Inlets

The inlet interchanges the organized kinetic and random thermal energies of the gas in an essentially adiabatic process. The perfect (no-loss) inlet would thus correspond to an isentropic process. The primary purpose of the inlet is to bring the air required by the engine from free stream conditions to the conditions required at the entrance of the fan or compressor with minimum total pressure loss. The fan or compressor works best with a uniform flow of air at a Mach number of about 0.5

The Requirements of the inlets:

- high total pressure ratio πd,
- controllable flow matching of requirements,
- good uniformity of flow,
- Low installation drag,
- Good starting and stability,
- Low signatures (acoustic, radar, etc.),
- Minimum weight and cost while meeting life and reliability goals.

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

- Inlet total pressure ratio and drag at cruise
- Engine location on wing or fuselage (avoidance of foreign-object damage,
- inlet flow up wash and downwash, exhaust gas re ingestion, ground clearance)
- Aircraft attitude envelope (angle of attack, yaw angle, cross-wind takeoff)
- Inlet total pressure ratio and distortion levels required for engine operation
- Engine-out wind milling airflow and drag (nacelle and engine)
- Integration of diffuser and fan flow path contour
- Integration of external nacelle contour with thrust reverser and accessories
- Flow field interaction between nacelle, pylon, and wing
- Noise suppression requirements.

Design considerations:

- The airflow entering the compressor or fan must have low Mach number, in the range 0.4 to 0.7, Part of this deceleration occurs upstream of the inlet entrance plane.
- The inlet must be designed to prevent boundary layer separation, even when the axis of the intake is not perfectly aligned with the streamline direction far upstream of the inlet.
- It is important that the stagnation pressure loss in the inlet be small.
- It is even more important that the flow velocity and direction leaving the inlet be uniform, since distortions in the velocity profile at the compressor inlet can severely upset the compressor aerodynamics and may lead to failure of the blades due to vibrations.
- Design of inlets that must operate efficiently in both supersonic and subsonic flight poses special problems;

Subsonic Inlets:

Internal flow and Stall in subsonic inlet sand Boundary layer Separation

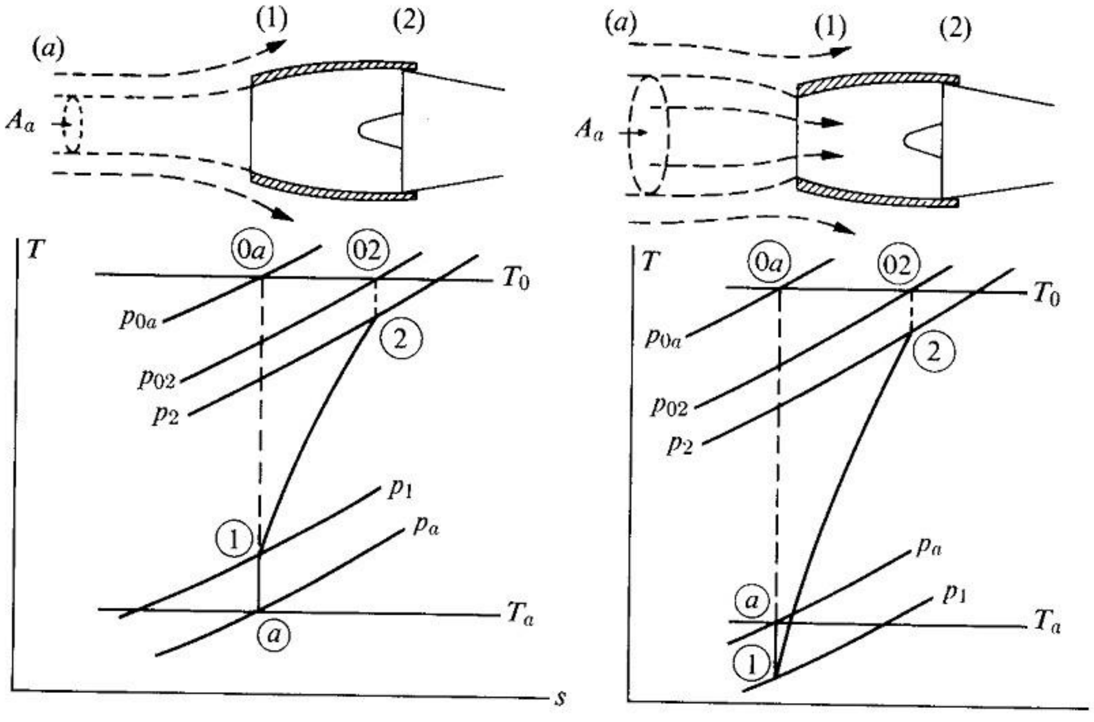
Depending on the flight speed and the mass flow demanded by the engine, the inlet may have to operate with a wide range of incident stream conditions. The Figure shows the streamline patterns for two typical subsonic conditions and the corresponding thermodynamic path of an "average" fluid particle.

During **level *cruise*** the streamline pattern may include some ***deceleration*** of the entering fluid External to the inlet plane and hence low mass flow rate[Fig. a]. During low-speed high-thrust operation (e.g., during takeoff and climb), the same engine will demand more mass flow and the streamline pattern may resemble Fig. b, which illustrates external acceleration of the stream near the inlet. .

For given air velocities external acceleration raises the inlet velocity and lowers the inlet pressure, thereby increasing the internal pressure rise across the diffuser. If this pressure increase is too large, the diffuser may stall because of boundary layer separation; stalling usually reduces the stagnation pressure of the stream as a whole.

Conversely, external deceleration requires less internal pressure rise and hence a less severe loading of the boundary layer.

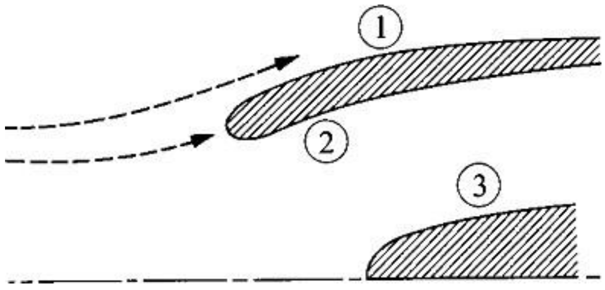
Therefore the inlet area is often chosen so as to minimize external acceleration during takeoff, with the result that external deceleration occurs during level-cruise operation. Under these conditions the "upstream capture area" Aa is less than the inlet area A1, and some flow is "spilled over" the inlet, accelerating as it passes over the outer surface.



(a) High speed or low mass flow (b) Low speed or high mass flow

In the actual engine inlet, separation can take place in any of the three zones shown in Fig. 2. Separation of the external flow in **zone 1** may result from local high velocities and subsequent deceleration over the outer surface. Separation on the internal surfaces may take place in **either zone 2 or zone 3**, depending on the geometry of the duct and the operating conditions.

