MODULE - 2

* INLET

⟹ SUBSONIC INLET

The inlet of an engine must be ~~defi~~ designed to prevent boundary ~~level~~ layer ~~eon~~ seperation even when, the axis of the intake is not perfectly alligned with the streamline direction. It is important that the stagnation pressure loss in the inlet should be minimum. The flow velocity and direction of the flow leaving the inlet, should be uniform

Depending on the flight speed and mass flow demanded by the engine the inlet mayhave to operate in a widerange of conditions. During level cruise (fig 1) the streamline pattern may include some deccelaration of the entering fluid external to the inlet.

During low speed operation (fig 2) ie during take off and climb, the same engine will demand more mass flow and the streamline pattern may resemble as shown in the figure, which is external accelaration of the stream near the inlet. In both the cases there is external change of state which is isentropic (no friction)

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In fig 2 the external accelaration rises, the inlet velocity and reduces the inlet pressure there by increasing the internal pressure ~~rise~~ rise across the diffuser. If this pressure rise is too large the diffuser may <u>stall</u> because of boundary layer seperation, the stalling usually reduces the stagnation pressure of the stream.

fig 1:

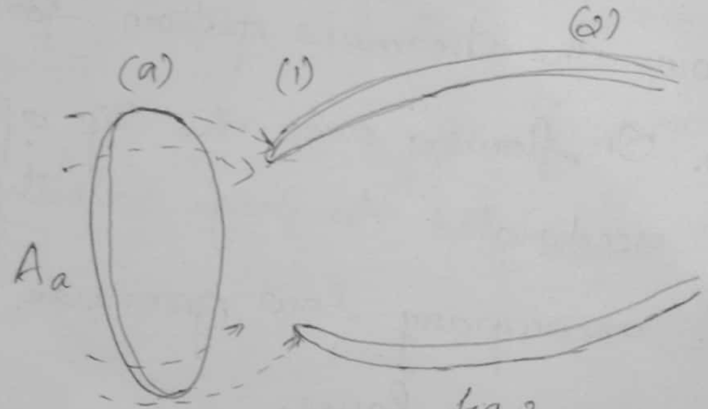

fig 2

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* INTERNAL FLOW

For subsonic aircraft inlets there is a requirement that the flow velocity entering the compressor must be steady and uniform. In actual engine inlet, the flow seperation can take place, in any of the three zones. The seperation of the external flow in zone 1, may be due to local high velocities and subsequent deccelaration over the outer surface. Separation

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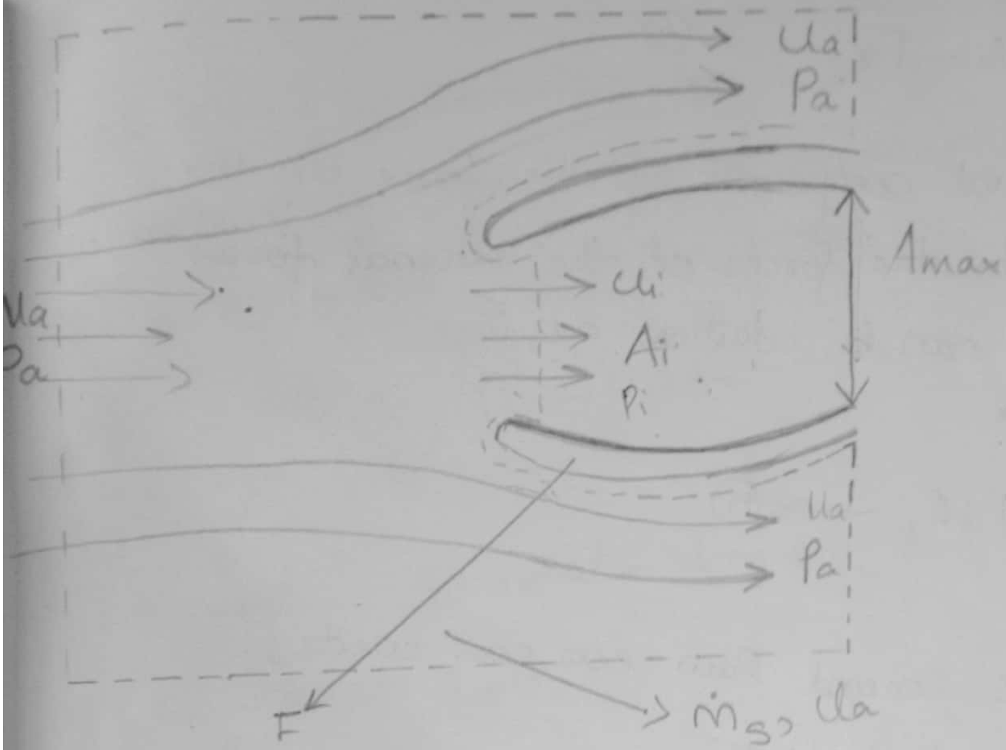
on the internal surface may take place either in zone 2 or zone 3 depending on the geometry of the duct and the operating conditions,

imp (10)

* The relation between minimum area ratio and external decceleration ratio

— The figure shows the streamline pattern for large external decceleration. On flowing over the lip of the inlet the external flow is accelarated to high velocity. This high velocity and the accompany low pressure can adversely effect the boundry layer.

