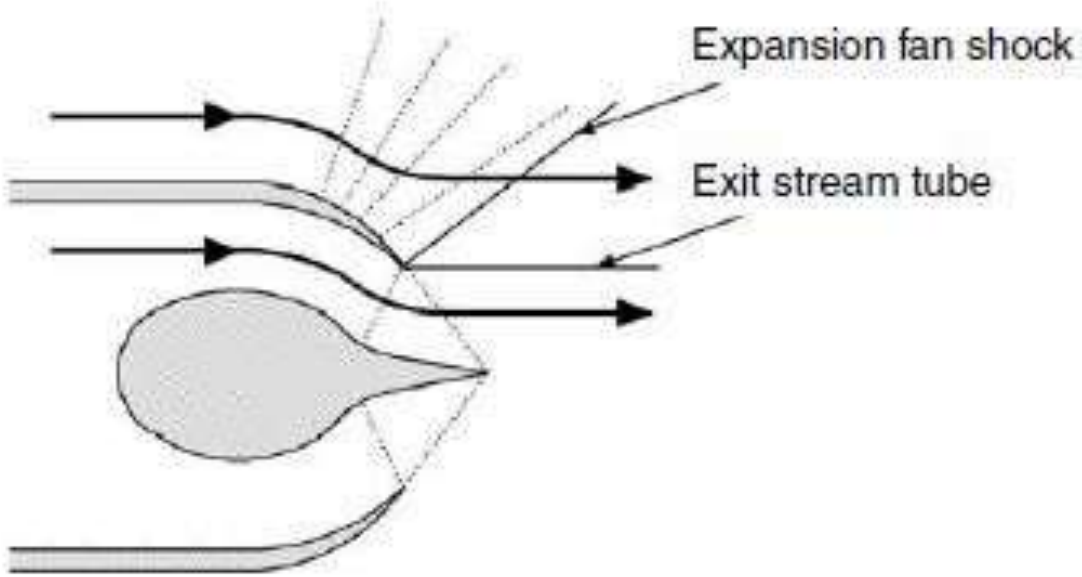


Behavior of convergent–divergent nozzle. (a) Design condition. (b) Exit pressure exceeds ambient pressure. (c)Ambient pressure exceeds exit pressure.

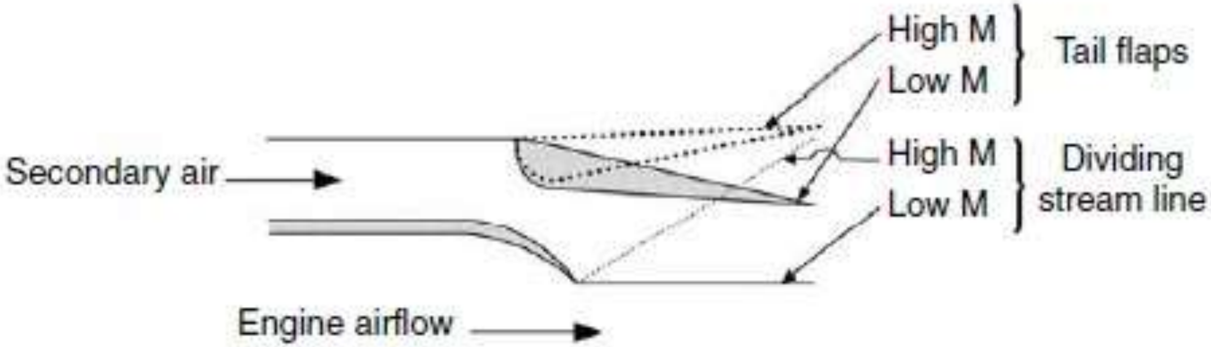
VARIABLE NOZZLE

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:

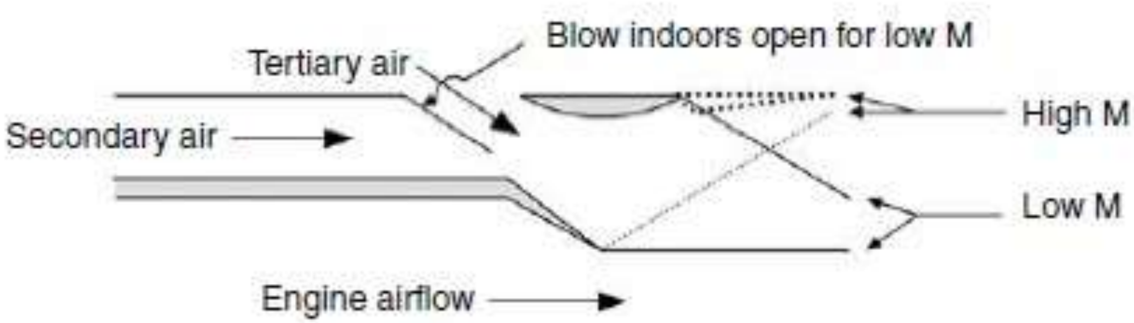
1. Central plug at nozzle outlet
2. Ejector type nozzle
3. IRIS nozzle



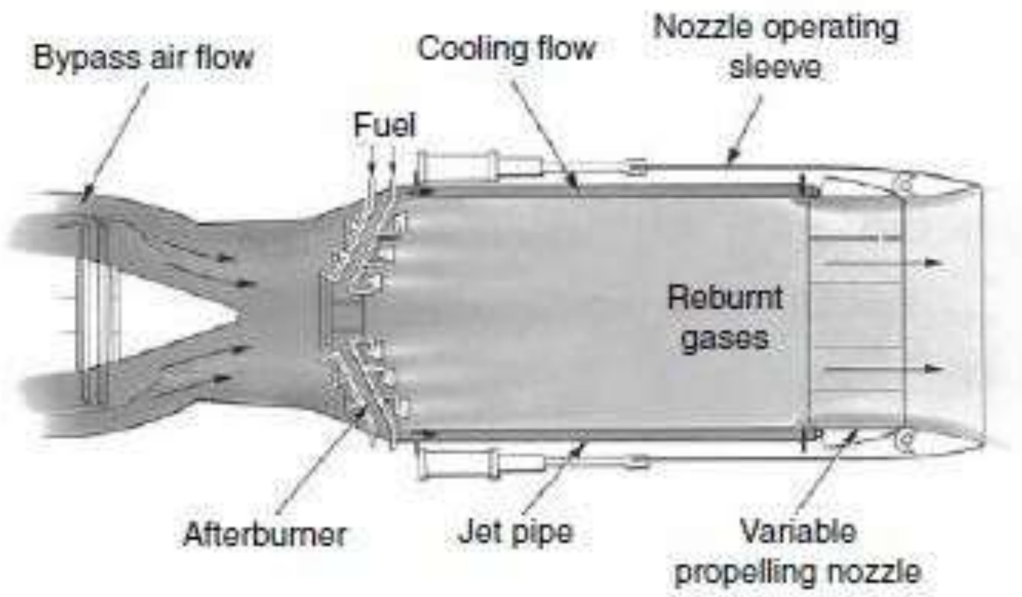
Plug nozzle at design point.