Zone 3 may be the scene of quite large adverse pressure gradients since the f low accelerates around the nose of the center body, then decelerates as the curvature decreases



Major features of external flow near a subsonic inlet

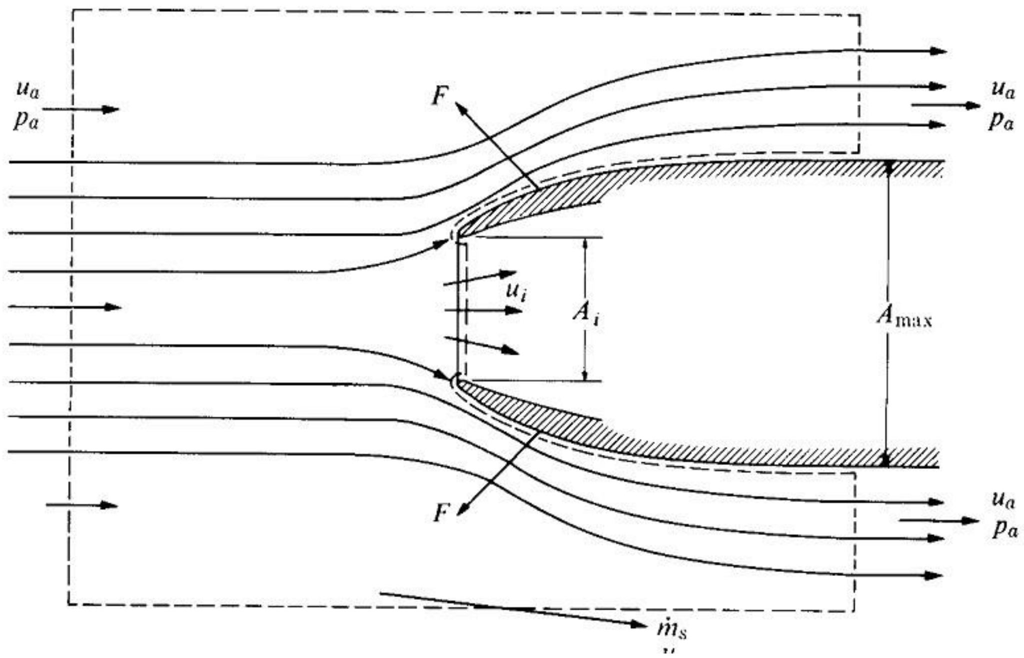
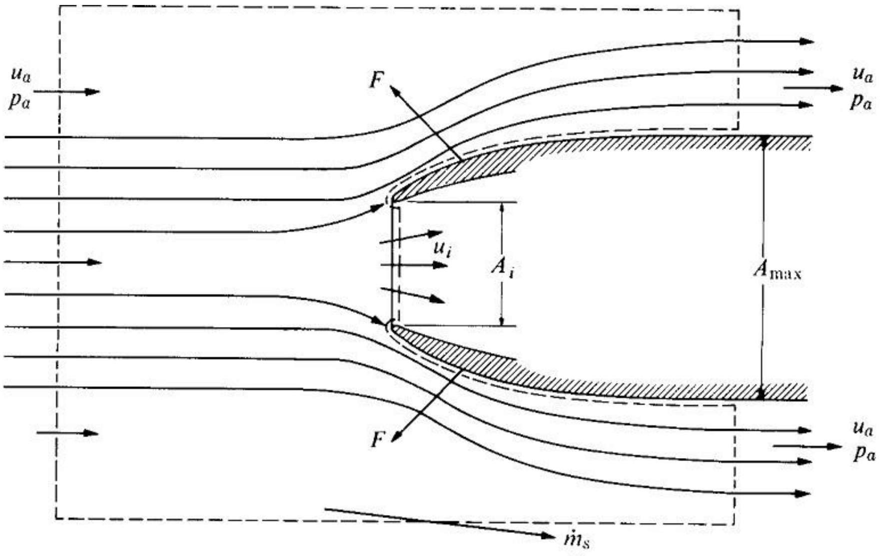


Figure shows a typical streamline pattern for large external deceleration. In flowing over the lip of the inlet, the external flow is accelerated to high velocity, much as the flow is accelerated over the suction surface of an airfoil. This high velocity and the accompanying low pressure can adversely affect the boundary layer flow in two ways:

For entirely subsonic flow, the low-pressure region must be followed by a region of rising pressure in which the boundary layer may separate. Hence one might expect a limiting low pressure $\mathbf{P_{min}}$ or, equivalently, a maximum local velocity $\mathbf{U_{max}}$**,** beyond which boundary layer separation can be expected downstream.

For higher flight velocities (or higher local accelerations), partially supersonic flow can occur. Local supersonic regions usually end abruptly in a shock, and the shock-wall intersection may cause boundary layer separation. One might expect a limiting local Mach number that should not be exceeded.

Relation for minimum area ratio (A_{max}/A_i) in terms of external deceleration (U_i/U_a)



Let A_{max} = Maximum area cross section of the inlet

A_i = Minimum area cross section of inlet

u_a = upstream or flight velocity

u_i = velocity inside inlet.

The net momentum flux out of the control volume

$$F = \dot{m}_i u_i + \dot{m}_s u_a - \dot{m}_e u_a \quad \text{— (1)}$$

$$F = \rho u_i^2 A_i + \dot{m}_s u_a - \rho u_a^2 A_{max} \quad \text{— (2)}$$

From continuity the side flow rate

$$\dot{m}_s = \rho u_a A_{max} - \rho u_i A_i \quad \text{— (3)}$$

$\because \dot{m} = \frac{m}{t} = \rho A \frac{l}{t}$, $\boxed{\dot{m} = \rho A V}$

3 in 2 $\Longrightarrow$

$$F = \rho u_i^2 A_i + \rho u_a^2 A_{max} - \rho u_i u_a A_i - \rho u_a^2 A_{max}$$

$$\rho u_i^2 A_i - \rho u_i u_a A_i$$

$$F_s = \rho A_i (u_i^2 - u_i u_a) \quad \text{— (4)}$$

Let P_s = Pressure on surface.

P_i = Pressure at inlet

P_a = Pressure at exit

The total Force

$$F_s = P_a A_{max} - P_i A_i - \int_{A_i}^{A_{max}} P_s \, dA \quad \text{— (5)}$$

$$F = P_a A_{max} - P_i A_i - P_s A_{max} + P_s A_i$$

$$F = (P_a - P_s) A_{max} - A_i (P_i - P_s)$$

$$F = (P_a - P_s) A_{max} - A_1 (P_i + P_a - P_a - P_s)$$

$$F = (P_a - P_s) A_{max} - (P_a - P_s) A_i - A_i (P_i - P_a)$$

$$F = \int_{A_1}^{A_{max}} (P_a - P_s)\, dA - A_i (P_i - P_a) \quad \text{—— (6)}$$

equating 4 and 6

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

$$\int_{A_i}^{A_{max}} (P_a - P_s)\, dA = \rho A_i (u_i^2 - u_i u_a) + A_i (P_i - P_a) \quad \text{—— (7)}$$

$$\Delta T_i = \rho A_i (u_i^2 - u_i u_a) + A_i (P_i - P_a) \quad \text{—— (8)}$$

where $\Delta T_i = \int_{A_i}^{A_{max}} (P_a - P_s)\, dA$

= Thrust component

Applying Bernoulli eqn to the external deceleration of internal flow

$$P_i - P_a = \rho \left(\frac{u_a^2 - u_i^2}{2} \right) \quad \text{—— (9)}$$

equation 9 in 8 $\Rightarrow$

$$\Delta T_i = \rho A_1 (u_i^2 - u_i u_a) + \rho A_i \left(\frac{u_a^2}{2} - \frac{u_i^2}{2} \right)$$

$$\Delta T_1 = \rho A_1 (u_i^2 - u_i u_a) + \frac{\rho A_1 u_a^2}{2} \left(1 - \frac{u_1^2}{u_a^2} \right)$$

$$\frac{\Delta T_i}{\frac{1}{2} \rho A_1 u_a^2} = \left(1 - \frac{u_i}{u_a} \right)^2 \quad \text{—— (10)}$$

The average pressure difference on the outer surface on the nacelle

$$\overline{P_a - P} = \frac{\Delta T_i}{A_{max} - A_i} = s (P_a - P_{min})$$

where s is a factor between 0 and 1

$$\Delta T_i = s (P_a - P_{min}) (A_{max} - A_1) \quad \text{—— (11)}$$

Equation 11 in equation 10

$$\frac{s (P_a - P_{min}) (A_{max} - A_i)}{\frac{1}{2} \rho A_i u_a^2} = \left(1 - \frac{u_i}{u_a} \right)^2 \quad \text{—— (12)}$$