The relation between minimum area ratio and the external decceleration ratio, is as shown below. A control surface is considered which extends far from the inlet on sides, ~~and~~ upstream, and downstream as shown in the figure. The external flow enters and leaves the control volume, with velocity 'Ua'. The internal flow enters the control volume with the velocity 'Ua' and leaves with a velocity 'Ui'



The net momentum flux out of the control volume can be written as,

[-ve ∵ only control volume considered, so separating out inlet air]

$$= \dot{m}_s U_a + \rho u_i^2 A_i - \rho U_a^2 A_{max} \longrightarrow (1)$$

From the mass balance equation [continuity eqn] we can write,

$$\dot{m}_s + \rho u_i A_i - \rho U_a A_{max} = 0$$

$$\dot{m}_s = \rho U_a A_{max} - \rho u_i A_i \longrightarrow (2)$$

Now the momentum flux can now be expressed as [substituting (2) into (1)] we get,

$$= \left(\rho U_a A_{max} U_a - \rho u_i A_i U_a\right) + \rho u_i^2 A_i - \rho U_a^2 A_{max}$$

$$= \rho u_i^2 A_i - \rho u_i A_i U_a$$

$$= \rho A_i \left[u_i^2 - u_i U_a\right] \longrightarrow (3)$$

The net force in the axial direction on the control volume can be written as,

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$$= P_a A_{max} - P_i A_i - F_x \longrightarrow (4)$$

where F_x, is the axial component of the force on the control volume due to the forces on the external surface of the inlet, and it can be written as,

$$F_x = \int_{A_i}^{A_{max}} P \, dA_x \longrightarrow (5)$$

Now from Newton's second law we can write,

net force = momentum flow flux.

$$P_a A_{max} - P_i A_i - \int_{A_i}^{A_{max}} P \, dA_x = \rho A_i (u_i^2 - u_i u_a) \longrightarrow (6)$$

Now adding the term, $\int_{A_i}^{A_{max}} P_a \, dA_x$ on to the LHS & RHS of eqn (6)

$$\int_{A_i}^{A_{max}} P_a \, dA_x + P_a A_{max} - P_i A_i - \int_{A_i}^{A_{max}} P \, dA_x = \rho A_i (u_i^2 - u_i u_a) + \int_{A_i}^{A_{max}} P_a \, dA_x.$$

$$\int_{A_i}^{A_{max}} (P_a - P) \, dA_x = \rho A_i (u_i^2 - u_i u_a) + P_a (A_{max} - A_i) + P_i A_i - P_a A_{max}$$

$$\int_{A_i}^{A_{max}} (P_a - P) \, dA_x = \rho A_i (u_i^2 - u_i u_a) + (P_i - P_a) A_i \longrightarrow (7)$$

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In the above equation the left hand side can be considered as a component of thrust which acts on the front external surface of the inlet, and it can be written as,

$$\Delta F_i = \int_{A_i}^{A_{max}} (P_a - P)\, dA_x \longrightarrow (8)$$

Now applying Bernoullis theorem to external deccelaration area of the internal flow,

$$P_a + \frac{1}{2}\rho C_a^2 = P_i + \frac{1}{2}\rho C_i^2$$

$$P_i - P_a = \rho\left(\frac{C_a^2 - C_i^2}{2}\right) \longrightarrow (9)$$

Now substituting eqn (8) & (9) into (7) we get,

$$\Delta F_i = \rho A_i (C_i^2 - C_i C_a) + \rho\left(\frac{C_a^2 - C_i^2}{2}\right) \frac{A_i}{A_{max}}$$

$$\Delta F_i = \frac{\rho A_i}{2}\left[2C_i^2 - 2C_i C_a + C_a^2 - C_i^2\right]$$

$$= \frac{\rho A_i}{2}\left[C_a^2 - 2C_a C_i + C_i^2\right]$$

$$= \frac{\rho A_i}{2} C_a^2 \left[1 - \frac{C_i}{C_a}\right]^2$$

$$\frac{\Delta F_i}{\frac{1}{2}\rho A_i C_a^2} = \left(1 - \frac{C_i}{C_a}\right)^2 \longrightarrow (10)$$

where $\frac{C_i}{C_a}$ is refered as external deccelaration ratio.

The pressure coefficient for the external flow can be written as,

$$C_p = \frac{P_a - P_{min}}{\frac{1}{2} \rho_a U_{max}^2} \longrightarrow (11)$$

For the outersurface of the nacelle and average pressure difference can be defined with respect to the area as follows,

$$\overline{P_a - P} = \frac{\int_{A_i}^{A_{max}} (P_a - P)\, dA_x}{A_{max} - A_i} = \delta (P_a - P_{min}) \longrightarrow (12)$$

where δ depends on the shape of the nacelle and its value varies between 0 to 1

From (8) we have

$$\Delta F_i = \int_{A_i}^{A_{max}} (P_a - P)\, dA_x$$

On the above expression substituting the value of δ from (12) we're,

$$\Delta F_i = \delta (P_a - P_{min})(A_{max} - A_i) \longrightarrow (13)$$

Now substituting the value of ΔF_i from eq (13) into eqn (10)

we're,

$$\frac{\delta (P_a - P_{min})(A_{max} - A_i)}{\frac{1}{2} \rho A_i U_a^2} = \left(1 - \frac{U_i}{U_a}\right)^2$$

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Now multiplying the left hand side of the above eqn with U_{max}^2,

$$\frac{\rho(P_a - P_{min})(A_{max} - A_i)\, U_{max}^2}{\frac{1}{2}\rho A_i U_a^2\, U_{max}^2} = \left(1 - \frac{U_i}{U_a}\right)^2$$

from (11), we can write,

$$\frac{\rho C_{P_{max}}(A_{max} - A_i)}{A_i U_a^2} = \left(1 - \frac{U_i}{U_a}\right)^2$$

$$\frac{A_{max} - A_i}{A_i} = \frac{(1 - U_i/U_a)^2}{\rho C_{P_{max}}\left(\frac{U_{max}}{U_a}\right)^2}$$

$$\frac{A_{max}}{A_i} = 1 + \frac{(1 - U_i/U_a)^2}{\rho C_{P_{max}}\left(\frac{U_{max}}{U_a}\right)^2}$$

* The list of major design variable for the inlet and nacelle are as follows;
  (i) inlet total pressure ratio and drag during ~~the~~ cruising condition.
  (ii) Engine location on wing or fuselage.
  (iii) Aircraft attitude
  (iv) Integration of diffuser.
  (v) Flow field interaction b/w nacelle, pylon and wing
  (vi) ~~Flo~~ Noise suppression requirements.