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

THRUST REVERSAL 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

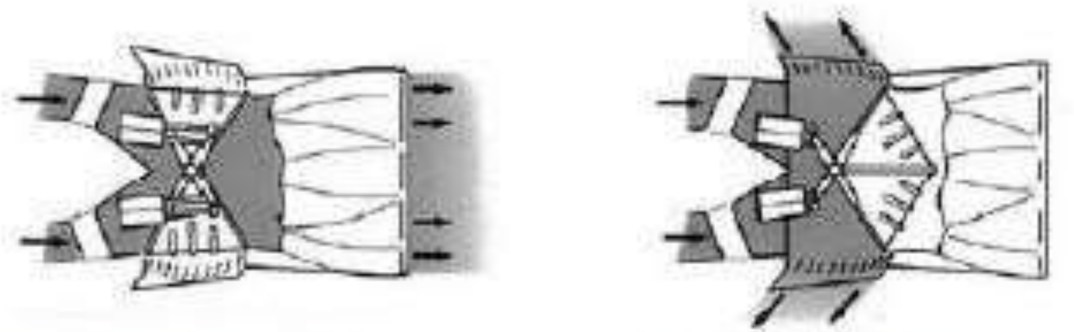
6. Reliable and fail safe

7. Cause few increased maintenance problems

8. Provide at least 50% of the full forward thrust

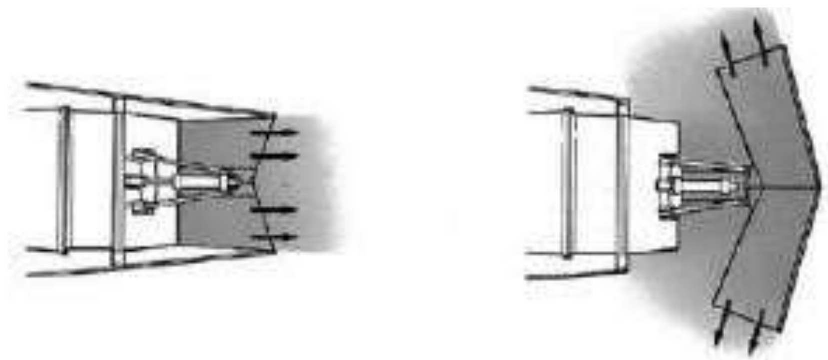
CLASSIFICATION

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

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.



Actuator extended and bucket doors in forward thrust position (B)

Actuator and bucket doors in reverse thrust position

NOZZLE COEFFICIENT

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 Cfg is the ratio of the actual gross thrust Fg actual to the ideal gross thrust $F_{gideal.}$

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

The following basic losses is accounted for:

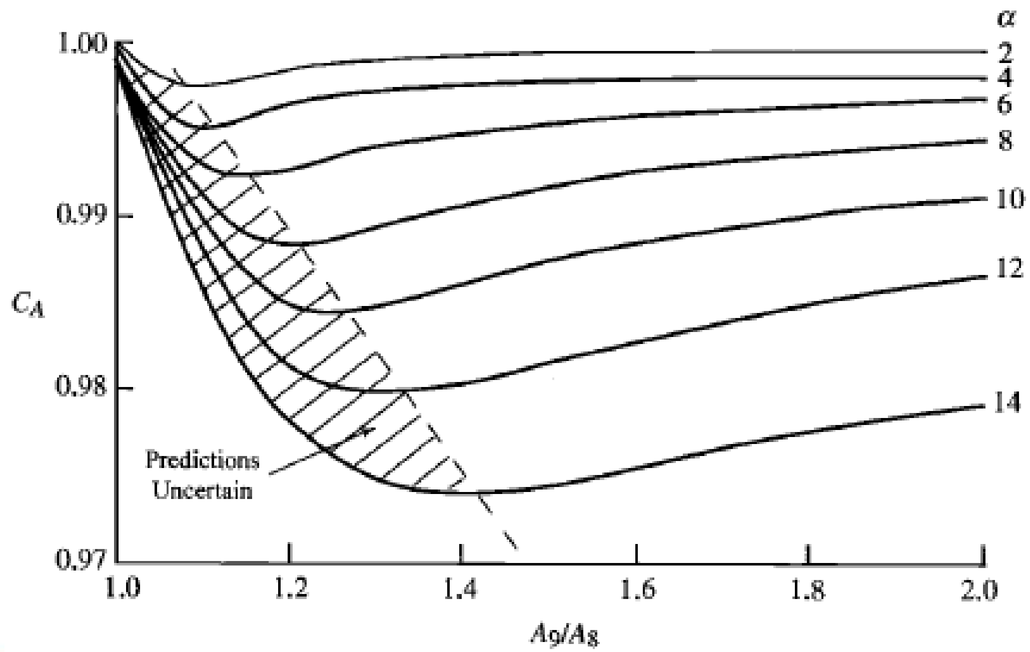
1) Thrust loss due to exhaust velocity vector angularity.

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

- 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.



UNIT - IV
THERMODYNAMICS OF REACTING SYSTEMS

Course Learning Outcomes

CLOs	Course Learning Outcome
CLO1	Understand different types of combustion chamber and functions of all the components.
CLO2	Describe the effect of operating variables on performance.
CLO3	Analyze combustion chamber performance and parameters influencing them.
CLO4	Describe the effect of flame tube cooling and its applications.

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
- Relightability
- 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.*

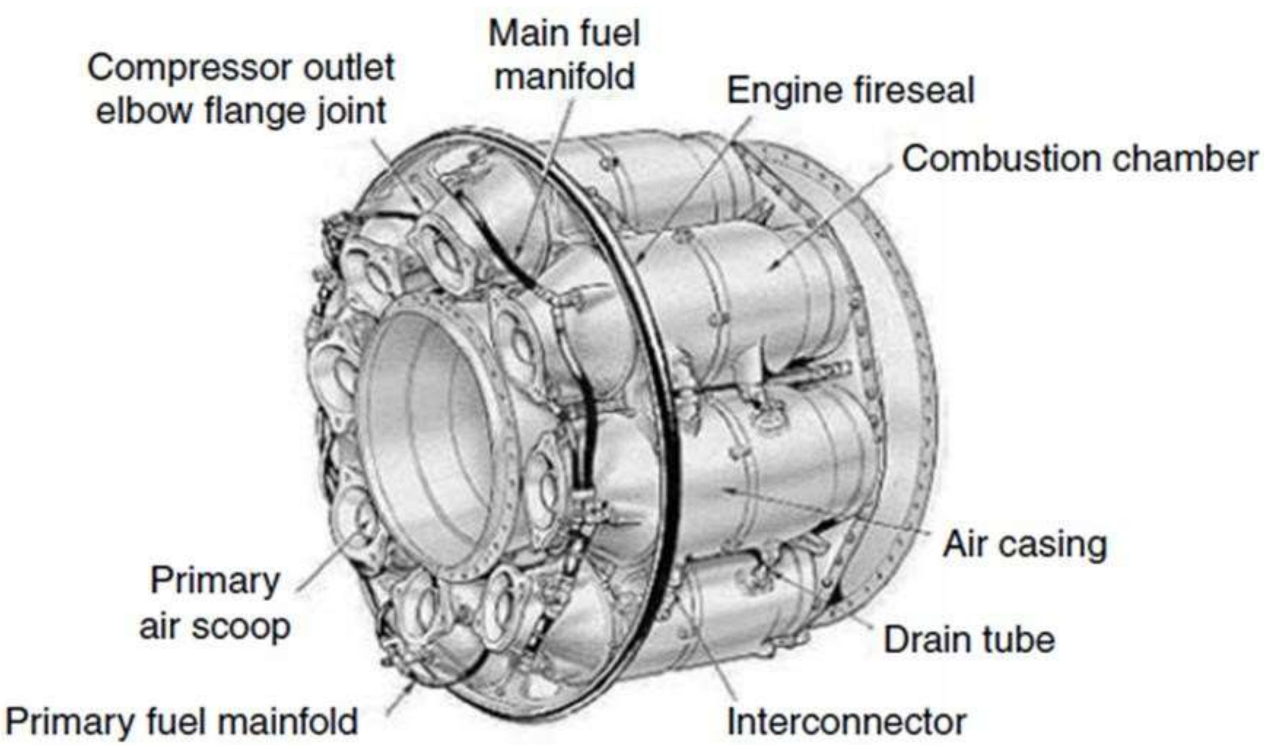
- Stable, efficient combustion can be upset by the

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.