The pressure coeffecient is given by

$$C_p = \frac{P_a - P_{min}}{\frac{1}{2} \rho_a u_{max}^2} \quad \text{—— (13)}$$

Equation 13 in equation 12 →

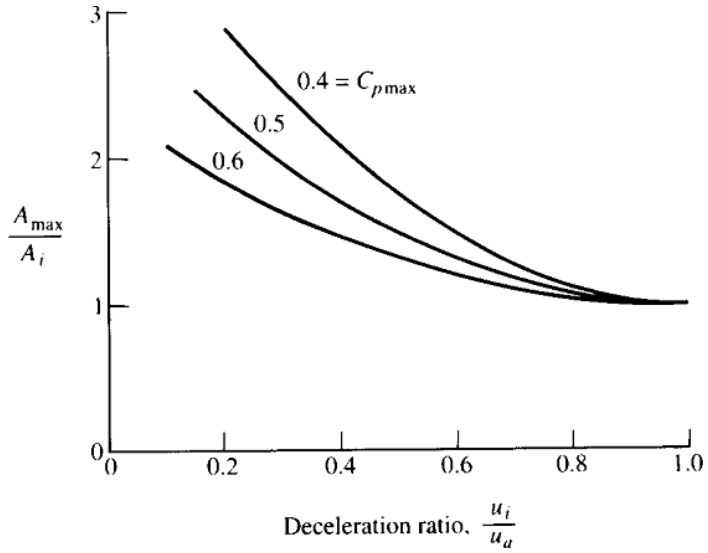
$$S\,C_p\left(\frac{A_{max}}{A_i}-1\right) = \left(1-\frac{u_i}{u_a}\right)^2 \frac{u_a^2}{u_{max}^2}$$

$$\frac{A_{max}}{A_i}-1 = \frac{\left(1-\frac{u_i}{u_a}\right)^2 \left(\frac{u_a}{u_{max}}\right)^2}{S\,C_p}$$

$$\frac{A_{max}}{A_i}-1 = \frac{\left(1-\frac{u_i}{u_a}\right)^2}{S\,C_p\left(\frac{u_{max}}{u_a}\right)^2}$$

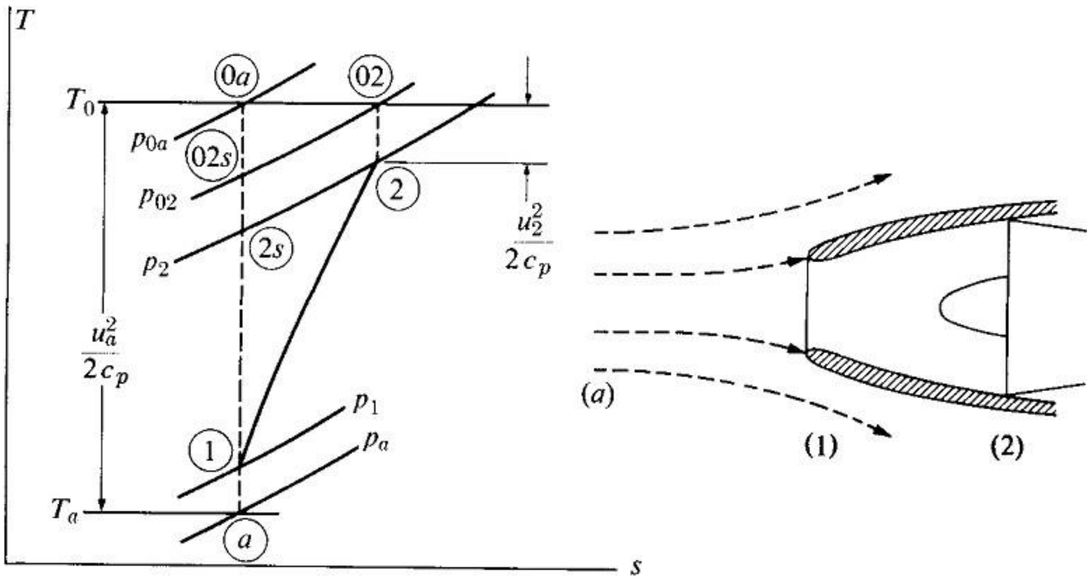
$$\boxed{\frac{A_{max}}{A_i} = 1 + \frac{\left(1-\frac{u_i}{u_a}\right)^2}{S\,C_p\left(\frac{u_{max}}{u_a}\right)^2}}$$

The main point here is that the larger the external deceleration (i.e., the smaller the value of Uj/Ua), the larger must be the size of the nacelle if one is to prevent excessive drag. Even in the absence of separation, the larger the nacelle, the larger the aerodynamic drag on it. But if the external deceleration is modest (e.g., U/ISa > 0.8), its effect on minimum nacelle size is quite small.



it shows that the performance of an inlet depends on the pressure gradient on both internal and external surfaces. The external pressure rise is fixed by the external compression and the ratio, A_{max}/Ai of maximum area to inlet area. The internal pressure rise depends on the reduction of velocity between entry to the inlet diffuser and entry to the compressor (or burner, for a ramjet). Nacelle size required for low drag can be quite strongly dependent on the degree of external deceleration

Difusser Effeniency



Isentropic efficiency: we can define the isentropic efficiency of a diffuser in this form:

$$\eta_d = \frac{h_{o2s} - h_a}{h_{oa} - h_a}$$

$$= \frac{T_{o2s} - T_a}{T_{oa} - T_a}$$

$$= \frac{\frac{T_{o2s}}{T_a} - 1}{\frac{T_{oa}}{T_a} - 1} \quad \text{—— } (1)$$

But $$\frac{T_{o2s}}{T_a} = \left(\frac{P_{o2}}{P_a}\right)^{\frac{\gamma-1}{\gamma}} \quad \text{— } (2)$$

$$\frac{T_{o2}}{T_a} = 1 + \frac{\gamma-1}{2} M^2 \quad \text{— } (3)$$

3 in 1 $\Longrightarrow$

$$\eta_d = \frac{\left(\frac{P_{o2}}{P_a}\right)^{\frac{\gamma-1}{\gamma}} - 1}{\left(\frac{\gamma-1}{2}\right) M^2}.$$

Typical

Typical

Also stagnation Pressure Ratio is given by

$$r_d = \frac{P_{o2}}{P_{oa}}$$

$$\frac{P_{o2}}{P_a} = \frac{P_{o2}}{P_{oa}} \times \frac{P_{oa}}{P_a} \quad \text{—— (5)}$$

$$\frac{P_{oa}}{P_a} = \left(\frac{T_{oa}}{T_a}\right)^{\frac{\gamma}{\gamma-1}} = \left(1 + \frac{\gamma-1}{2} m^2\right)^{\frac{\gamma}{\gamma-1}} \quad \text{—— (6)}$$