The airflow entering the compressor must have a low mach number in the range of 0.4 to 0.7 with the upper part of range suitable only for transonic compressor. If the engine is designed for subsonic cruise (mach no = 0.85), the inlet must be designed to act as a diffuser with diffusion from mach no > 0.85

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to 0.6. Part of this deceleration occurs, upstream of the inlet. The performance of the inlet should not be sensitive to pitch and yawning motion of the aircraft. Any distortion in the velocity profile at the compressor inlet can severely affect the compressor stages and may lead to the failure of the blades due to vibration.

* Supersonic Inlets:

For supersonic flight it is neccesary that the flow leaving the inlet must be subsonic. The compressors are capable of ingesting subsonic airflow at high pressure ratios. per stage. The difficulty in passing a fully supersonic stream through the compressor without excessive shock losses (especially at off design condition) has so far made the development of fully supersonic compressors a possibility. The mach no of the axial flow approaching a subsonic compressor should not be higher than 0.4, for a transonic stage it can be upto 0.6.

* Reverse Nozzle Diffuser

The deceleration of supersonic flow to subsonic speeds can be achieved by a simple normal shock. with small stagnation pressure loss, if the upstream mach no is close to 1. For high mach numbers, the losses across a single normal shock will be excessive. In this case, it will be better to use a combination of oblique shocks.

The isentropic deceleration will be still better, from 1-D. analysis, the supersonic nozzle [converging-diverg operated in reverse condition will be the ideal device to produce merely isentropic deceleration. The difficulties in implementing such a concept is the problem of success operation ~~to~~ a reverse nozzle inlet over a wide range of flight mach number are the interaction of the shocks with internal shock waves, boundary layer. Another difficulty is flow instability, under certain condition, the flow can become oscillatory.

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* Starting Problem On Supersonic Inlet. (case I - overspeeding the inlet)

The internal supersonic deceleration in a converging passage is generally difficult to establish. The design condition cannot be achieved without momentarily overspeeding the inlet air or varying the diffuser geometry. This difficulty is due to the shocks that arise during the deceleration process. If the boundary layer effects are neglected and the starting behaviour of converging - diverging diffuser is considered one dimensional and isentropic except for losses, that occur due to the presence of normal shockwave. It can be a valid representation of a real flow from which the wall boundary layer fluid is carefully removed by suction through porous wall.

The figure illustrates the successive steps of the operation of converging - diverging inlet.

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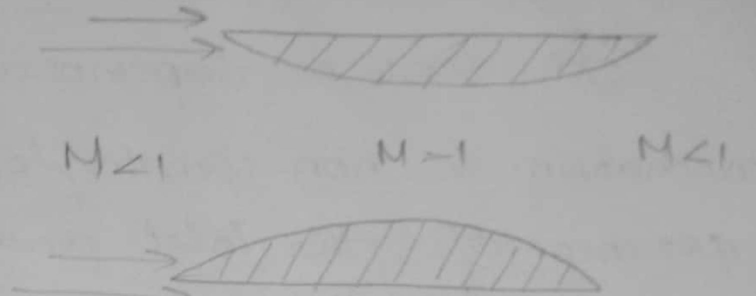
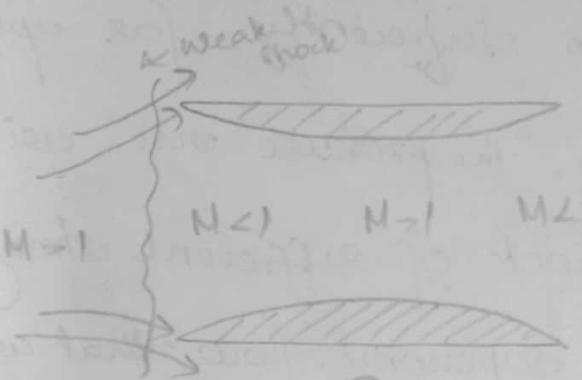
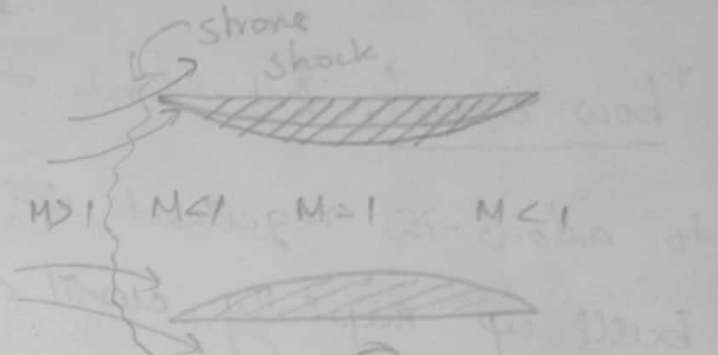
(a)



(b)



(c)



(d)

To isolate the inlet behaviour from that of the rest of the propulsion device, we assume that whatever is attached to the diffuser exit, is always capable of injesting the entire diffuser mass flow.