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

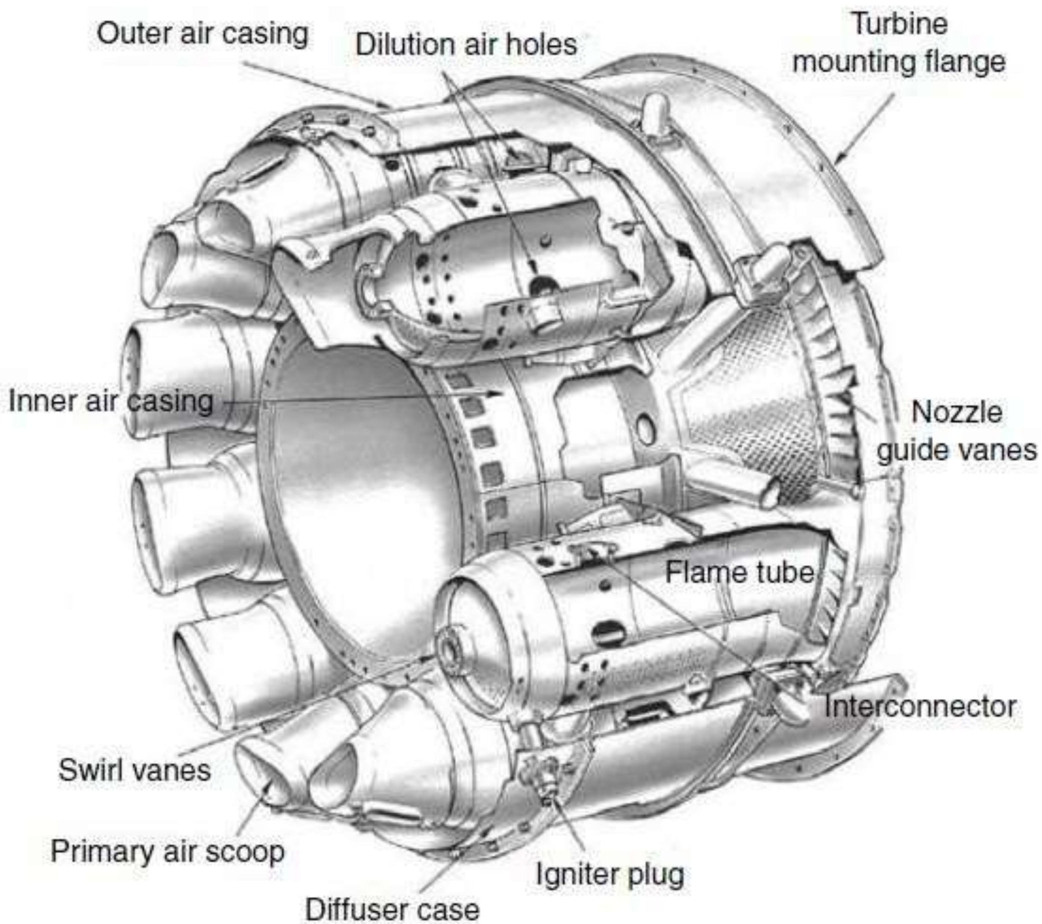
- Contemporary main burner systems may be broadly classified into one of the three types schematically illustrated in figure below: can, can annular, or annular.