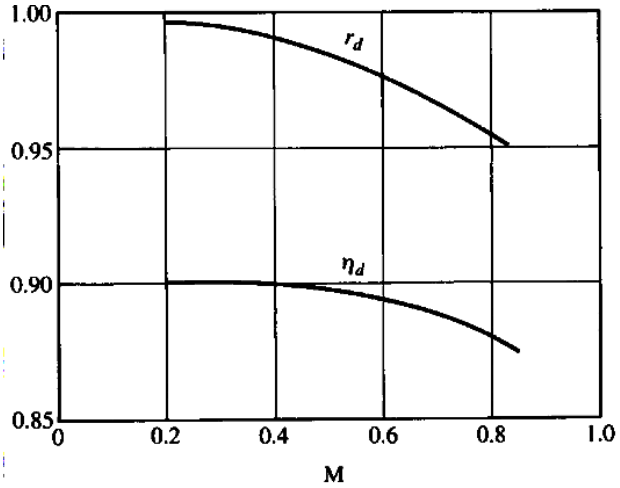
6 in 5 $\Rightarrow$

$$\frac{P_{o2}}{P_a} = r_d \times \left(1 + \frac{\gamma-1}{2} m^2\right)^{\frac{\gamma}{\gamma-1}} \quad \text{—— (7)}$$

7 in 3 $\Rightarrow$

$$\eta_d = \frac{\left(1 + \frac{\gamma-1}{2} m^2\right) r_d^{\frac{\gamma-1}{\gamma}} - 1}{\left(\frac{\gamma-1}{2}\right) m^2}$$

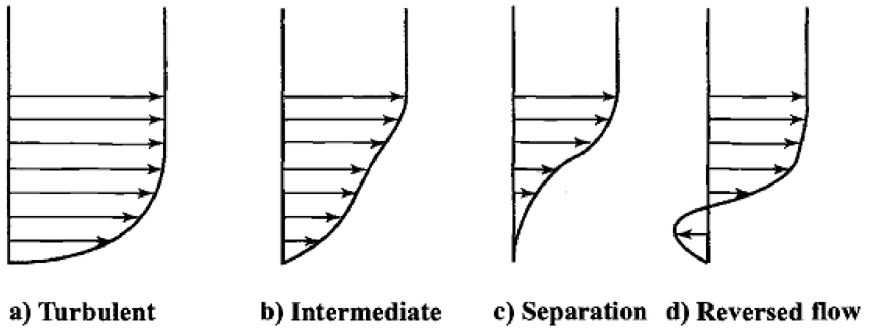
<u>subsonic diffuser performance is shown as</u>



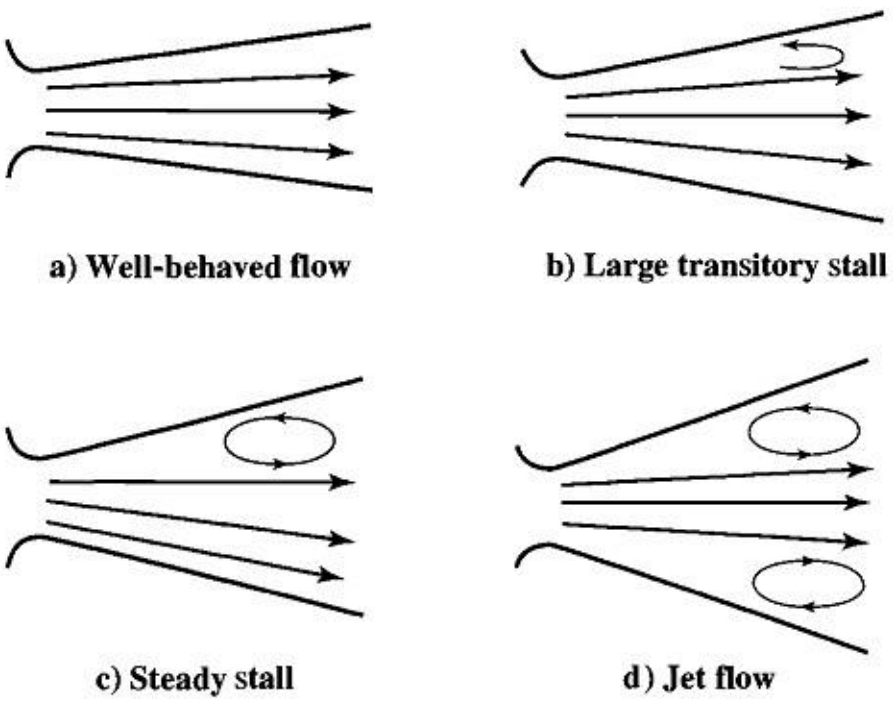
Typical subsonic diffuser performance; $\gamma = 1.4$.

Difusser:

The flow within the inlet is required to undergo diffusion in a divergent duct. This reduction in flow velocity creates an increase in static pressure that interacts with the boundary layer. If the pressure rise due to diffusion occurs more rapidly than turbulent mixing can reenergize the boundary layer, the boundary layer will assume the configurations shown in Fig. .

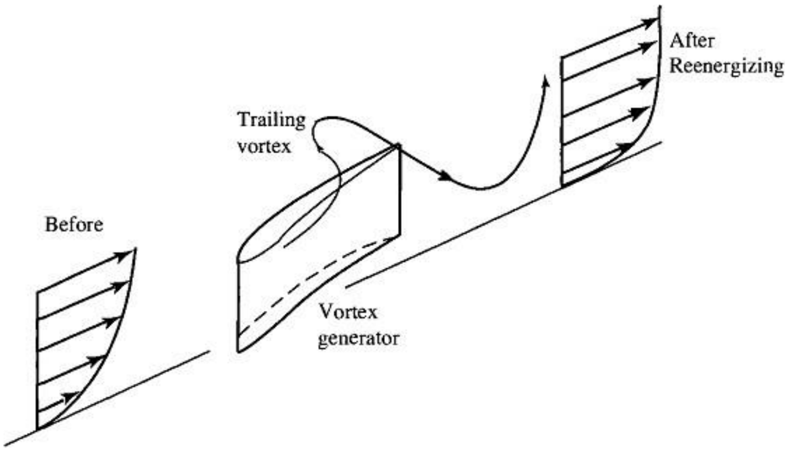


The rate of area increase in a diffuser has a direct effect on the behavior of flow in the diffuser, as shown in Fig.



. If the rate of area increase is greater than that needed to keep the boundary layer energized and attached, the flow may be characterized by unsteady zones of stall. The turbulent mixing is no longer able to overcome the pressure forces at all points in the flow, and local separation occurs at some points. The total pressure decreases markedly due to the irreversible mixing of a fairly large portion of low-velocity fluid with the main flow. If the diffuser walls diverge rapidly, the flow will separate completely and behave much as a jet, as shown in Fig. d. The rate of area increase without stall for a diffuser depends on the characteristics of the flow at the entrance and on the length of the divergent section

<u>Use Of Vortex Generators As A Mechanical Mixing Device To Supplement The Turbulent Mixing</u>



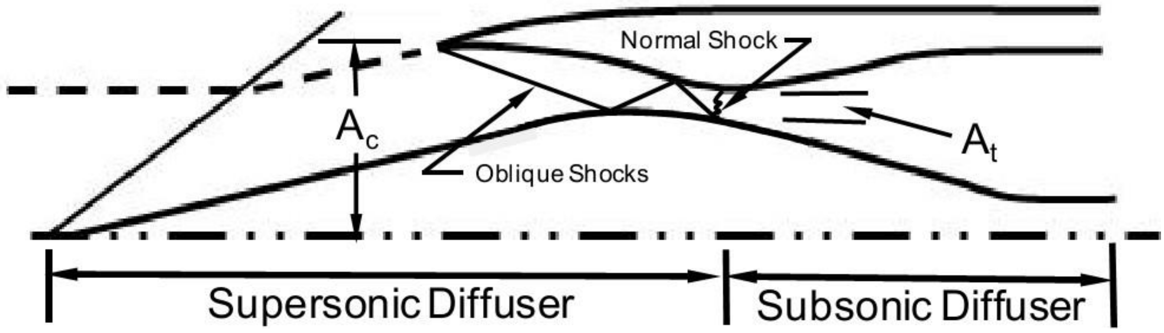
In the presence of an adverse pressure gradient (static pressure increasing in the direction of flow), boundary layers tend to separate when the boundary layer is not reenergized rapidly enough by turbulent mixing. Taylor proposed the use of vortex generators as a mechanical mixing device to supplement the turbulent mixing. If vortices are generated by vortex generators in pairs, regions of inflow and outflow exist. These carry high-energy air into the boundary layer and low-energy air out. Figure shows how vortex generators reenergize a boundary layer.