The fig (a) illustrates, low subsonic speed operations for, which the inlet is not chocked. In this case the airflow through the inlet and the upstream capture area is determined by conditions downstream of the inlet.

In fig (b) though the flight velocity is still subsonic the flow is assumed to be accelarated to sonic velocity and the inlet mass flow rate is limited by chocking condition. It can be seen from the figure that a spillage will occur in the inlet.

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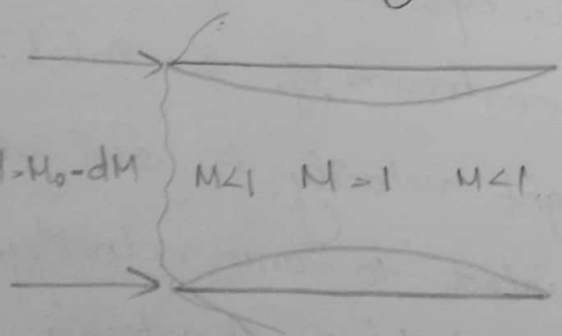
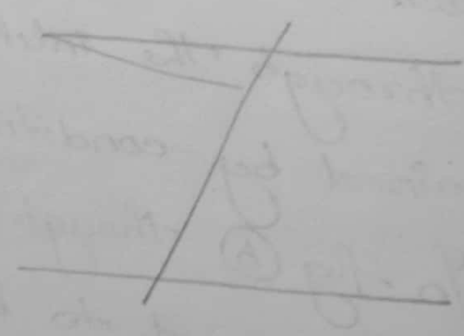
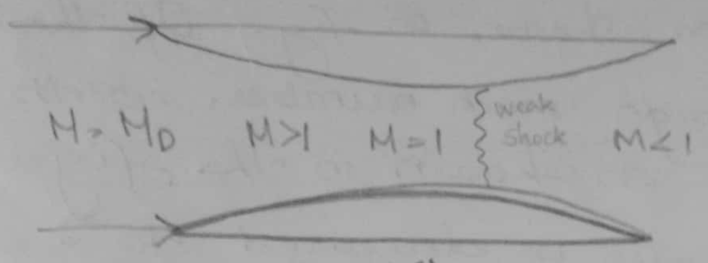
For sonic or supersonic flights speed the spillage mechanism is non-isentropic, ie, inorder to sense the presence of the inlet and flow around of the spilled air must be reduced to subsonic velocity upstream of the inlet. The mechanism for this deceleration is a detached "bow wave", that stands ~~safien~~ sufficiently far upstream to allow the required spillage. The pressure rise will built up rapidly until a shock of sufficient strength moves upstream against the supersonic flow that will allow the required spillage. Once the shock is established the flow entering the inlet is no longer isentropic

7/3/19 Hence when the ~~missile~~ designed mach number is reached as shown in figure (d) the reverse isentropic nozzle mass-flow cannot pass through the throat area. In this case the fluid suffers a stagnation pressure loss while by passing through the shock wave.



(e)





(f)

The spillage will continue unless the inlet can be overspeeded to a mach number (M_o) as shown in fig (e). At this mach number or just below it the inlet is capable of injesting the incident mass flow without spillage. The position of the shockwave will be at the lip of the inlet. A slight incement in the speed will cause the shockwave to enter into the converging section of the diffuser.

Since the shockwave cannot attain a stable position with in the converging section, it will move quickly to downstream and occupy the diverging section of the diffuser, its exact position is determined by downstream conditions as shown in fig (f). A slight decrease of the flight speed will require spillage and the shockwave will move outward through the inlet.

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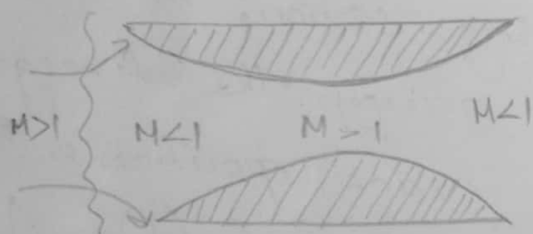
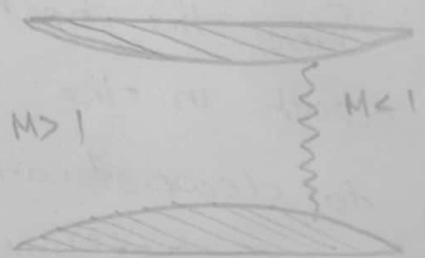
Case 2: Shock swallowing by area variation.

Overspeeding the inlet is one operation by which the supersonic inlet can be operating. If overspeeding is not a fesible, then it is possible to swallow the shockwave by varying the geometry at constant flight speed. This principle can be considered by a 1D analysis of the inlet as shown in fig. If the inlet is accelerated to the design mach number, with the starting shock present as shown in the figure, now if the area ratio is decreased to a value that can injest the entire inlet flow behind the shock then the shockwave will be swallowed to take up a position downstream of the throat. This variation will involve a momentary increase in the throat area, then achieving to an isentropic flow within the convergance, and a relatively strong shock occurs downstream. A complete isentropic flow can be achived by returning the area ratio to its original value.





A geometric variation such as shown in the figure will be difficult to achieve mechanically however the geometris the permit the axial motion of the a center plug between non-parallel walls can be used. With shockwave positioned external, a high static pressure within the converging section, causes considerable leakage then effectively increasing the throat area to permit shock swallowing, once this occurs the lower static pressure within the converging section

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decreases the flow losses.