CAN TYPE



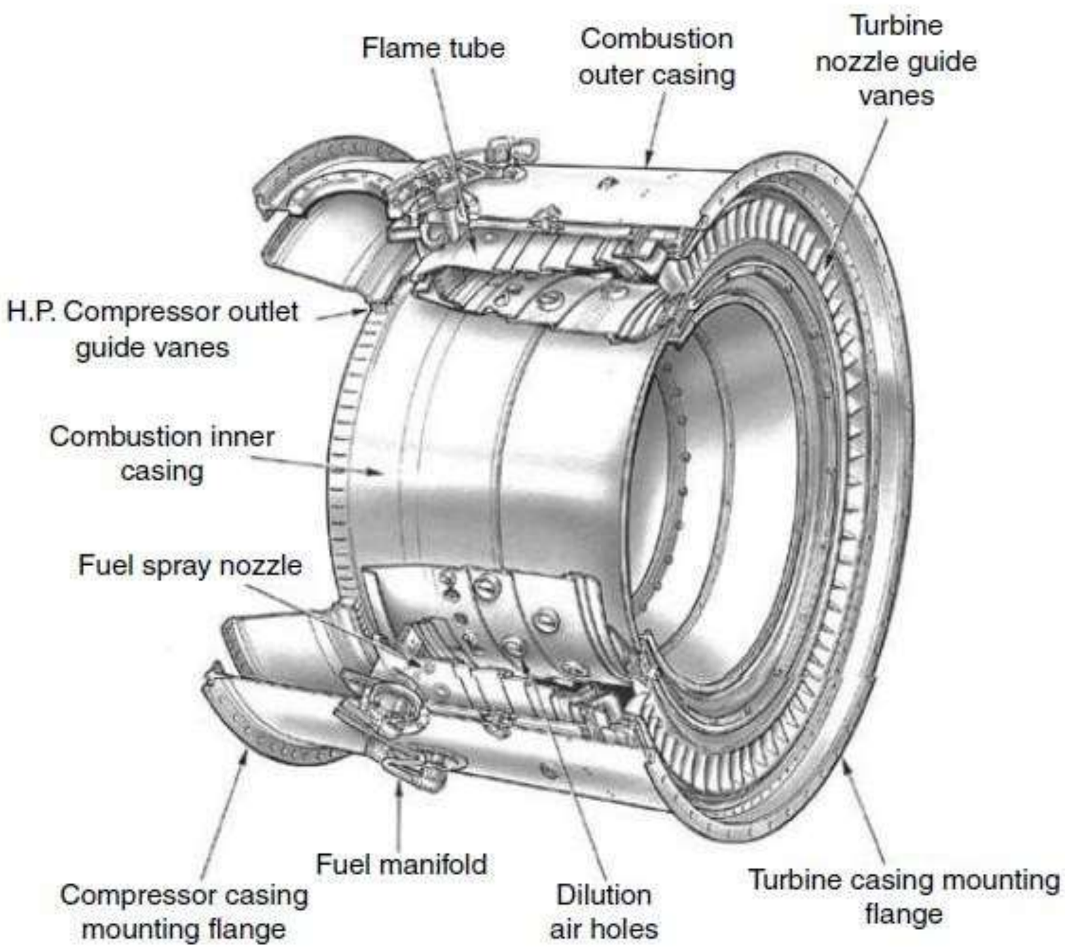
Mutiple can

CANNULAR

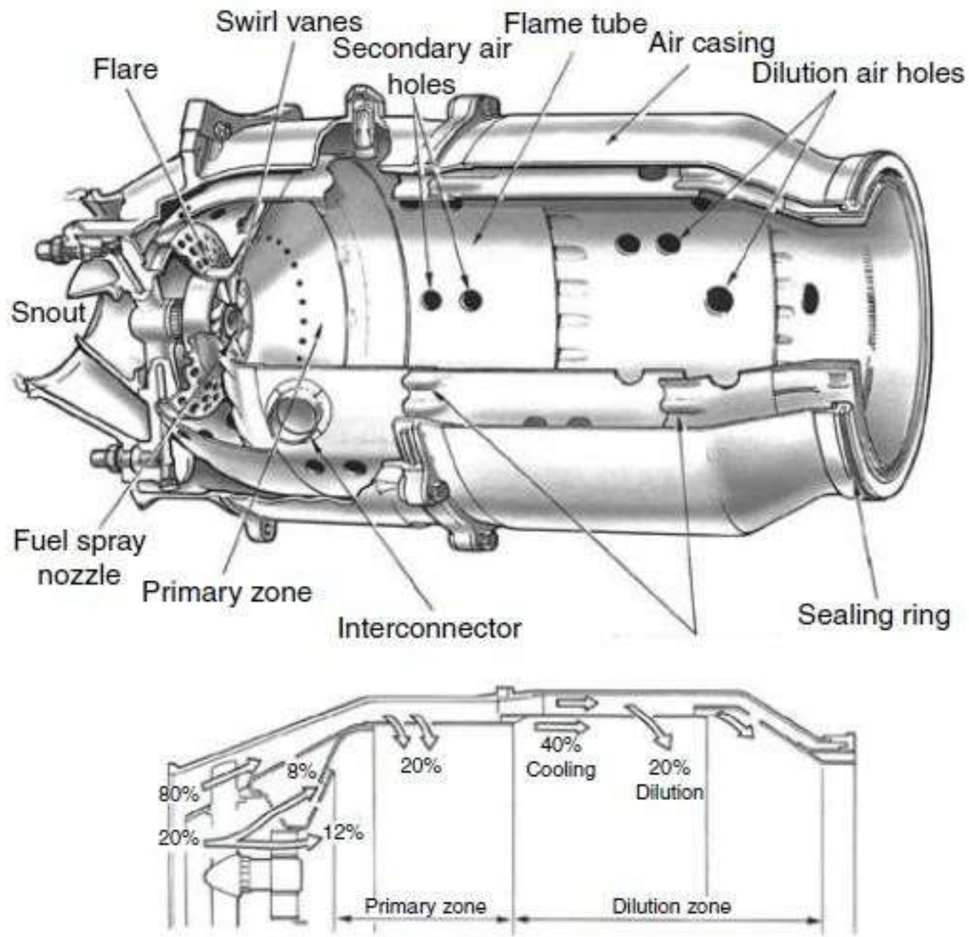


Outer air casing
Dilution air holes
Turbine mounting flange
Inner air casing
Nozzle guide vanes
Flame tube
Interconnector
Swirl vanes
Primary air scoop
Igniter plug
Diffuser case

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



Swirl vanes
Flame tube
Secondary air holes
Air casing
Dilution air holes
Flare
Snout
Fuel spray nozzle
Primary zone
Interconnector
Sealing ring
80%
20%
8%
12%
20%
40% Cooling
20% Dilution
Primary zone
Dilution zone

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.

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

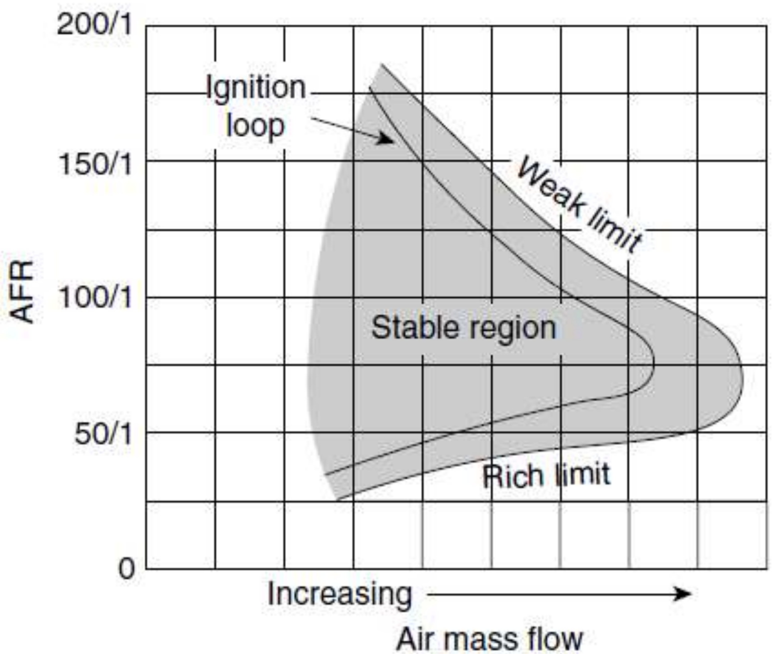
COMBUSTION EFFICIENCY

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.

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

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 maneuvering for military aircraft fitted with thrust vectoring systems.

19. 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

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

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

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

UNIT - V
PREMIXED FLAMES

Course Learning Outcomes

CLOs	Course Learning Outcome
CLO1	Understand different types of premixed flames.
CLO2	Explain the significance of flammability limits during combustion process.
CLO3	Describe theory of droplet combustion and turbulent combustion.
CLO4	Analyze the numerical methods of LNS & DNS and explain the parameters influencing them.

Applications:
Heating appliances
Bunsen burners
Burner for glass product manufacturing

Importance of studying laminar premixed flames:
Some burners use this type of flames as shown by examples above
Prerequisite to the study of turbulent premixed flames. Both have the same physical processes and many turbulent flame theories are based on underlying laminar flame structure.

Physical characteristics

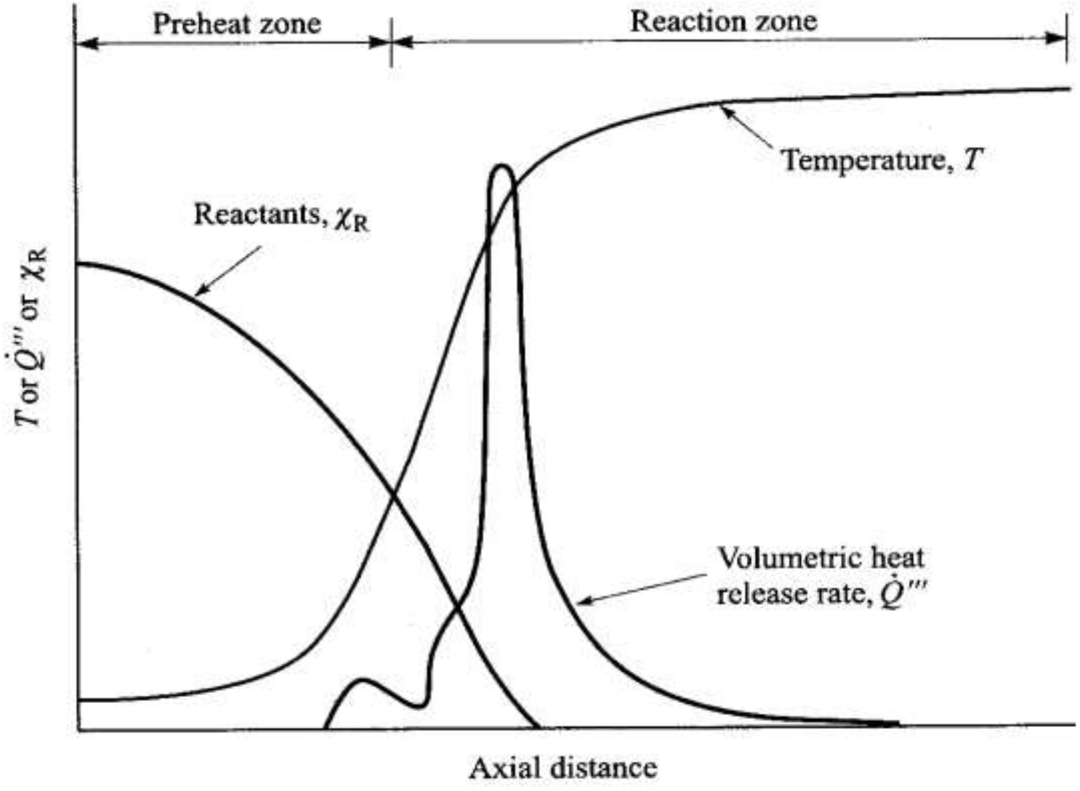
Figure 8.2 shows typical flame temperature profile, mole fraction of reactants, χ_R, and volumetric heat release, .

