

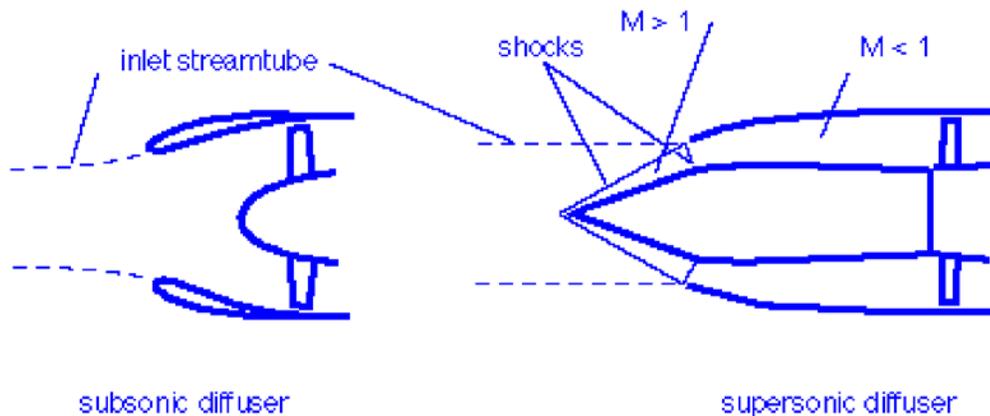
16.50 Lecture 22

Subjects: Inlets or Diffusers

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 M_0 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.

Inlets or Diffusers

These two titles are used interchangeably for the component that captures the oncoming propulsive streamtube and conditions it for entrance to the engine. The function of the inlet is to adjust the flow from the ambient flight condition, to that required for entry into the fan or compressor of the engine. It must do this over the full flight speed range, from static (takeoff) to the highest M_0 the vehicle can attain. Comparing the simple diagrams of subsonic and supersonic inlets, we can appreciate that the subsonic inlet has the simpler task:



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For any inlet the requirements are two:

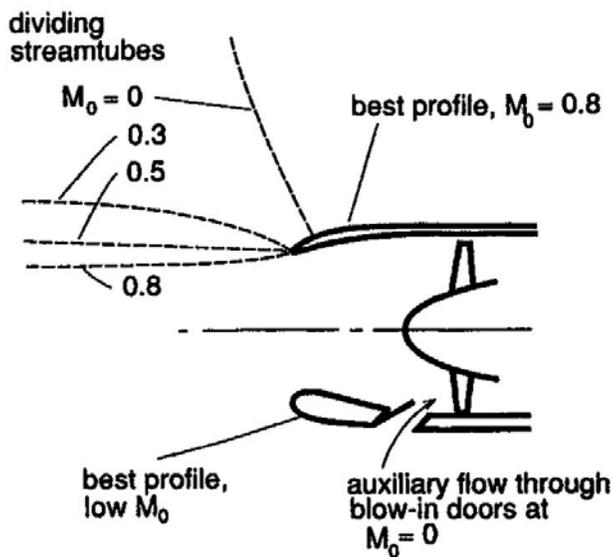
- a) To bring the inlet flow to the engine with the highest possible stagnation pressure. This is measured by the Inlet Pressure Recovery, π_d .
- b) To provide the required engine mass flow. As we shall see the mass flow can be limited by choking of the inlet.

Subsonic Inlets:

The subsonic inlet must satisfy two basic requirements:

- Diffusion of the free-stream flow to the compressor inlet condition at cruise.
- Acceleration of static air to the compressor inlet condition at takeoff.

There is a compromise to be made because a relatively thin "lip" aligned with the entering flow is best for the cruise condition. This is to avoid accelerating the flow, already at a Mach number of the order of 0.8, to supersonic speeds that will lead to shock losses. But a more rounded lip will better avoid separation for the takeoff condition because the air must be captured from a wide range of angles:



The minimum area, A_1 is set to just avoid choking ($M = 1$) in the inlet when M_2 has its largest value (this is set by the maximum blade Mach number and the blade shape).

A well designed subsonic inlet will produce a stagnation pressure recovery π_d in the order of 0.97 at its design condition.