* COMBUSTION CHAMBER. (Burner)

The Combustion chamber is a space where the chemical reaction between fuel and pressurised air occurs. The chemical reaction by which the chemical energy with in the fuel is converted into thermal energy is refered as chemical combustion. The efficiency of the combustion depends upon various operating parameters and operating conditions, eg; pressure ratio of a compressor, efficient mixing of fuel and air, proper ignition, ~~low subsonic~~ velocity of air (low subsonic)

⟹ Types of Combustion chamber

① Can Type

In this type the individual burners or cans are mounted around a circle with respect to engine axis. With each one, receiving air through its own cylindrical shroud. One of the main disadvantage is that this type of combustion chamber do not make use of the available space and which results in larger diameter engine. On the other hand this type of combustion chamber can be easily removed individually for maintanance

② Anular type

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This type of combustion chamber is essentially a single chamber made up of concentric cylinders, mounted coaxially

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with respect to ~~x axis~~ engine axis. The anular type of combustion chambers has a lesser Surface to volume ratio than the Can type combustion chamber, therefore the amount of coled air required is reduced. This arrangement make the complete utilisation of the available space and has a lesser pressure loss.

The disadvantage is that, the structural problems that arise due to larger diameter. The entire burner must be removed from the engine for inspection and repair.

(3) Can- Annular type.

This type of combustion chamber makes a good use of the available space and employes a number of replacable cylindrical inner liners, which recieves air through a common annular housing, which has a good control of the fuel and air flow pattern. This type of combustion chamber has an added advantage of greater structural stability and redused pressure losses, compared to the can type combustion chamber

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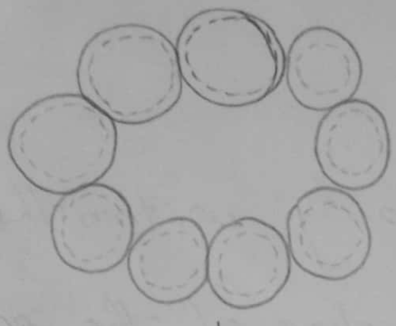
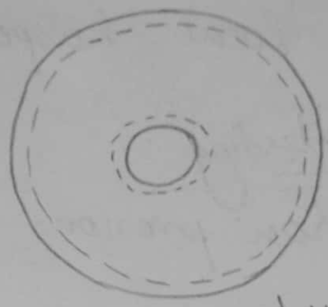
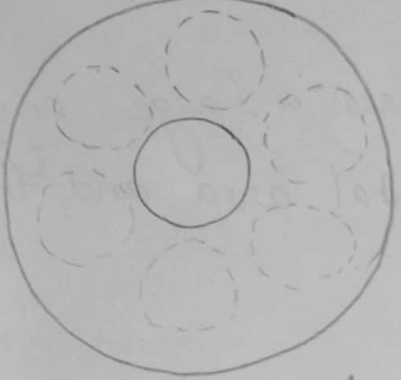


Can type



annular type



Can - annular type

* Combustion chamber Performance

The combustion chamber requires the following parameters

i) High combustion efficiency:-
This means that the fuel and air has to be
effectively mixed. and the combustion has to be complete

(ii) Stable operation.
This means the combustion process must be free from blow out at air flows from idle to maximum condition.

(iii) Low pressure losses:-
It is desirable to have as much as pressure as possible so as to accelerate the gases backward.

(iv) Uniform temperature distribution:-
This means that the temperature distribution throughout the combustion chamber must be uniform so as to

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avoid any local hotspot.

(v) Easy starting :-

Low pressure and high velocity within the combustion chamber makes the starting of the combustion process, a difficult task.

(vi) Size of the burner :-

A large con burner requires a large engine housing there by increasing the frontal area and the drag force.

(vii) Low Smoke burner

This relates with tracking of a military aircraft, it has a more chance of exposure with the enemy aircrafts.

(viii) Low Carbon formation

The carbon deposits can block the critical air passages and desrupt the air flow along the liner wall.

The main function of a gas turbine combustion chamber is to achieve efficient combustion with a uniform temperature distribution, during operation there will be some pressure drop in the combustion chamber, this loses should be minimise, the pressure loses are caused by two factors, i.e, pressure drop due to friction, and pressure drop due to accelaration accompanying heat addition

The size of the combustion chamber is primarily determined by the rate of heat release required. A parameter known as combustion intensity is defined to take into account of the heat released rate with respect to the combustion chamber placed. It is defined as the ratio of heat released rate to that of the product of combustion volume and pressure.

$$\boxed{\text{Combustion intensity} = \frac{\text{heat released rate}}{\text{Combustion volume} \times \text{Power}}}$$

The combustion efficiency is usually defined as the ratio of actual temperature rise to that of theoretical temperature rise.

$$\boxed{\text{Combustion efficiency} = \frac{\text{actual temperature rise}}{\text{theoretical temperature rise}}}$$

* Effect Of Operating Variables on burner performance

The various parameters which are considered as operating variables and which affects the burner performance are as follows;

(1) Pressure;
(2) Inlet air temperature
(3) Fuel air ratio
(4) Flow velocity.

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(a) Combustion efficiency:-

- As the pressure increases the combustion efficiency increases and it levels off to a relatively constant value.
- As the inlet air temperature is increased the combustion efficiency increases.
- If the fuel air ratio is increased the efficiency first increases then it levels off when the mixture reaches an ideal value and then the efficiency ~~def~~ decreases if the fuel air ~~b mi~~ ratio become too rich.
- Increasing the flow velocity beyond a certain value ~~reducing~~ reduces the combustion efficiency, this is because it reduces the available time for burning.