Velocity of reactants entering the flame, ν_u = flame propagation velocity, S_L

Products heated ⇨ product density (ρ_b) < reactant density (ρ_u). Continuity requires that burned gas velocity, ν_b >= unburned gas vel., ν_u

$$\rho_u \nu_u A = \rho_b \nu_b A \qquad (8.1)$$

For a typical hydrocarbon-air flame at P_{atm}, $\rho_u/\rho_b \approx 7$ ⇨ considerable acceleration of the gas flow across the flame (ν_b to ν_u).



8.2 Laminar flame structure. Temperature and heat-release-rate profiles

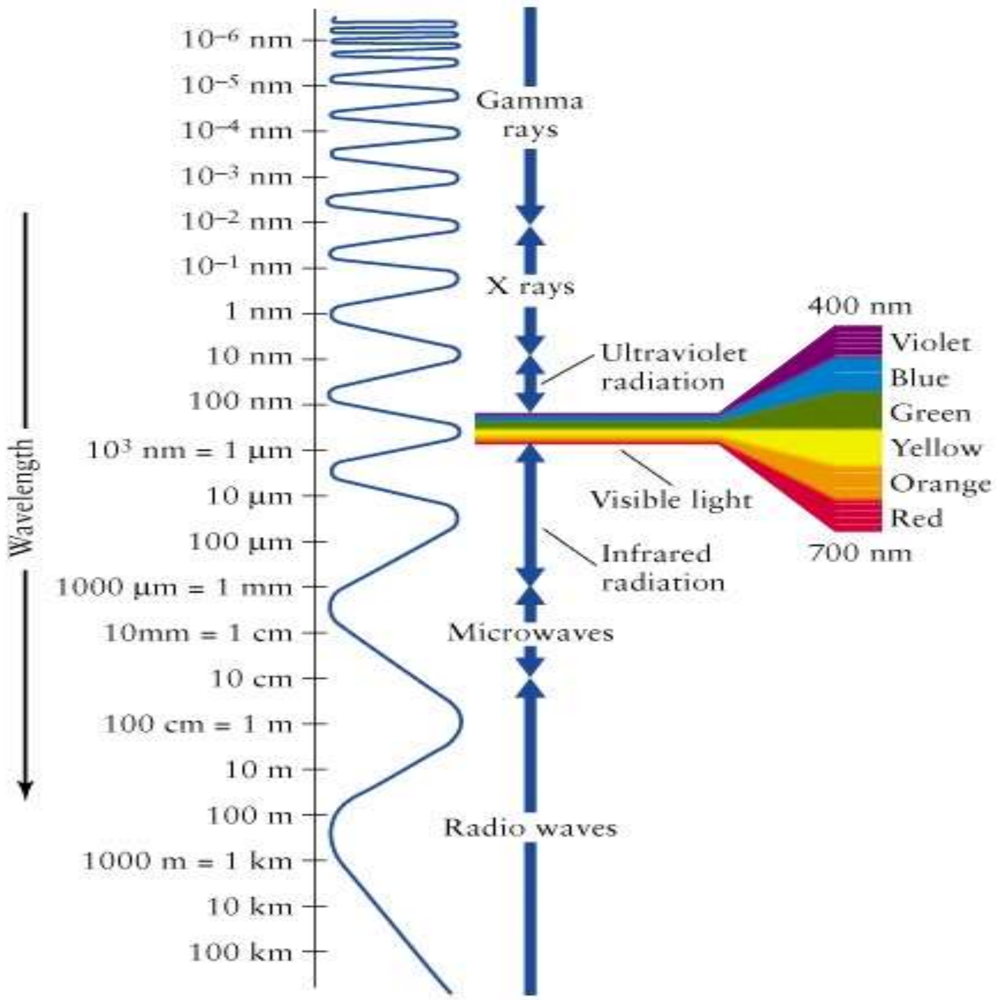
A flame consists of 2 zones:
Preheat zone, where little heat is released
Reaction zone, where the bulk of chemical energy is released
Reaction zone consists of 2 regions:
Thin region (less than a millimeter), where reactions are very fast
Wide region (several millimeters), where reactions are slow

In thin region (fast reaction zone), destruction of the fuel molecules and creation of many intermediate species occur. This region is dominated by bimolecular reactions to produce CO.
Wide zone (slow reaction zone) is dominated by radical recombination reactions and final burnout of CO via $CO + OH \rightarrow CO_2 + H$

Flame colours in fast-reaction zone:
If air > stoichiometric proportions, excited CH radicals result in blue radiation.
If air < stoichiometric proportions, the zone appears blue-green as a result of radiation from excited C_2.

In both flame regions, OH radicals contribute to the visible radiation, and to a lesser degree due to reaction $CO + O \rightarrow CO_2 + h\nu$.

If the flame is fuel-rich (much less air), soot will form, with its consequent blackbody continuum radiation. Although soot radiation has its maximum intensity in the infrared (recall Wien's law for blackbody radiation), the spectral sensitivity of the human eye causes us to see a bright yellow (near white) to dull orange emission, depending on the flame temperature



Spectrum of flame colours

Typical Laboratory Premixed Flames

The typical Bunsen-burner flame is a dual flame: a fuel rich premixed inner flame surrounded by a diffusion flame. Figure 8.3 illustrates a Bunsen burner.

The diffusion flame results when CO and OH from the rich inner flame encounter the ambient air.

The shape of the flame is determined by the combined effects of the velocity profile and heat losses to the tube wall.

For the flame to remain stationary,

S_L = normal component of ν_u = $\nu_u \sin\alpha$ (8.2). Figure 8.3b illustrates vector diagram.