By using vortex generators together with a short, wide-angle diffuser, it may be possible to have a lower total pressure loss than with a long diffuser without vortex generators. Here, the reduced skin friction losses associated with flow separation are traded against vortex losses. The use of shorter diffusers may reduce weight and facilitate engine installation.

Supersonic Inlets

The supersonic inlet is required to provide the proper quantity and uniformity of air to the engine over a wider range of flight conditions than the subsonic inlet is. In addition, the nature of supersonic flow makes this inlet more difficult to design and integrate into the airframe. In supersonic flight, the flow is decelerated by shock waves that can produce a total pressure loss much greater than, and in addition to, the boundary-layer losses.

Working Principle of Supersonic Inlets:



A supersonic inlet is made up of two distinct parts. First the flow is compressed supersonically from the velocity of the flight vehicle or, in other words, the free stream Mach number. This is done by reducing the flow area as the flow proceeds downstream. In this region the flow velocity is reduced through a series of compression waves and/or oblique shocks. Flow velocity is reduced to a minimum speed at the duct minimum area, called the throat of the inlet, where the flow approaches sonic velocity or a Mach number of one. At this point the flow Mach number will be reduced from supersonic, above one, to subsonic, below one, through a normal shock. This begins the second part of the inlet, the subsonic diffuser. In this region the velocity is reduced as the flow area is increased. The result of this process is conditioned air, smooth, subsonic air at high pressure, which is then delivered to the engine.

Supersonic Inlet Types:

Internal compression inlet: The internal compression inlet shown in Figure achieves compression through a series of internal oblique shock waves followed by a terminal normal shock positioned downstream of the throat (its stable location). This type of inlet requires variable throat area to allow the inlet to swallow the normal shock (during starting). Fast reaction bypass doors are also required downstream of the throat to permit proper positioning of the normal shock under varying flight and engine conditions.

External compression inlet. The compression of the external compression inlet is achieved through either one or a series of oblique shocks followed by a normal shock, or simply through one normal shock

Mixed compression inlet. : At flight Mach numbers above 2.5, the mixed compression inlet is used to obtain an acceptable total pressure ratio (by utilizing the required number of oblique shocks) while obtaining acceptable cowl drag. The mixed compression inlet is more complex, heavier, and costlier than the external compression inlet. The typical mixed compression inlet achieves compression through the external oblique shocks, the internal reflected oblique shocks, and the terminal normal shock. The ideal location of the normal shock is just downstream of the inlet throat, to minimize total pressure loss while maintaining a stable operating location of this shock. Similar to the internal compression inlet, the mixed compression inlet requires both fast-reacting bypass doors (to maintain the normal shock in a stable location) and variable throat area

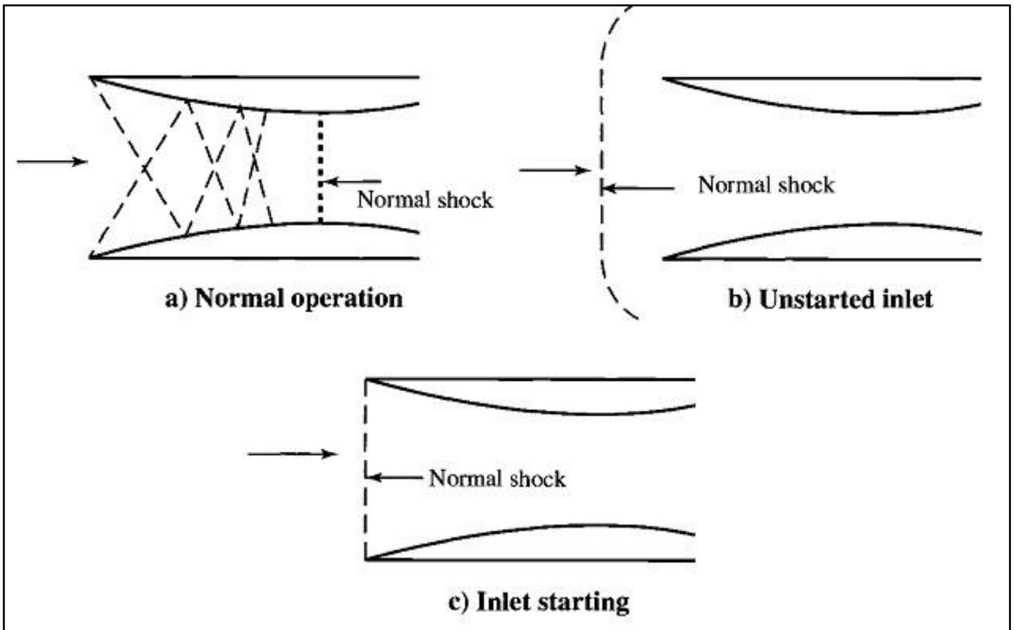


Fig. 10.24 Internal compression inlet.

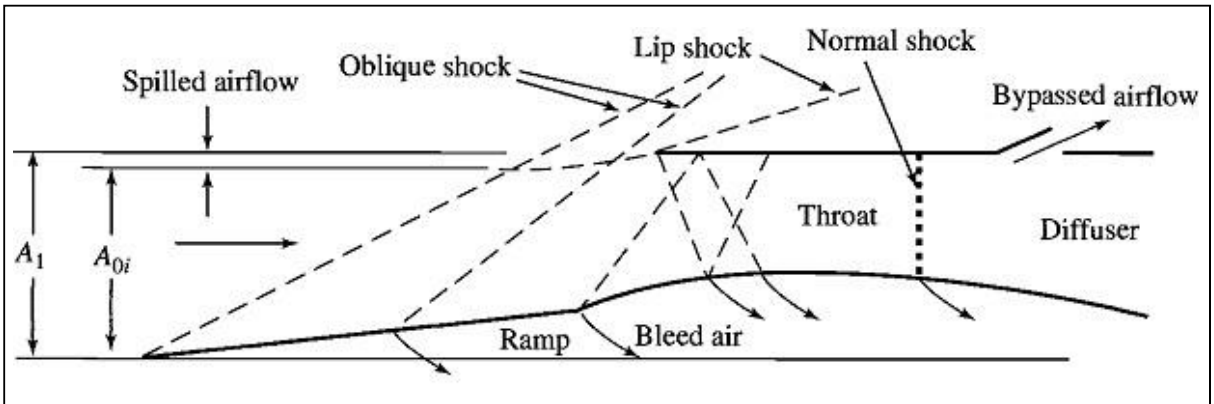


Fig. 10.30 Two-dimensional mixed compression inlet.