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(b) Stable operating range:

- As the pressure decreases the stable operating range becomes narrower until a point is attained below which burning will not take place.
- Increasing the temperature of the incoming charge usually increases the fuel air ratio, range, of stable operation.
- As the flow velocity increases the stable operating range becomes narrower until a critical velocity is attained above which combustion will not take place.

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(c) Temperature distribution

- Reducing the pressure below a point tends to affect the uniformity of temperature distribution.
If the fuel air ratio and the flow velocity is increased the exit temperature tends to become non-uniform, this is because more heat is released and the available time for mixing is reduced.

(d) Starting

- Starting is usually easier with higher temperature, higher pressure and low velocity. In addition there is an optimum fuel-air ratio.

(e) Carbon deposites

- The carbon deposites increases with increasing in temperature and pressure until a point where they begin to burn-off
- Increasing the fuel air ratio has a tendency to increase the carbon deposites.

imp
* Flame Stabilisation.

Flame stabilisation is of fundamentalthe importance ~~by~~ while calculating the performance of combustion chamber. In gas turbines the velocities at which the gases flow are much higher than the flame speed. The burning velocity should be equal to the flow velocity so as to ensure a stable flame front. Flame stabilisat

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is usually accomplished by causing some of the combustion products to recirculate and hence ignite the air fuel mixture continuously. The hot recirculating gases transfers the heat energy to relatively colder air fuel mixture and ignite it there by spreading the flame.

Following are the various methods of flame stabilisation;

1) Flame stabilisation by bluff body

This method is used to stabilise the flame in high velocity flow in various systems. The shapes generally utilised are cylindrical rods, rectangular disc, baffles, cone or V-gutter type which produces a wake, a low velocity recirculated flow in which the combustion can be initiated and maintained

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2) Flame stabilisation using pilot flame:-

A pilot flame (hot inert gas) is held adjacent to the relatively cold reactant mixture which is in the form of high velocity jet. Heat and mass are transfered across the boundary of the stream by the process of diffusion and mixing. The blow off occurs if the flow rate is greater than the reaction rate of the mixture.

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(3) Flame stabilisation using Swirler.

A swirler has a number of curved swirler vane which promotes the formation of recirculation of zone which is utilised for stabilisation. The swirl angle determines the size and strength of the recirculation zone.

(4) Flame stabilisation by counter flow

Here the direction of the fuel eject is made opposite to that of air fuel, this method reduces the flow velocity compared to the burning velocity and there by ensuring good mixing.

* EXHAUST NOZZLE:

(1) Equillibrium and frozen flow in angle:

The thrust produced by an engine depends upon the combustion and temperature of the combustion ~~chamber~~ products. The products of combustion in a well designed combustion chamber will be nearly in equilibrium, i.e at typical combustion chamber, temperatures and pressures the time required for mixing of fuel and air and the subsequent chemical reaction is generally small in comparison to the average residence time of the fuel and air within the combustion chamber. The combustion temperature are so high that the products of combustion are dissociated. The Equilibrium composition of the combustion products is dependent on the pressure and temperature and also at combustion

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Chamber operating condition, these products may include large quantities of dissociated materials.

In the exhaust nozzle the dissociated components will tend to recombine because of large drop in temperature. For the gases to maintain equilibrium composition as it expands, it is necessary that the recombination process occurs in a time that is short relative to the expansion time. Since the expansion process is very rapid this condition is not satisfied. If the reaction rates are low, no recombination occurs at all and the gases are said to be "frozen" at constant composition. For a given equilibrium composition, the final kinetic energy before expansion and at a given expansion ratio is higher for equilibrium flow than the frozen flow.

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(ii) 2 phase flow in nozzle

In some engines the gaseous working fluid contains small liquid droplets or solid particles that must be accelarated by the gas. The liquid droplets giving heat to the gas during expanasion in a nozzle. If the particles are small they'll have almost the same velocity as the gas and will be in thermal equilibrium with the ~~no~~ ~~the~~ nozzle gas flow. As the gases giveup the kinetic energy to accelarate the particles, they gai

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thermal energy from the particles. The larger particles do not move as fast as the gas and do not give heat to the gas as readily as the small particles. The larger particles have smaller momentum and they reach the nozzle exit at a higher temperature than the smaller particles, thus giving up less thermal energy.

* **Nozzle**

The purpose of the exhaust nozzle is to collect and straighten the ~~exhaust~~ gas flow and to increase the velocity of the gas before discharge from the nozzle for large values of specific thrust, the kinetic energy of the exhaust must be high, which requires a high velocity exhaust. The expansion process is controlled by the pressure ratio (or expansion ratio) across the nozzle. The maximum thrust for a given for engine is obtained when exit pressure (P_e) is equal to the ambient pressure $P_e = P_a$.

The function of an exhaust nozzle are as follows

i) accelarate the floco to a high velocity with minimum total pressure loss.

ii) Match the exit pressure and atmospheric pressure as closely as possible.

iii) Permit after burner operation without affecting the main engine operation, this function requires a variable area nozzle.

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iv) allow cooling of nozzle wall.

(v) Mix the core and bypass stream of turbofan engines if neccenary.

(vi) Allow for thrust vectoring if desired.