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Supersonic Inlets:

At supersonic flight speeds the pressure and temperature rise in the inlet can be quite large. For the best thermodynamic efficiency it is important that this compression be as nearly reversible (isentropic) as possible. At a flight Mach number of 3 the ideal pressure ratio is

$$\delta_0 \equiv \frac{P_{t0}}{P_0} = \left(1 + \frac{\gamma-1}{2} M_0^2\right)^{\frac{\gamma}{\gamma-1}} = 36.7$$

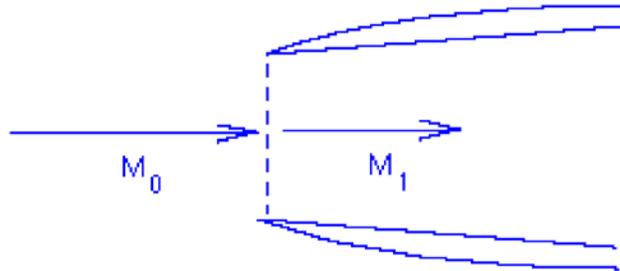
while the temperature rise is

$$\theta_0 = \frac{T_{t0}}{T_0} = 1 + \frac{\gamma-1}{2} M_0^2 = 2.8$$

For the turbojet cycle the compression is partly in the inlet and partly in the compressor:

Normal-Shock diffuser:

All existing compressors and fans require subsonic flow at their inlet with $0.5 < M_2 < 0.8$ at high power conditions. So the inlet must reduce the flow Mach number from $M_0 > 1$ to $M_2 < 1$. The simplest way to do this is with a Normal Shock.

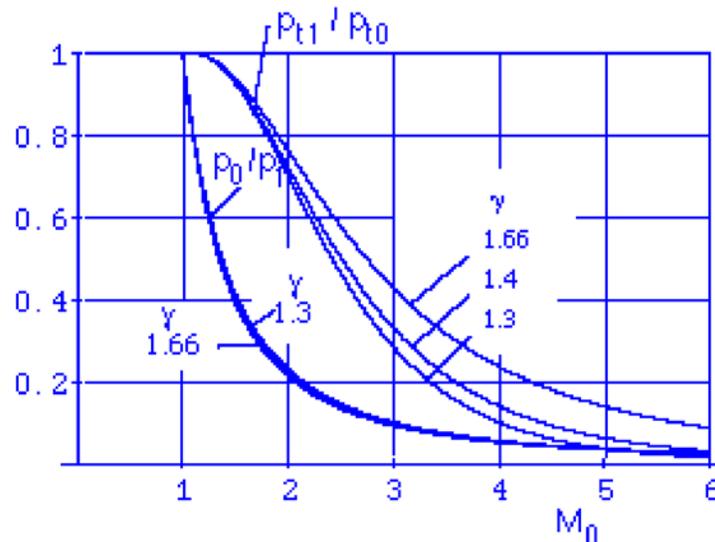


Here $M_1 < 1$ is entirely determined by M_0 , according to the normal shock relation

$$M_1^2 = \frac{1 + \frac{\gamma-1}{2} M_0^2}{\gamma M_0^2 - \frac{\gamma-1}{2}}$$

The stagnation and static pressures are also determined by M_0 :

$$\frac{p_{t0}}{p_{t1}} = \left(\frac{2\gamma M_0^2}{\gamma+1} - \frac{\gamma-1}{\gamma+1} \right)^{\frac{1}{\gamma-1}} \left(\frac{(\gamma-1)M_0^2 + 2}{(\gamma+1)M_0^2} \right)^{\frac{\gamma}{\gamma-1}} \quad \frac{p_1}{p_0} = 1 + \frac{2\gamma}{\gamma+1} (M_0^2 - 1)$$