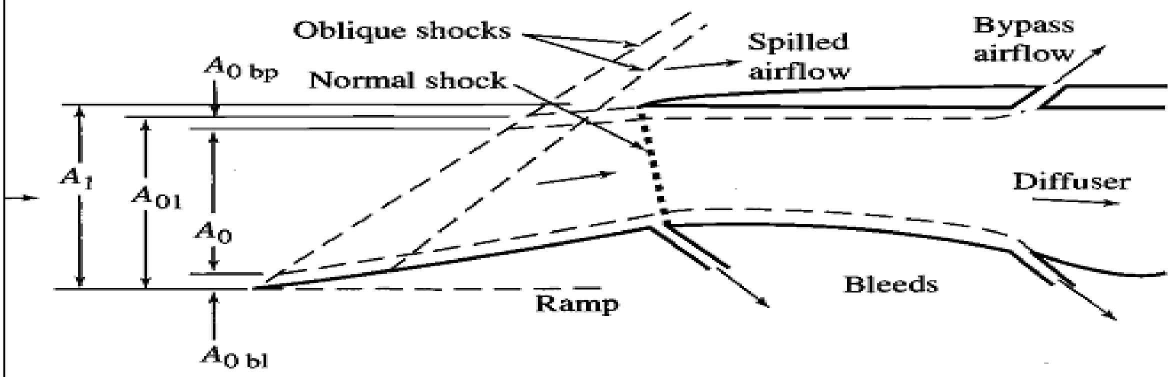
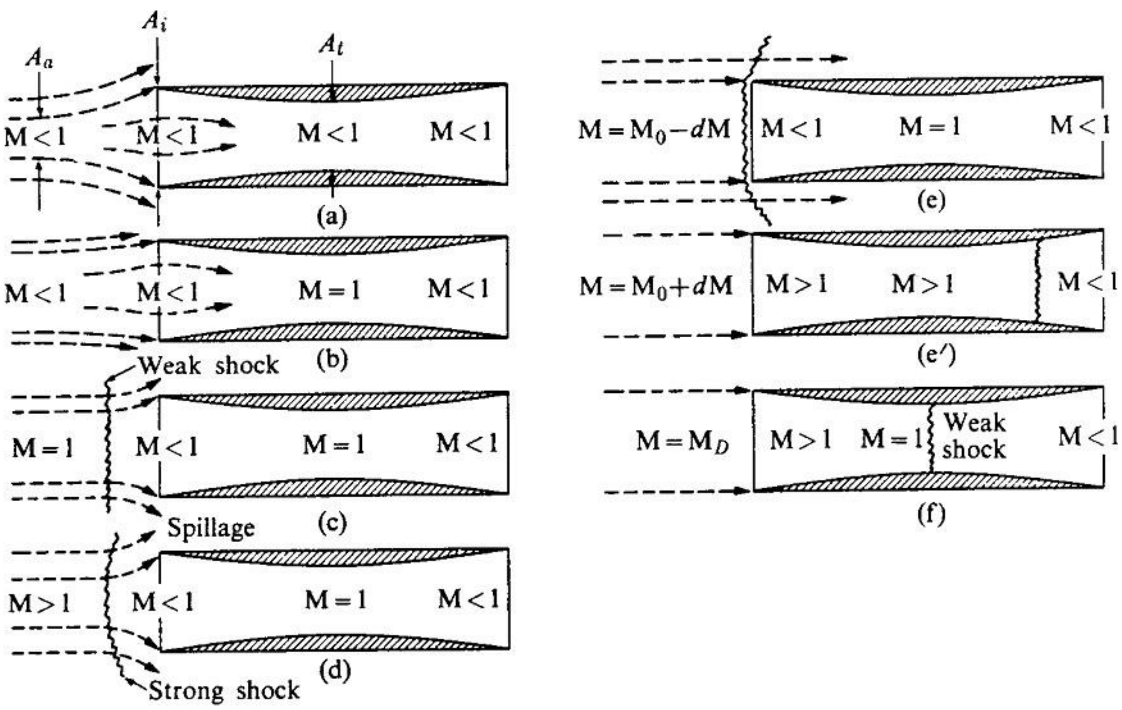


Fig. 10.26 External compression inlet and flow areas.

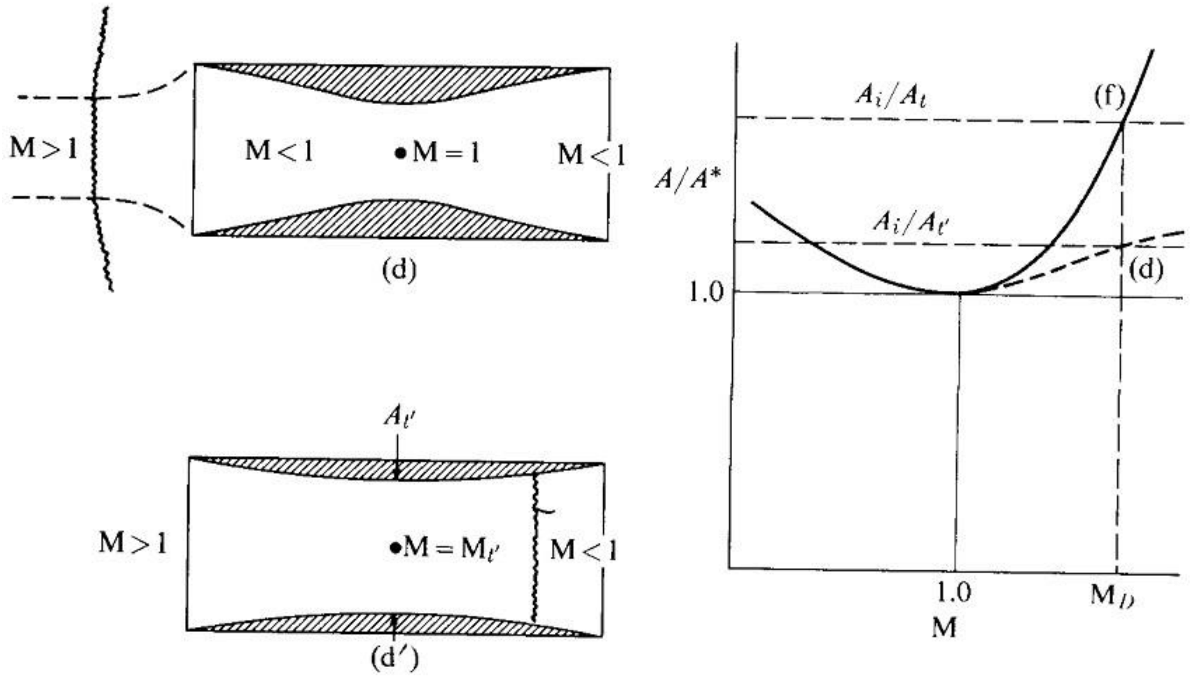
<u>Successive steps in the acceleration and over speeding of a one-dimensional supersonic inlet.</u>



Condition (a) illustrates low subsonic speed operation, for which the inlet is not choked. In Condition (b), though the flight velocity is still subsonic, the flow is assumed to be accelerated to sonic velocity at the minimum area A_t and the inlet mass flow rate is limited by the choking condition at A_t. Once the shock is established, the flow entering the inlet is no longer isentropic. Hence when the design Mach number of the aircraft is first reached, as at condition (d), the "reversed isentropic nozzle" mass flow cannot pass through the throat area A_t. At the Design Mach number , the inlet is capable of ingesting the entire incident mass flow without spillage. The shock position will be just on the lip of the inlet, as in condition (e), and a slight increment in speed, as to condition (e'), will cause the shock to enter the convergence. Since a shock cannot attain a stable position within the convergence it will move quickly downstream to come to rest within the divergence, at a position determined by downstream conditions. Having thus attained isentropic flow in the inlet, the Mach number may be reduced from M_0 to M_D, as at condition (f). At exactly the design speed, the throat Mach number would be just unity and isentropic deceleration from supersonic to sub sonic flow would exist. Even for this simplified model, however, this condition.