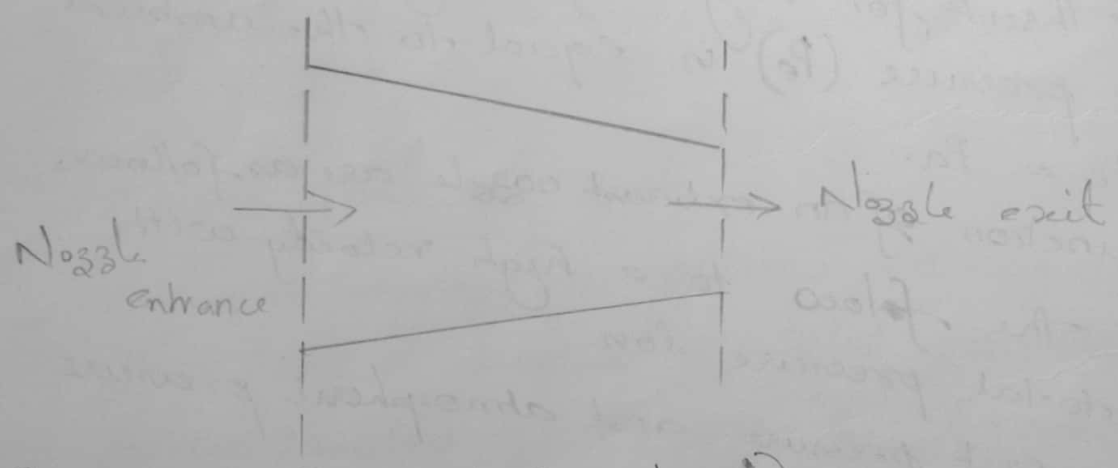
vii) Reduce the jet noice and infrared variation if ~~sea~~ desired.

viii) Allow for thrust reversal.

15/03/19

* Convergent Nozzle.

The convergent nozzle is a simple convergent duct as shown in the figure. When the nozzle pressure ratio is lesser (less than 4) the convergent nozzle is used. The convergent nozzle is generally used in subsonic aircraft.



Convergent - Divergent (CD Nozzle).

The C-D nozzle has a convergent duct and a divergent duct, Section of the nozzle when the cross sectional area is the minimum is refered as throat of the nozzle This type of nozzle utilises variable geometry and other aerodynamic features, The C-D nozzle is used if the nozzle pressure ratio is

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high [greater than 6] - High performance engines in supersonic aircrafts generally uses CD nozzle, if the engine has an after burner, the nozzle throat is made in such a way that the operating conditions of the engine. Upstream of the after burner remains unchanged. The exit area ~~must~~ must be varied to match the flow conditions so as to produce maximum available thrust.

* Real Flow in Nozzle.

- The real nozzle flow deviate from the isentropic flow by the following aspects

(i) Non- adiabic effect:-
- This is due to the cooling of the walls so as to keep the strength of the materials which results in non isentropic condition.

(ii) Viscous effects:-
The viscous effects and the effect of boundary layer interaction with the shockwaves can result in flow separation a flow losses.

(iii) The errosion of the walls which can be critical if the errosion widens the throat ~~corr~~ crossectional area

(iv) The real flow has energy losses which is not available for conversion into kinetic energy.

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* Ejector Nozzle.

The ejector refers to the pumping action of very hot, high speed engine exhaust, ejecting the surrounding air flow which together with the internal geometry of the nozzle, controls the expansion of engine-gas exhaust. At Subsonic speeds the air flow constricts the exhaust to a convergent shape. When after burning is selected an the aircraft speeds up the two nozzles dilate, which allows the exhaust to form a convergent divergent shape. The ejector nozzle is also able to utilise the air which has been ingested by the intake but which is not required for the engine. The efficient use of this air in the nozzle is a prime requirement for the aircraft which has to cruise ~~effeca effecti~~ efficiently at high supersonic speeds for long time.

* After Burner Nozzle or Variable Area Nozzle

The after burners on compact aircrafts requires a larger nozzle so as to prevent the adverse ~~effect~~ operation of the engine. The variable area nozzle consists of a series of moving overlapping ~~met~~ petals of with near circular cross section. If the aircraft flys at supersonic speeds then the after burner nozzle is followed by a seperate divergent nozzle.

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* Losses in Nozzle.

The majour losses in a nozzle are as listed below;

(i) The divergance of the flow in nozzle exit section causes a loss which varies as a function of the divergance angle.

(ii) Lesses the flow velocity in the boundary layer it can reduce the effective ~~exit~~ exhaust velocity

(iii) Unsteady combustion and oscillating flow can result in flow losses,

(iv) Chemical reaction in nozzle flow if any can cause losses by changing the gas properties and temperature.

(v) The performance will be reduced during to transient pressure operations, ie, during starting, stoping, etc..

(vi) For uncooled materials the throat region can be eroded, this result in reduction of the thrust can chamber pressure.

(vii) Non-uniform gas composition can reduce the performance due to incomplete mixing or combustion.

(viii) Operation at non-optimum nozzle expansion ratio can reduce the thrust, there will be no losses if the nozzle operates at opimum expansion ratio.

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* ~~Thrust Aug~~ Reversal.

The need for thrust reversing and thrust vectoring is normally determined by the required aircraft and engine system performance. The thrust reversers are used on commercial transport aircrafts to suppliment the breakes. In flights the thrust reversal enhances the compact effectiveness of fighter aircraft.

The two basic types of thrust reversers are;

i) Cascade blocker type &

(ii) clambshell type.

In cascade blocker type the primary nozzle exit is blocked-off and the cascades are open in the upstream portion of the nozzle duct to ~~revere~~ revesse the flow.