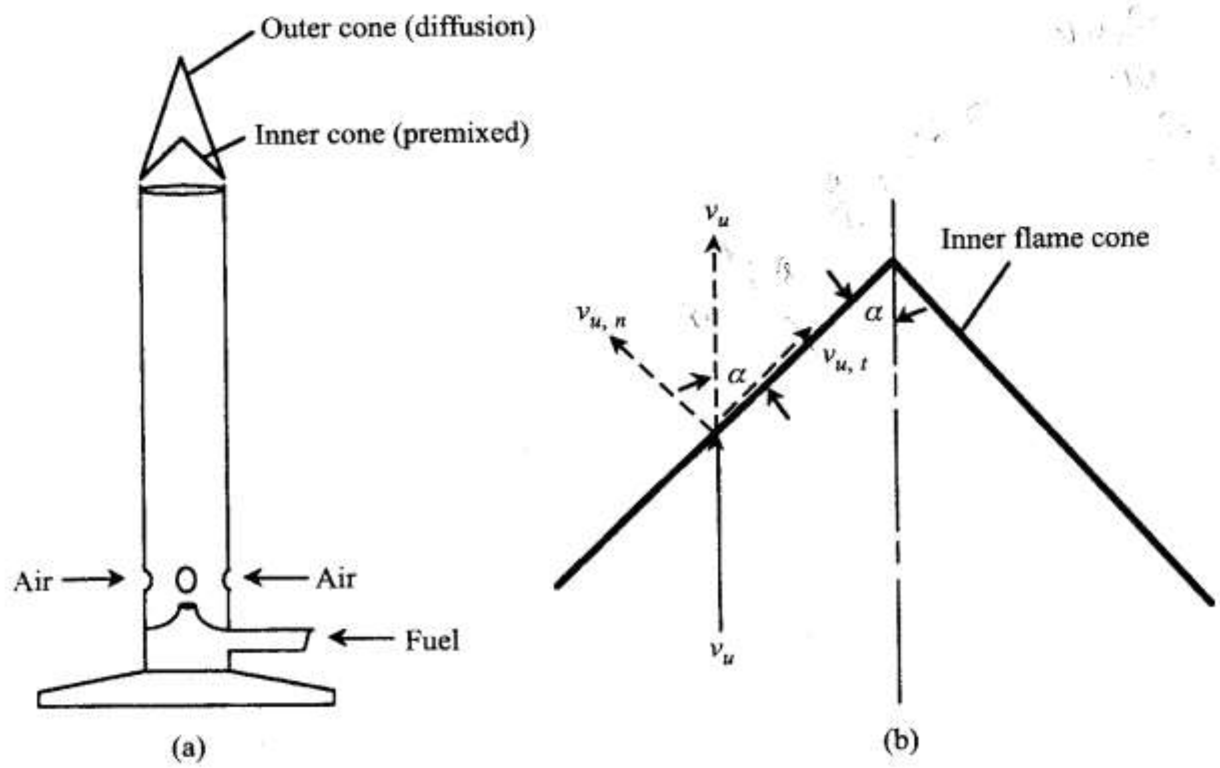


Figure 8.3 (a) Bunsen-burner schematic. (b) Laminar flame speed equals normal component of unburned gas velocity, $v_{u,n}$.

Example. A premixed laminar flame is stabilized in a one-dimensional gas flow where the vertical velocity of the unburned mixture, ν_u, varies linearly with the horizontal coordinate, x, as shown in the lower half of Fig. 8.6. Determine the flame shape and the distribution of the local angle of the flame surface from vertical. Assume the flame speed S_L is independent of position and equal to 0.4m/s (constant), a nominal value for a stoichiometric methane-air flame.

From Fig. 8.7, we see that the local angle, α, which the flame sheet makes with a vertical plane is (Eqn. 8.2)

α = arc sin (S_L/ν_u), where, from Fig. 8.6,

ν_u (mm/s) = 800 + (1200 – 800)/20 x (mm) (known).

ν_u (mm/s) = 800 + 20x.

So,

α = arc sin (400/(800 + 20x (mm))

and has values ranging from 30^o at x = 0 to19.5^o at x = 20 mm, as shown in the top part of Fig. 8.6.

To calculate the flame position, we first obtain an expression for the local slope of the flame sheet (dz/dx) in the x-z plane, and then integrate this expression with respect to x find z(x). From Fig. 8.7, we see that

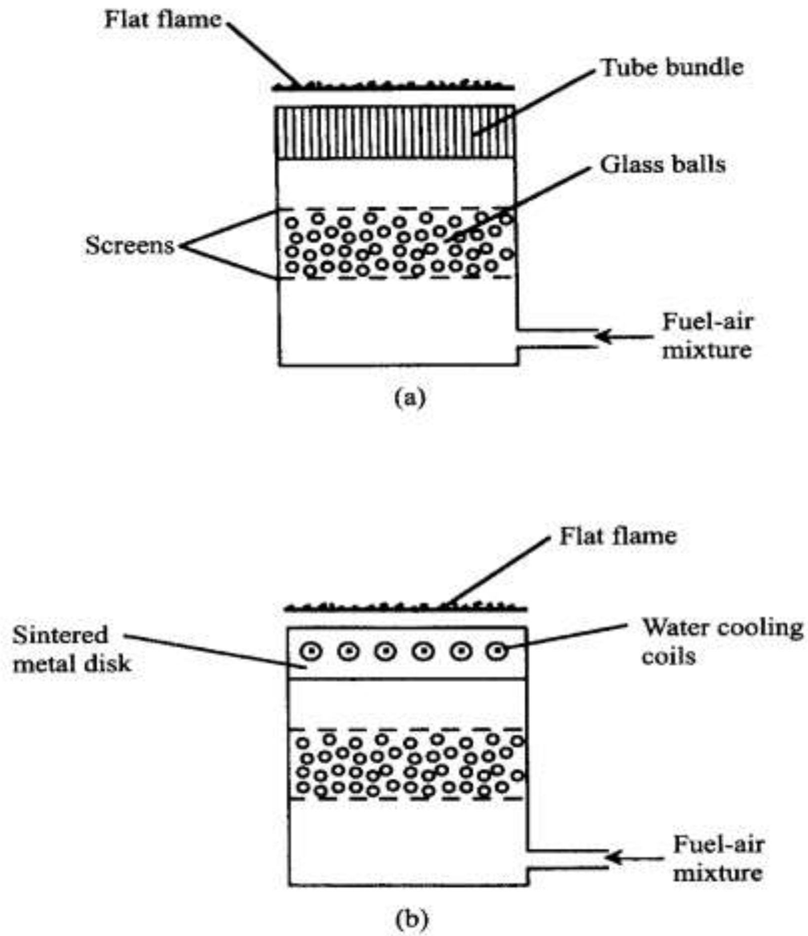


Figure 8.5 (a) Adiabatic flat-flame burner. (b) Nonadiabatic flat-flame burner.

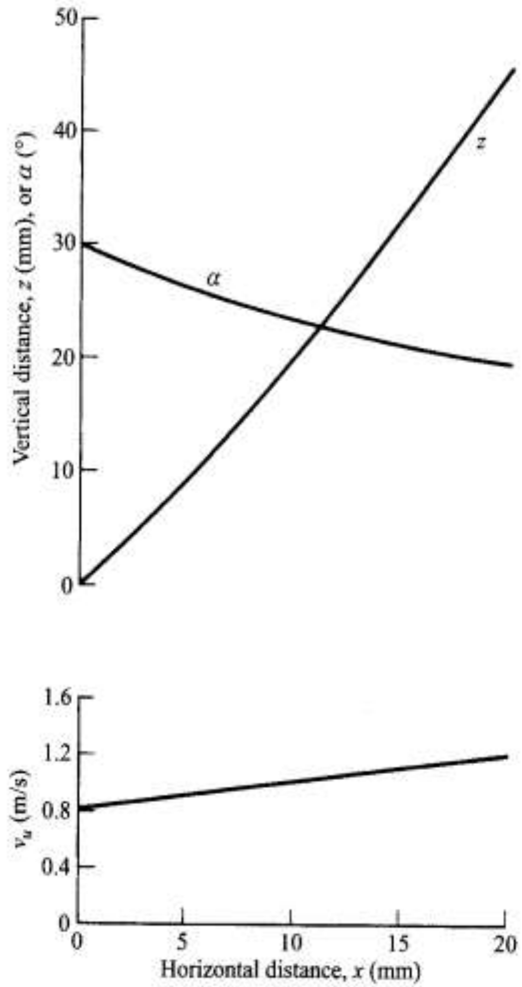


Figure 8.6 Flow velocity, flame position, and angle from vertical of line tangent to flame, for Example 8.1.