The Starting Problem / Shock swallowing by area variation

Internal supersonic deceleration in a converging passage is not easy to establish. In fact design conditions cannot be achieved without momentarily over speeding the inlet air or varying the diffuser geometry. This difficulty is due to shocks that arise during the deceleration process, while we examine the starting behavior of a converging-diverging diffuser



An inlet having A_i/A_t, greater than 1 ($A_i > A_t$) will always require spillage upon reaching supersonic flight velocities, since A_a/A_t, will always pass through a minimum of 1 just as sonic flight velocity is attained. It is necessary to perform some operation other than simply accelerating to the design speed in order to "swallow" the starting shock and establish isentropic flow. Over speeding is one such operation, but there are others.

If over speeding is not it might be possible to swallow the shock by a variation of geometry at constant flight speed. The principle is easily seen in terms of simple one-dimensional analysis.

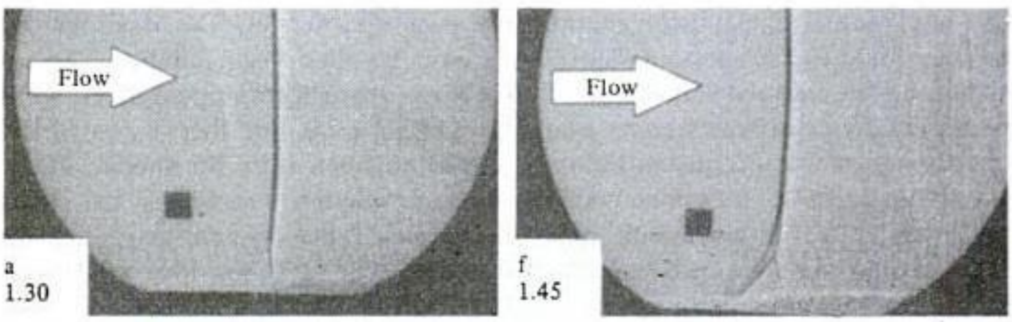
Suppose the inlet is accelerated to the design Mach number M_D with the starting shock present, as at point (d) in the above figure, if the actual area ratio can be decreased from A_i/A_t to the value that can ingest the entire inlet flow behind the shock, the shock will be swallowed to take up a position downstream of the throat. This variation would normally involve a momentary increase of throat area from A_t to a new value that we will call A_t* Having thus achieved isentropic flow within the convergence, the throat Mach number M, is greater than 1, and a relatively strong shock occurs farther downstream. Completely isentropic flow can then be achieved by returning the area ratio to its original value, while the operating conditions move from (d) to (f).

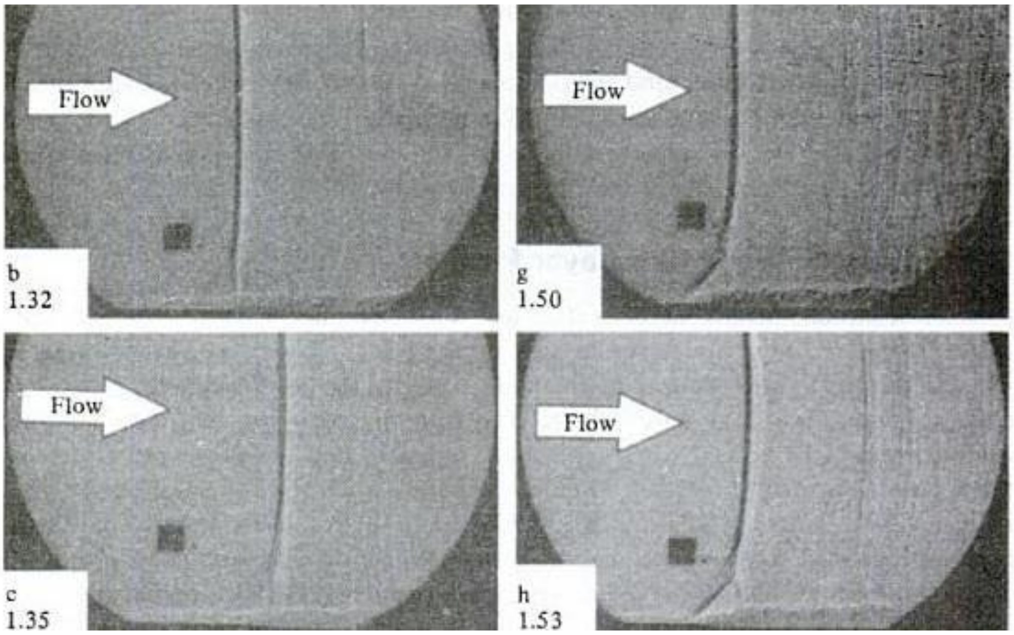
If the shock should undergo a momentary motion into the converging section of the diffuser the shock Mach number will be lowered and the down-stream stagnation pressure increased. This will increase the mass flow through the diffuser throat, lowering the density and the static pressure downstream of the shock. To accommodate this, the shock must move further down the converging section. From these arguments there is no location in the converging section at which the shock will be stable so the shock will move through the throat. If no adjustments are made in conditions downstream of the diffuser, the shock will move to a location in the diverging section of the diffuser at an area corresponding to the test section area, where it will then be stably positioned. This process is known as **swallowing the shock.** Once it occurs the shock can be positioned by changing the operating conditions of the exhauster.

In practice the shock must be maintained somewhat downstream of the diffuser throat because the shock is unstable in the converging part of the diffuser. If the shock moves upstream slightly, the shock Mach number increases, increasing the stagnation pressure loss and decreasing the mass flow capacity of the diffuser throat.

The Shock—Boundary Layer Problem

Across a shock wave of appreciable strength, the boundary layer separates, and this separation may have a large effect on the structure of the shock. Consider the interaction of a weak shock with a boundary layer .The following figure shows a series of schiieren photographs of the interaction of a shock wave and a boundary layer for upstream free-stream Mach numbers.





For M > 1, the shock wave near the boundary layer has taken up a “lambda” shape, with an oblique shock reaching ahead of the main shock location. Because a large fraction of the boundary layer is subsonic, the pressure rise due to the shock is sensed (near the wall) some distance ahead of the main shock wave. This upstream pressure gradient causes the boundary layer to grow rapidly,

As the free-stream Mach number increases above 1.25, the boundary layer thickens very rapidly under this lambda shock system, and causes the boundary layer to separate. The pressure gradient near the wall has become too large for the slow-moving fluid near the wall to continue moving in the main flow direction.

The above figure shows the effect of a strong shock in a diverging duct. The interaction between the “shock” and the boundary layer results in Large separation zones .Large separation zones cause a highly distorted, and probably unsteady, flow field that may require an axial distance of 10 duct widths or more to return to reasonable uniformity of flow. The flow field disturbances and distortions shown in above figure would have seriously harmful effects on the behavior of a compressor or combustor placed immediately downstream.

Unless one makes a strenuous effort to remove the wall boundary layer, strong shocks may have disastrous effects on duct flow. If a shock wave must be placed in a supersonic stream of given Mach number, then:

a. An oblique shock is much better than a normal one because the pressure rise is less;

b. The shock should interact with the wall at the point where the boundary layer is thinnest — preferably at the leading edge for the simple diverging inlet

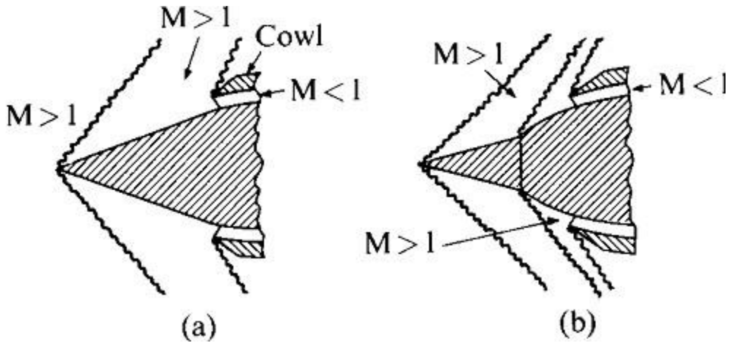
External Deceleration

External deceleration must occur upstream of the inlet plane in order to reduce the Mach number of the normal shock to a suitable value. The simplest and most practical external deceleration mechanism is an oblique shock or, in some cases, a series of oblique shocks. Though such shocks are not isentropic, the stagnation pressure loss in reaching subsonic velocity through a series of oblique shocks followed by a normal shock is less than that accompanying a single normal shock at the flight velocity. The losses decrease as the number of oblique shocks increases, especially at high flight Mach numbers.