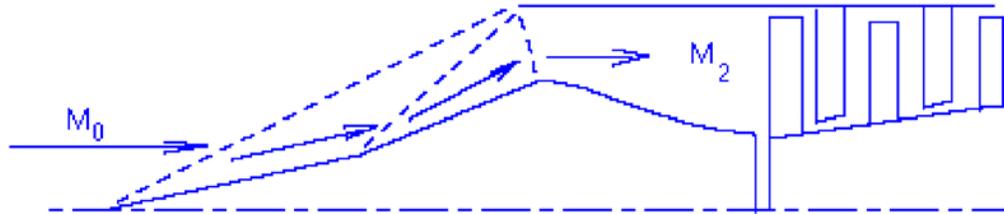


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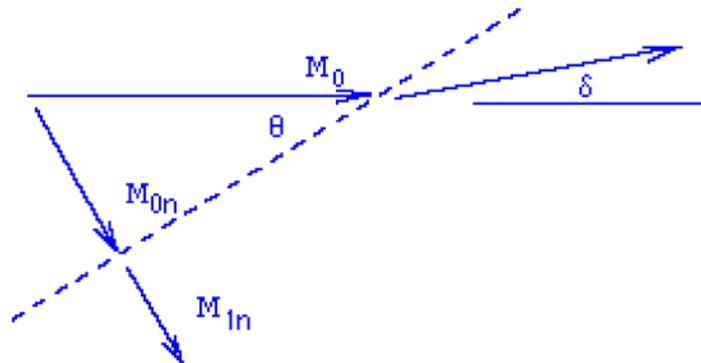
For low supersonic speeds, such diffusers are adequate because the stagnation pressure loss is small, but at $M_0 = 2$, $p_{t2}/p_{t0} \approx .71$, a serious penalty, and at $M_0 = 3$ $p_{t2}/p_{t0} \approx .32$. For example the F-16 fighter has a simple normal shock diffuser, while the F-15 has an oblique shock diffuser such as will be discussed next.

Oblique - Shock diffusers:

The losses can be greatly reduced by decelerating the flow through one or more oblique shocks, the deflection and the pressure rise of each being small enough to be in the range where the stagnation pressure ratio is close to unity. It is very important to understand that an Oblique Shock is in fact just a normal shock standing at an angle to the flow.



All of the change across the shock takes place because the Mach number normal to the shock is larger than unity. The velocity normal to the shock decreases with consequent increases in temperature and pressure, but the velocity parallel to the shock is unchanged:



M_{1n} is given in terms of M_{0n} by the same relation given for M_1 as a function of M_0 . But M_{0n} can be made close to 1. The condition for a weak or sound wave is just $M_{0n} = 1$, or

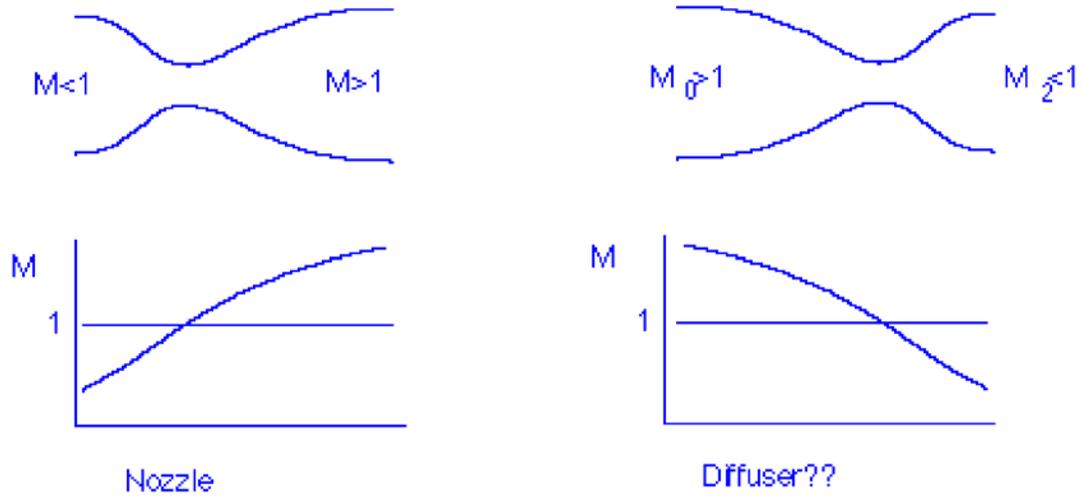
$$\sin\theta = \frac{1}{M_0} \quad (\text{weak compression})$$

By choosing the wedge angle (or deflection angle) δ we can set the shock angle.

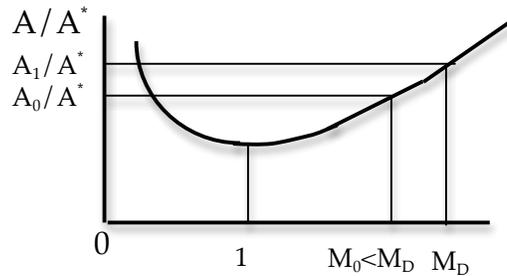
A series of weak oblique shocks, for each of which the M_n is near unity, hence all lying in the range of small p_t loss, can yield an efficient diffuser.