In the clambshell type the exhaust jet is splitted and reversed by the clampshell. Since both the types usually provide a change in the effective throat area during diploiment, most of the reversers are designed such that the effective nozzle throat area increases during the transition period, thus preventing the compressor ~~stock~~ stall

The vectoring nozzles have been utilized during ~~vtb~~ vertical take-off and landing [VTOL] and are proposed for future fighter aircrafts to improve the maneuvering and

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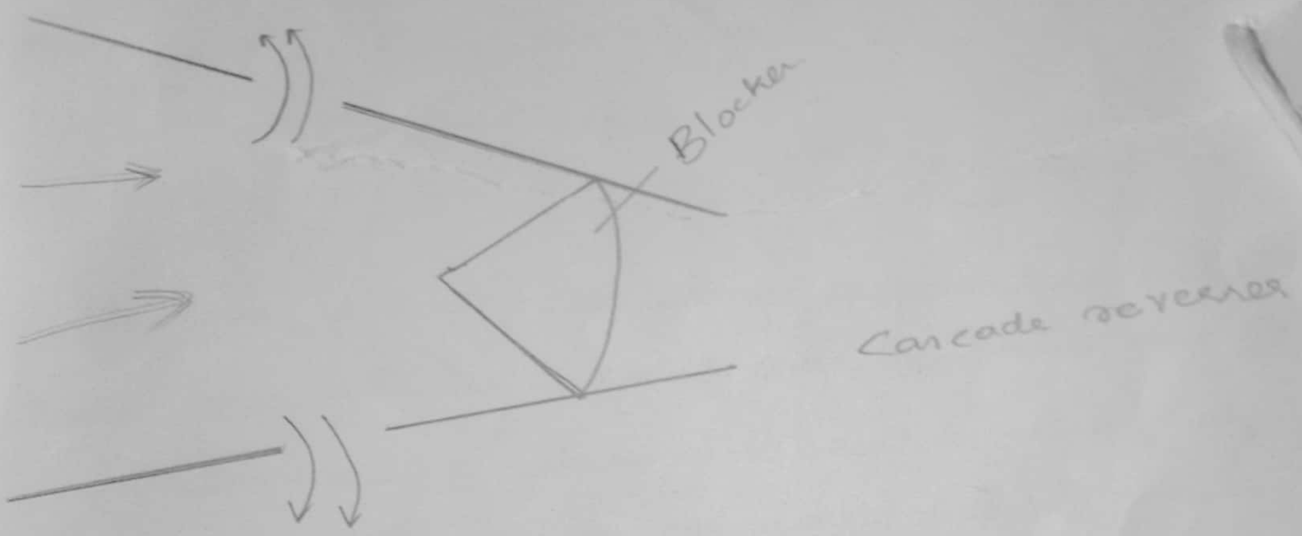
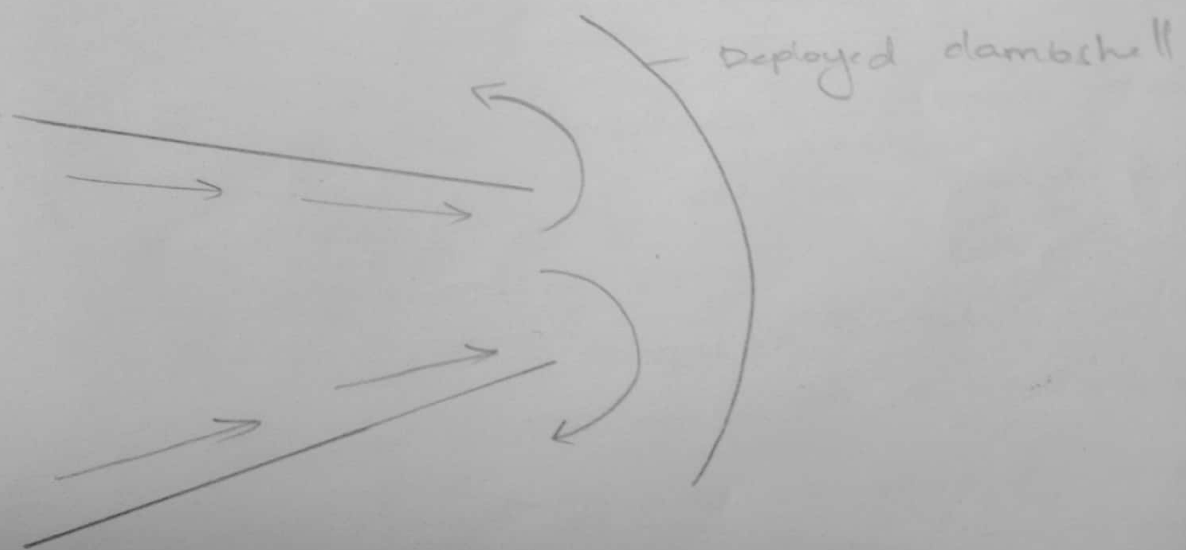
augment the lift during combat. The thrust vectoring at augmented power setting is being developed for utilising also in fighter jets. The cooling of the nozzle walls in contact with the hot gases is difficult in this case and demands increased amount of nozzle cooling airflow.





Clamshell reverser

It can be seen from the figure that, in the clamshell type thrust reverse, the exhaust jet is splitted into two after the gases exit from the nozzle. The clampshell can be deployed whenever the thrust reverser is required. In the cascade blocker type

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It can be seen from the figure that the exhaust gas are utilized for thrust reversal before it exits from the nozzle. This is done by positioning a blocker at the nozzle as shown in the figure.

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