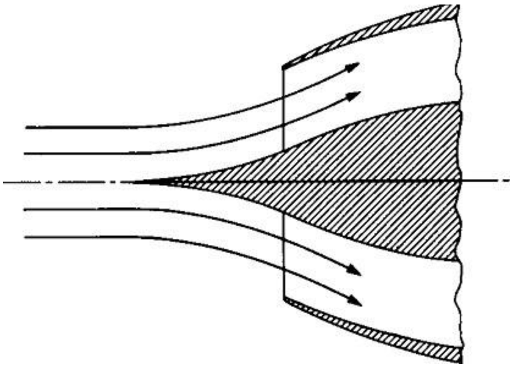
In the external compression process, shocks and boundary layers may inter act strongly, so that it is highly desirable to locate the oblique shocks at points where boundary layers are absent

The shape, size, and the number of the oblique planes influences the normal shocks

Considering the typical single oblique shock system and the double oblique shock systems. The double shock systems theoretically give better performance. . If the deceleration had been achieved by a single normal shock, the overall stagnation pressure ratio would have been only 0.33. But in case of double oblique shock systems the overall stagnation pressure ratio would have been only 0.875

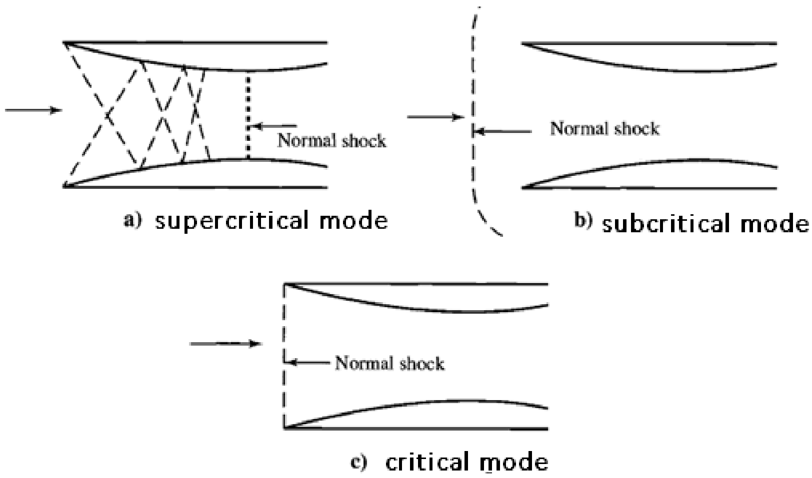


The following figure provides a ideal geometry to achieve low losses, while at the same time avoiding the starting problems of an internal convergence.



However, several practical difficulties would be encountered in the operation of such an inlet. This geometry, like that of the isentropic internal flow diffuser, would function properly at only one Mach number, and performance would be very sensitive to angle of attack. Furthermore, the boundary layer along the curved surface, unlike that along plane or conical surfaces, would be subject to a high adverse pressure gradient, which might cause separation. Finally, for high flight Mach numbers it would be necessary that the flow turnthrough large angles before reaching sonic velocity

The Flow Stability Problem



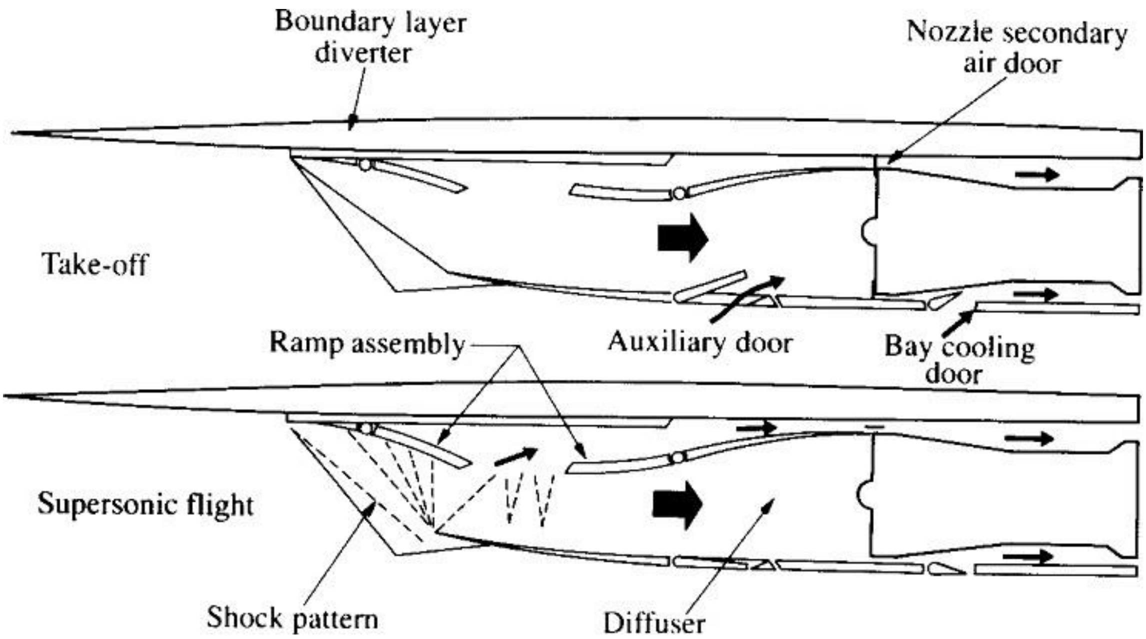
Typical Modes of Inlet operation

. Consider a fixed geometry inlet designed for shock-free operation at Compressible internal flow. At supersonic Mach numbers below the design value the inlet cannot pass the flow in the upstream stream tube and the excess must be diverted around the inlet. A shock therefore stands in front of the inlet, as in Figure (a). This mode is known as subcritical mode

As the Mach number is increased towards $\mathbf{M_D}$ **,**the corrected flow per unit area of the incoming stream decreases, reducing the flow that must be spilled round the inlet, and allowing the shock to move closer to the inlet. At the design Mach number, the shock will sit on the inlet lip. In this position it is unstable, because a small perturbation that moves it into the inlet causes a decrease in shock Mach number, this mode is known as Critical mode

With achieving shock swallowing in the diffuser, the consequence of the transient is shock motion through the throat to a downstream position determined by the variable nozzle. To achieve the best recovery, the nozzle is adjusted to position the shock at the throat. The mode is known as supercritical Mode.

Working Principle of Supersonic Inlet of Concorde Aircraft



The above figure shows the design of the two-dimensional intake adopted for the Concorde aircraft, whose design flight Mach number is 2; the Concorde is also required to cruise over certain land areas at subsonic speeds. The Figure shows the intake geometry during takeoff; here the ramp assembly is raised to allow as much air as possible to the engine. Shock waves are of course absent as the air enters the engine with a Mach number of about 0.5. As the Concorde reaches a flight Mach number of about 0.7, the auxiliary door closes. Above a flight Mach number of 1.3, the ramps are progressively lowered; the forward ramp controls the position of the oblique shock waves that decelerate the airstream from supersonic to subsonic speeds at the engine intake.