SIMPLIFIED ANALYSIS

Turns (2000) proposes simplified laminar flame speed and thickness on one-dimensional flame.

Assumptions used:

One-dimensional, constant-area, steady flow. One-dimensional flat flame is shown in Figure 8.5.

Kinetic and potential energies, viscous shear work, and thermal radiation are all neglected.

The small pressure difference across the flame is neglected; thus, pressure is constant.

- The Cp mixture ≠ f(temperature, composition). This is equivalent to assuming that individual species specific heats are all equal and constant.
- Fuel and oxidizer form products in a single-step exothermic reaction. Reaction is

$$1 \text{ kg fuel} + \nu \text{ kg oxidiser} \rightarrow (\nu + 1)\text{kg products}$$

- The oxidizer is present in stoichiometric or excess proportions; thus fuel is completely consumed at the flame.

FACTORS EFFECTING FLAME SPEED AND THICKNESS

1. Temperature (T_u and T_b)

Temperature dependencies of S_L and δ can be inferred from Eqns 8.20 and 8.21. Explicit dependencies is proposed by Turns as follows

where α is thermal diffusivity, T_u is unburned gas temperature, T_b is burned gas temperature.

where the exponent n is the overall reaction order, R_u = universal gas constant (J/kmol-K), E_A = activation energy (J/kmol)

Combining above scaling's yields and applying Eqs 8.20 and 8.21

For hydrocarbons, $n \approx 2$ and $E_A \approx 1.67.10^8$ J/kmol (40 kcal/gmol). Eqn 8.29 predicts S_L to increase by factor of 3.64 when T_u is increased from 300 to 600K. Table 8.1 shows comparisons of S_L and δ

The empirical S_L correlation of Andrews and Bradley [19] for stoichiometric methane-air flames,

$$S_L \text{ (cm/s)} = 10 + 3.71.10\text{-}4[T_u(K)]^2 \qquad (8.31)$$

which is shown in Fig. 8.13, along with data from several experimenters.

Using Eqn. 8.31, an increase in T_u from 300 K to 600 K results in S_L increasing by a factor of 3.3, which compares quite favourably with our estimate of 3.64 (Table 8.1).

1. Quenching by a Cold Wall

Flames extinguish upon entering a sufficiently small passageway. If the passageway is not too small, the flame will propagate through it. The critical diameter of a circular tube where a flame extinguishes rather than propagates, is referred to as the quenching distance.

Experimental quenching distances are determined by observing whether a flame stabilised above a tube does or does not flashback for a particular tube diameter when the reactant flow is rapidly shut off.

Quenching distances are also determined using high-aspect-ratio rectangular-slot burners. In this case, the quenching distance between the long sides, i.e., the slit width.

Tube-based quenching distances are somewhat larger (~20-50 percent) than slit-based ones

Ignition and Quenching Criteria

Williams [22] provides 2 rules-of-thumb governing ignition and flame extinction.

Criterion 1 -Ignition will only occur if enough energy is added to heat a slab thickness steadily propagating laminar flame to the adiabatic flame temperature.

Criterion 2 -The rate of liberation of heat by chemical reactions inside the slab must approximately balance the rate of heat loss from the slab by thermal conduction. This is applicable to the problem of flame quenching by a cold wall.

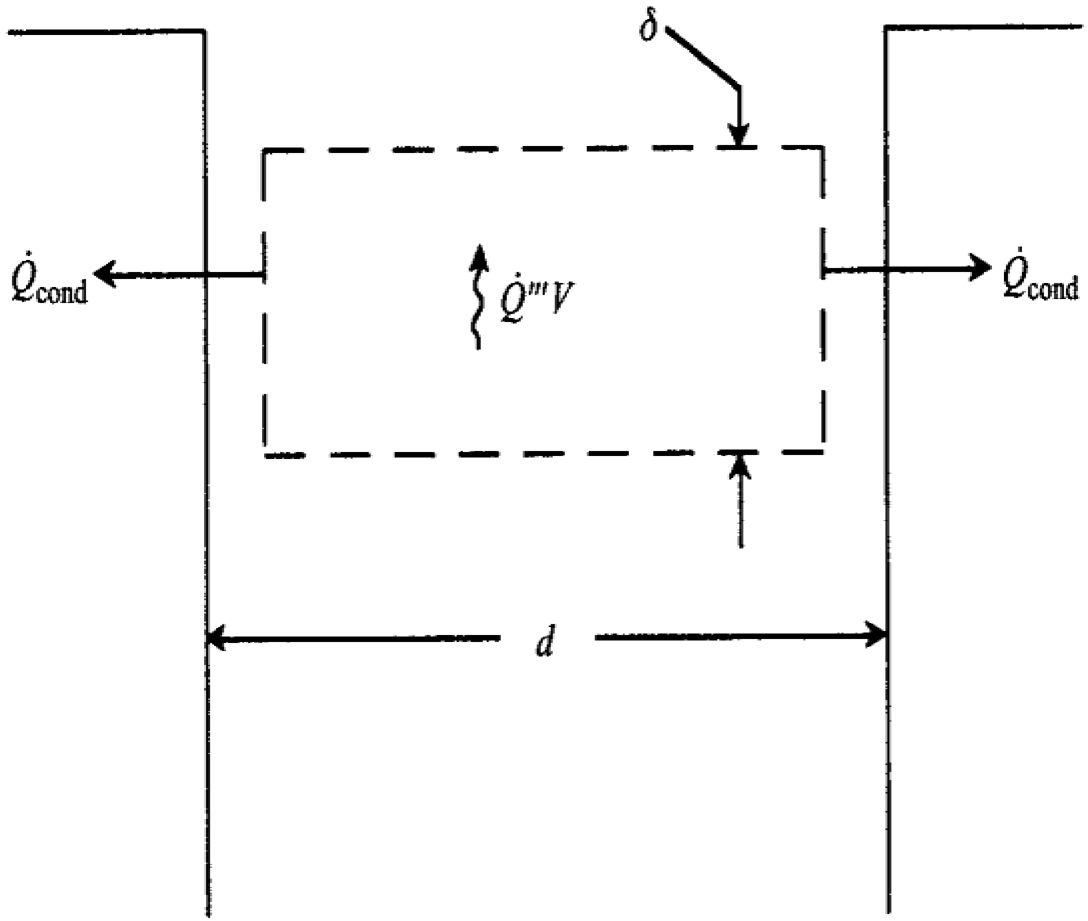


Figure 8.18 Schematic of flame quenching between two parallel walls.

2. Flammability Limits

A flame will propagate only within a range of mixture the so-called lower and upper limits of flammability. The limit is the leanest mixture ($\Phi < 1$), while the upper limit represents the richest mixture ($\Phi > 1$). $\Phi = (A/F)_{stoich} / (A/F)_{actual}$ by mass or by mole

Flammability limits are frequently quoted as %fuel by volume in the mixture, or as a % of the stoichiometric fuel requirement, i.e., (Φ x 100%). Table 8.4 shows flammability limits of some fuels

Flammability limits for a number of fuel-air mixtures at atmospheric pressure is obtained from experiments employing "tube method".
In this method, it is ascertained whether or not a flame initiated at the bottom of a vertical tube (approximately 50-mm diameter by 1.2-m long) propagates the length of the tube.
A mixture that sustains the flame is said to be flammable. By adjusting the mixture strength, the flammability limit can be ascertained.

Although flammability limits are physio-chemical properties of the fuel-air mixture, experimental flammability limits are related to losses from the system, in addition to the mixture properties, and, hence, generally apparatus dependent .