Diffusers with internal contraction

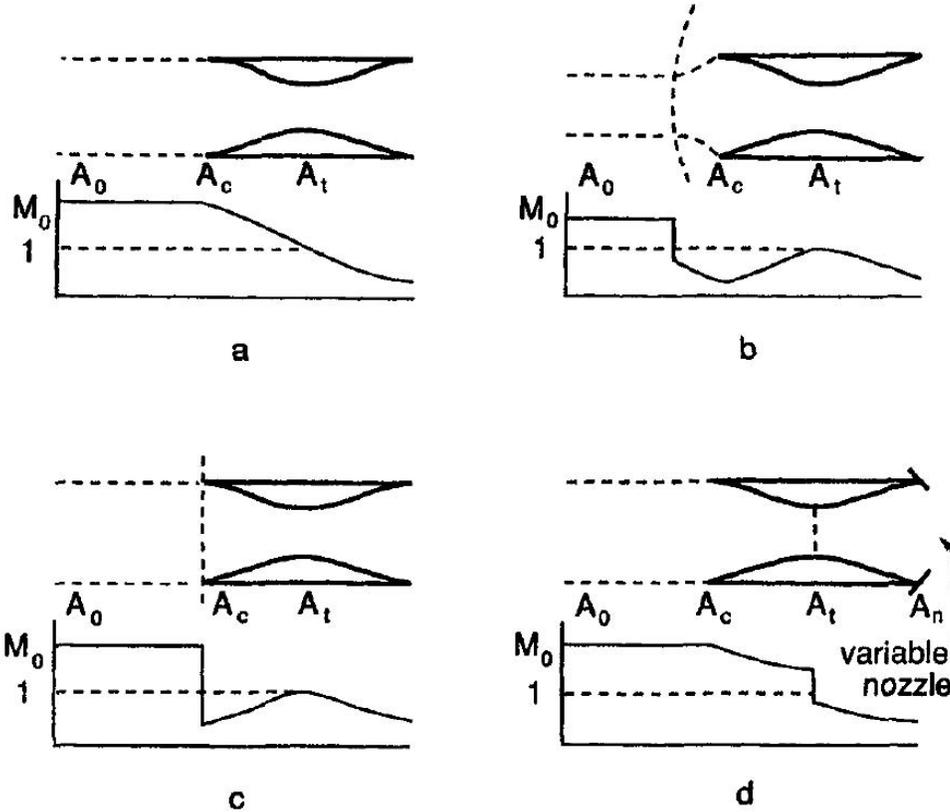
One might ask why we do not just use a convergent-divergent nozzle in reverse as an inlet:



This would work at one design Mach number, the one for which the isentropic area ratio between the incoming supersonic flow and the sonic throat is exactly the as-built area ratio A_1/A_{throat} . But during the acceleration to this Mach number the fully supersonic flow cannot be established in the inlet without varying the geometry. To see this, imagine the inlet flying at M_0 , lower than the design Mach number. The flow will look as depicted in the top right diagram below. This is because at the lower M_0 the flow area that would decelerate isentropically to sonic at the throat is smaller than the built A_1 :



So, if the flow did arrive undisturbed at the inlet, it could only occupy a fraction of it, the rest of the flow into the frontal area A_1 having to be somehow “disposed of”. This “spillage” is accomplished by the detached normal shock; behind it the flow is subsonic and it can turn around the inlet. Unfortunately, the shock at the full flight Mach number is very lossy, and it is not practical to simply force the plane to continue accelerating to the design condition (there may not even be enough thrust left to do it). What can be done is to manipulate the geometry to swallow the shock and reduce its strength. This is called “starting” the diffuser.



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To "start" the diffuser, meaning to pass the shock through the convergent portion, we must increase the throat area until the normal shock is just at the lip. At that point, any further small increase in throat area causes the shock to jump rapidly to a position in the divergent part of the nozzle where the area is again A_1 ; this is because the shock is unstable in the converging section, but stable in the divergent section. So far nothing has been accomplished, because the shock is still as strong as it was. But if the pilot now reduces fuel flow, the compressor moves down the operating line to a lower value of \bar{m}_2 and since the actual flow rate \dot{m} does not change any more, this means a higher total pressure p_{t2} into the compressor, and hence a weaker shock. This is accomplished by the flow through a repositioning of the shock to a location nearer the throat, on the supersonic side. The process can continue until the shock is almost at the throat (if too near, it may accidentally pop all the way to the front (an "unstart", which is a violent transient event). The final condition is depicted in the diagram at the bottom right.

This is an adequate solution for modest values of M_0 , but for $M_0 > 1.5$ the loss due to the shock at the throat is still excessive.

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