Aircraft Engines and Gas Turbines

second edition

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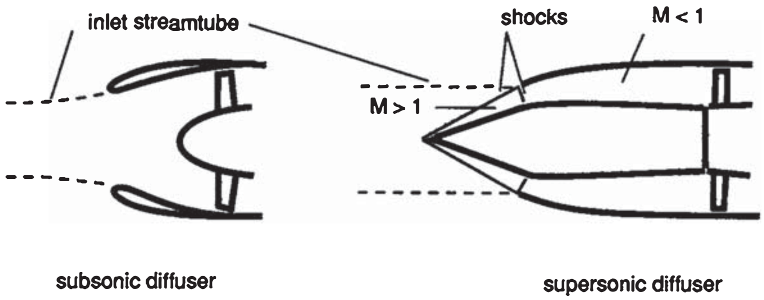

Figure 3.1
Schematic diagrams of diffusers.

be used in chapter 10. A different procedure will be followed here to retain as much as possible of the simplicity of the ideal cycle analysis. A mean c_{pc} and a mean γ_c will be defined for the compressor, and another pair of values, c_{pt} and γ_t, for the turbine. The first pair of values will be used for all processes occurring in the air ahead of the burner. The second set will be used for all processes in the combustion gases downstream of the burner. A mean specific heat, $\bar{c}_p$, will be defined for the range of temperatures in the burner.

3.2 Diffuser Pressure Recovery

As the engine airflow is brought from the free-stream conditions ahead of the aircraft to the conditions required at entrance to the engine, it may be smoothly decelerated as in the subsonic inlet at the left in figure 3.1, or it may be decelerated through shock waves, then further decelerated in a divergent passage as in the supersonic inlet shown at the right in the figure. In the subsonic inlets, viscous shear on the wall results in the growth of boundary layers that for this purpose may be thought of as regions in which the stagnation pressure of the fluid is low. Mixing this fluid with the inviscid core flow results in some reduction in the average stagnation pressure, below the value p_{t0} of the free stream. The ratio of this average stagnation pressure at the entrance to the engine (denoted p_{t2}) to the free-stream value will be termed the *diffuser pressure recovery* and denoted π_d. Thus,

$$\pi_d = \frac{p_{t2}}{p_{t0}} = \frac{p_{t2}}{p_0 \delta_0}.$$

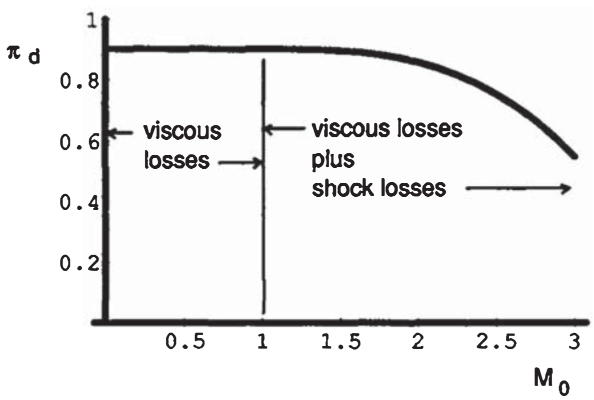

Figure 3.2
Typical diffuser pressure recovery as a function of flight Mach number.

When the flight velocity is supersonic, a further mechanism for loss of stagnation pressure exists in the compression through a series of shocks. Such losses vary markedly with M_0, and for $M_0 > 2$ they may constitute the principal source of diffuser pressure drop.

A typical variation of π_d with M_0 is shown in figure 3.2 for $0 < M_0 < 3$. For $M_0 > 1$, it includes the loss through a single oblique shock and a normal shock, as sketched in figure 3.1.

3.3 Compressor and Turbine Efficiencies

Losses in compressors and turbines originate primarily in regions of viscous shear on the blades and on the walls of the flow passages; these regions represent flows of lower stagnation pressure than the inviscid flow, as in the diffuser. The low-energy fluid becomes mixed into the base flow, and at the compressor (or turbine) outlet there is an average stagnation pressure and an average stagnation temperature.

Shock losses are also important in fan stages and in the first stage of modern transonic compressors.

For a given stagnation pressure ratio from inlet to outlet, the result of losses in a compressor is to require more energy input than for an ideal (isentropic) compressor. The efficiency is therefore defined as

$$\eta_c = \frac{\text{Ideal work of compression for a given } \pi_c}{\text{Actual work of compression for a given } \pi_c}.$$

The details of the design of a subsonic inlet depend on the way these compromises are struck. Modern computational tools make possible detailed calculations of the boundary layer behavior at all operating conditions, so that the compromises can be made quite rationally. With wing-mounted engines, they also deal quite effectively with the interaction between the external flow over the nacelle and the flow over the wing. An example of great success in this area is the nacelle installation for the Boeing 737-300. For this version of the 737, the original low-bypass JT8D engines were replaced by high-bypass CFM-56 engines of much larger airflow, consequently requiring larger-diameter nacelles. To avoid lengthening the landing gear it was necessary to mount the nacelles higher on the wing than early practice would have allowed without serious interference drag arising. Through the extensive use of computational fluid dynamics, a design was evolved with less interference drag than the original low-bypass installation.

When fully developed, a good inlet will produce a pressure recovery π_d between 0.95 and 0.97 at its optimum condition. For a more detailed discussion of subsonic inlet design, see reference 4.10.

4.2.2 Supersonic Diffusers

Flight at supersonic speeds complicates the design of the diffuser for three reasons. The most fundamental is the existence of shock waves, which introduce a wholly new loss mechanism that can lead to large decreases in stagnation pressure, even in the absence of viscous effects, and to bistable operation, with large changes in losses and in mass flow between the two modes. Much of the emphasis in discussions of supersonic diffusers has been on this aspect of their behavior. A second reason is that the variations in capture streamtube diameter between subsonic and supersonic flight for a given engine are very large (as much as a factor of 4 between $M_0 = 1.0$ and $M_0 = 3.0$), and an aircraft that is to fly at $M_0 = 3.0$ must also operate at $M_0 = 1$! Finally, as M_0 increases, the inlet compression becomes a larger fraction of the overall cycle compression ratio; as a result, the specific impulse and thrust per unit of mass flow become more sensitive to diffuser pressure recovery. This is especially evident for hypersonic air-breathing propulsion systems.

A typical diffuser, such as the one at the right in figure 3.1, is made up of a supersonic diffuser, in which the flow is decelerated by a combination of shocks and diffuse compression, and a subsonic diffuser, which reduces the