Although our calculations show that in the fully mixed state the mixture is not flammable, it is quite possible that, during the transient leaking process, a flammable mixture can exist somewhere within the room.

C_3H_8 is heavier than air and would tend to accumulate near the floor until it is mixed by bulk motion and molecular diffusion.

In environments employing flammable gases, monitors should be located at both low and high positions to detect leakage of heavy and light fuels, respectively.

3. Ignition
Most of ignition uses electrical spark (pemantik listrik). Another means is using pilot ignition (flame from very low-flow fuel).

Simplified Ignition Analysis
Consider Williams' second criterion, applied to a spherical volume of gas, which represents the incipient propagating flame created by a point spark.
Using the criterion:
Find a critical gas-volume radius, $\mathbf{R_{crit}}$, below which flame will not propagate
Find minimum ignition energy, $\mathbf{E_{ign}}$, to heat critical gas volume from initial state to flame temperature (T_u to T_b).

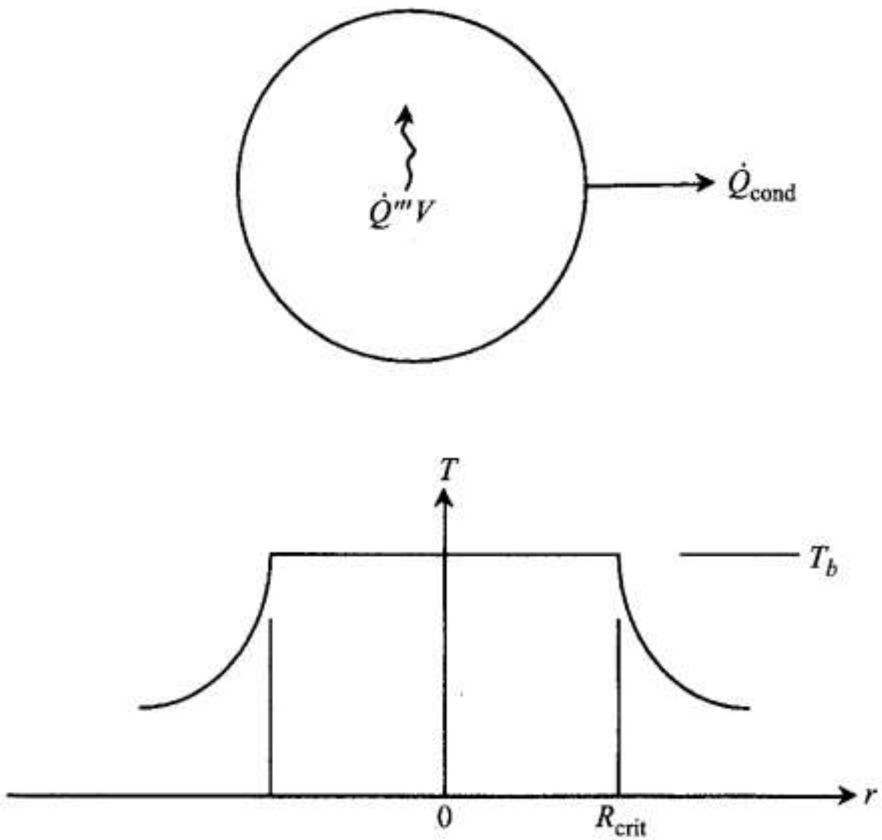
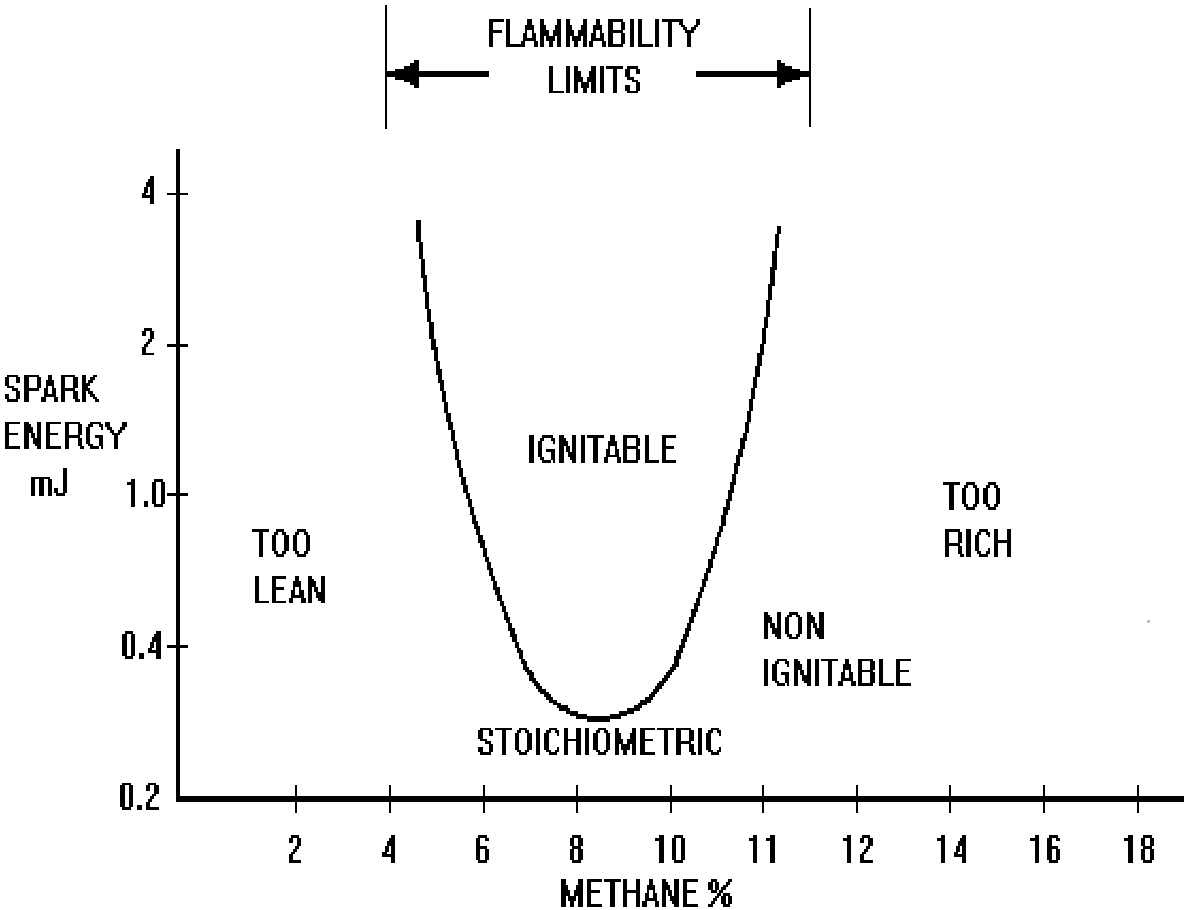


Figure 8.20 Critical volume of gas for spark ignition.

Effect of methane composition on E_{ign}



Temperature influence on spark-ignition energy

Fuel	Initial temp (K)	E_{ign} (mJ)
n-heptane	298	14.5
	373	6.7
	444	3.2
Iso-octane	298	27.0
	373	11.0
	444	4.8
n-pentane	243	45.0
	253	14.5

Fuel	Initial temp (K)	E_{ign} (mJ)
n-heptane	298	7.8
	373	4.2
	444	2.3
propane	233	11.7
	243	9.7
	253	8.4
	298	5.5
	331	4.2
	356	3.6
	373	3.5
	477	1.4